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ESTIMATE OF LIMIT LOADS FOR DESIGN OF HELICOPTER DYNAMIC STRUCTURES

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ABSTRACT

Helicopter dynamic structures (main rotors, tail rotors and dynamically loaded parts of the control system) are mainly designed by fatigue since the preponderant loading is by oscillatory loads of large magnitude and high loading frequency. A considerable effort is expended to evaluate these loads in various flight regimes and establish safe service lives for the components.

Static limit loads which occur only once or a small number of cycles are more difficult to define. Their importance in assuring structural integrity in violent maneuvers or emergency conditions has not received a comparable amount of attention. Underestimation of these loads can make the structure vulnerable to low cycle fatigue such as the ground-air-ground cycle. A study of peak loads generated in recorded maneuvers during military structural demonstrations on various two-blade rotor systems is helpful in establishing limit loads for new designs.

I. INTRODUCTION

The Structures Engineer faced with the task of sizing dynamic structures for rotorcraft for minimum weight and structural adequacy must pay close attention to analysis of the structures for fatigue. By dynamic structures is meant those components of the rotorcraft which are subjected almost continuously to loads of a strongly varying nature, of large amplitude and a frequency which is equal to or a multiple of the rotational frequency of the rotors. Such structures encompass the rotor blades and hubs of both main and tail rotors, the control systems rotating with those rotors and the control systems not rotating with the rotors but on the output side of control system power actuators. Pylon or rotor and transmission mounting structure can also be included. This structure may contain vibration attenuating devices.

Methods to analyze these structures for fatigue have been developed to a fairly high degree. The magnitude of the oscillatory loads which are developed in normal flight conditions can be established quite accurately. High speed level flight is used as a basis for this fatigue analysis with factors to be applied to the loads appropriate for the amount of maneuvers and the types of material (Reference 1). The stabilized condition of high speed level flight can be analyzed using computer programs that take most relevant parameters as well as dynamic response of the structure into account.

Correlation checks on the results from analysis can be obtained from actual flight tests of prototypes or scaled from previous tests on similar structures.

The same computer programs used for analysis of loads in level flight can be used for analysis of maneuvers. Experience shows however that more refinements of the analysis are required and results are often less accurate. Correlation is more difficult to obtain even for maneuvers of a trimmed, steady nature which are of importance for fatigue analysis.

Maneuvers of a non-steady and non-trimmed character which approach or exceed aerodynamic or other limits do not correlate at all well. In many ways simplifying assumptions must be made and the actual aerodynamics and dynamics are not clearly understood. Correlation with flight measurement becomes extremely difficult since executions of such maneuvers cannot be reproduced exactly. For purposes of fatigue analysis these maneuvers are of less interest as the number of cycles accumulated at the peak load condition is extremely small. For this reason this type of maneuver is usually not included in the normal flight spectrum used for fatigue evaluation.

This type of maneuver is of interest to the Structures Engineer since the peak loads generated approach limit conditions. The structure should be substantiated for these conditions.

2. LIMIT LOADS

It is good practice to design helicopter structures in such a way that a failure can be tolerated by making the design redundant. In many cases this is difficult to accomplish and it may lead to excessively heavy designs.

A non-redundant structure must be substantiated for the highest peak load that can be expected in service. To establish what is the highest peak load that can be expected it is possible in some cases to define limits that are physically impossible to exceed. One such limit is the maximum lift that can be generated by an aerodynamic lifting surface. As a helicopter rotor is a rotating lifting surface it follows that there will be a definite limit to the thrust that can be generated by the rotor.

Besides lending itself well to analysis, extensive measured data are available on the thrust that can be developed by a helicopter rotor. Reference 2 mentions a collection of data on the load factor which is the ratio of thrust to gross weight. Load factors on AH-1G gunships recorded in operation under combat conditions in Southeast Asia have shown that this helicopter can transiently achieve load factors well above the stall limits (Reference 2). Figure 1 shows peak values of the blade load coefficient as measured during structural demonstrations of various military helicopters at BHT.

A structural demonstration consists of a number of specified maneuvers intended to explore the handling and performance limits of the helicopter. The measured load factor peaks, both positive and negative, give a good indication of thrust limits if it is kept in mind that they are conservative to the extent that they include airloads on airframe and wings. This conservatism could be avoided if a device could be perfected that measures rotor thrust direct instead of through the load factor. At the same time the measured peaks do not always represent aerodynamic limits since the maneuver may have been called off for other reasons such as vibrations, feedback loads or rate limits.

