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Helicopter Rotor Behaviour after Engine Failure in Forward Flight

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Summary.—Calculations have been made to find the changes in rotor speed following engine failure in forward flight. Several particular examples were investigated, based on the parameters of the S–51 helicopter. The effect of pilot's control movement was included.

A rapid loss of rotational speed occurs with practically no change in forward speed of the helicopter, thus the tip-speed ratio increases rapidly. This may lead to stalling of the retreating blade and/or interference of the blades with the droop stops, either of which are dangerous conditions.

The time available to the pilot to reduce the collective pitch after engine failure is very short throughout the speed range and engine failure constitutes a danger to safety on this type of helicopter. Some form of automatic pitch reduction or power failure warning system is necessary.

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1. *Introduction.*—In a previous report[1], the behaviour of a helicopter rotor after engine failure in hovering flight was calculated and compared with the one available flight case. The present report extends the work to engine failure during forward flight. The method of calculation is somewhat similar to that used in Ref. 1, but the attitude of the disc has to be considered as an additional variable. Rotor disc attitude can be controlled to some extent by the pilot and, whereas attitude has practically no effect in hovering conditions, the effects in forward flight can be significant.

Since the attitude of the disc can be controlled by the pilot, some attempt must be made to take into account his possible control movements following engine failure and before he has realized that he should reduce the collective pitch. As the pilot's reaction by means of control movements is bound to vary from one pilot to another and also with the various flight conditions, several particular cases have been considered to determine whether the effect of the pilot's control on the behaviour of the rotor is of importance.

In forward flight, the rotor speed drops very quickly after engine failure but during this time the forward speed remains approximately constant. Thus, the tip-speed ratio will increase rapidly. This results in a backward tilting of the disc increasing the incidence of the blades. The changes in rotor speed and disc incidence give a change in thrust which produces a rate of climb or descent and this in turn alters the disc incidence.

The tilting of the thrust vector gives a moment about the centre of gravity of the helicopter and when the fuselage responds to this the rotor follows. Hence, this response has its effect on the disc incidence. In the calculations, it was found that it was not possible to use approximations for the response conditions to give sufficient accuracy, except at low tip-speed ratio, and that the vertical forces, horizontal forces and pitching rotation of the helicopter had all to be

* R.A.E. Tech. Note Aero. 2119, received 15th December, 1951.

taken into account. It was therefore necessary to use step-by-step methods of solution throughout the calculations.

The loss of rotor speed is not in itself serious but it does lead to other serious conditions. It can bring about stalling of the retreating blade and/or sufficient blade flapping to cause interference with the droop stops. Either of these conditions is dangerous. A third limitation to be considered is undue coning but as the upward flapping freedom is generally very large this conditions is unlikely to be the critical limitation. In high-speed flight, an engine failure can take the helicopter beyond these limitations in an extremely short time, unless immediate action is taken by the pilot to reduce the collective pitch.

It was hoped to do a series of wind-tunnel tests on a three-bladed rotor model to compare with the calculations. After two tests had been completed one rotor blade was broken and the tests were discontinued.

2. *Method of Calculation*.—2.1. *General Method*.—The rotational deceleration of the rotor at any instant is given by

$$\dot{\Omega} = \frac{Q}{I} \quad \ldots \quad (1)$$

where Q is the instantaneous torque and I is the moment of inertia of the rotor, together with the equivalent inertia of the tail rotor and rotating parts.

The vertical acceleration of the helicopter neglecting the small angle of tilt of the disc is given by

$$\frac{\dot{v}_a}{g} = \frac{W - T}{W} \quad \ldots \quad (2)$$

where W is the weight of the helicopter and T is the instantaneous thrust.

The values of thrust and torque in forward flight were taken to depend on the rotor speed, collective pitch, tip-speed ratio and disc incidence. The thrust and torque coefficients were obtained from charts, similar to those of Ref. 2, which were based on theoretical estimations, or from the wind-tunnel tests of Ref. 5.

The disc incidence is a function of the initial trimmed conditions, the tilt of the disc due to the change in tip-speed ratio, the angular response of the helicopter together with the associated following of the rotor, the change in flight path and movements of the pilot's fore-and-aft control.

The change in tilt of the rotor disc due to tip-speed ratio is well known and the value, assuming linear lift characteristics for the blade, was estimated, as from Ref. 6, by the equation,

$$a_1 = 2\mu \frac{\frac{4}{3}\vartheta_0 - \lambda}{1 + \frac{3}{2}\mu^2} \quad \ldots \quad (3)$$

Alternatively, this angle of tilt could be taken from the wind-tunnel tests of Ref. 5. In the use of the wind-tunnel work, the shaft axis should be interpreted as the no-feathering axis of the helicopter.

Pitching of the helicopter is caused by the moment of the thrust vector about the centre of gravity, *i.e.*, to the product of thrust, distance of rotor centre above centre of gravity and tilt of the rotor disc relative to the fuselage. This tilt of the thrust vector depends mainly on the position of the pilot's control, on the value of a_1 as given by equation (3) and, to a lesser extent, on the rotor following behaviour. The latter feature arises from the fact that, as the fuselage pitches, the rotor disc will also pitch but with an angle of lag which depends on the pitching velocity. This characteristic is described in Ref. 2 and the angle of lag is obtained from the derivative

$$a_{1q} = \frac{16}{\gamma \Omega} \quad \ldots \quad (4)$$

All the calculations reported in this note were done using step-by-step methods.

2.2. *Effect of Pilot's Control.*—Movement of the pilot's fore-and-aft control has little effect on the deceleration of the rotor after engine failure but it does have a large influence on the disc incidence, blade flapping motion, etc., and it must be taken into account. It is difficult to estimate what instinctive control movements the pilot may make after engine failure and before he reduces the collective pitch. Such movements would vary with the conditions of flight at the time of the engine failure and there may be considerable difference in action between different pilots.

