FIG. 1—CH-47A CHINOOK—AIRCRAFT WEIGHT: 8·9 TONS; LOAD: 10·5 TONS
The long A.S.I. probe is fitted for test purposes only.

HELICOPTER STABILITY AND CONTROL

BY

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Introduction

The handling qualities of a piloted vehicle assume greater significance as the diversity of application rises and each variant is produced in larger numbers. The establishment of suitable criteria and uniform characteristics for handling is essential for flight safety, for the training of pilots, for ease of conversion from one variant to another and for the diversion of attention to operational functions. In the past ten years the helicopter force in the Royal Navy has expanded to encompass anti-submarine warfare and commando operations, in addition to the original search and rescue duties, and it is evident that the future force may equal or exceed the fixed-wing strength. Increasing emphasis must therefore be placed upon handling qualities and upon standardization of automatic control facilities and displays. The purpose of this paper is to consider some of the factors affecting the stability of helicopters and to indicate current trends in auto-stabilization and flight path control.

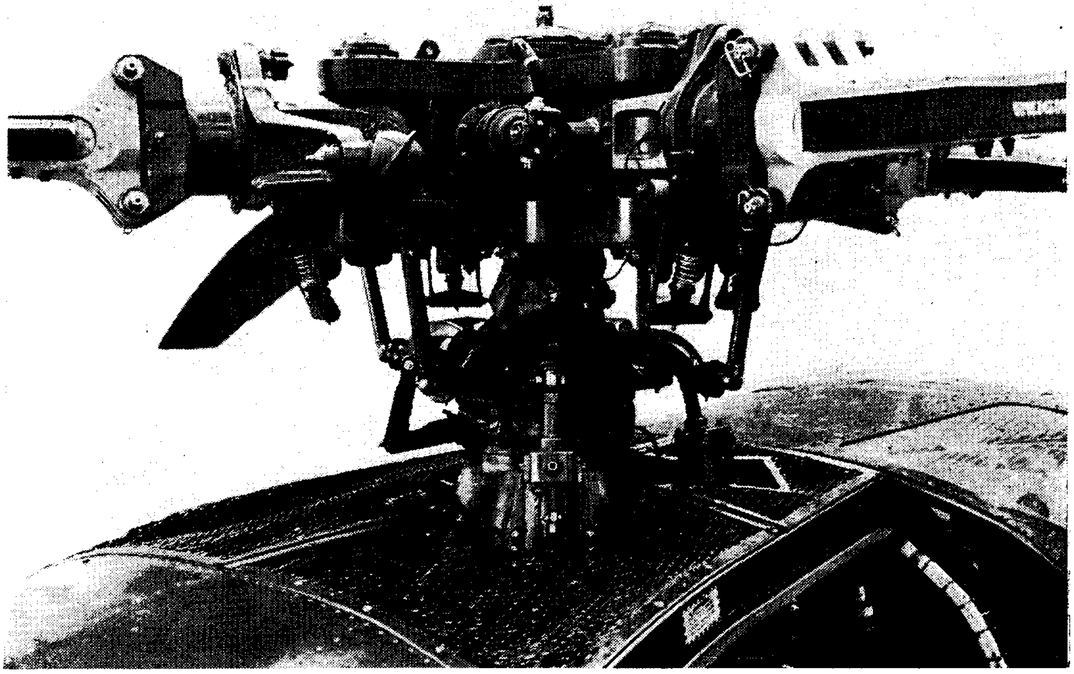


FIG. 2—THE SIKORSKY S.58 ROTOR HEAD AS EMPLOYED IN ALL MARKS OF WESSEX

Vehicle Configuration

Four configurations of helicopter are in current production but only two have been extensively developed, namely, the single rotor machines, pioneered by Sikorsky, and the tandems by Piasecki. The co-axial and the synchropter both represent attempts to exploit contra-rotation within a more compact geometry than can be achieved with the tandem: they present their own peculiar problems of stability and control, particularly in yaw, but these will not be discussed.

At first sight the power required to drive a tail rotor might appear to place the single rotor vehicle at a disadvantage. This is not the case, however, since the other configurations all suffer from deficiencies associated with main rotor interaction; in the tandem, the rear rotor operates in a downwash in forward flight. The tandem does possess an advantage in the hover in that its lower rotor disc loading enables it to lift greater loads. Complexity, stability characteristics, and vibration all weigh against this configuration and it is not until the all-up weights approach 20,000 lb that the lower loading, greater centre of gravity range and lower rotor tip speed ratio become significant. Where overall dimensions are important, a tandem with large overlap provides the more compact vehicle. This is particularly true in the folded condition since the modern single rotor machine, with its engines under the rotor head, must have the head set well back on the fuselage. The price paid for an overlap of 40 per cent diameter, as in the Chinook, may lie in the region of a 6 per cent loss in thrust and a 6 per cent increase in the power required; pitch and yaw stability and control are also adversely affected by the reduction in fuselage length.

The Articulated Rotor

The blades of the rotor are hinged to permit variation of pitch, thereby providing direct control over the magnitude of the lift vector. The wings of some missiles are hinged to provide similar control but the conventional aircraft achieves this in two stages, namely, development of a pitching moment at the tail and rotation of the whole fuselage. The practical effect of this difference lies in the nature of the loads which are reflected back on to the controls.

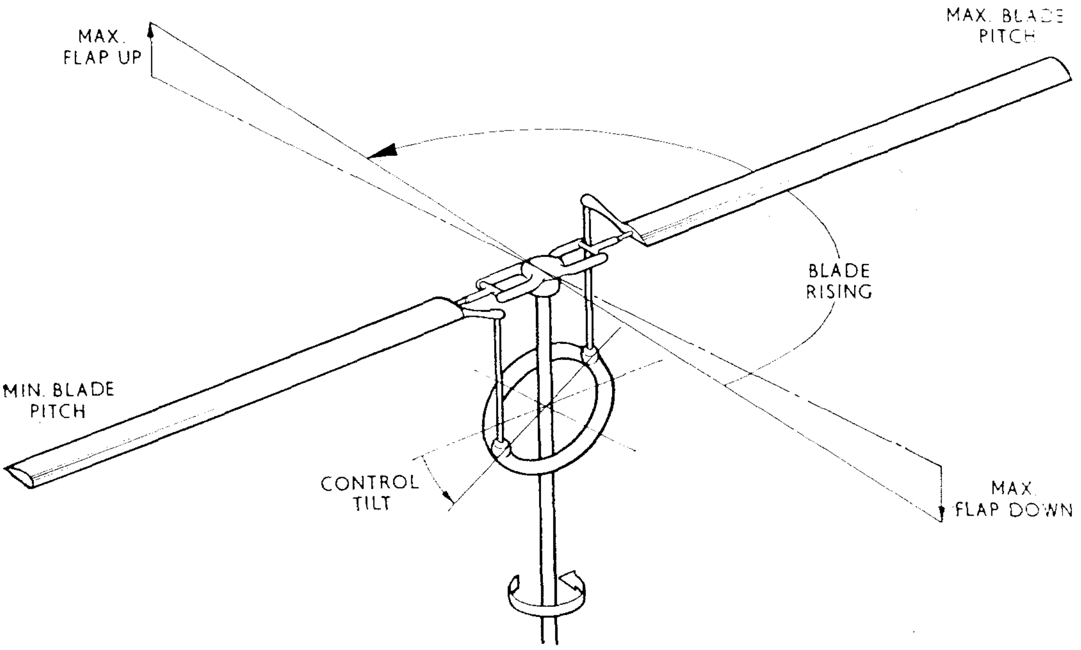


FIG. 3—EFFECT OF FLAPPING ON CYCLIC PITCH CONTROL

The blades are also hinged to permit flapping, thereby relieving the bending loads on the transmission arising from cyclic variations of lift. This greatly reduces vibration and leaves the blades free to seek instantaneous equilibrium between centrifugal force and lift. In turn, the flapping motion gives rise to oscillatory changes in the radius of gyration and hence to Coriolis accelerations which must be relieved by lead/lag hinges and dampers. The magnitude of the flapping motion can be reduced by deliberate cross-coupling of the blade pitch control mechanism, or inclination of the flapping hinge, so that the pitch of a rising blade is automatically decreased.

The fully articulated rotor is a central feature of most helicopters; research into rigid rotor dynamics continues in the United States and Germany, but articulation solves the problem of alleviating the severe oscillatory aerodynamic loads, gusts and retreating blade stall.

Flying Controls

Collective pitch is employed to vary total lift and cyclic pitch to control the direction of the lift vector. This vector can be regarded, to a first approximation, as being perpendicular to the tip path plane: that plane does not, however, tilt in phase with the changes in cyclic pitch. When the orbit of the control mechanism is displaced from the trim condition (FIG. 3), cyclic pitch is increased throughout the half revolution corresponding to the raised half of the control orbit; hence the blades must climb throughout this half revolution, reaching their highest point 90 degrees beyond that of the control orbit. The geometry of the controls is configured to cancel this effect but there is an inherent lag in control response of one-quarter of a revolution: this amounts to about 0·07 seconds for a rotor speed of 220 r.p.m.