3. OUT-OF-PLANE LOADS

Maximum thrust load can be applied to the blades as a static load together with the centrif-

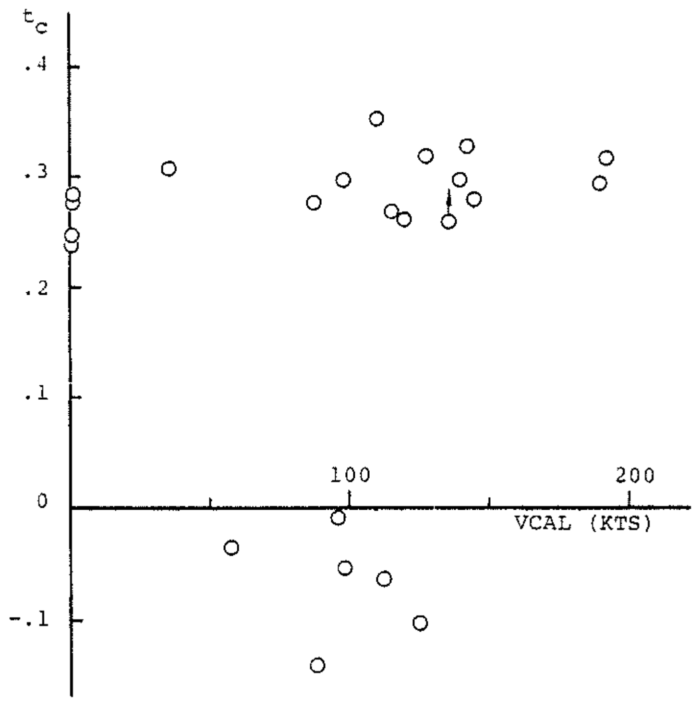

Figure 1 Peak Blade Load Coefficients Derived From Load Factors Measured in Maneuvers

ugal loading. As the blade is flexible in this direction the deflections will modify these out-of-plane bending moments. Figure 2 shows the results of such analysis for a twin engine attack helicopter. The analysis for the limit condition results in a load distribution shown by the dotted line. To arrive at limit loads it is necessary to include dynamic effects. Limit loads are derived by adding oscillatory loads to the static load distribution. Oscillatory loads can be added as obtained from analysis of maneuvers or as derived from factored level flight loads.

As a check on how this load distribution compares with peak loads measured in a structural demonstration, data points from the structural demonstration of this attack helicopter are shown. There seems to be a good confirmation that the limit load distribution determined through the described process is close to the actual limit load envelope.