However, in order to investigate the effect of pilot's fore-and-aft control, some particular examples of possible pilot action were examined, covering what were thought to be practical conditions. In selecting these examples, attention was paid to the likely behaviour of the helicopter after engine failure.

First it was assumed that the pilot kept his control fixed after engine failure had occurred. In this case, the drag forces on the blades cause the latter to lose rotational speed. During this period, there is practically no change in the forward speed and the tip-speed ratio increases. This causes the rotor to tilt backwards changing the flow through the rotor disc and giving an increase in thrust coefficient. The inclination of the thrust vector gives a moment about the centre of gravity of the helicopter making the fuselage pitch. As the fuselage rotates, the rotor follows except for a small angle of lag proportional to the angular velocity. This response of the helicopter causes further changes in the flow through the disc which, together with the change in tip-speed ratio, increases the backward tilt of the disc and so on. Thus if the pilot makes no control movement after engine failure the helicopter will pitch nose-up (fairly rapidly at high forward speed). The effects of all these changes were included in the step-by-step calculations made.

Turning to cases involving positive pilot action, for the second example it was considered that the pilot might attempt to keep the attitude of the helicopter constant. To do this, the pilot must prevent the nose-up pitching moment discussed above. Thus, as the helicopter rotor disc tends to tilt backwards due to the changes in tip-speed ratio and flow through the disc, the pilot must make the necessary cyclic pitch corrections, *i.e.*, he will have to move his fore-and-aft control continuously forward to maintain the attitude trim. In this case the calculations were much simpler as helicopter fuselage movements were not involved.

For the third case, it was assumed that the pilot might attempt, by use of his control, to maintain constant forward speed. If the magnitude of the thrust decreases or if the thrust is tilted backwards there is a horizontal deceleration; in order to maintain constant forward speed, the forward component of the thrust must be maintained. As the thrust decreases, therefore, the pilot must use his control to incline the thrust axis further forward and as this forward tilt alters the flow through the disc in such a way as to decrease the thrust, the forward inclination of the thrust becomes appreciable. Also, this tilting of the thrust vector produces movements of the helicopter fuselage with resultant movements of the rotor, in a similar manner to those discussed in section 2.1. In this case, the pilot must make the appropriate cyclic pitch applications such that, with helicopter response, etc., included, the forward component of the thrust is constant. It was found in these calculations that quite an appreciable forward stick movement was required and as the fuselage pitching response built up, the stick had to be brought back to prevent exceeding the intended conditions.

In the above examples, the stick was kept fixed in one case and moved forward in the other two. In order to complete the picture, an example was calculated in which the stick was moved back. In this case, an arbitrary stick movement was selected such that the ensuing response was not excessive. Again the calculations were made by step-by-step methods as discussed above.

2.3. *Blade Incidence.*—For the purpose of this note, it was not necessary to know the general distribution of incidence along the blade and, in the calculations made, thrust coefficients were estimated as in Ref. 6 or could be taken from the charts of Refs. 2 and 5.

However, it was necessary to know the conditions at which stalling would appear on the blade as this might cause a limitation to flight safety. Also, estimation of the rotor characteristics *e.g.*, rotor tilt due to tip-speed ratio, was based on the assumption that no blade stalling occurred and calculations extending beyond these limitations are liable to error.

The angle of incidence at any point on the rotor disc was taken from equation (8) of Ref. 6, *viz.*,

$$\alpha = \vartheta_0 - A_1 \cos\psi - B_1 \sin\psi - \frac{a_0\mu \cos\psi + \lambda}{x + \mu \sin\psi} . \qquad (5)$$

We were interested only in blade stalling and this will occur initially on the retreating blade somewhere about the $\psi = 270$-deg position. Hence, using the $\psi = 270$-deg position, we may simplify the above equation

$$\alpha_{270} = \vartheta_0 + B_1 - \frac{\lambda}{x - \mu} \qquad (6)$$

ϑ_0 is the collective pitch

B_1 is a measure of the pilot's longitudinal control

λ is a parameter representing the flow through the rotor disc and may be expressed in two parts, the component of forward speed through the disc and the induced velocity.

The component of forward speed through the disc can be obtained directly from the disc incidence and this in turn may be obtained from the disc attitude and angle of descent. The disc attitude is in turn dependent on the fuselage attitude and the disc tilt due to tip-speed ratio, together with any correction for lag between the rotor disc and the fuselage. The other component, the induced velocity, may be estimated with sufficient accuracy at the forward speeds considered from the equation of momentum

$$v_i = \frac{T}{2\pi R^2 \rho V} . \qquad (7)$$

It is therefore simple from equation (6) to evaluate the blade incidence at any position x along the retreating blade or alternatively if stalling limitations are known to evaluate the extent of the stalled area.

2.4. *Blade Flapping.*—The blade flapping angle relative to the shaft is given by

$$\beta_s = a_0 - a_{1s} \cos\psi - b_{1s} \sin\psi - a_{2s} \cos 2\psi - b_{2s} \sin 2\psi . \qquad (8)$$

Replacing the coefficients relative to the shaft by the equivalent coefficients relative to the no-feathering axis

$$\beta_s = a_0 - a_1 \cos\psi - b_1 \sin\psi - a_2 \cos 2\psi - b_2 \sin 2\psi + B_1 \cos\psi - A_1 \sin\psi . \qquad (9)$$

For the purpose of the present work, we were interested mainly in the maximum and minimum flapping angles. These will occur in approximately the fore-and-aft positions of the blade and using $\psi = 0$ or 180 deg we may neglect b_1, A_1 and b_2. The evaluation of the other coefficients is discussed below.

