PAPER Nr. : 7

HELICOPTER TAIL CONFIGURATIONS TO SURVIVE TAIL ROTOR LOSS

by

P. Taylor
Department of Aeronautics & Astronautics
University of Southampton, U.K.

FIFTH EUROPEAN ROTORCRAFT AND POWERED LIFT AIRCRAFT FORUM
SEPTEMBER 4 – 7TH 1979 - AMSTERDAM, THE NETHERLANDS

HELICOPTER TAIL CONFIGURATIONS TO SURVIVE TAIL ROTOR LOSS

P. Taylor
Department of Aeronautics & Astronautics
University of Southampton, U.K.

ABSTRACT

Recently the US Army have specified that a helicopter must be capable of returning from its mission after suffering a tail rotor loss. The helicopter should possess sufficient directional stability to fly at the minimum power speed with a sideslip angle of not more than 20°. A simple theory, describing the yawing oscillation of a helicopter, has been applied to a typical helicopter in order to identify the stability implications on the aerodynamic design of meeting the above tail rotor loss criterion. The fin area required, for a fin and single tail rotor configuration, to meet both the above criterion and to ensure adequate lateral stability characteristics was large even if camber and incidence were used. The same helicopter but with twin tail rotors and no fin was investigated. This configuration has additional advantages including the unique ability to land in confined places after the loss of a rotor. The work presented in this paper is part of a research activity at W.H.L. into tail rotor configurations.

NOTATION

A_T	Tail rotor disc area
a_1	Fin lift curve slope
C	Helicopter yawing inertia
$\bar{\bar{c}}$	Fin mean aerodynamic chord
C_D	Fin drag coefficient
C_L	Fin lift coefficient
C_M	Fin pitching moment coefficient
C_T	Tail rotor thrust coefficient
l_F	Distance from centre of gravity to the fin aerodynamic centre
l_T	Distance from centre of gravity to the tail rotor centre
N	Yawing moment
N_ψ	Yawing moment derivative with respect to yaw
N_r	Yawing moment derivative with respect to yaw rate
Q	Main rotor torque

r Yaw rate, $\dot{\psi}$

S/A Ratio of blocked disc area to total disc area

S_F Fin area

t Time (seconds)

V_T Tail rotor tip speed

V_o Forward speed

$\hat{v}$ Sideslip non dimensionalised with respect to tip speed

α_F Fuselage attitude (positive nose down), (degrees)

α_S Fin setting angle (positive L.E. left), (degrees)

ρ Air density

σ Relative density

ϕ Side wash angle, (degrees)

ψ Yaw angle (positive nose right), (degrees)

1. INTRODUCTION

Interest in the flight dynamics of a helicopter without a tail rotor began during the late 1960's following the experience of the US Army in Vietnam. The ability to return safely from a mission after having sustained damage to or suffering the complete loss of the tail rotor system maximizes the chance of survival and increases the effectiveness of a military helicopter. The ability of a civil helicopter either, with complete safety, to arrive at its destination, or to return to its point of departure or to divert following a tail rotor loss is absolutely desirable.

In the past, little attention has been directed towards the overall airframe directional stability because this directional stability was completely dominated by that of the tail rotor. To date, several military helicopters in the US have been designed to survive a tail rotor loss: some more successfully than others. Also, it is interesting to note that one civilian helicopter, Ref.1, designed to meet the FAA IFR requirements has a fin sized to increase the survivability of a tail rotor loss.

There are several methods of proceeding:

a. The overshoot and initial acceleration in yaw should be contained and the autorotative characteristics should be such as to allow an immediate power-off landing. The local terrain may not be suitable. It is also extremely inconvenient.

b. Additional protection and strengthening of the tail rotor system can be provided. This incurs a weight penalty and reduces the possibility of a tail rotor failure. There is no capability for surviving a tail rotor loss when it occurs.

c. The directional stability of the airframe can be increased.

d. An additional tail rotor can be provided. This is the only solution if it is necessary to be able to land safely in a confined space such as the helipad on an oil rig or the rear of a frigate and which obviously requires controllability into the hover.

This paper is concerned with the last two options.

The possibility of increasing the directional stability of the airframe by a combination of increasing the fin area and offsetting the fin was investigated. This solution has been accepted by one manufacturer, Ref.1 and 2, but rejected by another, Ref.3, on the grounds of incurring a weight panelty and excessive blockage of the tail rotor. It is cost effective to have the ability to estimate the required fin area at a pre design stage. This was achieved by proposing a simple model employing linear aerodynamics. The fin area predicted by this first model was used to determine the preliminary fin geometry which was then entered into a second model employing non linear aerodynamics. This second model was constructed in modular form such that, as the design proceeds and wind tunnel data becomes available, the 'analytical' modules can be replaced by 'data' modules. The fin geometry determined by this second model can be used for full simulation studies and for wind tunnel models.

At an early stage in the design iteration, it became clear that the fin area required by the example helicopter to survive a tail rotor failure was large. This has serious repercussions on low speed handling, which for specific applications, may prove to be too severe to permit such a solution to be used. A solution which overcomes this problem is the provision of twin tail rotors which offer increased survivability at all speeds and controllability in the hover. Landings are therefore possible in confined places and the complete elimination of tail rotor blockage by a fin enhances agility in low speed manoeuvres.