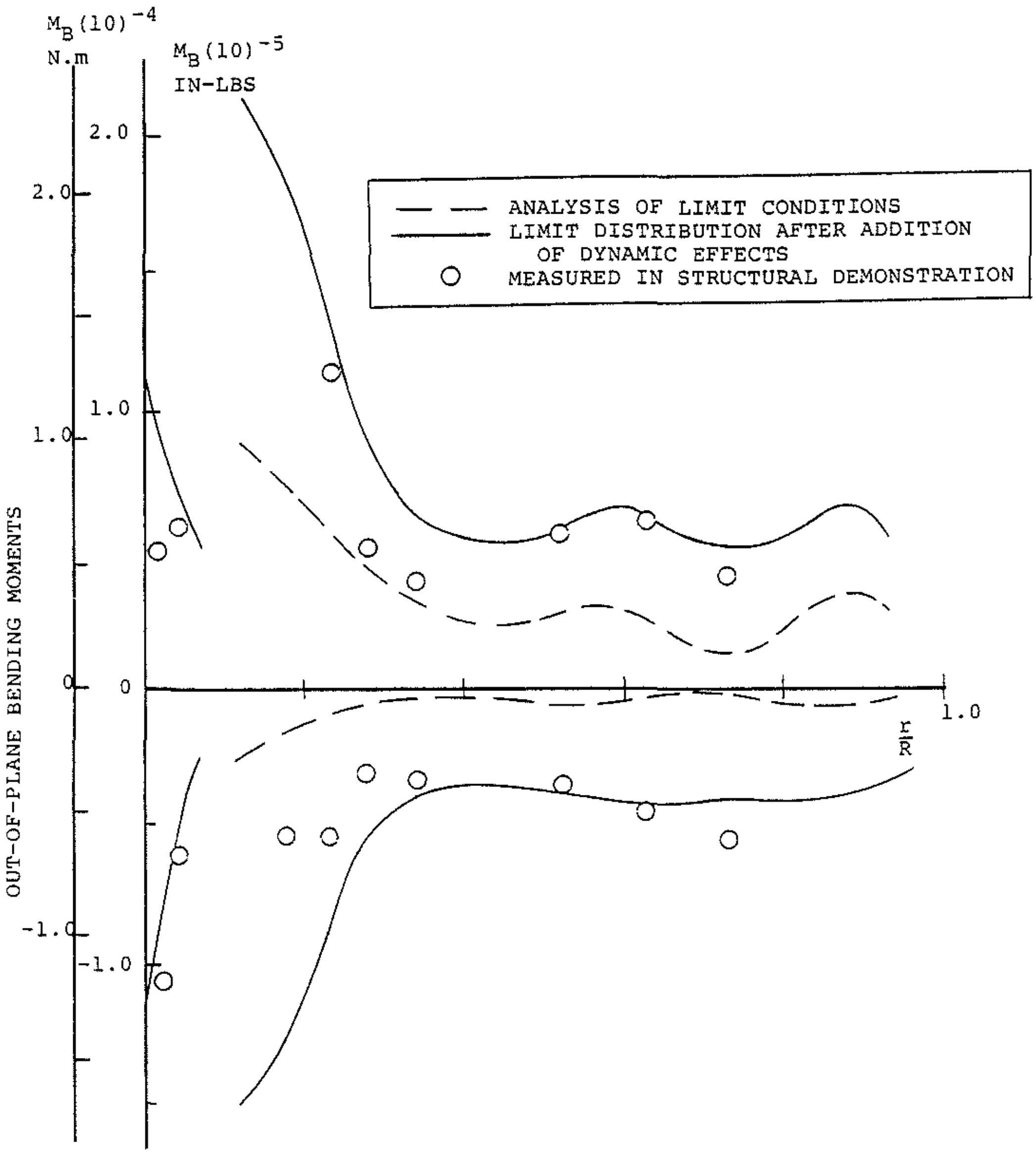

Figure 2 Out-of-Plane Limit Bending Moments on Main Rotor of Twin Engine Attack Helicopter

4. IN-PLANE OR CHORDWISE LOADS

The steady state condition for in-plane loads can be either powered or unpowered. In the powered condition the drive torque is distributed equally over the blades and is absorbed along the blade by aerodynamic loads. In the unpowered condition drive torque is zero or slightly negative.

Oscillation of in-plane loads occur in addition to the steady state loads. Semi-rigid two-bladed rotors act as free-free beams when aerodynamic impulses cause this beam to oscillate. This motion is very lightly damped. The oscillations caused by forward flight are aggravated by coupling with the three/rev beamwise mode (also called "S"-ing mode which often is close to resonance near operating conditions).

The bending moment distribution along the blade as a result of the oscillation is not too different from the steady state power-on distribution, the maximum moment being at the drive shaft and the moments dissipating along the blade by dynamic loads. To arrive at a limit load distribution it becomes convenient to express the peak moments at station zero in terms of main rotor drive torque $Q_{M/R}$ as follows:

$$M_{C\ Lim} = \frac{Q_{M/R}}{B} (1 \pm K) \text{ power on}$$

$$M_{C\ Lim} = \frac{Q_{M/R}}{B} (0 \pm K) \text{ power off}$$

Figure 3 shows values of the factor K plotted against drive torque $Q_{M/R}$ for a number of two-bladed main rotors as measured in structural demonstrations.