Cyclic pitch is directly applied for pitch, roll, and translational flight in the single rotor machines. In these machines, tilting the rotor forwards initiates forward flight: as the vehicle accelerates the fuselage swings nose down tilting the transmission and accentuating the forward tilt of the disc. In the tandem, forward tilt on both discs provides a nose down pitching moment, but forward flight results in serious front-rotor downwash effects on the rear rotor. The tandem therefore employs a combination of cyclic and differential collective pitch to tilt the whole vehicle nose down for forward flight. Differential

Control	*Limit of control*	*Limit of force for breakout (threshold)*	
	Force (lb)	*Min. (lb)*	*Max. (lb)*
Longitudinal cyclic .	8	0·5	1·5
Lateral cyclic . .	7	0·5	1·5
Collective . . .	7	1	3
Rudder (yaw) . .	15	3	7

Control	*Control power in still air hover: angular displacement of vehicle (deg.) achieved one second after step displacement of control from trim*	
	Control displaced 1 inch	Max. control displacement
Longitudinal cyclic .	$45\sqrt[3]{W + 1000}$	$180\sqrt[3]{W + 1000}$
Lateral cyclic . .	$27\sqrt[3]{W + 1000}$ (in half	$81\sqrt[3]{W + 1000}$ a second)
Rudder (yaw) . .	$110\sqrt[3]{W + 1000}$	$330\sqrt[3]{W + 1000}$

W = Max, all-up weight.

FIG. 4—CONTROL FORCE AND POWER REQUIREMENTS OF MIL-H-8501A

collective pitch confers very powerful longitudinal control on the tandem; this can be a significant advantage in operations such as towing.

The tandem also employs differential lateral cyclic pitch to obtain yaw control. Although all control movements are presented to the pilot in the same manner in both vehicles, the principal significance of the coupling arrangements in the tandem lies the fact that the rotor can only possess a given range of blade pitch: this range must suffice for all composite control movements in pitch, yaw and collective. Yaw control is generally poor in the tandem owing to the large moment of inertia of the vehicle in this axis and the lag involved in main rotor response. The single rotor vehicle has a low moment of inertia and is limited only by the stalling characteristics of the tail rotor and the available torque.

Control loads are generally unacceptable for manual handling of the helicoptor, owing to non-linear hinge moments, vibration and cyclic variations at a frequency equal to the rotor speed multiplied by the number of blades. Manual control is feasible in some machines below 10,000 lb all-up weight: the early Belvedere achieved this at a much higher weight and accomplished the feat of placing the spire on Coventry Cathedral, in this condition. However, machines as small as the Sioux, at 3,000 lb, are now provided with power assisted controls, in which the loads are only partially reflected in order to retain some natural 'feel'. For ease of handling the control loads should exhibit a smooth opposing gradient with a predictable threshold, in the region of 1 lb (FIG. 4); as in the fixed wing aircraft, this is most commonly provided by

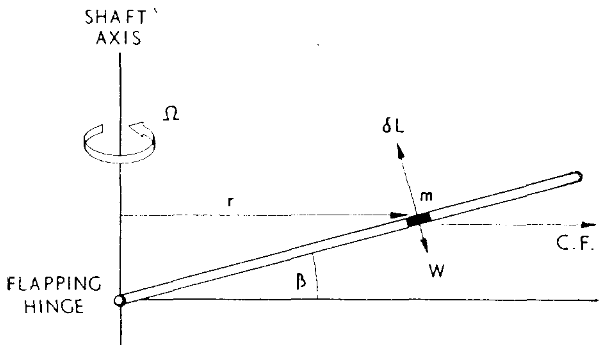


FIG. 5—THE PARAMETERS OF BLADE FLAPPING

divorcing the loads completely from the controls, through irreversible hydraulic servos, and applying springs to the control column to give artificial 'feel'. Manual reversion, in the event of hydraulic failure, is then impossible, hence duplication of the servos and their supplies is highly desirable. Powered flying controls are, in any case, a pre-requisite of automatic control systems.

Stability Analysis

Analysis of the motions of a helicopter in space is complicated by the fact that the normal degrees of freedom apply to both the rotor and the fuselage, thereby yielding to a total of twelve differential equations: theoretically this then implies 144 stability derivatives. The total problem is clearly unmanageable hence the longitudinal (pitch) plane is usually considered in isolation from the lateral (rolling) and directional (yaw) planes, despite inherent coupling. The latter two planes must be considered together in the case of the tandem owing to the particularly strong coupling which occurs in this machine. Reference 3 describes a method of obtaining a qualitative assessment of static and dynamic stability, in one plane, from the relevant equations of motion. A determinant is formed from the coefficients and this yields a characteristic equation to which Routh's criterion can be applied.

Fuselage forces and moments are difficult to describe owing to the irregular body shapes employed and the complex patterns of turbulent air flow resulting from downwash and the relative wind. Analysis is almost entirely based, therefore, on wind tunnel assessments. In general, the moments contributed by the fuselage have a de-stabilizing effect on the overall vehicle, in both pitch and yaw. Fins and tailplanes are frequently added to counter these effects in forward flight and these surfaces are more amenable to theoretical treatment.

For the purposes of stability analysis it is sufficient to consider small perturbations and hence to ignore many of the coupling effects. Since the motions considered are long-term in relation to the rotor speed, it is also valid to ignore the harmonics of blade motion: the overall vehicle cannot respond to them. For similar reasons a constant induced velocity over the rotor disc, as derived from simple momentum theory, is considered valid. Rotor speed is also assumed to be constant; in practice the larger vehicles incorporate a rotor governing system with a tolerance in the region of $\pm$ 5 per cent.

Rotor Stability

The blades of the articulated constant speed rotor are subject to lift, weight and centrifugal force. Invariant forces have no influence on stability hence weight and the steady state value of lift can be ignored. Cyclic variations of lift, arising from the influence of the relative wind, cause the blades to rise and fall, thereby introducing flapping accelerations and variations of blade centrifugal force. Coriolis accelerations and variations of blade drag give rise to secondary effects on blade motion in azimuth, and therefore on lift.

The principle parameters are shown in FIG. 5. Taking moments about the flapping hinge, and assuming that β is small,

The moment due to lift $= \delta L \times r$

$\delta L = \frac{1}{2}\rho v^2 S C_L \delta\alpha = \text{constant} \times \delta\alpha$

$\delta\alpha$ is the increment in angle of attack arising from flapping velocity, hence

$$\delta L \text{ varies as } \dot{\beta} \text{ and } \int_0^R \delta L r \, dr = K\dot{\beta} \text{ where K is an aerodynamic constant}$$

$$\text{The moment due to flap} = m\ddot{\beta}r \times r$$

$$\text{and} \int_0^R m\ddot{\beta}r^2 \, dr = I\ddot{\beta}, \text{ where I} = \text{moment of inertia}$$

$$\text{The moment due to C.F.} = m\Omega^2 r \times \beta r$$

$$\text{and} \int_0^R m\Omega^2 r^2 \beta \, dr = I\,\Omega^2\beta$$

$$\text{Hence, summing these forces, } \ddot{\beta} + \frac{K}{I}\dot{\beta} + \Omega^2\beta = 0$$

This describes a simple, second order system in which the natural frequency is equal to the angular velocity of the rotor and the damping depends upon the ratio of the aerodynamic and dynamic properties of the blade. It is evident therefore, that a cyclic, sinusoidal variation of lift, with maximum amplitude at an azimuth angle ψ, will result in maximum flap at $\psi + 90°$. Thus the effect of forward speed is to cause rearward flapping of the rotor.

This treatment explains the forward speed stability of a lifting rotor: as speed increases, the rotor flaps rearward, deflecting the thrust vector aft, thereby reducing forward speed. The rotor is also stable with respect to changes in vertical velocity since an increase in vertical velocity reduces the angle of attack of the blades. The rotor is, however, unstable with respect to changes in its overall angle of attack to the relative wind, in forward flight: if the rotor is tilted aft the advancing blades increase in angle of attack causing rearward flapping.

When an angular displacement is applied to the rotor shaft, swinging it about the hub, the rotor disc is constrained by inertia to remain in its original plane of rotation. The control orbit shifts, however, with the shaft and a cyclic pitch correction occurs, providing an unbalance of lift which causes the disc to follow the shaft. At a steady angular velocity of the shaft, ω, the lag of the rotor, φ, is approximately,

$$\frac{16\,\omega}{\gamma\Omega(1 + \frac{1}{2}\mu^2)} \text{ radians}$$

where γ = the ratio of aerodynamic and dynamic blade properties.

and μ = ratio of forward speed to tip speed (advance ratio).