2. ANALYTICAL MODELS TO DETERMINE THE FIN SIZE

The recent design competitions for the US Army AAH and UTTAS specified the following requirements for continued flight after the loss of a tail rotor:

a. The helicopter shall maintain level flight at the minimum power speed with no more than 20 degrees of yaw.

b. The helicopter shall be able to land with power off at a speed of 35 kts without exceeding a sideslip of 6kts i.e. the sideslip angle, power off, shall be less than 10°.

Although these requirements are only concerned with steady state yaw angles, they do have a major influence on the fin size. In addition, it was also stated that the overshoot in yaw shall be limited to ensure that the helicopter will recover without any immediate control input. In the author's view, in order to limit the yaw acceleration to a level which will assure recovery, the overshoot in yaw should be limited to 20 degrees and not the steady state yaw angle.

It was anticipated that the yawing response would be the dominant mode. Consequently, it was decided to represent the yawing response by a simplified model which described the oscillation of a helicopter pivoted about a fixed axis through its centre of gravity. Subsequently, results at Westland Helicopters from a complete six degrees of freedom simulation have validated this approach. Initially, the aerodynamic forces and moments were considered to be linear functions of yaw displacement and rate. This approach had the advantage of introducing the fin area into the calculation explicitly. Having obtained an initial estimate of the fin area, the details of the fin geometry were determined, based on this estimate of fin area, and were incorporated with the details of the operating environment of the fin into a second model. In this case, the aerodynamic forces and moments became non-linear functions of the yaw displacement.

2.1 Linear Aerodynamic Model

The aerodynamic yawing moment N, about the yawing axis through the centre of gravity, is given by

$$N = N_r r + N_\psi \psi$$

where r is the angular velocity in yaw or yaw rate, $\dot{\psi}$, and N_r and N_ψ are aerodynamic derivatives.

If Q is the main rotor torque then the complete simplified equation of motion after the tail rotor has failed is

$$C\ddot{\psi} - N_r\dot{\psi} - N_\psi\psi = Q - N_{FIN,\psi = 0}$$

where C is the yawing inertia of the helicopter and $N_{FIN,\psi = 0}$ is the yawing moment provided by the fin at zero yaw and which offloads the tail rotor in trimmed cruise flight. The fin and fuselage were considered to contribute to N_ψ but only the fin was considered to contribute to N_r. The analytical forms of the fin contributions to N_ψ and N_r are:

$$N_{\psi FIN} = -\tfrac{1}{2}\sigma\rho V_o^2 S_F l_F a_1$$

and

$$N_{rFIN} = -\tfrac{1}{2}\sigma\rho V_o S_F l_F^2 a_1$$

where S_F is the fin area

l_F is the distance of the fin aerodynamic centre aft of the centre of gravity

a_1 is the lift curve slope of the fin.

Note that the fin stiffness, $N_{\psi FIN}$ depends on V_o^2 and the fin damping, N_{rFIN}, depends on V_o, where V_o is the forward speed. As the forward speed decreases, both the fin stiffness and damping are reduced: the stiffness more rapidly than the damping. This has serious implications on the ability to maintain powered forward flight without a tail rotor at low speeds and when attempting a landing especially in gusty conditions.

The fuselage yawing moment was empirically derived based on past experience (Fig.2). The value of the derivative, $dN/d\psi$, at $\psi = 0$ was used to represent the fuselage contribution to N_ψ. As can be seen, this is a conservative estimate. A further assumption was that the main rotor torque remained constant throughout the following motion after the tail rotor had failed. The main rotor torque will change in response to the constant speed rotor governer's attempts to maintain the rotor speed at a constant value relative to the fuselage. Also, because of the increase in drag in yawed flight, the forward speed will be reduced. It was felt that, although these effects could be incorporated in the non linear aerodynamics model to be discussed next, the scale of programming would reach that of a full simulation and the incorporation of these effects was beyond the scope of this paper.

2.2 Non Linear Aerodynamics Model

Useful results were obtained from the linear aerodynamics model. At this time some very limited wind tunnel data became available. This data indicated that at small angles of yaw either side of the trimmed position (typically $-8^\circ < \psi < 8^\circ$) the fin yawing moment was anything but linear (Fig.3). The reason for this non-linearity was the partial shielding of the fin by the fuselage and rotor hub upstream (Fig.4). This effect varies with fuselage incidence, i.e. the more nose down the fuselage attitude, the less shielded the fin becomes. This can have serious implications on the initial motion and, in particular, the overshoot because the fin is the only means of counteracting the destabilizing fuselage contribution to the static directional derivative N_ψ. Because the fuselage wake reduces the effective fin area, the initial motion will be less stable locally, becoming more stable as the fin clears the fuselage wake (Fig.4). Hence, the fin size must be sufficient to overcome these initially larger accelerations (larger than those occurring if the fin was completely clear of the fuselage wake) due to this locally reduced stability in order to stabilize the resultant following motion and not to exceed the overshoot criterion. This is one reason why the author believes that the design requirement should stipulate a value for the maximum allowable yaw displacement.