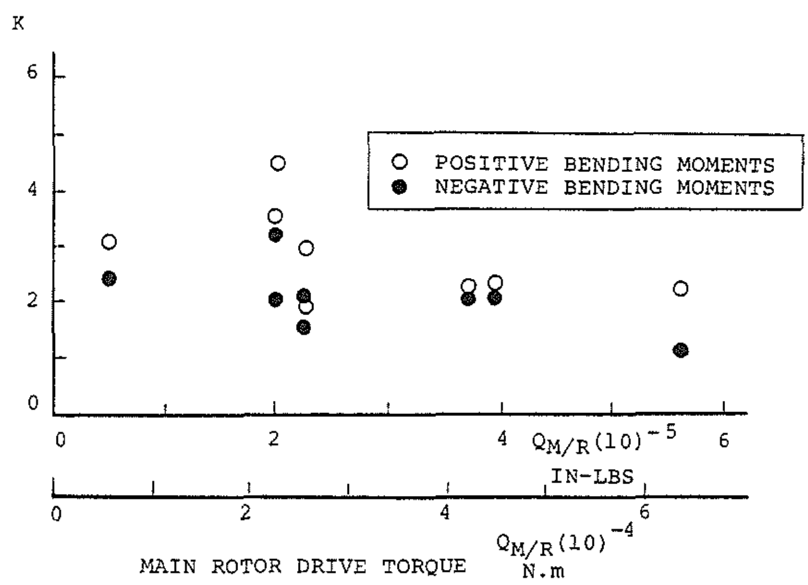

Figure 3 In-Plane Moment Factor K

Once a value of K for design has been decided upon a limit bending moment distribution along the blade can be found by balancing the root chord moment with inertia loads along the blade. To this the contribution of chord moment due to centrifugal load has to be added. This contribution is due to the fact that in each section the resultant of centrifugal load of the blade outboard of this section does not necessarily coincide with the section neutral axis.

Figure 4 shows the chord moment distribution for analysis on the Cobra rotor together with the peak loads envelope measured during load surveys and during structural demonstration.