This lag is the source of damping and, as such, is of considerable significance to the overall stability of a helicopter in pitch and roll.

It will be seen that μ has a comparatively small effect on the value of this expression; for a constant rotor speed, the normal range of forward speeds would not cause φ to vary by much more than 10 per cent. However, while the rotor is following the shaft, the thrust vector is not perpendicular to the tip path plane but leads this position by an amount dependent upon the expression, $\dfrac{\theta_C}{C_T/\sigma}$

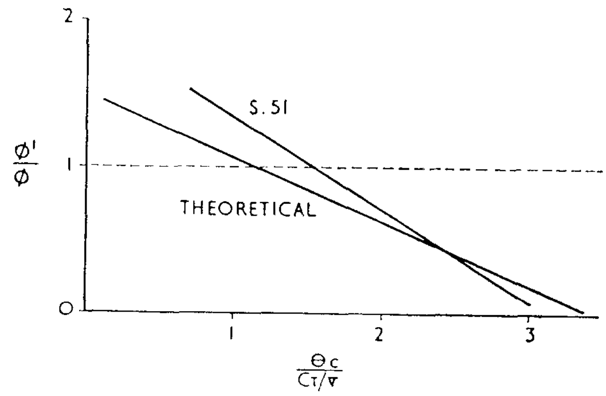


FIG. 6—LEFT VECTOR DEFLECTION

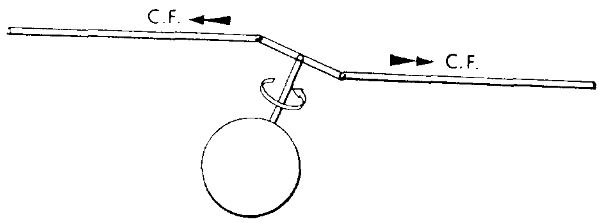


FIG. 7—DAMPING DUE TO OFFSET FLAPPING HINGES

Where,

θ_C = collective pitch
C_T = the thrust coefficient
σ = rotor solidity ratio (blade area/disc area)

and the actual lag of the vector, behind the shaft, is reduced to φ^1. FIG. 6, taken from Reference 6, shows the relationship between these functions. It can be seen that high collective pitch, required during a climb or for high forward speed, results in low damping; conversely, damping is high during autorotation.

Damping disappears as $\frac{\theta_C}{C_T/\sigma}$ approaches a value of about 3·3 and this can be a significant design limitation.

Offset flapping hinges are employed to improve centre of

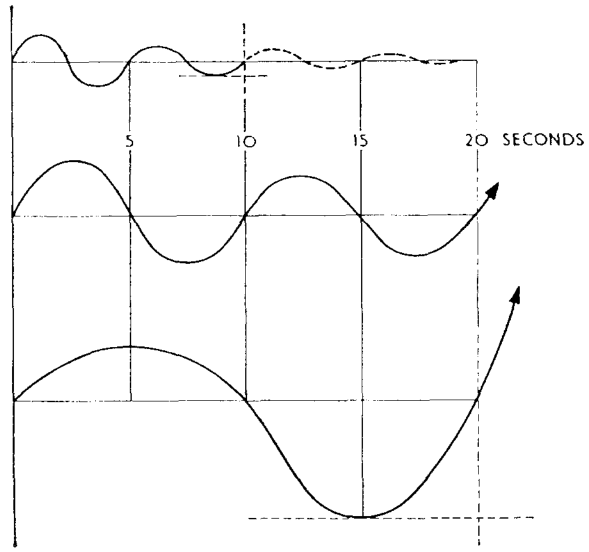


FIG. 8—DYNAMIC STABILITY—LONGITUDINAL PLANE—MIL-H-8501A REQUIREMENTS
Oscillations—Up to 5 secs. period—to reduce to half amplitude within 2 cycles. No residual
5–10 secs. period—to be lightly damped
10–20 secs. period—not to exceed double amplitude within one cycle

gravity range, but they are also a source of damping since the inertial forces on the blades apply a restraining torque to the shaft (FIG. 7). Damping also varies with the hub height above the vehicle c.g.; the vehicle pitches and rolls about the latter point, causing the rotor to be displaced horizontally, thereby generating flap back, as in the case of speed stability. Flapback also results from sideslip and, in this case, is known as 'effective dihedral'.

Stability of the Vehicle

The fuselage is generally statically unstable with respect to both speed and angle of attack. Although these effects can be partially countered by the addition of fins and tailplanes, the rotor characteristics are a prime source of static

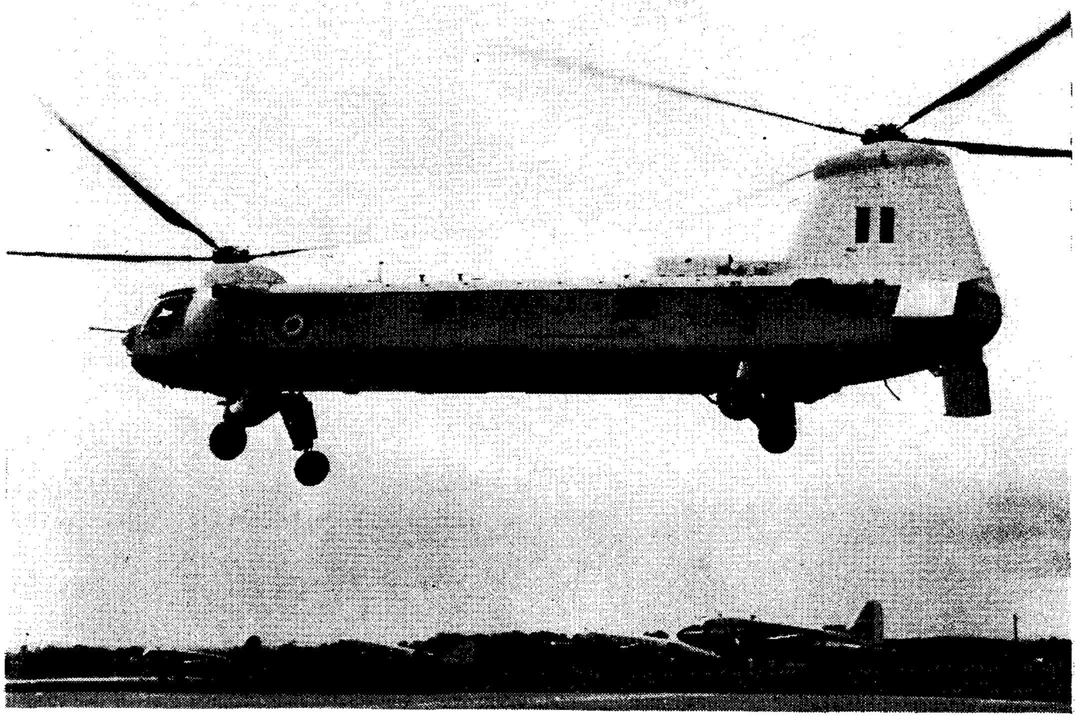


FIG. 9—THE BELVEDERE

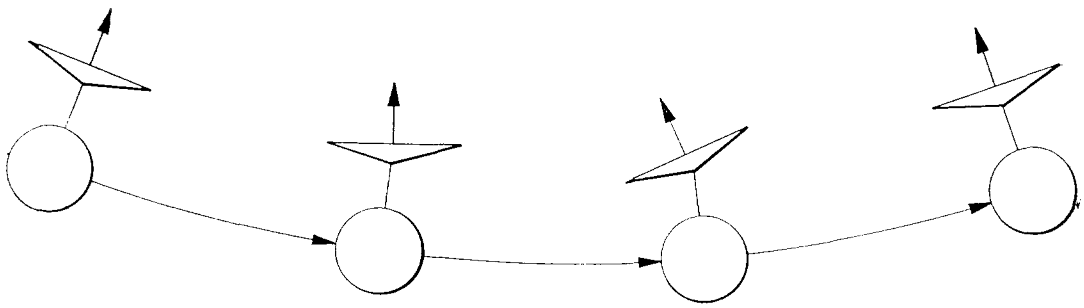


FIG. 10—OSCILLATION FOLLOWING DISTURBANCE FROM TRIM (UNCORRECTED BY FIG. 6)

stability. Dynamic stability cannot be achieved without the addition of artificial devices. Given static stability, a pilot can control dynamic divergence provided that it is of a reasonable period and does not diverge within the first cycle (FIG. 8). However, the pilot is obliged to maintain continuous, positive control over the vehicle and this is the fundamental difference between the handling qualities of the helicopter and the fixed wing aircraft. Many modern tandem machines cannot be handled without artificial stabilization: a notable exception to this is the Belvedere.