The equation of motion was identical to that quoted above. The representation of the aerodynamic forces and moments was more involved. The fuselage yawing moment was given by the expression in Fig.2. The fin yawing moment was given by

$$N_{FIN} = (1 + \frac{d\phi}{d\psi})\tfrac{1}{2}\sigma\rho V_o^{\,2} S_F\ (l_F(C_L\cos\psi - C_D\sin\psi) + \bar{\bar{c}}C_M)$$

where ϕ is the sidewash at the fin

C_L, C_D, C_M are the lift, drag and pitching moments of the fin

and $\bar{\bar{c}}$ is the mean aerodynamic chord of the fin.

As shown in Fig.4, S_F depends on ψ. C_L and C_D were derived from the two dimensional data for a NACA 0012 section, this being the only 2D data available for large incidences beyond the stall. A typical analytically derived fin yawing moment curve is shown in Fig.5.

The yawing moment derivatives were obtained by locally differentiating the yawing moment expressions. The yaw response was solved by proceeding from time $t = 0$ in a piecewise manner, solving the equations at the end of each time step Δt.

3. RESULTS

The fin area required such that the overshoot in yaw did not exceed 20° after a tail rotor failure was determined at a forward speed of 100 kts. Because of the lack of information about the sidewash at the fin and its variation with yaw, it was decided to put $d\phi/d\psi$ to zero giving a conservative estimate of the fin yawing moment and its derivative.

From the outset to avoid having an excessively large fin with consequent weight penalties incurred not just from the additional fin structure but also from the need to strengthen the tail boom to support this additional fin weight and accommodate the fin root bending moment, it was necessary to offload the tail rotor at 100 kts by about 55%. This was achieved by setting the fin at 8° incidence. In practice this can be achieved by a combination of camber and incidence.

The linear aerodynamics model predicted a fin area requirement of 64 ft^2 at 100 kts forward speed (Fig.6). The fin area chosen as a starting point, sized by present day rules and not aimed at tail rotor loss designs, was 35 ft^2. Examples of the response at 75, 125 and 150 kts are also presented in Fig.6. These figures show that because the stiffness increases faster than the main rotor torque as the forward speed increases, the overshoot and steady state yaw are reduced as speed increases. As speed decreases, the stiffness decreases and the main rotor torque increases below the minimum power speed. The ability to survive a tail rotor loss at low forward speed becomes doubtful although an emergency landing may be possible. The same arguments apply to a reduced fin area.

The non linear aerodynamics model predicted a fin area requirement of 82 ft^2 at 100 kts (Fig.7). This increase in fin area compared to the linear model is mainly the consequence of the blocked fin at small angles of yaw. Fig.7 also shows the response at 75 kts, 125 kts and 150 kts for this fin. The results are similar to those obtained for the linear aerodynamics model. Note that the damping of the motion has been increased compared to the linear model by the inclusion of the drag term in the fin yawing moment expression. Comparing the linear aerodynamics model with a fin area of 64 ft^2 to the non linear aerodynamics model with a fin area of 82 ft^2 (Figs.6 and 7), apart from the already mentioned damping increase, the responses at 125 kts and 150 kts are remarkably similar. This is not surprising because the displacements involved do not exceed the fin stall value i.e. the relevant fin aerodynamics are essentially linear in the non linear model. As forward speed decreases, the differences in response become more discernible as the linear model greatly over predicts the fuselage yawing moment especially at the large displacements involved at 75 kts.

The steady state sideslip power-off condition is given by

$$N_\psi \psi = N_{FIN\psi} = 0$$

where ψ must be less than -10° and is only met by the non linear model. This requirement needs careful consideration because, at the stall, N_ψ changes sign twice: at C_L max and at the bottom of the C_L break.

To meet the tail rotor loss design objectives, the use of a fin is possible but large fin areas are required even with offloading the tail rotor by about 55%. To achieve this increase in area whilst maintaining the same fin moment arm, the height of the fin would have to be increased: adding area in front of the fin decreases the moment arm and main rotor clearance. However, increasing the height implies increasing the height of the fin aerodynamic centre causing an increase in roll-yaw coupling to add to the pilot's problems. Unless a rudder is fitted, controllability after tail rotor loss is doubtful at low speeds and a run-on landing power-off is necessary. These disadvantages may not be acceptable. An ability to land in confined spaces may be required.

4. OPERATIONAL ASPECTS OF HELICOPTERS WITH LARGE VERTICAL TAIL SURFACES

The preceding sections have shown that survival after a tail rotor loss is enhanced by the use of a large, cambered fin, Fig.8. One advantage of a large fin is that it can be incorporated into the design of a conventional single tail rotor helicopter with the minimum of effort. Also, the tail rotor may be off-loaded, by approximately 55% in the example considered here, by the use of fin incidence. This causes a reduction in tail rotor flapping which decreases the Coriolis lag bending moment and consequently improves the fatigue life of the tail rotor blades.

The major disadvantages of incorporating a large fin centre on the low speed handling problems.

Lynn, Ref.4, has shown that the percentage thrust loss of a pusher tail rotor due to the presence of a fin decreases approximately as the reciprocal of the fin-rotor separation distance, Fig.8. From the limited data available, the 'fin blockage' appears to be approximately a linear function of the ratio between the blocked disc area and the total tail rotor disc area. Hence, the thrust loss will be increased by the use of a fin which is between two and three times as large as it might have been. The results presented in Ref.5 imply that the increase in fin blockage due to, say, 35 kts sideways flight, Fig.8, remains approximately independent of fin rotor separation. Although the fin blockage can generally be reduced by increasing the fin tail rotor separation distance, there will still be a blockage effect in side and quartering flight which can reduce the thrust by as much as 15% of the tail rotor thrust. An additional problem is the interference between the main rotor tip vortex and the tail rotor. For a given pitch, the tail rotor can suffer a thrust loss as high as 20% on certain headings even with top blade aft rotation, Fig.8. It is highly desirable to reduce the fin blockage contribution to the tail rotor thrust requirement in order to diminish the low speed quartering flight handling problems. A further problem on some designs is the restriction placed on the available tail rotor pitch due to the collective interlink system which may result in the pilot running out of pedal.

