The Structural Airworthiness of Helicopters with particular reference to Fatigue Failure

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SUMMARY

The importance of reliable airworthiness criteria is stressed in an introductory paragraph Practice in aeroplane design is reviewed briefly, noting particularly the current tendency to permit increasingly severe fatigue loads The principal loads on the elements of a helicopter are enumerated, and consideration of their repetitive nature leads to an extended discussion of fatigue failure and methods of correlating fatigue loads Suggestions are offered for a rationale of provisional strength requirements and the amendments desirable following the establishment of additional data The need in this connection for extensive V-g and blade stress records taken in operation is emphasised

ESSAY

Many people in Great Britain are now confident that helicopters will shortly be in general use for a variety of commercial purposes Some of these anticipated functions are not new to air transport undertakings—certain feeder-line services within the British Isles, for example, may well be operated by helicopters—but the fundamental ability of the helicopter to take-off and land in confined spaces gives it an additional scope in circumstances which preclude the use of fixed-wing aircraft Many applications in the latter category occur to the mind Only two of the possibilities are the establishment of passenger and freight services to remote parts of the country, and express mail services between the principal cities Some districts in Scotland, for example, are at present virtually isolated during part of the year, or can only be reached after a tedious and difficult journey Little or no preparation would be necessary in arranging landing sites for helicopters Operational research is already being conducted by the Helicopter Unit recently formed by B E A

Two important facts arise from even the most superficial examination of the field of operations Firstly, that the successful commercial use of helicopters will depend largely upon the reliability of the machines themselves, and secondly, that a high standard of airworthiness will have to be maintained Operation by relatively unskilled persons—both private owners and commercial employees—and from sites having only meagre maintenance and repair facilities, will demand an exceptional freedom from mechanical troubles Rotary-wing aircraft cannot be permitted to land freely in the densely populated centres of the principal cities until the safety of such operation has been proved to the regulatory authorities

The American C A A has already established airworthiness requirements for rotary-wing aircraft Such requirements, however, have not yet been published in this country For fixed-wing aircraft, the Air Registration Board's Civil Airworthiness Requirements and the Design Requirements for Military Aeroplanes (A P 970) of the Air Ministry, cover the field It is relevant, before considering the problems of helicopter airworthiness, to study briefly certain aspects of the fixed-wing requirements

The principal fixed-wing loads are specified by a flight envelope, which defines the permissible limiting combination of acceleration and forward speed The Load-Factor sub-committee of the A R C recommended (1930) that, in general, any manoeuvre which does not leave a factor of safety of 2 on the aircraft structure, during normal execution by a good pilot, should be prohibited Consideration of acceleration records available at that time led to the specific recommendation of an ultimate load factor of 8 The fatigue effects of manoeuvring loads were thus very slight in a good design, and failure from this cause was likely to occur only once in 10^7 or 10^8 flying hours—a negligible proportion Since 1930 the increase in aeroplane operating speed and altitude, and in wing-loading, has tended to reduce the manoeuvring load factor In addition, more accurate methods of stress calculation permit a lower factor of safety This overall reduction of the maximum normal acceleration factor and the factor of safety, means that the loads imposed by gusts may be critical The total (ultimate) factor is now specified as 5 25 for aircraft in the normal civil category weighing under 8 tons, and higher for military and aerobatic types

The frequency of gusts varies with their intensity, the maximum velocity encountered being about 70 ft /sec —such a case being very rare If the load imposed on an aircraft structure by a gust of 50 ft /sec is of the same order as the ultimate load, then that imposed by one of 15 ft /sec — probably occurring more than a million times during 10,000 hours' flying— may cause fatigue failure during the lift of the aircraft

These facts, together with the present tendency to increase the flying life of transport aircraft, make it important to take fatigue into account The P I C A O has recommended that " the strength and fabrication of the " aeroplane shall be such as to ensure that the probability of disastrous " fatigue failure of the primary structure under repeated loads anticipated " in operation, is extremely remote during the expected life of the aircraft " or parts thereof When a type of construction is used for which experience " is not available to show that compliance with static strength requirements " is sufficient to ensure the strength of the structure under repeated loads, " its strength under such loads shall be substantiated by suitable investiga- " tions " Such tests have, in fact, been performed on at least one aircraft, and may be expected in time to become a usual feature in proving the strength of airframes

The question of rotary-wing aircraft airworthiness may now be studied in the light of aeroplane practice