The pitch and roll stability characteristics of the single rotor machine in the hover differ only by virtue of the moments of inertia; FIG. 10 gives an indication of the oscillatory behaviour arising from a disturbance from trim. As for rolling of a fixed wing aircraft, a pure change of attitude results in neutral aerodynamic static stability, i.e., damping without any restoring force; however, sideslip, with 'effective dihedral', produces oscillatory motion. The pitch stability of the tandem in the hover is mainly derived from the stability of the rotors with respect to vertical velocity.

Stability in Forward Flight

The longitudinal dynamic instability of the single rotor helicopter in forward flight is characterized by an oscillation of 10 to 20-second period which diverges

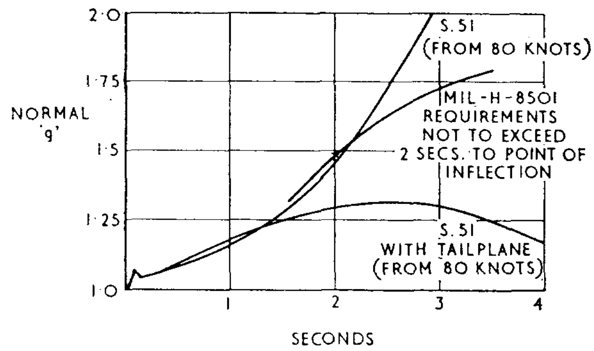


FIG. 11—NORMAL ACCELERATION—'PULL AND HOLD'

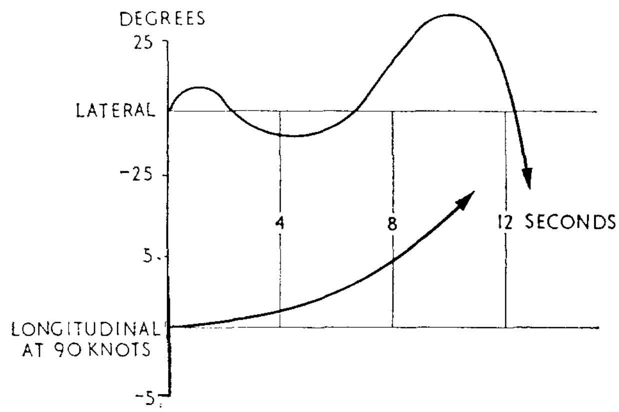


FIG. 12—LATERAL AND LONGITUDINAL INSTABILITY OF HRP TANDEM

completely during the second cycle. This motion can be easily contained by the pilot: it arises from the static instability of rotor and fuselage with respect to angle of attack, which overcomes the speed stability. The addition of a tailplane can be sufficient to cancel out this instability at high speed; at low speed all tail surfaces are ineffective owing to low dynamic pressure. The quick stop or 'pull and hold' manœuvre can be a handling problem in that it is difficult for the pilot to anticipate the nature of the build-up of normal acceleration following stick displacement. This is illustrated for the S.51 by FIG. 11 from Reference 7, which also indicates the improvement provided by a tailplane. Pilots can detect changes of 0·02g and react to this sensation for forward flight control.

The longitudinal instability of the tandem is more serious and is, in many cases, statically divergent, as indicated in FIG. 12 from Reference 8. This poses a severe handling problem and calls for artificial aids as an essential part of the vehicle. The behaviour arises from the effects of rotor interaction: the front downwash moves aft to the rear rotor in $\frac{1}{4}$ to $\frac{1}{2}$-second, its effect varying with induced velocity, wake angle and rate of climb. As forward speed increases the wake angle decreases, rear rotor angle of attack increases and a nose down pitching moment develops. The remedies are conflicting in that it is advantageous to have higher thrust from the front rotor at high speed, whereas higher rear rotor solidity is desirable at low speed. Swash plate dihedral, i.e., tilting the cyclic control orbits inwards, is effective only at low speed.

As in the fixed wing aircraft, a fin is essential for most helicopters to provide directional stability and damping. In a single rotor vehicle, the tail rotor adds to the damping in that the angle of attack of the blades changes favourably as the tail swings. Tandem directional stability is generally poor, dynamically divergent and heavily cross-coupled with the lateral plane, thereby giving rise to a 'Dutch roll' of characteristic period in the region of 8 seconds. (See FIG. 14 from Reference 9.) There are two principal sources of the coupling, firstly the nose down inclination of the inertial axes and secondly the inequality of lateral cyclic displacement of the front and rear rotors. Factors contributing to the instability are the large moment of inertia in yaw, the position of the centre of gravity, i.e., near the centre of the vehicle, the relatively low damping in roll arising from low disc loading and the short moment arm at which the fin can be placed. Lateral/directional instability becomes more serious as the rotor overlap is increased; it can be countered in some cases by decreasing effective dihedral or applying greater flapping hinge offset to the rear rotor but the principal remedy lies in the design fins, tailplanes and stub wings. The tailplane for the Belvedere passed through four stages of development involving dihedral, anhedral, wing tip fins and cranked span before the present

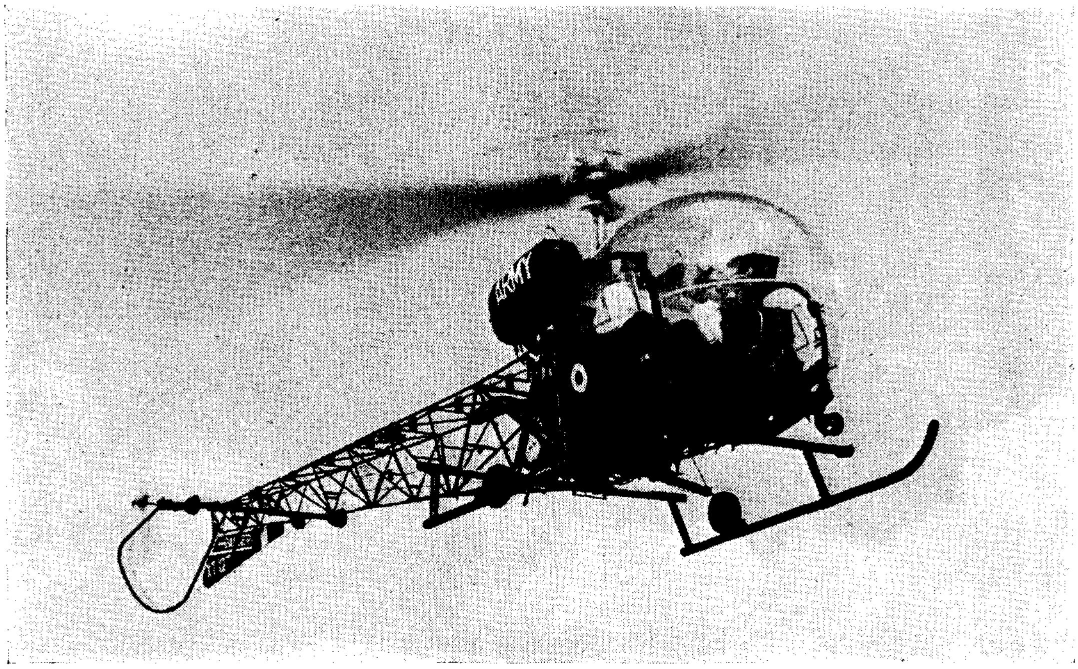


FIG 13—THE BELL 47G–3B1, TO BE BUILT UNDER LICENCE AS THE 'SIOUX'

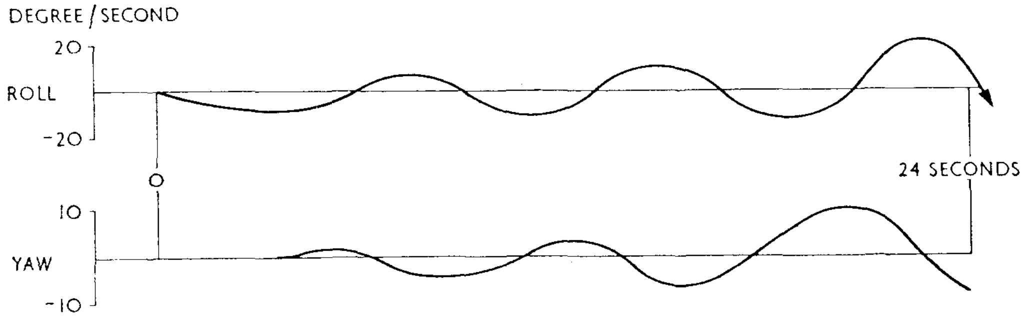


FIG. 14—'DUTCH ROLL' OF HRP TANDEM AT 70 KNOTS

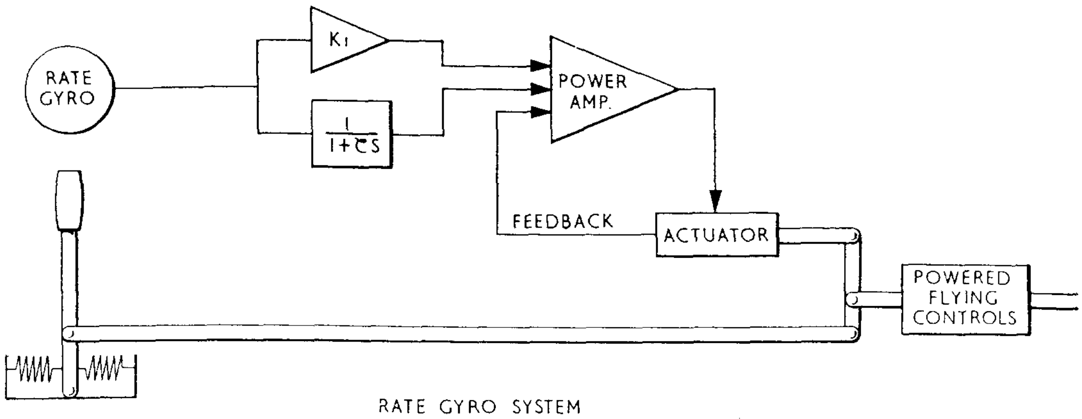
form was adopted. It can be argued that the low degree of directional stability in the tandem is advantageous for the purpose of cross-wind manœuvres.