The above mentioned problems are reasonably well known. However, these problems do incur power penalties in the low speed flight regime. As in Ref.3, this can lead to the updating of the tail transmission system on large fin helicopters, especially if the fin blockage is initially underestimated as is often the case.

The tail rotor system having the lowest weight will be that found on helicopters having the smallest fin area possible, i.e. the minimum structure required to support the tail rotor gearbox. For this design, the likelihood of a tail rotor failure is diminished by improving the reliability of the tail rotor system. This provides no capability for surviving a tail rotor failure when it occurs.

5. TAIL ROTOR LOSS AT SPEEDS LESS THAN THE MINIMUM POWER SPEED

It was demonstrated in an earlier section, that the use of a 82 ft^2 cambered fin at incidence on the example helicopter should ensure the survival of a tail rotor loss at speeds as low as the minimum power speed (taken to be 100 kts). As the speed decreases below the minimum power speed, the main rotor torque increases and the aerodynamic stiffness decreases such that the overshoot in yaw increases significantly. Under these circumstances, it is doubtful if the pilot is capable of taking the correct action of reducing main rotor collective and entering autorotation. At these low speeds, the helicopter may enter a spiral drive after initially suffering large displacements in sideslip, pitch and roll which are accompanied by significant lateral and longitudinal accelerations.

Naturally, it can be argued that the major part of a mission is spent at high speed cruise. This may also coincide with the lowest probability of encountering ballistic damage in the case of a military helicopter. This argument is of small consolation if the tail rotor fails just after take-off or just before landing.

Even if the low speed handling penalties of a large fin are acceptable, it is obviously highly desirable to have the capability of surviving a tail rotor loss at all speeds including the hover.

6. THE TWIN TAIL ROTOR AS A SOLUTION

A yaw stabilizing device which is insensitive to forward speed is required in order to maintain some control in the hover after a tail rotor failure, such that a safe landing can be executed. A rotor is such a device. It is, therefore, worthwhile considering the use of twin tail rotors such that if one should fail, the remaining rotor offers good yaw stability and sufficient control for emergency manoeuvring and landing.

As is shown in Fig.9, where the response of the twin tail rotor is compared to that of a fin with area 82 ft^2 at a speed of 100 kts, the dynamic overshoot is readily contained, especially if some stability augmentation is used on the remaining tail rotor. Figure 10 shows that the loss of one tail rotor can be contained in the hover. Control with one tail rotor will be limited depending on the design philosophy but it should be adequate for a safe landing in a confined space.

It has been assumed that each tail rotor has two blades of the same characteristics as the single tail rotor, the geometry of which had been determined by present day design rules for the example helicopter. The analytical forms of the tail rotor contributions to N_ψ and N_r are:

$$N_{\psi_T} = -\tfrac{1}{2}\sigma\rho A_T V_T V_o l_T \frac{dC_T}{d\hat{v}}$$

and

$$N_{r_T} = -\tfrac{1}{2}\sigma\rho A_T V_T^{\,2} l_T^{\,2} \frac{dC_T}{d\hat{v}}$$

where A_T is the tail rotor disc area

l_T is the distance of the tail rotor centre aft of the centre of gravity

V_T is the tail rotor tip speed

C_T is the tail rotor thrust coefficient and

$\hat{v}$ is the sideslip non dimensionalized with respect to tip speed.

Note that in this case, the stiffness, N_{ψ_T}, depends on V_o but the damping N_{r_T} is not directly dependent on V_o. Consequently, for a tail rotor loss whilst hovering, a steady yaw rate can be achieved. This is not possible with the fin solution.

In general, a single tail rotor drive system has a final gearbox at the tail rotor head and an intermediate gearbox at the foot of the fin. It may be possible to use one central final gearbox having one input and one output shaft provided frangible couplings are incorporated in the final drive to each tail rotor. This reduces the number of gearboxes by one. Alternatively, for reliability reasons, it may be decided to use separate drive shafts to each rotor. The number of gearboxes has not been increased and the weight of the extra drive shaft is a small fraction of the total drive system weight. The drives to each rotor will only need to carry half the total torque required by the equivalent single tail rotor thus reducing the weight of the final gearboxes. Vulnerability to drive failure from ballistic damage is also reduced.

Elimination of the fin structure also reduces weight. In addition, the weight of the tail boom could be reduced since the torsional loads generated by a single tail rotor and fin configuration have been almost eliminated by adopting a shallow 'V' tail layout in which the final rotor shafts are in line with the tail boom counter line.

A more complex control system will incur a weight penalty. This should be designed with the loss of one tail rotor in mind.

The possibility of overcoming some of the low speed handling problems discussed earlier exists. With a suitable tailplane span, the problem of the trailing vortex from the main rotor interfering with the tail rotor should be alleviated by the use of twin tail rotors, Figure 11. Figure 11 also shows how this interference effect is expected to become less severe and to be distributed over the low speed flight envelope.