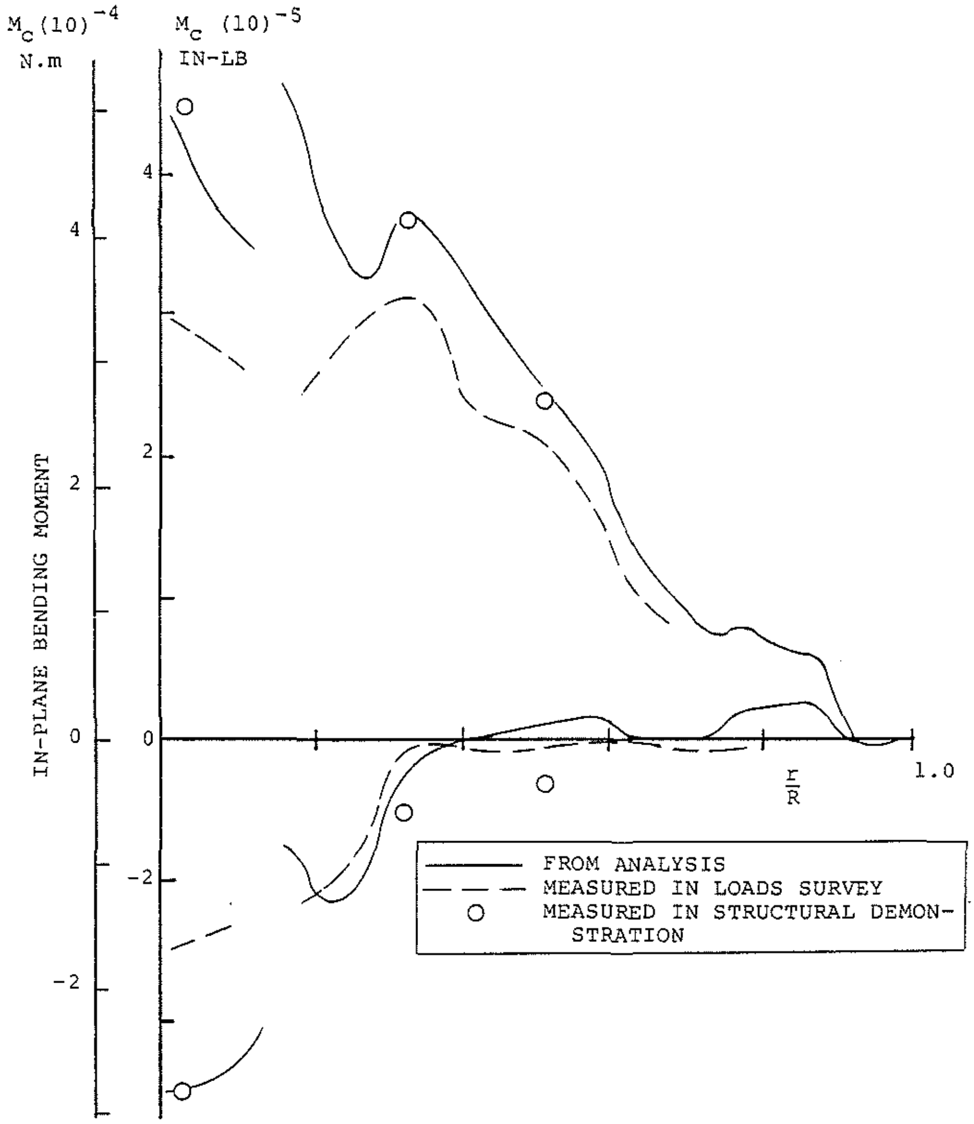

Figure 4 In-Plane Limit Bending Moment Distributions for Cobra Main Rotor

5. TORSION LOADS

Torsion loads in rotor blades result from a variety of causes. At the root of the blade they become apparent as control system loads. The loads have a steady component which is mainly dependent on collective blade pitch angle and inertia about the blade lengthwise axis (important for large chord blades) also on gross weight and blade radius. The other component of blade torsion load is oscillatory. Besides being also dependent on gross weight, blade radius and chord length the oscillatory torsion loads are dependent on forward speed.

Figure 5 shows the character of measured torsional moments along the blade on a medium utility helicopter. From this it can be seen that nose-down pitching moments (negative) are amplified considerably in the structural demonstration. If level flight oscillatory moments are taken as a base, peak torsional moments can be expressed in terms of level flight oscillatory moments. Level flight oscillatory feathering moments can be estimated by means of simple formulae and comparison with measured data on comparable rotors.

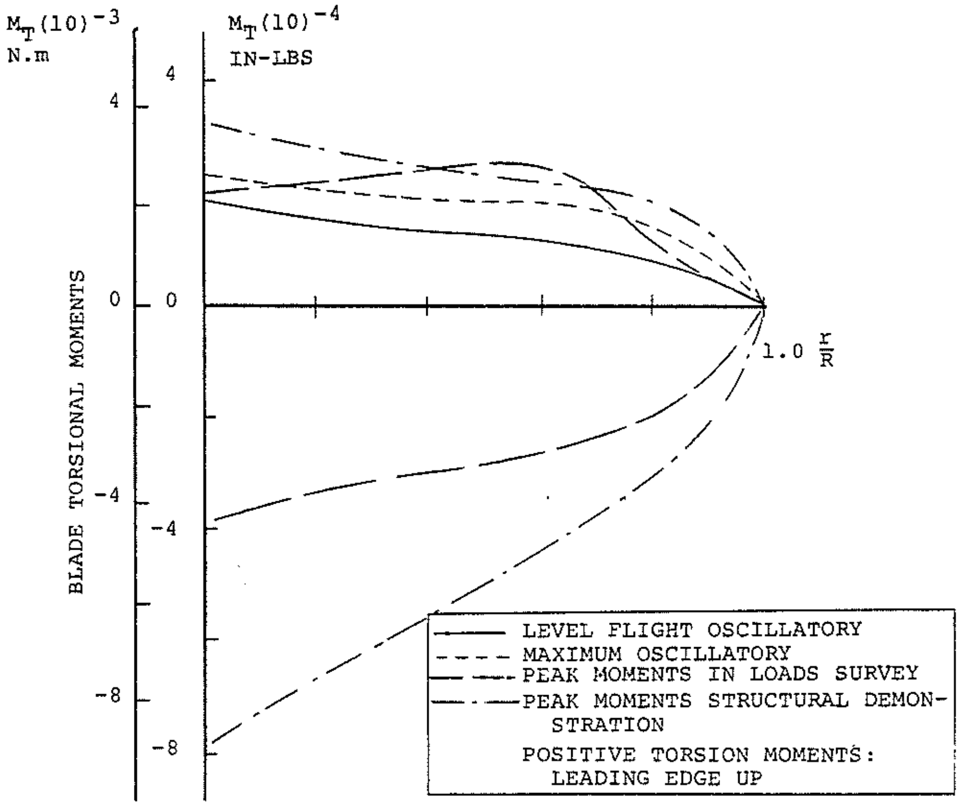

Figure 5 Torsional Moment Distribution on Main Rotor of Medium Utility Helicopter

Figure 6 shows the ratio of peak feathering moments at the control arm to level flight oscillatory moments for a number of helicopters. The ratios are plotted against blade chord which is an important parameter for control loads. It can be seen that peak nose-down feathering moments can reach a value of six times level flight oscillatory moments while nose-up feathering moments peak at only half that value.