Dynamic Stabilizing Devices

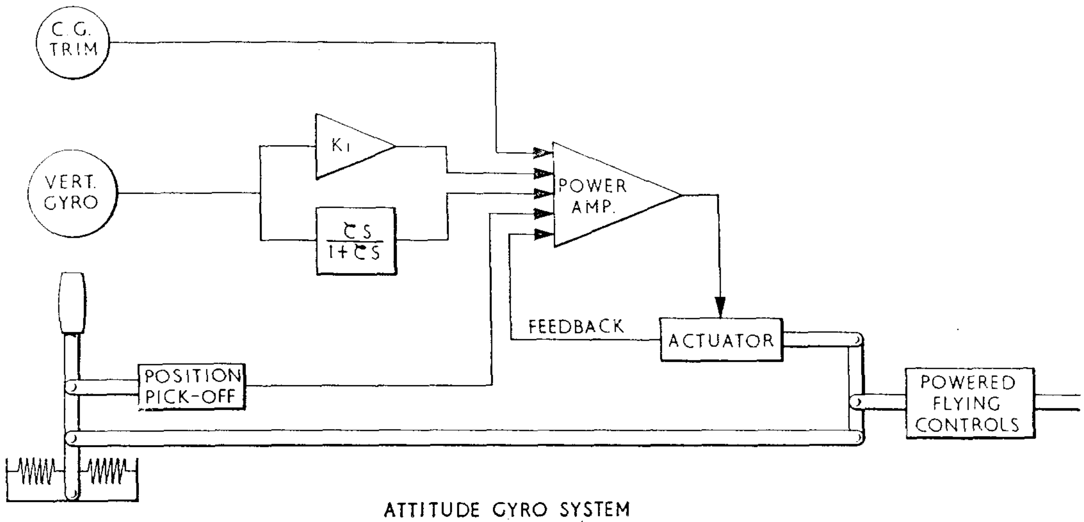
Three devices have been employed to improve rotor stability but they are principally applicable to the smaller two-bladed rotors. The Bell gyroscopic stabilizing bar is the most successful of these devices. The bar carries two weights at its extremities and is mounted below, and at right angles to, the two bladed rotor; it is hinged to flap about the shaft hence, when the shaft is tilted, it remains in the original plane of rotation. The bar is linked to the blade pitch mechanism so that cyclic pitch correction is applied to the rotor as the shaft swings. Dampers are linked to the bar to give a controlled rate of follow up. This device is fitted to the Sioux: the paths traced by its components are just visible in FIG. 13.

The Hiller control rotor employs two small paddle shaped aerofoils in place of the weights on the stabilizing bar. All cyclic control demands are linked to the control rotor, the main rotor blade pitch being varied by the flapping of the paddles. The system is reversible in that flapping of the main blades also imparts corrective pitch changes to the paddles, thus there is a form of closed-loop control of the main rotor motion.

The Kaman servo tab is a conventional trailing edge control employed in conjunction with torsionally flexible blades. Cyclic and collective blade pitch



RATE GYRO SYSTEM



ATTITUDE GYRO SYSTEM

FIG. 15

are applied by deflection of tabs operated by linkages passing down the inside of the blades. The linkage is configured to modify pitch in relation to flapping motion.

Auto-Stabilization

Auto-stabilization in a fixed-wing aircraft is only required to augment the inherent aerodynamic stability, whereas in the helicopter it must counteract the modes of instability discussed earlier. Since the fuselage is the principle de-stabilizing influence on the helicopter, the parameter to be controlled is fuselage attitude with respect to the horizon (θ in the longitudinal plane), and the control equation should be of the form,

$$b_1 = K_1\theta + K_2\dot{\theta} + K_3\ddot{\theta}$$

θ and $\ddot{\theta}$ can easily be detected by erected and spring-restrained gyroscopes respectively. Theoretically, any two terms can be derived from the third hence it should be sufficient to sense only one. In practice, only one term is sensed but the other two are imperfectly derived owing to the practical difficulties associated with electronic differentation and integration (FIG. 15). The principal significance of this deficiency lies in the fact that only the attitude gyro system can maintain a fixed fuselage attitude over the long term. However, with a good rate gyro system the deviations can lie within 1 degree over several minutes leaving the pilot with the simple task of long term trimming.

There is one fundamental difference between the two systems, and this concerns their axes of reference. The gravity erected gyro always reads errors from the local vertical, whereas the rate gyro reads instantaneous angular velocities of the fuselage axes; its rotor is constrained to follow these axes. This has an unfortunate effect, during a banked turn, in that the rate gyro experiences cross coupling between pitch and roll: it reads an apparent nose-up motion and demands a nose-down correction. Nose-up control correction in a banked turn is a fundamental requirement in fixed-wing aircraft, hence a pilot transferring to a 'rate stabilized' helicopter might not find this objectionable. It is, however, an undesirable feature, particularly when blind flying close to the sea.

The advantages of the rate gyro system are principally ones of engineering but are, nevertheless, considerable. The gyros are very much smaller, lighter and more rugged. The system also requires less overall authority over the aircraft, and this is of particular significance in limiting the effect of a 'runaway' failure. There is no requirement for matching of the stabilizer to the pilot's controls since there is no fixed space datum in the system: it merely stabilizes the aircraft about the position chosen by the pilot, after any integral terms, arising from a manoeuvre, have decayed to zero (generally, 2 to 3 seconds).

The attitude gyro system demands control corrections proportional to attitude errors from the local vertical, whereas the longitudinal cyclic control bears a non-linear relationship to attitude, dependent upon forward speed. For example, at speeds up to 10 or 15 knots the fuselage may actually swing nose up, due to downwash on the tail with the rotor tilted forwards. Furthermore, when changing speed, the stick can be displaced by large increments much more rapidly than the fuselage can follow. If the stabilizer is to stabilize the helicopter about any chosen attitude, there must be continuous cancellation of the gyro signals representing error between that attitude and the true vertical. For the reasons stated such cancellation is, however, subject to transient mismatching which causes the actuators to sit off-centre or even to saturate. This is one reason for increasing the overall authority of the system.

The problem of stick position cancellation has been overcome in the French SFIM autopilots by configuring the system so that the attitude signals are switched out whenever the pilot moves the control column. These systems sense rate only during manœuvres and the penalty is, therefore, loss of datum. When the pilot releases the controls the aircraft will be stabilized about whatever attitude he has then attained and not about the original trimmed condition.

In roll, stick position cancellation changes the nature of the basic control law of the helicopter: displacement becomes proportional to angle of bank in a turn whereas the unstabilized response approximates more nearly to a rate demand system. In yaw, attitude stabilization has no meaning since there is no datum to which the fuselage can be returned. Autopilots generally contain a heading 'lock' which can be released for manœuvring but, if the auto-stabilizer has a yaw channel it is provided purely to improve the damping in this plane and may, therefore, be fed by a rate gyro sensor.