Apart from a small dihedral effect, fin blockage is eliminated. The tail rotors' thrust and power requirements inside and quartering flight are minimised. The dihedral angle can be chosen to alleviate or remove the mutual interference of the two rotors in hovering flight.

The twin tail configuration also ensures that the fuselage wake passes clear of the tail rotors so that, for small sideslip angles depending on the tailplane span, the tail rotors yawing moment will not suffer any shielding effects as in the case of a central fin and tail rotor, Figure 12. This also helps to contain the dynamic overshoot after the loss of one tail rotor because the initial local dynamics are important as discussed previously.

7. CONCLUSIONS

In order to limit the severity of the motion which results from the loss of a tail rotor, the author suggests that a more realistic criterion for the determination of fin area is:-

*The yaw overshoot should be no greater than 20° at the minimum power speed.

A fin capable of satisfying this criterion may give rise to handling problems in low speed flight and consequently may only be a valid solution for certain types of helicopters, where high agility near the hover is not a specific mission requirement and where landings are not required to be within the confines of an oil platform or ship deck.

A solution which offers to overcome these problems and can do better than the above criterion, even in the hover, is the use of a twin tail rotor configuration which has the following major advantages with regard to tail rotor loss:-

*Minimal yaw disturbance following the loss of one rotor, especially if stability augmentation is used.

*Ability to survive at all speeds.

*Control retained in the hover to give the capability of landing in confined spaces.

REFERENCES

1) M.L. Hester, et al., Handling Qualities Aspects of the Bell Model 222 Design and Development Program, Paper presented at 34th Annual National AHS Forum, 1978.

2) T.J. Horst and R.J. Reschak, Designing to Survive Tail Rotor Loss, Paper presented at 31st Annual National AHS Forum, 1975.

3) K.B. Amer, et al., Handling Qualities of Army/Hughes YAH-64 Advanced Attack Helicopter, Paper presented at 34th Annual National AHS Forum, 1978.

4) R.R. Lynn, et al., Tail Rotor Design, Part I: Aerodynamics, Paper presented at 25th Annual National AHS Forum, 1969.

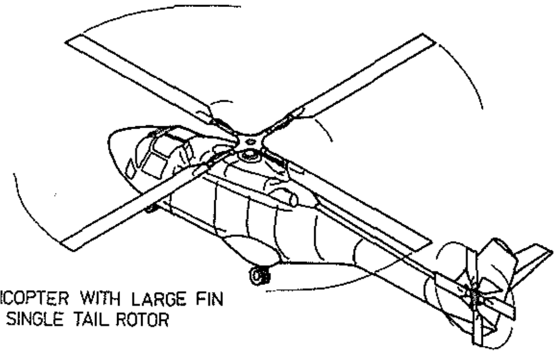
5) W. Wiesner and G. Kohler, Tail Rotor Performance in Presence of Main Rotor, Ground and Winds. Paper presented at the 29th Annual National AHS Forum, 1973.

ACKNOWLEDGEMENT

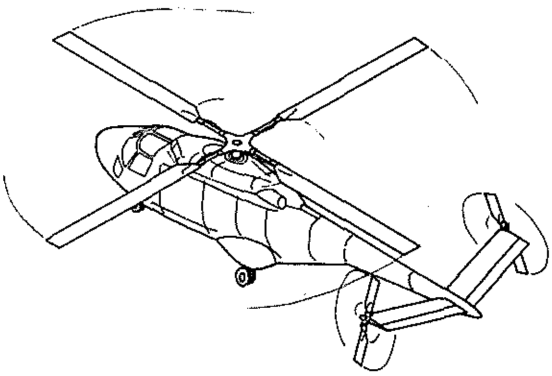
The author would like to acknowledge the support of the Aerodynamics Department, Westland Helicopters Limited, Yeovil, Somerset, and the help of A. Brocklehurst, Westland Helicopters, in preparing this paper.