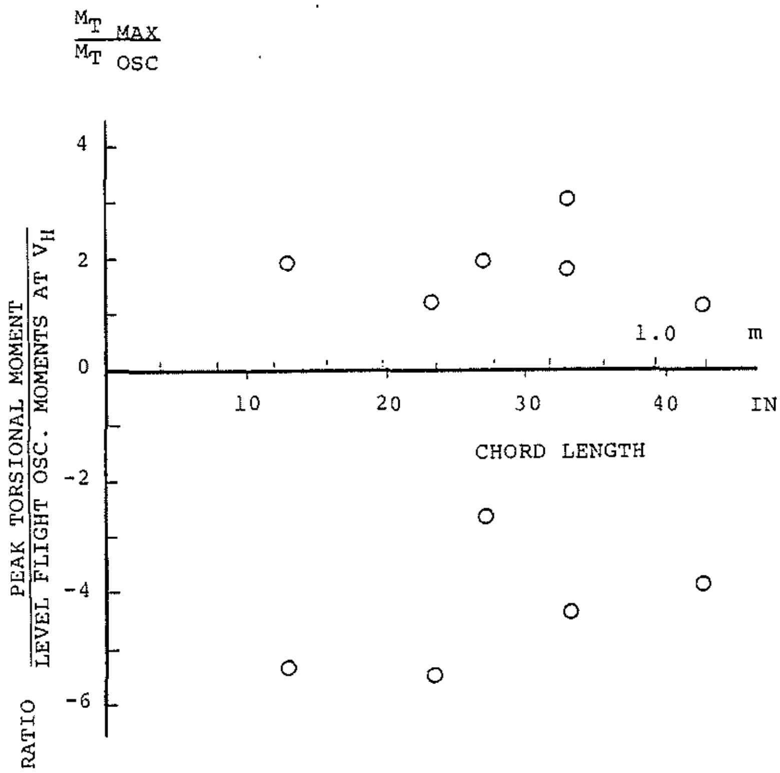

Figure 6 Ratio of Peak Blade Control Moments to Level Flight Oscillatory Moments

A word of caution should be interjected here. In most cases the torsional distribution is such that the feathering moment to the controls is higher than the moments along the blade. In conditions of deep stall on the blade this is not necessarily true.

Figure 7 shows a distribution measured on a Cobra gunship during its structural demonstration in such a condition. It shows a moment at about 40% of radius, roughly 25% greater than the control moment at the blade root. This distribution peak was attributed to stall of the inboard portion of the blade in a left rolling pull-out.