Conventional compensation techniques are employed for stabilizing the minor servo loops and for countering resonances and improving damping of the servos, the flying controls and the aircraft. The actuators are electro-mechanical or electro-hydraulic and are normally introduced in series with the pilots inputs to the powered flying controls. Pilots generally prefer a dead beat, or critically damped, response but, with highly geared systems, some overshoot must be accepted: the decay time may be in the region of 3 seconds. Since the actuators of a stabilization system are essentially fast acting, the authority

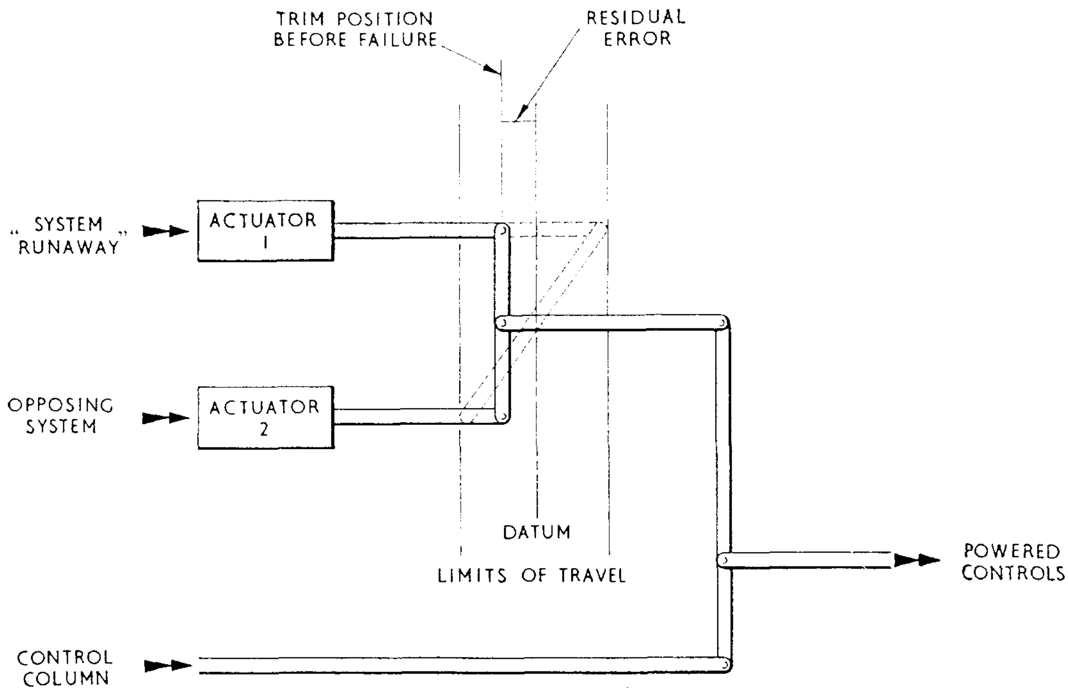


FIG. 16—'RUNAWAY' IN A DUPLEX SYSTEM

must be limited to avoid catastrophic failures; the range employed is commonly within $\frac{1}{4}$ of that available to the pilot. A duplex system, giving the mean output of the two channels, can provide some alleviation of failures but the effectiveness of the serviceable channel depends upon the trim position of the servos at the instant of failure (FIG. 16). In a duplex rate gyro system it is feasible to double the gain in the serviceable channel when the defective one is switched out; this restores the overall sensitivity at the expense of range of authority. An attitude gyro system cannot afford this loss of authority.

Application of the stabilizer to the helicopter must be integrated with any autopilot or flight control equipment. If such equipment is fitted then it will, of necessity, contain at least one vertical reference and also an independent set of full-authority actuators. The full advantages of the rate gyro system are therefore more evident in a vehicle which does not possess the additional facilities. The American Autonetics Stability Augmentation System is a rate gyro system applied principally to tandem helicopters as a part of the basic flying machine; since the system is essential for aircraft handling, simplicity and reliability are paramount considerations for these aircraft and, in the Chinook, the system is duplicated.

The Cruise Autopilot

Whereas auto-stabilizers are generally designed to govern the short term behaviour of a vehicle, autopilots are intended to provide facilities for long-term automatic flight. This differentiation has led to some authorities classifying the attitude stabilization system as a 'stick-steering autopilot'. In the cruise condition, however, the autopilot requires further information in addition to a vertical reference; it must be able to control heading, height and airspeed (FIG. 17). Heading lock can be achieved, without difficulty, by employing signals from a gyro-magnetic compass system but the measurement of height and airspeed presents problems particular to the helicopter.

The sensing of both dynamic and static pressure is complicated by the constantly varying conditions all round the fuselage, resulting from rotor downwash. No attempt was made to control airspeed in the Sikorsky/Lear HSS-1N system but this was not a serious deficiency in view of the fact that

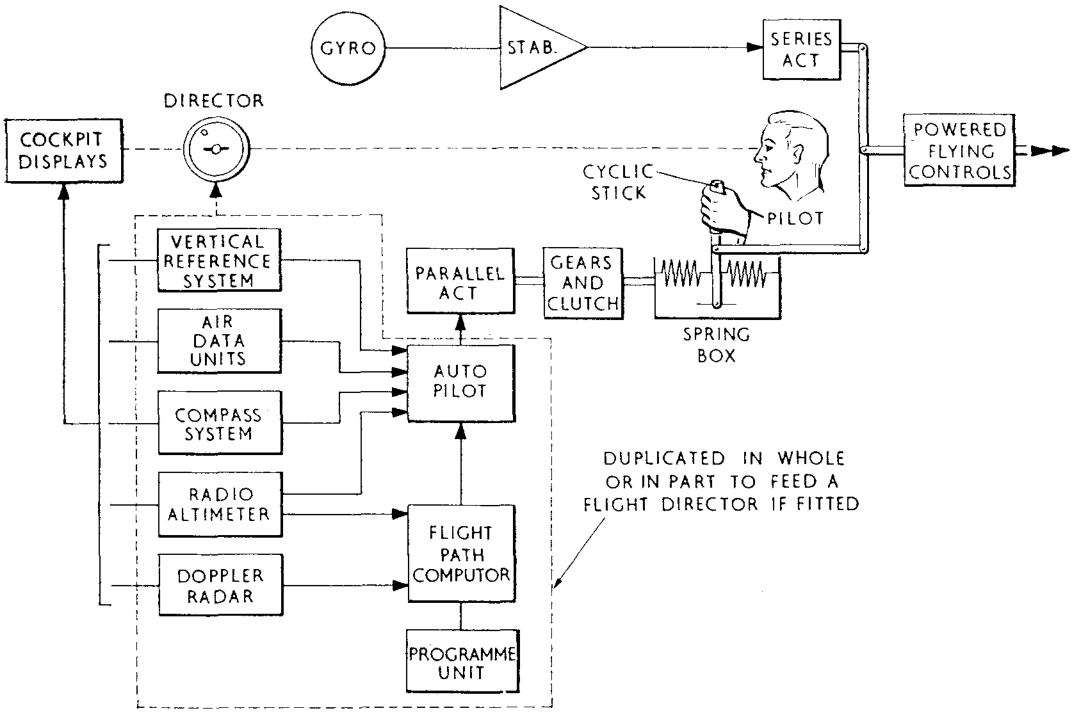


FIG. 17—A FLIGHT CONTROL SYSTEM

it could be maintained within reasonable tolerances merely by holding the appropriate pitch attitude. For the Wessex 3 system, currently under development, an airspeed lock has been designed and has necessitated the provision of a nose boom to clear the rotor wake; this is satisfactory at high speed but still presents problems below about 20 knots. Static pressure measurement has been used as a signal source for height holding but no entirely satisfactory position for the vent has yet been found and turbulence has dictated a rather low gain system with accuracy limited to $\pm$ 25 feet.

Instruments for the measurement of static and dynamic pressures at low speed and low altitude have yet to be developed to the required accuracy. Force balance air data systems are beset by problems of spring linearity and sensitivity, vibration and temperature stability. The alternative system, now being adopted, employs a specially designed synchro, of almost negligible driving torque, geared directly to the capsules.

Height can fortunately be measured by other means, namely, a radio altimeter, though this equipment actually measures terrain clearance. In the low-flying helicopter, in blind flying conditions, it is important that the equipment should give the best possible resolution close to the surface. No British equipment is at present available designed to change its sensitivity with height, as is the case with the American APN. 117. However, APN. 117 merely compresses the upper part of the range, by scaling down the sweep of the modulation on the transmitted signal, without giving any basic improvement in threshold or discrimination at low level. For the purposes of the height lock facility, all modern radio altimeters can provide signals accurate to within $\pm$ 2 feet, hence the value of the expanded scale lies only in presention to the pilot. One change, brought about by the blind landing requirements of civil airliners, is the introduction of altimeters giving current rather than voltage outputs; this signal current can be passed directly through the pilot's instrument before being coupled to the autopilot, thereby ensuring that the pilot reads the actual commands received by the autopilot and is made immediately aware of a failure. This arrangement is employed in the Trident and the Wasp.