FIGURE 1

ALTERNATIVE HELICOPTER TAIL CONFIGURATIONS



HELICOPTER WITH LARGE FIN AND SINGLE TAIL ROTOR



HELICOPTER WITH TWIN TAIL ROTORS

FIGURE 2

FUSELAGE YAWING MOMENT

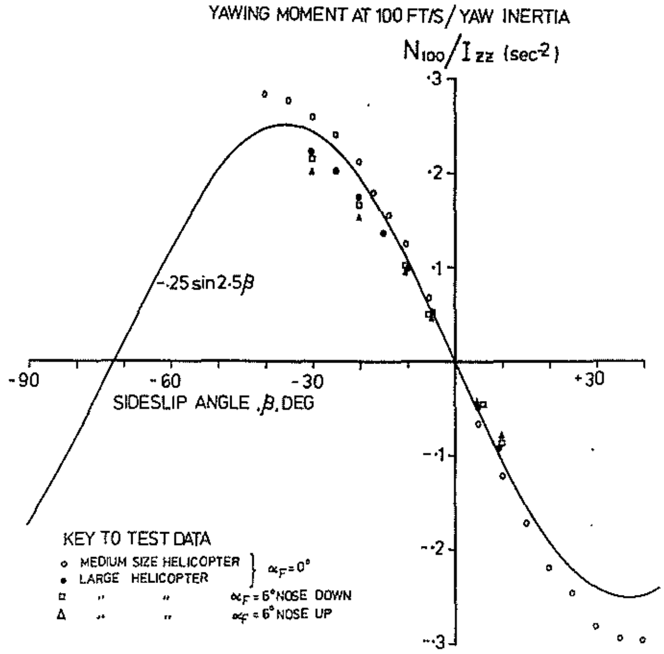
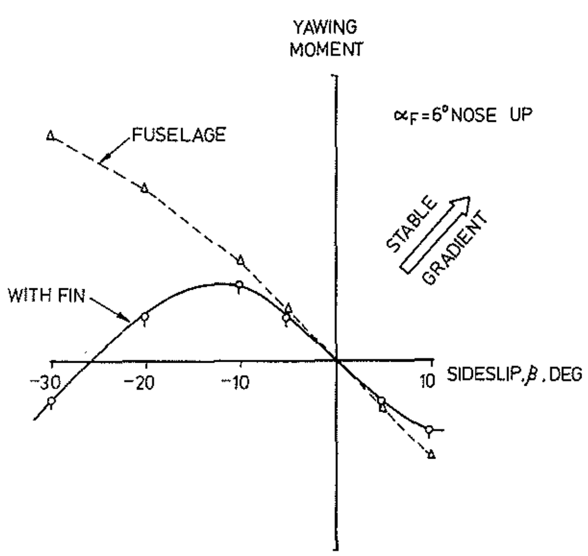


FIGURE 3

WIND TUNNEL DATA

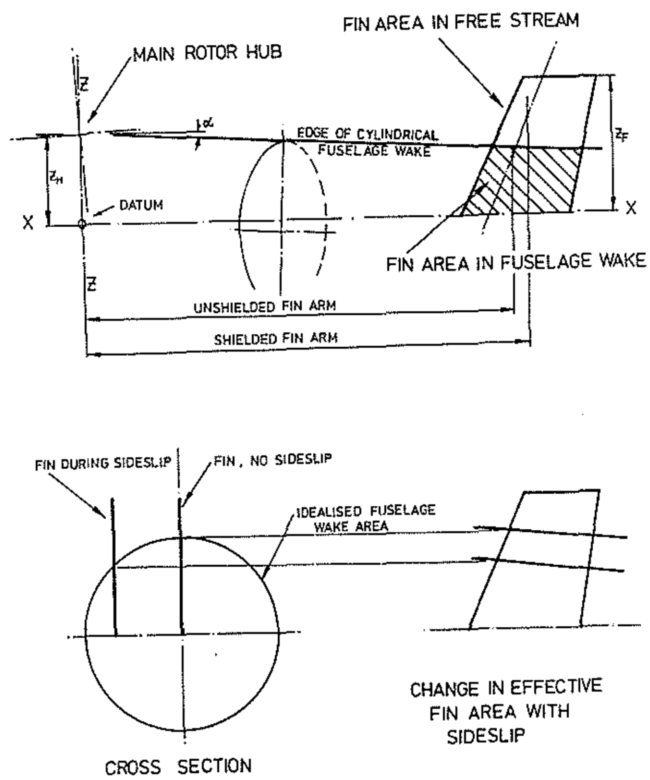
FUSELAGE AND FIN YAWING CHARACTERISTICS



TYPICAL HELICOPTER CHARACTERISTIC

FIGURE 4

DIAGRAM OF FIN SHIELDING



CROSS SECTION (FROM REAR)