The coning angle a_0 was evaluated as in equation (13) of Ref. 6 but was approximated in the following form

$$a_0 = \frac{\gamma}{2}\left[\tfrac{1}{4}\vartheta_0 \frac{1 - \mu^2}{1 + \frac{3}{2}\mu^2} - \tfrac{1}{3}\lambda \frac{1 - \frac{1}{2}\mu^2}{1 + \frac{3}{2}\mu^2}\right] . \qquad (10)$$

It should be noted that in cases where the incidence of the disc does not change to any great extent, the changes in a_0 may be estimated using the following simple argument. The aerodynamic moment about the flapping hinge is approximately proportional to the thrust coefficient t_c and to Ω^2, while the centrifugal-force moment is proportional to Ω^2. Thus, as the rotor slows down, the coning angle can be taken to be proportional to t_c.

The value of a_1 has already been discussed and is given in equation (3).

B_1 represents the cyclic pitch applied to the rotor. The required B_1 to trim is given in Ref. 6, section 4, and is a function of rotor characteristics, centre of gravity position and fuselage pitching moment.

a_2, the second harmonic of flapping, is very small but can be included by taking its value approximately $0 \cdot 07 a_1$.

2.5. *Safety Limitations.*—2.5.1. *Blade stalling limit.*—As tip-speed ratio increases the incidence of the retreating blade increases until stalling starts at the blade tip and the associated vibration occurs. As the stall spreads inboard from the tip, the vibration will increase in severity, eventually becoming unacceptably serious. It is expected that extreme difficulty of control will develop due to the irregularities in the flapping motion. In order to assess the danger in particular conditions, it is necessary to select a borderline, represented by a definite value of maximum incidence at a selected point along the blade, at which it is expected that the vibration and the interference with control effectiveness will become excessive.

It has been shown in Ref. 3 that the blade tip stalls at about 12 deg. When the stall has spread and the tip incidence has reached 16 deg the helicopter becomes very difficult to control and in Ref. 3 the 16-deg value at the tip is regarded as the limiting flight condition.

It is felt that for a short period in an emergency condition (as in the engine failure case under consideration) it might be possible to accept more severe conditions. Therefore, it is proposed in this note to use 16-deg incidence at the $0 \cdot 75$-radius position as the limit and this is regarded as an absolute limit beyond which the stalled area of the disc would be so large that vibration and irregular flapping motion would be likely to lead to disaster.

2.5.2. *Downward flapping limit.*—A limit is imposed on the downward flapping angle of the blade by the droop stop. For the S-51 helicopter this angle is $-2 \cdot 5$ deg and is not likely to vary greatly for different types of helicopter.

If the blade should flap sufficiently to cause interference with this stop in flight, the normal flapping motion is disturbed. This may lead to some loss of control and even to blade or fuselage structural damage. Thus, blade intereference with the droop stop may be a very dangerous condition.

The maximum and minimum flapping angles will occur at approximately the fore-and-aft positions of the blade and to simplify the calculation $\psi = 0$ or 180-deg positions are taken.

The minimum flapping angle is almost certain to occur at the aft position, i.e., $\psi = 0$ and is therefore given from equation (9) by

$$\beta_{s\,\psi=0} = a_0 - a_1 - a_2 + B_1 . \qquad (11)$$

However, in cases of low thrust and large forward movements of control, the flapping angle at $\psi = 180$ deg should be checked.

$$\beta_{s\,\psi=180} = a_0 + a_1 - a_2 - B_1 . \qquad (12)$$

2.5.3. *Upward flapping limit.*—Generally, there is no upward flapping stop but eventually, if sufficient upward flapping takes place, mechanical interference will occur. For the S-51 helicopter, the angle at which this would occur is about 40 deg.

It is therefore extremely unlikely that the upward flapping angle will constitute a critical limitation, but the possibility of this limitation on other helicopters with less mechanical freedom should not be forgotten.

3. *Range of Calculations.*—The behaviour of the helicopter rotor after engine failure depends on a number of variables:

(*a*) helicopter forward speed

(*b*) initial rotor speed

(*c*) collective-pitch setting

(*d*) rotor inertia

(*e*) pilot's stick position.

All these factors have a large influence on the loss of rotor speed, increase of incidence of the retreating blade and flapping of the blades. A number of specific cases, all based on the S–51 helicopter, have been taken to cover as much as possible of the range of variables concerned.

Most of the calculations were made from initial trimmed conditions at cruising speed and the effect of pilot's control movement has been studied by calculating the effect of a number of possible pilot's actions as discussed in section 2.2.

A comparison was made at low forward speed for initial conditions of maximum and minimum operating rotor speeds with the associated collective pitch settings. The influence of collective pitch and initial rotor speed is easier to deal with at a low forward speed as the effect of pilot's control is much less significant and the calculations can be done for attitude constant conditions to simplify the work.

A further example was to take the maximum permissible tip-speed ratio as the starting condition. Thus, the entire speed range of the helicopter has been covered in the range of calculations.

All calculations were made with the centre of gravity in the middle of the range. Correction to any other centre of gravity position is easily made by reference to the B_1 trim curves of Ref. 6.

The effect of moment of inertia on rotor deceleration has been shown in Ref. 1.

Details of the conditions for each of the calculations are given in Table 1.

4. *Wind-Tunnel Tests.*—It was hoped to do a number of wind-tunnel tests on a 12-ft diameter three-bladed rotor. Full details of this rotor are given in Ref. 5. However, after two test conditions had been completed, one of the blades fouled the droop stop and was broken.