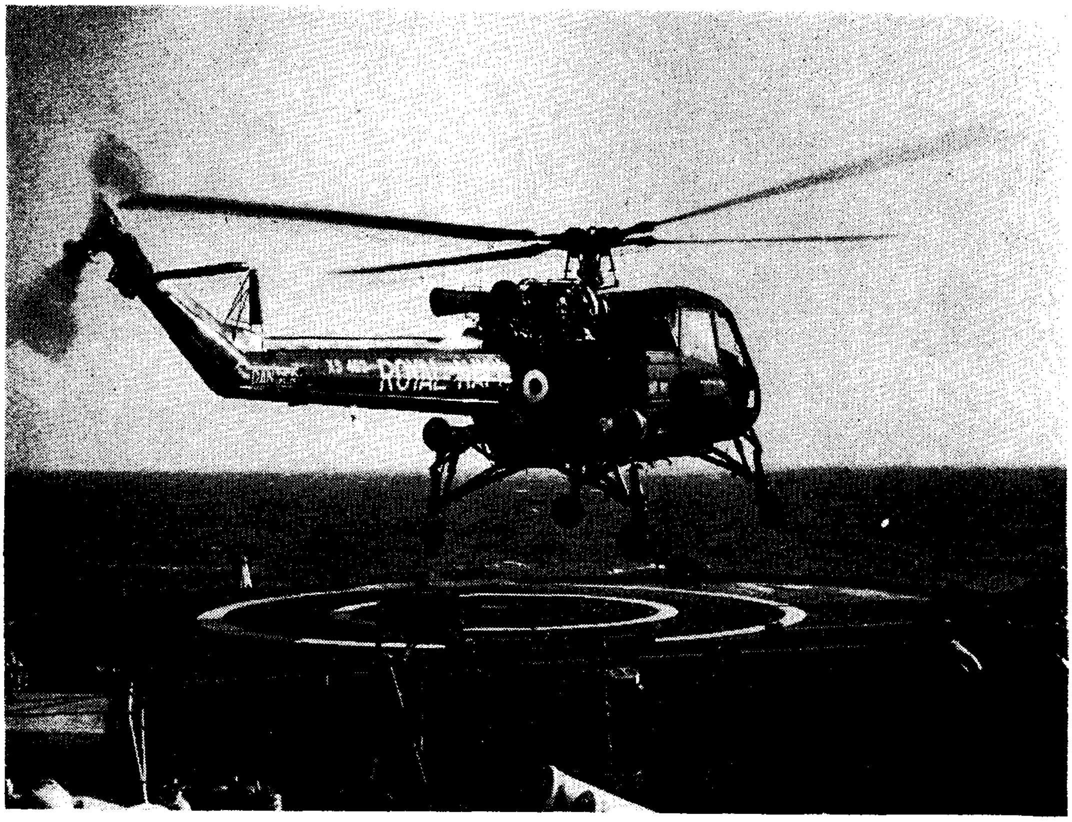


FIG 18—THE WASP HAS MK.1

The autopilot unit itself is principally comprised of signal matching, shaping and mixing amplifiers; the heading, height and speed signals are compared with the settings selected by the pilot. Additional damping may be required in the overall dynamic loop and this can be provided by approximate integration of the signals from accelerometers which read translational accelerations in the vertical and horizontal planes. In the HSS-1N autopilot, as incorporated into the Wessex 1, these accelerometers are not stabilized, but attitude errors are partially cancelled electrically; in the Wessex 3 system the accelerometers will be mounted on a stable platform.

Since the autopilot only applies long-term corrections to the flight path its actuators can move at low rates. This has the advantage that the pilot can contain any 'runaway' failures, without difficulty; furthermore the autopilot can be given the full range of authority over the aircraft controls. The actuators are generally introduced in parallel with the flying controls and, in the Wessex 1 and 3, are geared to drive the cyclic stick spring box so that the datum of the stick itself is displaced in both pitch and roll. The presence of the spring box between the actuators and the pilot facilitates over-riding of the system when required.

Navigation

To prescribe a unique flight path for the aircraft, a flight control system must possess further information on its motion with respect to earth, in addition to the data required by the cruise auto-pilot. Distance travelled from a given datum can be determined by measuring acceleration or velocity and integrating continuously, along the required axes, to determine position. The control system does, in any case, require velocity signals and these can be provided by doppler navigational radars. These equipments transmit 3 or 4 (the fourth is redundant) narrow beams of continuous wave energy outwards at angles of approximately 45 degrees from the aircraft centre line and 30 degrees from

FIG. 19—THE WESSEX HAS MK.3

the vertical; the doppler return signals are then resolved along aircraft or earth axes.

Helicopter doppler radars are subject to an additional problem in that they are required to measure positive and negative velocities, with a continuous output through zero. The nature of this problem can be appreciated by considering the magnitude of the signal as the velocity approaches zero. The frequency of transmission is 13,500 Mc/s in one equipment and 8,800 Mc/s in another; in the latter case the quoted accuracy of 1 knot, in the hover, represents a doppler shift of the order of 20 cycles and the sign of the output must be changed in passing between positive and negative returns of this value. A fundamental problem with reception of any doppler signal lies in the fact that, since the transmitted beams must be inclined to the surface in order to read any relative motion horizontally, the receiver only acquires the very small percentage of reflected energy which is returned as a result of surface irregularities. This situation is aggravated, for the helicopter, when attempting to hover over a dead calm sea.

The American APN. 97 doppler equipment measures velocities along aircraft axes only and is limited to readings greater than $\pm$ 5 knots. The British replacement, AD 580, contains certain design features which offer a radical improvement on this performance; in addition, it computes distances travelled with respect to earth axes. The American doppler equipments employ direct detection of the doppler signal, as it is received at the aerial, whereas the AD 580 equipment contains a 10 Mc/s I.F. stage with the mixing carried out at the aerial (FIG. 20). This feature makes possible a 10 to 1 reduction in the transmitted power and at least a 3 to 1 improvement in the signal to noise ratio. This, coupled with a more sophisticated tracking system capable of operating on smaller signals, yields the required accuracy.

The receivers of both equipments employ two channels for each beam, with the reference frequency shifted by 90 degrees between them, to ensure that the sign of the doppler signal is preserved after detection: it is then indicated by the $\pm$ 90 degrees phase difference between the two outputs. In the APN 97 these outputs are fed to a phase comparator which yields a polarity representing direction of movement (FIG. 20). This technique is not sufficiently precise for controlling the helicopter through zero velocity. The sign information is not

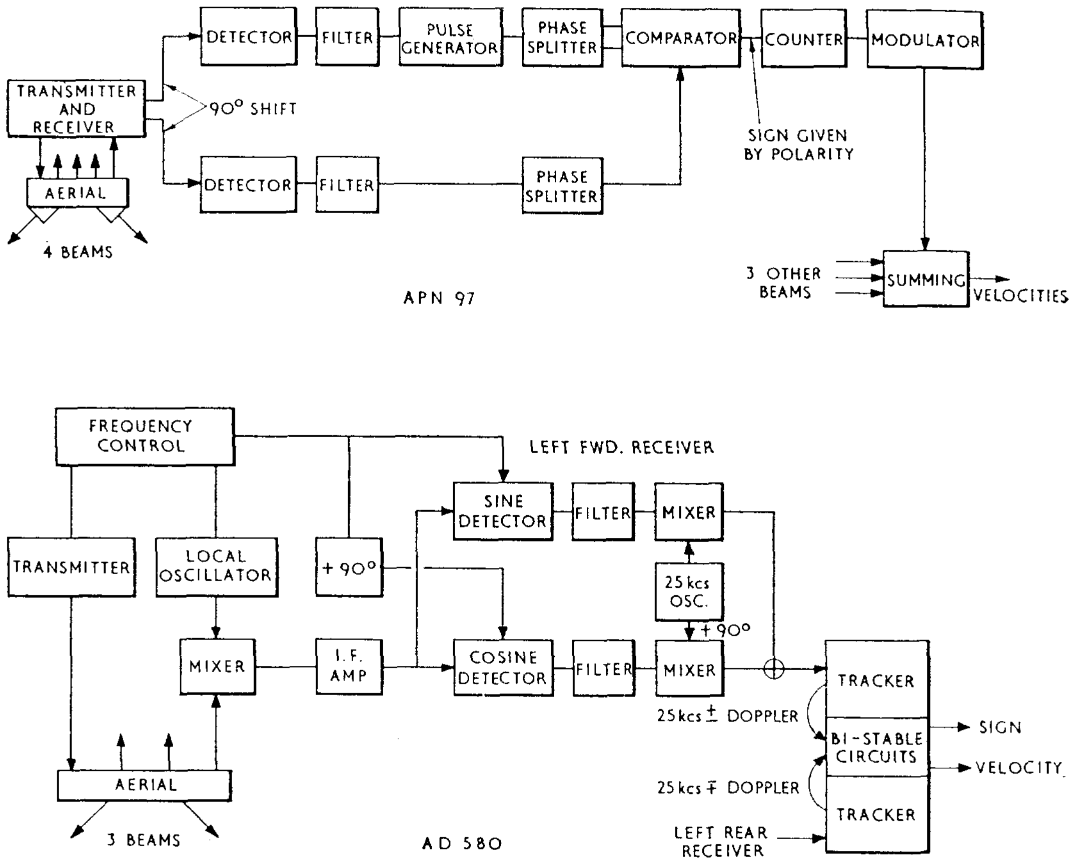


FIG. 20—HELICOPTER DOPPLER RADARS

therefore extracted at this stage, in the AD 580, but is obtained from a circuit which subtracts the movements in a pair of beams. A train of pulses is fed in from the left forward beam and another from the left rear: with forward motion the forward beam will give rise to the more rapid train of pulses and, as soon as the circuit sees successive pulses on one line, it can identify the faster train and indicate direction. This indication is so precise that the equipment will probably register the direction of movement when the helicopter is being pushed along the deck.