CHANGE IN EFFECTIVE FIN AREA WITH SIDESLIP

FIGURE 5

COMPARISON OF THEORETICAL AND EXPERIMENTAL FIN NON-LINEAR YAWINGING MOMENT

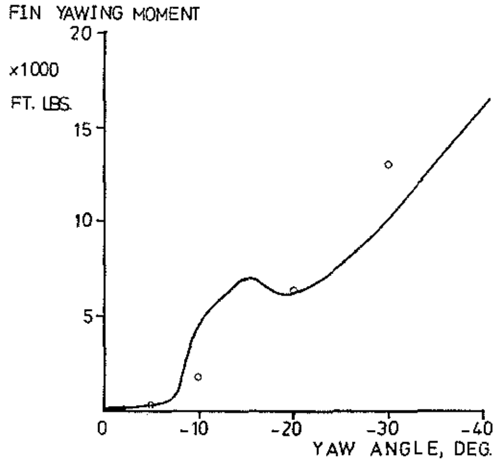


FIGURE 6

YAW RESPONSE AFTER TAIL ROTOR LOSS WITH FIN LINEAR MODEL

$S_F = 64\ FT^2$, $\alpha_S = -8°$

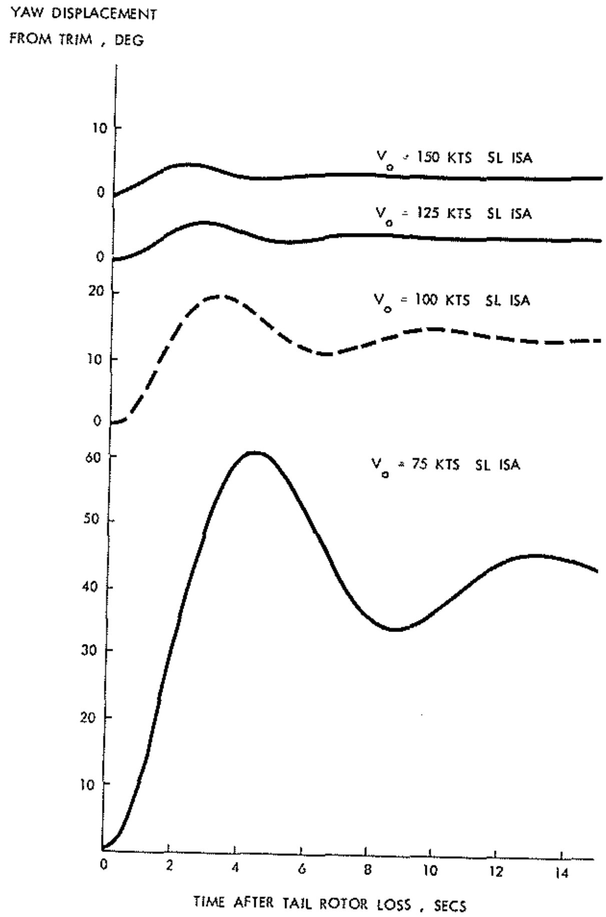


FIGURE 7

YAW RESPONSE AFTER TAIL LOSS WITH FIN NON LINEAR MODEL

S_F 82 FT^2 , α_S - -8°, α_F - 3°

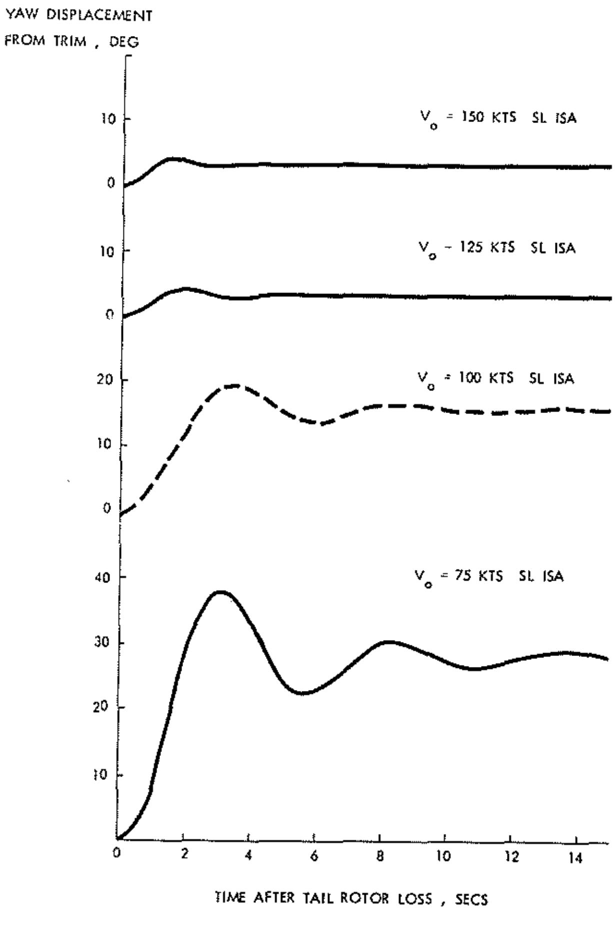
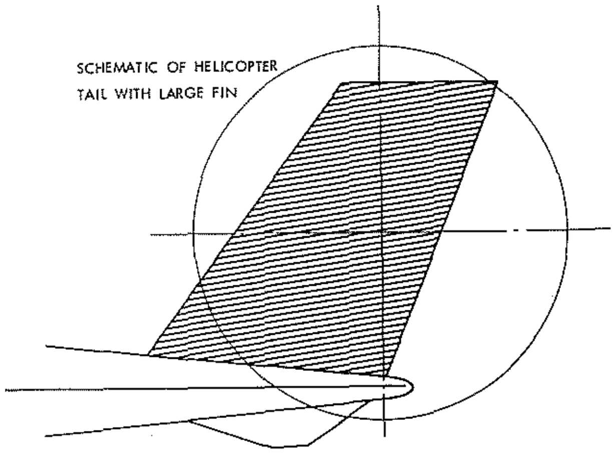