The fuselage and landing gear do not present any special problems which cannot be dealt with by established methods The fuselage loads imposed by manoeuvres are similar to those on fixed-wing aircraft, although at present few data are available upon which to base a normal acceleration factor The contribution of V-g records in establishing the value of this factor for aeroplanes should not be forgotten As applied to helicopters the factor has no direct relation with forward speed (V), and it may be considered instead as a function of rotor r p m This is more logical, but the form of the function may vary considerably between different machines, as it is dependent upon the method of control and the governing mechanism if one is fitted Although it is desirable to record acceleration on a time base, this is difficult under ordinary operating conditions, and the apparatus required is unduly complex Useful data can, however, be obtained from a standard V-g recorder, which has the merit of separating manoeuvres at zero forward speed Aircraft fitted with the conventional type of flexible, fully-articulated blades are not very sensitive to gusts, and the maximum acceleration induced in the fuselage does not usually exceed that caused by control operation An ultimate factor of 3 5 on the normal 1g fuselage loads should be adequate, though this figure should not be employed without investigation unless the design is similar to that of an existing machine of proved airworthiness

First thoughts may suggest that a light undercarriage is sufficient Normal power-on landings—a category which may be expected to include nearly all landings during the life of a helicopter—should be treated as a possible cause of fatigue failure, the maximum vertical velocity of such an airborne landing being taken as 4 ft /sec Power-off landings present a more severe case The vertical velocity at touch-down may exceed 12 ft / sec during an emergency, or because of specially adverse conditions The maximum load developed in these circumstances depends not only upon the characteristics of the helicopter, but to a very great extent upon the pilot's experience and skill in timing A rational approach to this problem may only be made by the analysis of acceleration records Until adequate data of this nature are available, attention should be directed to ensuring that failure of the undercarriage will not result in injury to the passengers or crew Since the forward speed will usually be low during a heavy landing, this condition is not difficult to fulfil

Unlike the elements so far considered, the rotor has no equivalent in an aeroplane , but it is unquestionably the most important part of a helicopter The present discussion will be confined to the most usual type of rotor now in use, namely the hub-driven rotor with flexible, articulated blades The behaviour of rigidly mounted blades is similar, since they may be replaced as a first approximation by semi-rigid blades having a virtual hinge at some point in the spanwise axis The chief loads are the centrifugal force, and bending moments in both flapping and drag planes Shear forces are small, and torsion is not usually critical, except in the case of rotors controlled by ailerons The centrifugal force is virtually constant, and in most blade designs an ultimate factor of 4 or more may be obtained under this load alone The presence of a bending moment in the flapping plane is due to the fact that the resultant of the aerodynamic load components in this plane does not pass through the centre of inertia forces The centre of inertia may be adjusted by suitable blade design so that it coincides with the aerodynamic centre for a given lift distribution, but in forward flight this latter distribution varies with the position of the blade in azimuth, being concentrated further outboard when the blade is retreating than when it is advancing It follows that whatever the mass distribution, there will be a cyclic variation of bending moment at each point along the blade The magnitude of this moment is reduced by blade bending, since the centrifugal force component normal to the blade is modified, and acts as a relief The inertia forces associated with this bending vibration are comparatively small

Similar loads occur in the drag plane The centrifugal force acts at the centre of gravity of the blade , the distance between this point and the centre of drag is greater than that between the centre of inertia and lift, but since blades are invariably stiffer in the drag plane, the stresses due to bending are of the same order in both directions Other inertia loads may be present because of a periodic variation of angular velocity and flapping angle If oscillations in the drag plane are mechanically damped, or resilient hinges employed, the restraint will contribute to the drag moment

The steady operation of a helicopter in forward flight thus causes, at each point in the load-carrying member, a steady fibre stress and a fluctuating stress It is usual to state the stress in terms of its maximum and minimum values, by quoting their algerbaic mean and the semi-amplitude of their difference , that is, in the form $M \pm R$ The constant part of this stress is the sum of the tensile stress due to centrifugal force (which may be as much as $\frac{1}{4}$ of the ultimate stress at some points) and the mean fibre stress due to bending The frequencies of the principal components of the fluctuating part are once and twice the rotor speed If the rotor speed is 250 r p m then the first harmonic will complete a million cycles during 70 hours' flight The fatigue strength of the blade under this combination of loads must thus be studied, rather than the static strength