By modulating one of the beams, sufficient information can be obtained to permit computation of height, thereby providing an alternative to the radio altimeter. This facility has been added to the new equipment in order to feed a second height channel in the flight control system of the Wessex 3, thereby improving safety.

Automatic Manœuvres

The first requirements for automatic manœuvres in helicopters arose in connection with the anti-submarine role and were related to the problems of descending from the cruise to the hover, over water, and maintaining a steady hover while carrying out a search with the dipping sonar. These manœuvres are difficult for the pilot to perform with accuracy and consistency in visual conditions; they are impossible, without artificial aids, when flying blind.

The transition manœuvre is carried out in the HSS-1N system by coupling the radio altimeter and the doppler radar to the auto-pilot and simultaneously demanding the required hover height and zero surface speed. These are simple step inputs and the aircraft exhibits a transient response, in height, which is under-damped: a large step results in excessive overshoot and death by drowning if the pilot does not intervene. At the end of a transition, the aft cyclic correction, applied to arrest the forward motion, may also prove excessive, resulting in a tendency to drift backwards. Both these deficiencies can, however, be overcome by intelligent use of the system. Once the speed has been reduced to 5 knots and the sonar body has been lowered into the water, the system can be switched to the hover hold condition; cyclic pitch control is then related to the angle between the sonar cable and the vertical while height is obtained by subtracting sonar depth from cable length.

These manœuvres have been the subject of further development and the system for the Wessex 3 will incorporate electro-mechanical function generators to define the optimum flight path for the transition to the hover, from a given range of initial cruising conditions: the helicopter will then be constrained to follow this path. In the hover, the system will be similar except that increased depths of sonar operation will make it impracticable to obtain accurate height information from the cable: the radio altimeter will therefore be employed throughout. Special attention has been given to the effects of sea motion on the flight control system, as a radio altimeter cannot discriminate between this motion and movement of the helicopter. Since the period of the motion lies between 7 and 20 seconds, it is not feasible to eliminate it by filtering the radio altimeter output. The short term datum must therefore be established by inertial means: this has given rise to a unit known as the 'mean sea level computor' which carries out the double integration of an accelerometer signal and relates this to the altimeter, to correct long-term drift. The accelerometer will be mounted on a stable platform and will be temperature stabilized but accuracy still presents problems associated with vertical errors following a banked turn and the small magnitude of the accelerations to be detected.

These improvements, coupled with the extensive navigational facilities being provided for the observer, should make it possible to operate in almost any weather conditions. Provision is also being made for executing automatic banked turns, to improve the navigational accuracy.

Safety is of paramount importance in any flight control system: considerable study must be devoted to analysing the possible failure cases and their consequences. The problems are more acute for low flying aircraft and the systems must be designed either to fail safe or to contain the failure for sufficient time to permit corrective action by the pilot, a period normally assessed at 5 seconds. Complete protection can only be provided by triplication of all equipment, with full integrity for each system and automatic monitoring to eliminate the 'odd one out' when the failure occurs. Duplicated systems can, however, reduce the effects of failures. Since the height channel is the most critical when hovering over the sea this will be duplicated in the Wessex 3, the second channel being fed from the doppler radar: collective control will be governed by the mean position of the two actuators.

Flight Direction

In recent years considerable attention has been given to the development of display systems, for all types of aircraft, which compute the required flight path and present the pilot with steering instructions, instead of applying the corrections directly to the aircraft. Flight director systems fall basically into two categories, namely, those giving 'head-up' displays, in which information is projected on to the windscreen, and the 'head-down' instruments. Such systems have the advantage that the pilot is an integral part: he retains full authority at all times and can interpret the commands indicated. Furthermore, the power amplifiers and actuators, are eliminated.

Flight director systems are at present under development for helicopters and one is planned for the Wessex 3. When used in conjunction with the automatic control system it should provide a means of detecting failures, in addition to imitating all the automatic manœuvres, but this will depend upon the reaction of the pilots, which has yet to be assessed.

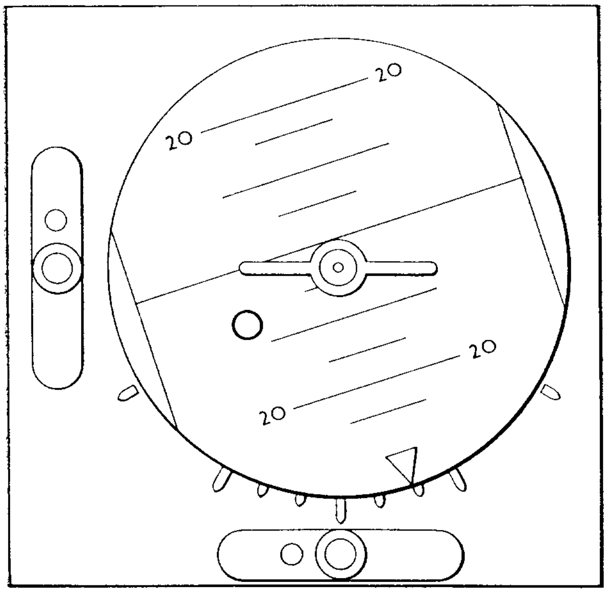


FIG. 21—A FLIGHT DIRECTOR

Flight directions as shown: Nose down
Roll left
Turn left
Climb
Attitude indicator depicting: Nose down
Roll left

The instrument presentation (FIG. 21) consists of three small white discs, one moving vertically, on the left of the instrument, one horizontally, at the bottom, and the third moving in both axes across the face of the display: these represent the collective, yaw and cyclic commands respectively and, in each case, the datum mark on the instrument case is 'flown' towards the disc, as though it represented a target moving in space. The background to the display is formed by a presentation of fuselage attitude in pitch and roll. When used in conjunction with a fully automatic control system the indices of the director can be nulled by feedback from that system so that the director will show zero error when the system is functioning correctly.

Conclusion

It can be assumed that all but the smallest of future helicopters will incorporate some form of auto-stabilization: accepting the basic division between attitude and rate gyro systems, this will yield a high degree of uniformity of flying qualities. It is, however, still significant that the single rotor helicoptors can be designed with a sufficient degree of natural stability, in all axes, to permit handling, over extended periods, with only the powered flying controls available, whereas, in tandems, the dynamic divergence is often too rapid to make this possible. Although the principle requirements for flight path control lie in the anti-submarine field, it is evident that facilities will be required in the future for blind approach and landing, in both civil and military transport helicopters, and for automatic transition and hover in rescue helicopters operating at night. As systems are re-engineered at lower weights, duplication must be extended for improved safety. The accuracy at present required of the vertical references and accelerometers suggests that an inertial navigation system may become the core of a future flight control system.

References:

1. 'Aerodynamics of the Helicopter'. Gessow and Myers.
2. 'Introduction to Helicopter Aerodynamics'. Stepniewski.
3. 'Fundamentals of Helicopter Stability and Control'. Vertol Division of Boeing Airplane Company. Report R.242 (1961).
4. 'Principles of Helicopter Engineering'. Jacob Shapiro.
5. 'An introduction to the physical aspects of helicopter stability'. Gessow and Amer. NACA TN 1982.
6. 'Theory of helicopter damping in pitch and roll and a comparison with flight measurements'. Amer. NACA TN 2136.
7. 'Longitudinal flying qualities of several single rotor helicopters in forward flight'. Gustafson, Amer, Haig and Reeder. NACA TN 1983.
8. 'An investigation of the handling qualities of tandem helicopters'. Gebhard and Goland. Inst. of Aero Sciences (No. 704).
9. 'Studies of lateral-directional flying qualities of a tandem helicopter in forward flight'. Amer and Tapscott. NACA Rep. 1207.
10. 'Studies of the speed stability of a tandem helicopter in forward flight'. Amer and Tapscott. NACA Rep. 1260.
11. 'Requirements for helicopter flying qualities'. MIL Spec. H-8501A.

Acknowledgements

Chinook photograph by courtesy of The Boeing International Corporation. All other photographs by courtesy of Westland Aircraft Ltd.