FIGURE 8



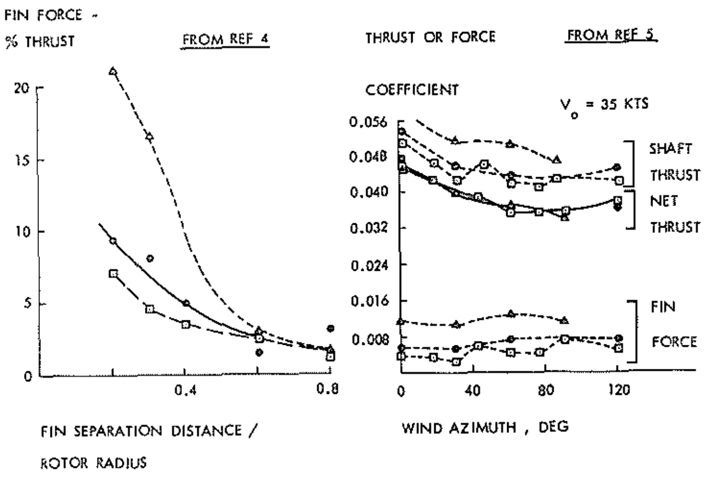


FIGURE 9

YAW RESPONSE AFTER TAIL ROTOR LOSS TWIN TAIL ROTOR

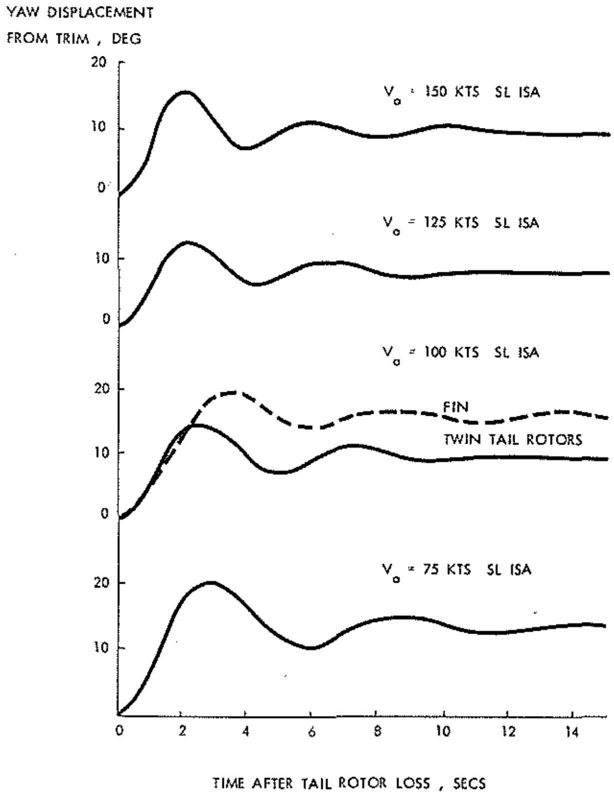


FIGURE 10

RATE OF YAW IN HOVER AFTER TAIL ROTOR LOSS TWIN TAIL ROTOR

(SL ISA)

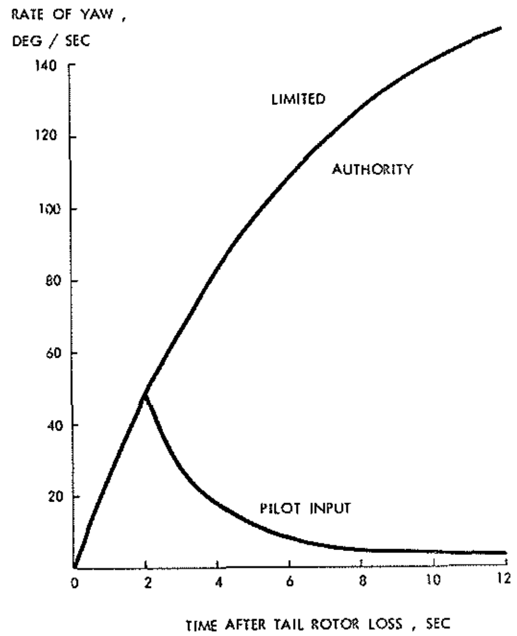
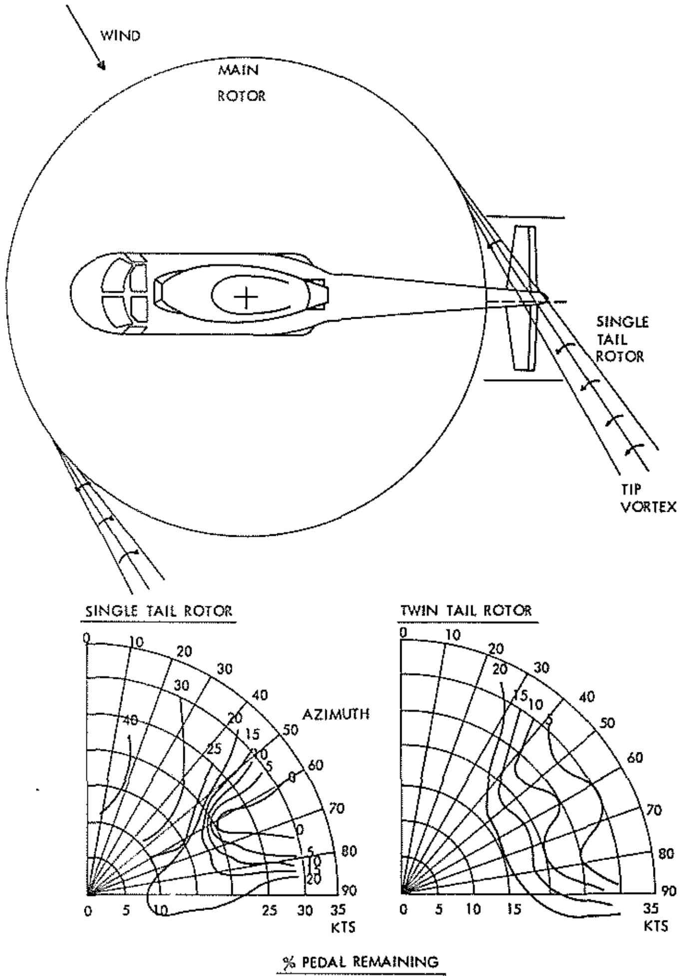


FIGURE 11



% PEDAL REMAINING

FIGURE 12