The results obtained are included for information in Fig. 8. As only tests at low tip-speed ratio had been done, the results do not contribute much and no comparison is drawn with calculations.

5. *Results.*—The results of the calculations are presented in Figs. 1 to 7 and the various parameters are plotted against the time elapsed from the instant of engine failure. The curves are arranged in groups to illustrate the influence of specific variables on the rotor behaviour.

5.1. *Rotor Speed.*—In Fig. 1, it is shown that, after engine failure, the rotor speed falls off at almost the same rate, despite the large variations in forward speed, as long as the initial rotor speed and pitch setting are constant. The three particular cases considered are:

(*a*) for climbing conditions (D)

(*b*) for high-speed flight (J)

(*c*) for the hovering case (A, Ref. 1) which is also at approximately the same pitch setting.

In Fig. 2, a comparison is made to show the effect of collective-pitch setting at constant rotor speed and helicopter forward speed. This is done at fairly low flight speed where any change of trim or pilot action does not influence the conditions to any extent. It will be seen that the higher the pitch setting, the greater the loss of rotor speed. In further comparisons for the various parameters, it will be found that the initial pitch setting is one of the most important aspects affecting the rotor deceleration.

In Fig. 3, the effect of different initial rotor speeds at constant collective-pitch setting is presented and it will be seen that the percentage rotor speed falls off more slowly for the higher initial rotor speed. The decelerating torque on the rotor is almost the same in both cases, giving similar loss of absolute rotational speed which shows up as a smaller percentage loss when the initial absolute rotor speed is higher.

5.2. *Blade Flapping at Low Flight Speed.*—The flapping angles of the blade in its aft position for the three conditions at low flight speed are plotted in Fig. 4. Both the changes in the tilt of the disc due to change in tip-speed ratio, at the low tip-speed speed ratios taken, and the changes in coning angle are comparatively small. It is also well known from previous work that any probable longitudinal control action which might be applied in these low flight speed conditions would not have any serious effect on this flapping motion. The question of limitations due to flapping interference does not arise therefore at this low tip-speed ratio.

5.3. *Blade Flapping at Cruising Conditions with Pilot's Control Effects.*—The effect of the pilot's fore-and-aft control action is investigated for cruising speed conditions and for the four cases of pilot's action as considered in section 2.2. Curves of the pilot's control movements necessary to obtain the assumed conditions are given in Fig. 5a. The loss in rotor speed after engine failure and the helicopter normal acceleration are shown in Figs. 5b and 5c, while the corresponding flapping angles of the blade and the fuselage response are given in Figs. 6a and 6b respectively. An examination of the normal acceleration and the fuselage attitude curves shows that the range of probable pilot's longitudinal control action has been adequately covered. In the stick-forward case (E) where control angles are such as to give constant forward speed, the loading reduces to $0 \cdot 6g$ and the fuselage pitches nose-down by 12 deg in less than 3 sec. In the stick-backward case (H) the loading has reached $1 \cdot 1g$ in $1 \cdot 25$ sec (and would increase rapidly but for the stall limitation) while the helicopter is pitching rapidly nose-up. It is unlikely that the pilot's control action would fall outside this range.

The loss of rotor speed does not depend to any extent on the control movements. However, the effects on loading, flapping motion and fuselage response are very significant. Backward movement of the stick causes initial backward tilting of the disc but the change in loading due to the changed flow through the disc soon increases the coning angle. Thus, the calculations show that in the later stages the increase in coning angle is much greater than the backward tilt of the disc. It will be seen that there is quite an appreciable fuselage rotation even when the pilot's control is kept fixed due to the changes produced by the increasing tip-speed speed ratio. The calculations indicate therefore, that while the effects of pilot's control action are appreciable in cruising flight, any reasonable fore-and-aft control movements are unlikely to have any serious influence on the safety of the helicopter due to blade flapping interference at this forward speed.

5.4. *Blade Flapping at High Flight Speed.*—Fig. 7 gives the conditions following engine failure at $\mu = 0 \cdot 35$ corresponding to the maximum permissible forward speed. In this case, the rotor speed decreases at a similar rate to other forward speed conditions at the same pitch setting. However, the backward tilting is much more important and the decrease in flapping clearance from the droop stop is serious. It must be remembered that the calculations are made with the centre of gravity central; if the forward limit is taken it is estimated that the clearance will be decreased by a further 2 deg. The greater amount of backward tilt at high forward speed can readily be appreciated by reference to the trim curves of Ref. 6 where it is shown that the slope of the trim curves becomes very much steeper at higher tip-speed ratios.

5.5. *Blade Stalling.*—On each of the curves for all the cases mentioned above, the time at which the blade stalling limitation is reached has been indicated. It will be seen that this appears to be the critical factor affecting the safety of the helicopter after engine failure in all cases and the interval of time from engine failure after which blade stalling sets in is extremely short throughout the speed range.

6. *Discussion.*—In order to obtain a general impression of the time available to the pilot after engine failure, during which he must reduce his collective pitch, the limitations obtained in the calculations are plotted together in Fig. 9. It will be seen that for all practical purposes, the blade stalling limit is the critical one at all flight speeds. Blade flapping limitations occur at high speed but the time limits are very close to each other and, in fact, the progressive blade stalling may increase the blade flapping motion and make the time limit even smaller.

The limiting conditions which will seriously affect the safety of the helicopter are experienced within 3 sec over the entire speed range. At high forward speed, the most severe conditions are encountered and the limitations may be exceeded in an extremely short time, perhaps *under one second*. It must be remembered that the blade-stalling conditions assumed as the limiting conditions for the purpose of calculation were very severe, and it is unlikely that the helicopter could continue beyond the blade-stalling boundary of Fig. 9 without encountering disaster.

