

THE PRACTICAL IMPLEMENTATION OF FATIGUE
REQUIREMENTS TO MILITARY AIRCRAFT AND HELICOPTERS
IN THE UNITED KINGDOM

By R. D. J. Maxwell
Royal Aircraft Establishment, Farnborough, Hampshire,
United Kingdom

SUMMARY

The paper describes the methods adopted in the United Kingdom to ensure the structural integrity of military aeroplanes and helicopters from the fatigue point of view. It describes the procedure adopted from the writing of the specification to the monitoring of fatigue life in service, and outlines the requirements to be met and the way in which they are satisfied. It also indicates some of the outstanding problems that remain to be solved.

INTRODUCTION

The formal airworthiness requirements for the design of military aircraft and helicopter structures against fatigue are contained in the Ministry of Aviation Supply's publication AvP 970. This document lists a number of mandatory requirements together with advisory leaflets as to how these requirements may be satisfied. Although the mandatory parts, which are written in fairly general terms, are still valid, the advisory leaflets, written mainly in 1958-1959, are now a little out of date and do not always agree with current practice. The object of this paper is to describe the existing process of ensuring an acceptable fatigue performance including both the satisfaction of the mandatory requirements and the subsequent monitoring of that performance in service.

However, before starting the main part of the paper it is worth indicating how the Structures Department of the Royal Aircraft Establishment, which is part of the Ministry of Aviation Supply, is involved in the various phases of an aircraft's development and operational use. Its activities can be summarized as follows:

- (a) Making critical comments on the initial specification from the Ministry of Defence, who is the customer, and the early brochures from the manufacturers. These comments are made through the Project team in the Ministry of Aviation Supply, which is the procurement authority.
- (b) Interpreting the aircraft usage in terms of load spectra by discussions with the Ministry of Defence and the manufacturers.

(c) Agreeing with the manufacturers, as required by AvP 970, on the extent of fatigue testing to be done.

(d) Agreeing with the manufacturers after completion of fatigue testing on the service life to be promulgated to the Ministry of Defence through the Project team.

(e) Acting as technical adviser to the Project team in discussions on fatigue arising in service.

It is clear therefore that Structures Department has a hand in every phase of the aircraft's life.

THE GENERAL PROBLEM

Throughout the sequence of operations, the general fatigue problem will be considered under three basic headings:

- (a) The determination of the loads/stress spectra experienced by various parts of the structure
- (b) The determination of the fatigue performance of various parts of the structure
- (c) The estimation and monitoring of the service life

In general, the procedure will be considered in two phases:

- (a) The design-development phase, that is, up to the aircraft's entrance into service
- (b) The production and service phase

Firstly, fixed-wing aircraft designed on safe-life principles will be considered; secondly, fixed-wing aircraft that are essentially fail-safe; thirdly, helicopters which are invariably designed safe-life; and lastly, fail-safe helicopters.

WRITING THE AIRCRAFT SPECIFICATION

When the fatigue life specification is written, two important aspects need to be covered. Firstly, the role or combination of roles for which the specified life is required must be described in sufficient detail to enable load spectrum estimates to be made. This description is extremely important, as contractual compliance with the fatigue life requirements will be determined by tests under these load spectra. Thus, the requirement should indicate

- (a) The types of role in which the aircraft will be operated (that is, route flying, ground-attack, marine reconnaissance, etc.) and the proportion of time spent in each role

- (b) Flight profiles anticipated (heights, speeds)
- (c) Operating weights and stores to be carried
- (d) Numbers of landings
- (e) Numbers of pressurizations

Secondly, it must be made clear whether the required life is the minimum to be achieved in the stated mixture of roles or whether it is the average life to be achieved. This definition of the life determines whether the life is to be achieved under the most severe spectrum or under an average spectrum, estimated from the stated usage.

DETERMINATION OF LOAD SPECTRA

At the beginning of the design-development phase, the load spectra for the aircraft are estimated for the specified utilisation. The estimates are obtained mainly from data collected from previous aircraft. Toward the end of this phase, the estimates may be modified by loads measured on prototype aircraft. The loads to be considered include those discussed in the following paragraphs.