The magnitude of the stress at each point depends principally (during unaccelerated flight) upon the forward speed and the rotor speed Additional loads are imposed by accelerated motion, which may be the result of ground running (take-off, landing and taxying), control operation, or gusts An understanding of the strength of materials subjected to fatigue loads is necessary before attempting to judge the adequacy of a particular design

The first experimental studies of fatigue failure were conducted about 80 years ago, and it was soon established—largely owing to the work of Wohler—that, for example, a steel specimen subjected to reversed bending will fail after a certain number of reversals, that depends on the range of the applied stress This information is usually presented in the form of a curve connecting the number of reversals (N) at which failure occurs under different values of the stress range (S) In the case of some steels it is found that stresses below a certain limiting value do not cause failure when repeated indefinitely Fatigue failure of these steels after 20 million reversals is extremely rare, and may be neglected for practical purposes There is no such fatigue limit for the majority of materials

The strength of specimens under repeated load is greatly impaired by the presence of irregularities causing stress concentrations, and by poor surface finish There is not, however, a constant relation between the maximum stress in a specimen caused by, say, a surface notch, and the number of reversals to failure An improvement in the fatigue life is observed beyond that which might be expected to correspond to the maximum elastic stress Improvement is partially due to local work hardening of the material, and depends upon the size and stress distribution of the specimen and upon the structure and elastic properties of the material Attempts to correlate these effects by a notch sensitivity index (defined as the ratio of the proportional change in fatigue strength of a similar plain specimen compared with the notched specimen, to the proportional increase of elastic stress due to the notch) have not been very successful, the definition itself is unsatisfactory in some respects An approach along different lines is more encouraging, but it will be sufficient at this stage merely to note that when an estimate of fatigue strength is made from data relating to polished laboratory specimens, an allowance must be included for the stress concentrations and defects of surface finish which occur in practice Some tests have been performed upon complete structures and structural elements, the fatigue strength of which is commonly found to be less than half that calculated from the nominal stress

So far consideration has been given only to specimens subjected to equal reversed stress As in the case of stress concentrations, no general relationship can be found for specimens subjected to a combination of steady and alternating loads, although many investigators have studied the problem A simple formula can be assumed which leads to conservative results, and is suitable for practical purposes The " modified Goodman Law " is such a relationship It is best expressed symbolically

$$R = R_o (1 - M/U)$$

where $2R_o$ is the safe range of stress at zero mean stress (corresponding, for example, to $\pm R_o$ in reversed bending)

$2R$ is the safe range of stress at mean stress M,

and U is the ultimate strength of the material

It should be emphasised that this is no more than a convenient rule which, when applied in design, gives a reasonable assurance of safety Used in conjunction with the S — N curve appropriate to the material in its actual form (*i e*, containing stress raisers and surface imperfections) it is evidently possible to derive a value for the permissible range at any given mean stress That is, the range of load which will not cause failure in less than the specified number of reversals

It now remains to establish the behaviour of a specimen subjected to a system of loads in which the values of M and R change from time to time Given a list of loads and the number of cycles corresponding to each, it might be asked whether the specimen will fail The question cannot be answered directly, except by appeal to a test in which the loading conditions are exactly reproduced A simple theory has, however, been proposed, which enables problems of this sort to be assessed The " Cumulative Damage " theory is based on the supposition that each cycle of stress does a certain amount of damage to the material, and failure occurs when the sum of all the damage reaches a critical value It is commonly observed that when a fatigue test is preceded by a few cycles of overstress the life of the specimen is prolonged, probably because of work hardening of the material The anomoly—and others—may be removed by modifying the assumptions, but for simplicity these variations will be neglected Expressed quantitatively, the damage contributed by p_n cycles of stress, under which load alone the specimen has a total fatigue life of c_n cycles, is p_n/c_n The cumulative damage under a system of varying loads is then given by the sum of expressions of the type p_n/c_n, each of which represents the effect of a particular load The material fails, according to this hypothesis, when the sum becomes unity It follows that the order of application of the loads does not affect the life of the part Stresses which are less than the limiting stress do not contribute damage, since c_n—the number of cycles to failure—is infinite in such cases The results of a few tests which have been conducted using a varying load have been checked with the cumulative damage theory Sufficient agreement is obtained to justify using the theory tentatively, and it is hoped that an extensive investigation will be made shortly, since such a simple method of correlating different loads is an invaluable tool in analysing fatigue effects