Thus, it is necessary to consider the above implications of engine failure in recommending the safe operating envelope for the helicopter and to restrict the flight envelope accordingly. The present flight envelope is produced mainly from blade-stalling and tip-speed ratio limitations in normal flight. It is shown here that these limitations are exceeded in an extremely short time after engine failure. Thus, these conditions must be taken into account if the standard of safety is to be maintained, even although the influence on helicopter design may be serious.

It would appear that there is a need for some form of automatic pitch reduction in the event of engine failure in normal flight. At low altitude, however, the pilot may wish to increase the collective pitch and make use of the rotor kinetic energy immediately. There is therefore the necessity for introducing some form of cut-out or over-ride for use at low altitude.

It is also felt, in view of the very short time intervals involved, that an engine failure warning system is essential. However, in the worst cases the time interval is so short that a warning would not be sufficient to ensure safety and the warning system cannot be considered as an alternative to an automatic pitch reduction in a really bad case.

These recommendations for automatic pitch reduction and for a power failure warning were also made in the report on hovering conditions (Ref. 1).

Each helicopter type should be considered on its own merits. The S–51 helicopter, which is the basis of this report, seems particularly bad from the engine failure point of view. Rotors with higher tip speed are beneficial in relation to engine-failure characteristics, since the kinetic energy in the blades is higher, the collective pitch setting lower and the same forward speed would correspond with a lower tip-speed ratio.

7. *Conclusions.*—7.1. The calculations show that the rotor blades lose rotational speed very rapidly when engine failure occurs. The loss of rotor speed is mainly dependent on the initial collective pitch setting and is roughly independent of forward speed.

7.2. Serious blade stalling occurs within 3 sec of engine failure in all cases.

7.3. At high forward speed, the flapping motion of the blades may also limit the time available for the pilot to reduce the collective pitch setting after engine failure.

7.4. The very short period of time available, throughout the speed range, in which the pilot must take appropriate action makes the case of engine failure on this type of helicopter a very dangerous condition.

7.5. The implications of engine failure should be taken into consideration in determining the safe operating flight envelope for the helicopter.

7.6. The development of a 'constant-speed rotor' or some form of automatic pitch reduction in the event of engine failure is required.

7.7. Some form of power-failure warning system is required, particularly if there is no means of automatic pitch reduction.

LIST OF SYMBOLS

a_0		Coning angle
a_1, b_1		Coefficients in Fourier series for flapping
A_1, B_1		Coefficients in Fourier series for feathering
b		Number of blades
c		Mean blade chord
I_1		Blade moment of inertia about flapping hinge
I		Moment of inertia of rotor, together with equivalent inertia of the tail rotor and rotating parts
q		Angular velocity of helicopter in pitch
Q		Torque of main rotor
R		Rotor radius
T		Thrust of main rotor
t_c		Thrust coefficient $T/bcR\rho(\Omega R)^2$
U		Velocity of flow through the rotor disc
V		Forward flight velocity
V_a		Rate of descent
V_i		Induced velocity
W		Weight of helicopter
x		Fraction of rotor radius
α		Incidence of blade section
β_s		Flapping angle measured from plane perpendicular to shaft
γ		Lock's inertia number $\rho acR^4/I_1$
ϑ_0		Collective pitch of blade
λ	=	$U/\Omega R$. Coefficient of flow through the disc
μ	=	$V/\Omega R$. Tip-speed ratio
ρ		Air density
ψ		Blade azimuth position (measured from downwind position in direction of rotation)
Ω		Angular velocity of rotor

REFERENCES

No.	*Author*	*Title, etc.*
1	W. Stewart and G. J. Sissingh	Notes on helicopter rotor behaviour after engine failure in hovering flight. R. & M. 2659. May, 1949.
2	J. K. Zbrozek	Introduction to dynamic longitudinal stability of single-rotor helicopter with hinged blades. A.R.C. 11,440. February, 1948.
3	F. B. Gustafson and G. C. Myers	Stalling of helicopter blades. N.A.C.A. Tech. Note 1083.
4	P. Brotherhood and M. F. Burle	Flight measurements of longitudinal and lateral trim of the S-51 helicopter. A.R.C. 12,879. September, 1949. (Unpublished.)
5	H. B. Squire, R. A. Fail and R. C. W. Eyre.	Wind-tunnel tests on a 12-ft diameter rotor. R. & M. 2695. April, 1949.
6	W. Stewart	Helicopter control to trim in forward flight. R. & M. 2733. March, 1950.

TABLE 1

Flight Conditions for Calculations

Case	Tip-speed ratio μ	Collective pitch ϑ_0	Rotor speed r.p.m.	Flight path	Remarks
A	0	11·0	192	Hovering	Ref. 1
B	0·1	9·1	172	Level	Attitude constant
C	0·1	12·0	172	Climb	Attitude constant
D	0·1	12·0	192	Climb	Attitude constant
E	0·25	8·6	192	Level cruising	Forward speed constant
F	0·25	8·6	192	,, ,,	Attitude constant
G	0·25	8·6	192	,, ,,	Controls fixed
H	0·25	8·6	192	,, ,,	Control pulled back
J	0·35	12	192	Level	Controls fixed

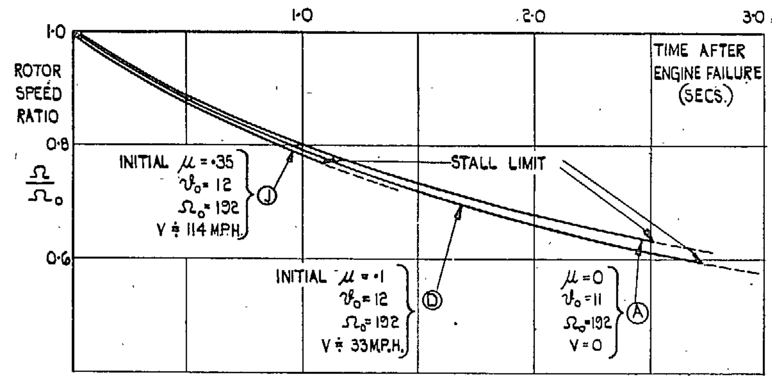


FIG. 1. Effect of forward speed on loss of rotor speed.