Gust Loads

The gust loads are estimated from the flight profiles quoted in the aircraft specification by using mainly discrete gust data with rigid-body response giving centre-of-gravity accelerations. For larger aircraft some allowance is made for flexibility. The use of power spectral methods to estimate gust loads in terms of centre-of-gravity accelerations is under consideration. The same discrete gust data are used to estimate tailplane and fin loads. Fin load frequencies are arbitrarily multiplied by 3 to allow for Dutch roll type of response and to allow for some manoeuvre content.

Manoeuvre Loads

Manoeuvre loads again are obtained in the form of centre-of-gravity accelerations. They are compiled mainly from the load spectra collected from fatigue (load) meters (counting accelerometers) on previous aircraft flying similar roles with some allowance where necessary for different design limit values of centre-of-gravity acceleration. If new types of role are envisaged, manoeuvre loads must be estimated by consultation with the operators. Manoeuvre loads are mainly of significance for the wing and fuselage but may also be important for the tailplane. Attempts have been made on some aircraft to calculate the tailplane loads required to initiate the centre-of-gravity accelerations of the manoeuvre spectra. In general, such calculations suggest that peak loads of about twice the magnitude of the tail balancing loads for the manoeuvre under consideration are obtained.

Ground-Air Cycle

Until recently this ground-air cycle, a once per flight cycle, which is mainly of importance for the wings has been considered to range from a lower limit given by the down load generated by a 1.2g acceleration while taxiing under maximum take-off load to an upper limit occurring in the 1g level flight condition. It is now considered that a more realistic allowance for the ground-air cycle is obtained for transport and heavy bomber aircraft if the upper limit of the cycle is taken as the 1g condition plus the positive load occurring once per flight. In addition, it is recognised that the once per flight down load for this class of aircraft is likely to be between 1.3g and 1.4g rather than 1.2g. The ground-air cycle is normally considered to be unimportant for fighter-attack aircraft where negative manoeuvres in flight give greater down loads than those experienced on the ground.

Ground Loads

Estimates of ground loads are, of course, of primary importance for the fatigue-life assessment of the undercarriage, but the loads transmitted to the rest of the structure can also be important for the top surfaces of wings and fuselages of large aircraft. Many modern transports and heavy bombers have undercarriages on or near the fuselage. Consequently, the top surfaces of the large-span, fuel-filled wings are in tension on the ground. The alternating stresses generated by ground loads can therefore cause fatigue problems in the top wing surfaces. Similarly, bending loads in the long fuselages can produce fatigue-prone regions. In general, little data analogous to the gust and manoeuvre data exist. At present, methods of measurement and analysis of such loads on development aircraft are difficult and no operational recorders are available. Although power spectral methods of analysis are giving some indication of the vertical loads likely to be experienced, little has been done to calculate side loads, which may be extremely important for the undercarriage.

Local and Acoustic Loads

Local loads include such loads as those due to flap and airbrake operations. Estimates of sound pressure levels for acoustic loads can be made once the engine type and configuration are known.

CONVERSION OF LOAD SPECTRA TO STRESS SPECTRA

The structure is examined in detail and at all stations considered to contain possible fatigue problems, the local stresses corresponding to the various parts of the load spectrum are calculated. In general, rigid body conditions are used, but some allowance for

dynamic effects is made if it is thought that the stress levels will be significantly affected. In the later stages of the design-development phase, the stress calculations are supplemented by flight measurements on prototype aircraft. The importance of knowing the utilisation pattern in some detail again becomes apparent since the centre-of-gravity accelerations of the load spectrum must be associated with the correct weight and flight conditions to obtain the corresponding stresses. Hence, an estimate must be made of where in the flight the accelerations are most likely to occur.

ASSESSMENT OF FATIGUE PERFORMANCE

The initial assessment of fatigue performance is by a calculation using Miner's hypothesis to evaluate the lives of those components for which the stress spectra have been determined together with S-N curves appropriate to the type of component and material considered. In general, manufacturers use their own S-N data based on tests on previous aircraft with components similar to those proposed for the new model. Where such curves are not available, either the basic material curves are used with some allowance for stress concentrations and other effects or some typical curve such as those in the Royal Aeronautical Society/Engineering Sciences Data Unit Data Sheets. In particular, the Heywood joint curve A, Data Sheet E.05.01 is regarded as a good starting point for calculations on aluminium alloy structures. Parts shown by the initial calculations to have marginally acceptable lives are tested under realistic load sequences of the required stress spectrum.