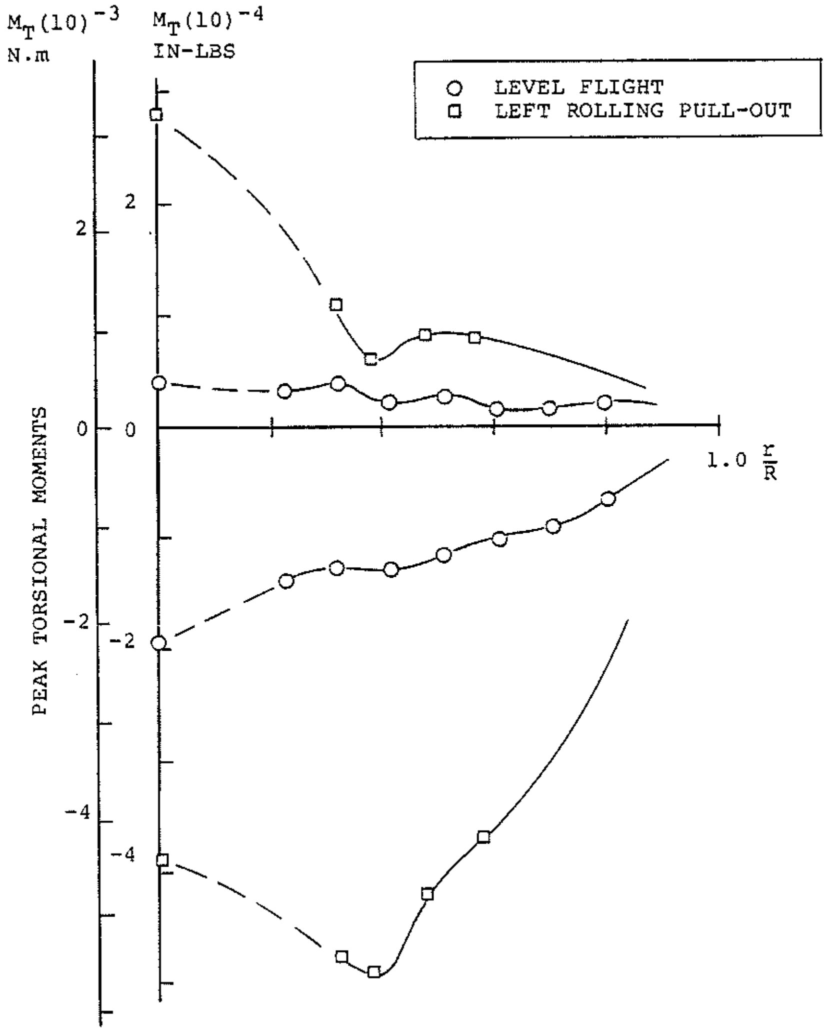

Figure 7 Peak Torsional Moment Distribution on Cobra Main Rotor Blades Due to Stall

6. TAIL ROTORS

The thrust of a tail rotor is subject to aerodynamic limits quite similar to main rotors. Blade load coefficients can be chosen identical to those for main rotor blades for a realistic estimate of limit thrust.

In the context of this paper the major difference between tail rotors and main rotors is caused by the fact that the power supplied to the tail rotor is not limited to installed power as is the case for the main rotor. Depending on flight conditions the power available to the tail rotor is virtually unlimited since the main rotor and its inertia can supply whatever instantaneous power the tail rotor needs. It is therefore quite possible to drive a tail rotor into stall.

As the power supplied to the tail rotor is a measure for the loads, it is of interest to know the magnitude of peak torques developed during maneuvers. Figure 8 shows peak power as measured during structural demonstrations as a ratio of main rotor power for a number of Bell helicopters. These are all of the more or less standard type with non-shrouded tail rotors and a tail boom length which allows the tail rotor to clear the main rotor. It shows that tail rotor power can reach a value of 35% of main rotor power and that even negative power can be developed up to 11%.

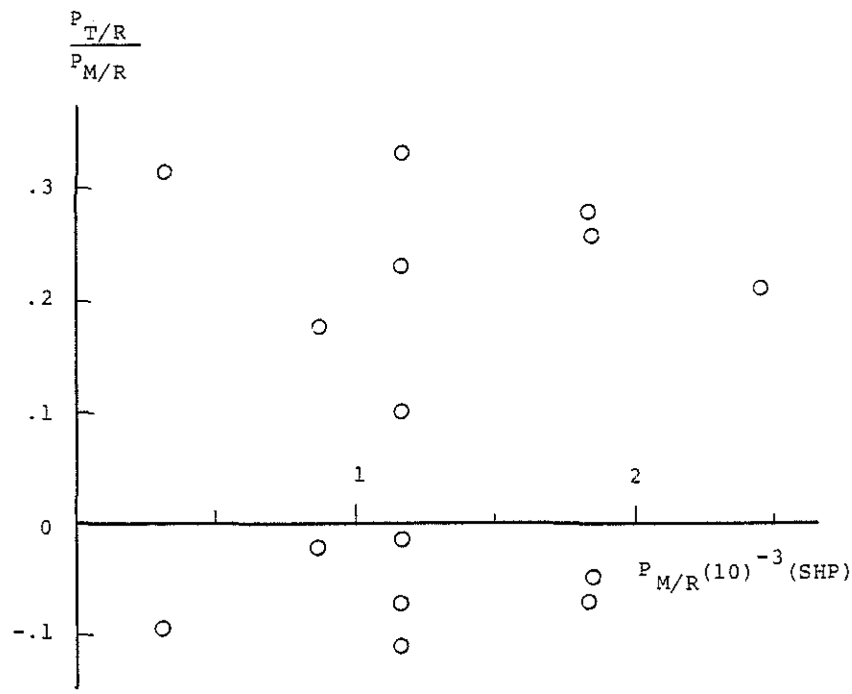

Figure 8 Ratio of Peak Tail Rotor Power to Main Rotor Power

Tail rotor in-plane loads can be expressed in terms of tail rotor drive torque similar to main rotors. However, the definition of basic design drive torque can vary between helicopters. To avoid this difficulty the peak tail rotor in-plane moments at the hub can be expressed in terms of main rotor drive torque. Figure 9 shows the results when this is done for a number of helicopters. This figure shows that peak in-plane bending moments on tail rotors can reach a magnitude of 6% of main rotor drive torque in positive as well as negative direction.