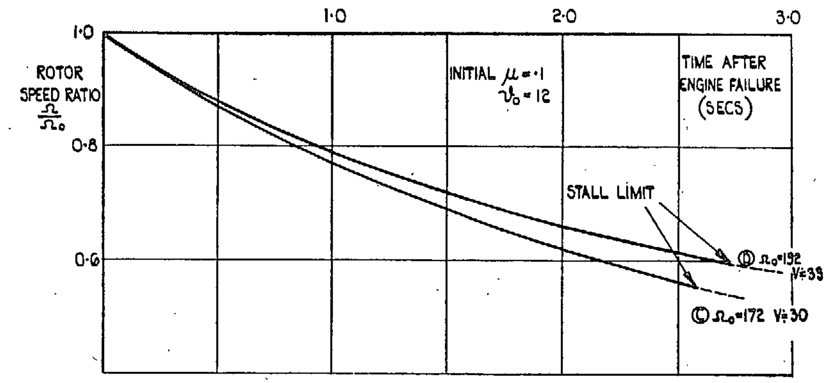


FIG. 3. Effect of initial rotational speed at same collective pitch.

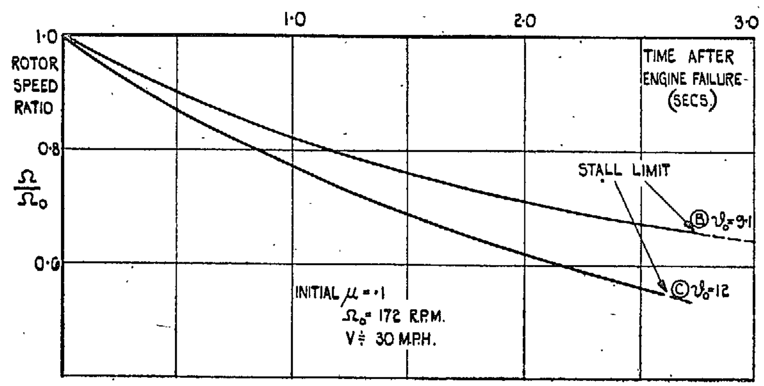


FIG. 2. Effect of collective pitch at same initial rotor speed.