The system of stresses at any particular point in a helicopter blade is very complicated, as has been seen, and must be simplified to make the problem manageable Consider firstly the steady flight condition , the aircraft may be expected to spend some part of its total flying life in the state represented by each point within the flight envelope (plot of limiting forward speed versus rotor r p m) If a helicopter spends 1% of 5,000 flying hours in a particular condition, the number of stress cycles completed will be in excess of 500,000 Most of the stresses involved will thus correspond to points on an S—N diagram in the region of which fatigue strength is low and almost constant Hence load is a more critical factor than the number of reversals Little will be lost, therefore, by dividing the flying life into a few steady flight conditions corresponding to maximum loads Such conditions will usually be limiting points on the flight envelope The cumulative damage due to this system of loads will be not less than that ooccurring in practice

The alternative is to integrate (over the whole envelope) the damage corresponding to each flight condition A similar process is inevitable in the case of the loads, due to accelerated flight, if fatigue damage from this cause is to be considered Information is thus needed giving the frequency of loads of each order due to control operation, landing, ground handling, and gusts

No mention has been made so far of how the required data are to be collected , how accurate are the assumptions , and what is the allowance to be made for departure from the " most probable " figures used in calculations To recapitulate it is necessary in order to assess the suitability of a particular structural element to know —

(i) the stress (in the form $M \pm R$) corresponding to each flight condition and manoeuvre,

(ii) the probable frequency of each occurrence (i) during the life time specified,

and (iii) the life of the element under various ranges of reversed stress

Theoretical analysis of the rotor will indicate the critical points of the steady flight envelope, and together with results from strain gauges, will enables the pattern of stress to be comprehended A conservative assumption such as that already suggested could be employed to reduce consideration of steady flight to a few cases only The stress cycle (at mean zero stress) equivalent to each load may now be found by the modified Goodman relation This is again conservative Great care must be exercised in the selection of S—N data corresponding to reversed stress The effect of notches, etc , makes it very desirable, if not essential, for at least a few control tests to be performed on the actual structural element, under a load distribution similar to that occurring in flight A margin of safety must be allowed because of the scatter associated with all experimental fatigue results Alternatively, an S—N curve may be used giving a number of reversals to failure that is rarely *not* exceeded (not more than 1 in 10^4 times, say) That it is very difficult to draw such a curve may be appreciated from the results of one series of laboratory tests which, although exceptionally carefully controlled, show a variation of life between 0 25 and 40 million cycles in 216 tests at the same load The margin may be provided by applying a factor of safety to either the load or the life The number of reversals in the case now being considered is large, and hence a factor on life is not convenient From the figures just quoted, such a factor might have to be greater than 100, but would vary with the expected life of the aircraft In some designs the rotor will have an indefinite life under steady flight conditions alone in this case a factor on life becomes meaningless The author prefers to apply a factor to the load—that is, to both the steady and alternating parts This factor should cover the error involved in the correlation of different loads by the concept of cumulative damage

A similar process must be followed in calculating the effect of acceleration loads, though in this case the basic information relating to the stresses and frequency of application is not available The stresses due to control operation and gusts may be calculated, although at some length and without much accuracy Data also exist from which the frequency of gust loads may be deduced, but these loads are not usually so severe as control-induced loads, the frequency of which is quite unknown

There is a very real need for information on acceleration stresses in blades , this could be provided by strain gauges in conjunction with a recording device similar to the V-g recorder in use on aeroplanes Such an instrument would enable allowance to be made for all transient stresses, whether due to taxying, starting, manoeuvring, or any other accelerated regime

At present, then, owing to the lack of essential knowledge, it is not possible to ensure absolutely that fatigue failure will not occur during the specified life It is suggested that a fatigue factor of 1 5 be applied to the steady flight loads, it being a requirement that the structure shall not fail in fatigue during the specified life of the aircraft under these fully factored loads alone In addition there should be an ultimate factor of 2 0 on the maximum loads occurring from any cause It is hoped that this will ensure that peak loads do not damage the structure to such an extent that it fails subsequently under the steady flight loads

In the future, when enough information has been collected, it will be possible to apply the cumulative damage theory to all the loads A factor must then be applied to life, because deviation in practice from the " most probable " frequency is essentially a change in the number of occurrences A factor of, say, 4 should be applied to the frequency of each stress, as derived from flight acceleration records The fatigue load factor should then be reduced, since the fatigue effects of all loads will have been included The factor will still have to account for error in calculated loads, for sub-standard strength due to material faults, corrosion and other service damage, for faulty workmanship and the spread of experimental results in constructing the S—N curve The author is of the opinion that this factor will be about 1 25