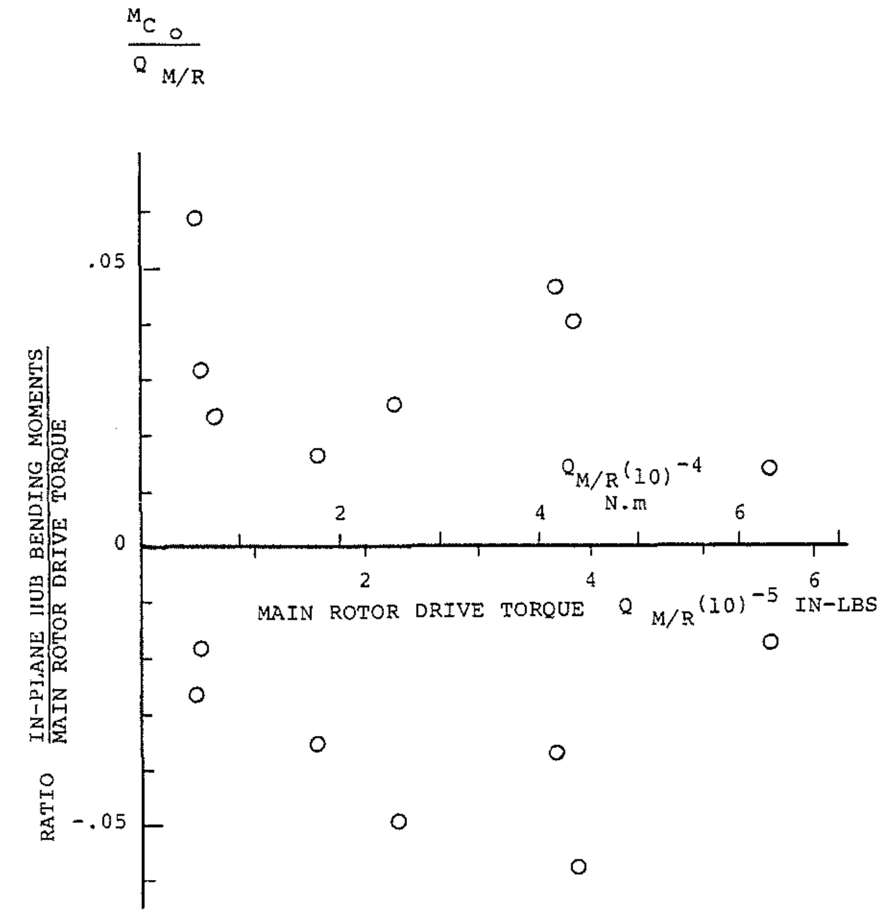

Figure 9 Ratio of Peak Tail Rotor In-Plane Bending Moments to Main Rotor Drive Torque

7. CONCLUSIONS

- Analysis of helicopter dynamic components to limit loads is a requirement which must be met for structural substantiation.

- Establishing limit loads is complicated by the necessity of including dynamic effects to steady limit conditions. Mathematical analysis of these dynamic effects is at present unsatisfactory because of difficulties in balancing the system and because of abnormal and unsteady aerodynamic conditions.

- The severe maneuvers leading to limit conditions of the dynamic components are normally outside the scope of the flight loads measured in the fatigue spectrum because their contribution to fatigue is negligible.

- Structural demonstrations which are a part of military qualification procedures are intended to explore the limits of helicopter performance, handling and structural characteristics. A study of time histories and peak loads developed during the maneuvers is helpful in determining practical limits for design of new structures.

- More emphasis on and analysis to actual peak loading conditions may make the structure less vulnerable to unorthodox operational uses and unexpected low cycle fatigue problems.

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