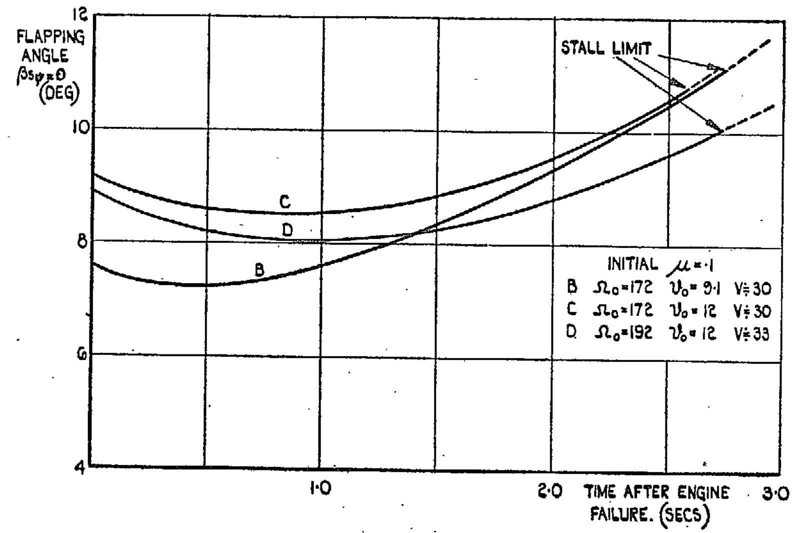
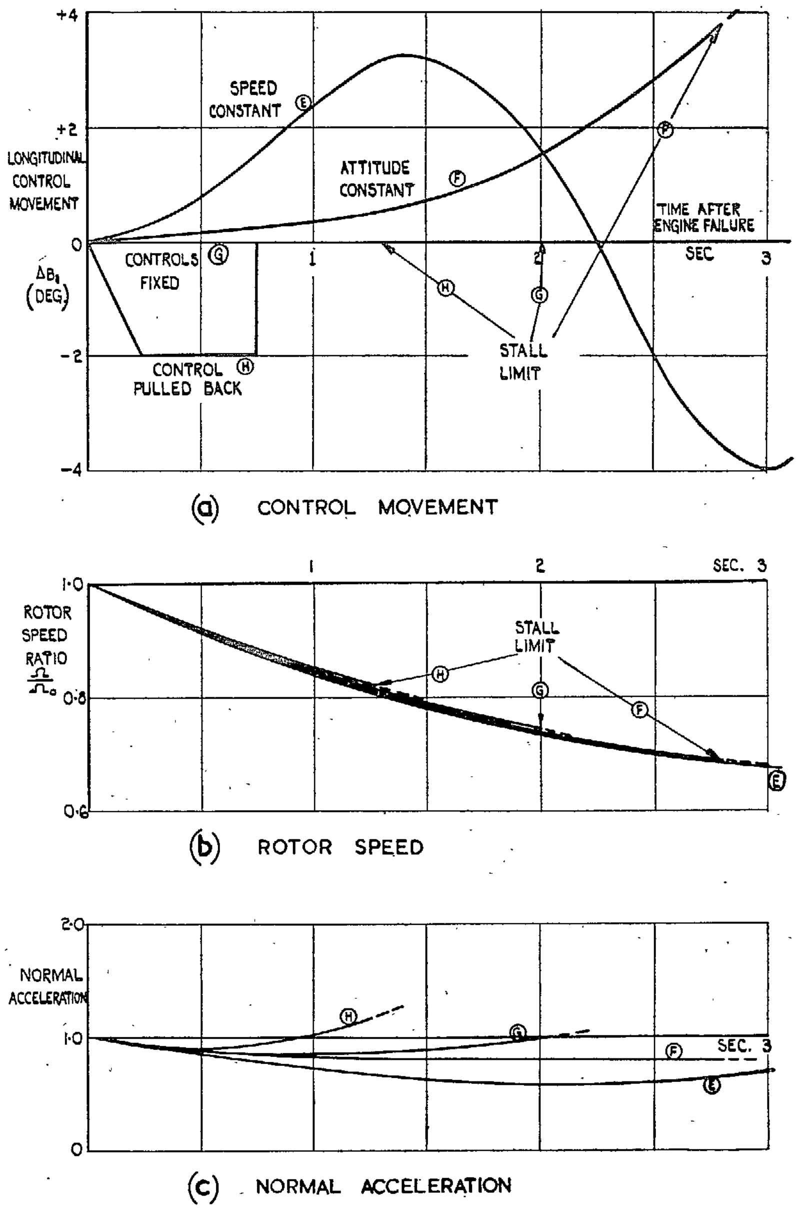
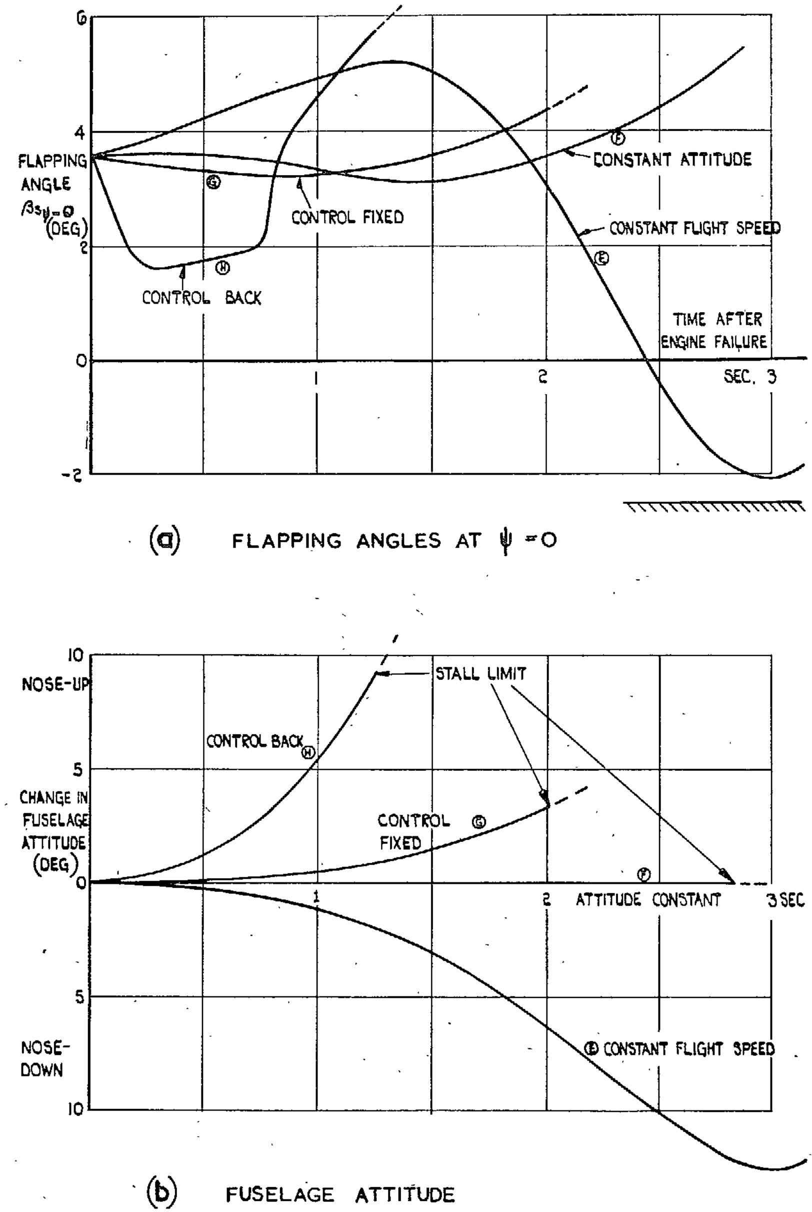


FIG. 4. Flapping angles for low-speed conditions.



FIGS. 5a, 5b and 5c. Effect of movements of pilot's control. $\mu = 0{\cdot}25$. $\vartheta_0 = 8{\cdot}6$ deg. $\Omega = 192$ r.p.m.



FIGS. 6a and 6b. Effect of movements of pilot's control. $\mu = 0{\cdot}25$. $\vartheta_0 = 8{\cdot}6$ deg. $\Omega = 192$ r.p.m.

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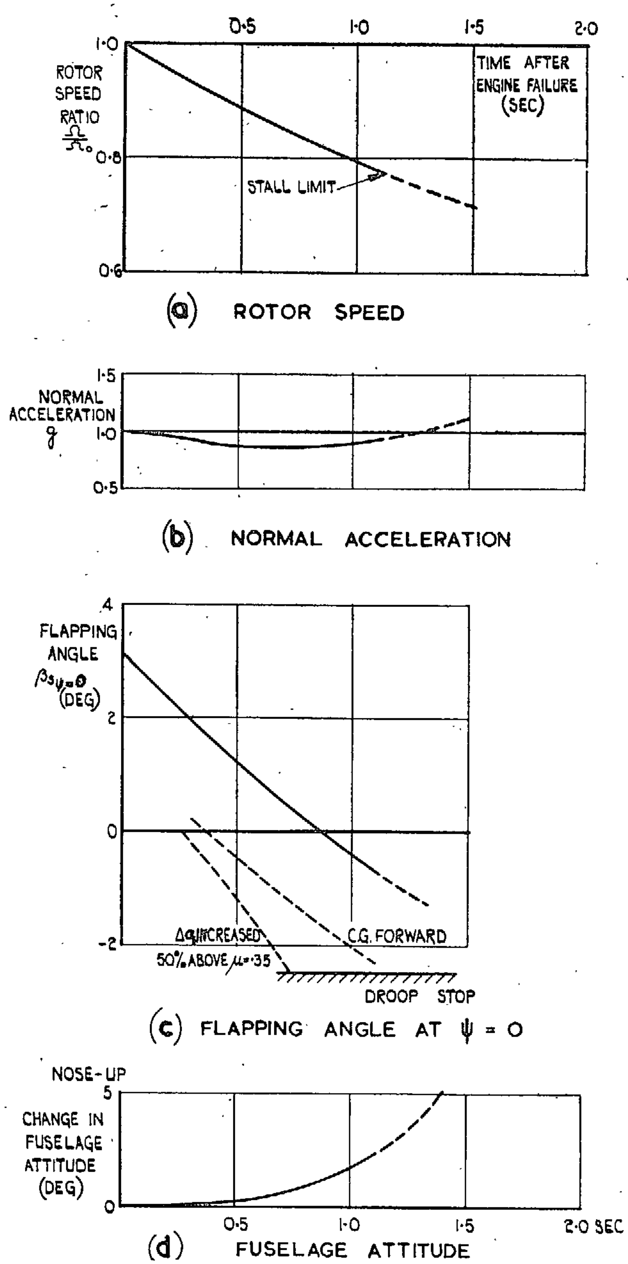

FIGS. 7a, 7b, 7c and 7d. Helicopter behaviour at high speed. $\mu = 0{\cdot}35$. $\vartheta_0 = 12$ deg. $\Omega = 192$ r.p.m.

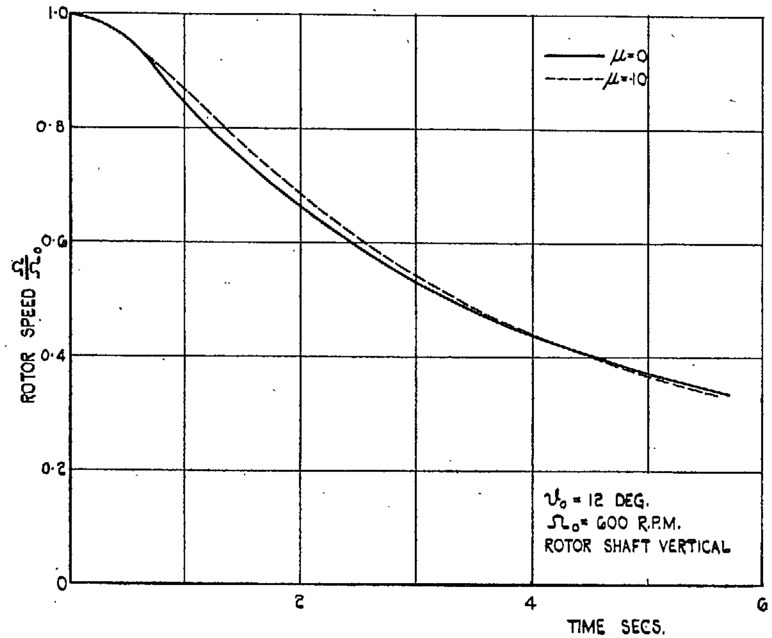

FIG. 8. 12-ft diameter model rotor tests.

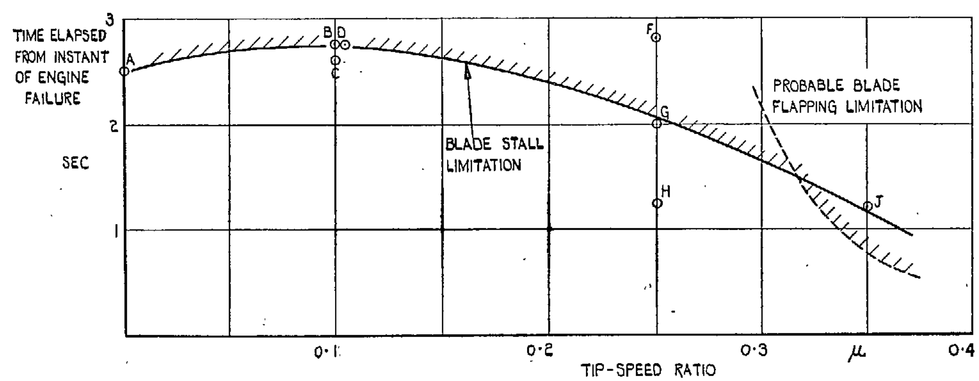

FIG. 9. Time available after engine failure.

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