ASSIGNMENT OF PROVISIONAL SERVICE LIFE

At this stage, the end of the design-development phase, there will be a number of prototype aircraft flying and production will be about to start. In order to provide some safeguard for early flying until the major fatigue test is completed, provisional fatigue lives are assessed on the basis of the calculations and test results available at this time. The life of the aircraft as a whole will be determined by the life of the most critical irreplaceable component. Within that life, other components may need replacing. In all cases lives will be calculated by using the average spectrum for the sortie, or mixture of sorties, required and either the standard S-N curves or the later component tests. These lives will then be the lives one would expect for average components, and must be divided by the following factors:

(1) By 2 to account for inaccuracies in calculation and component tests compared with full-scale (major) tests. This factor is based on a paper presented by Raithby to the I.C.A.F. in 1961 (ref. 1) which showed that lives based on component tests and calculations usually overestimated the lives subsequently achieved on the full-scale test.

(2) By a factor varying from $3\frac{1}{3}$ to 5 depending upon the number of specimens tested. This factor is essentially to allow for scatter. In this context, since the standard S-N curves are usually based on a large number of results a scatter factor of $3\frac{1}{3}$ is used. The greater uncertainty compared with results based on the component tests is usually allowed for by using what are thought to be conservative S-N data.

(3) By a factor of 1.5 to allow for variations in load spectrum from aircraft to aircraft flying the same role, when it is assumed that the calculated life is based on an average spectrum. This factor is not required if the provisional life is to be monitored for individual aircraft by the fatigue meter or some other method of recording individual variations of load spectrum. If it is decided to use the fatigue meter to monitor the provisional life, a formula will be derived as described subsequently, but unless the major fatigue test is likely to be delayed or the particular aircraft are going to fly consistently in a severe role, it is normal to wait for the results of the major fatigue test before developing the fatigue meter formula.

THE MAJOR OR FULL-SCALE TEST

It is now recognised that lives based on calculations or component tests are likely to be inaccurate. This condition exists partly because the loads on the particular components considered are difficult to assess accurately owing to the complex nature of the structure and partly because of the difficulty of predicting which are in fact the critical components. It has therefore become a matter of policy to carry out tests either on the complete structure or on the major components (complete wing, fuselage, fin, etc.). In the latter case, all parts of the structure must be covered.

The test specimen is normally an early production airframe to ensure that detail design and manufacturing standards are comparable with those of service aircraft. The load spectrum is again derived from the utilisation pattern in the specification. However, by this time some flight load measurements should have taken place on prototype aircraft so that more knowledge should be available, for example, on the dynamic response of the aircraft, and should lead to more realistic relationships between local stresses and centre-of-gravity accelerations.

Usually the loads are applied in realistic sequences by using many load levels. For transports and heavy bombers this procedure results in flight-by-flight loading so that ground-air cycles are interspersed with flight loads and in most cases, a ground load spectrum also is applied. In addition, on some of these aircraft, the manoeuvres and gust loads have been applied in a random order between the ground-air cycles. These realistic sequences are intended to ensure that the changing residual stress

patterns around the stress concentrations which are known to affect fatigue life, but which at this time are not taken into account in theoretical assessments, are reasonably accounted for on test. The random load sequence of gusts has another advantage over the more common block programme in that it is easier to use a large number of load levels because there is no fixed pattern for each flight and hence no need to choose intervals of load that result in finite numbers of each level per flight. This procedure enables a better representation of a continuous stress spectrum to be made than can be achieved with the usual block programme. The flight-by-flight representation is not always used on fighter-attack aircraft if the negative flight manoeuvres impose bigger down loads than those on the ground.

The test is normally carried on to the "factored" required life unless prior catastrophic failure occurs. If no such failure has taken place, a review is made and frequently the test is continued for another factored life or to failure to allow for any extension of life in service beyond that anticipated at the design stage.

INTERPRETATION OF MAJOR FATIGUE TEST

The failure or failures that have occurred under the known loading on test have to be related to the load spectra experienced in the various roles in service and safety factors applied to allow for scatter. For each failure, the following procedure is adopted:

(1) The S-N curve used to estimate the life of the failed item in the design-development phase is adjusted by factoring the stress scale until the calculation using the stress spectrum applied on the test gives the test life to failure.

(2) This adjusted S-N curve is then used to calculate the lives to be expected in the various service roles, using Miner's hypothesis and the anticipated spectra. The same curve is used to derive the coefficients of the fatigue meter formulae which are obtained by the method described by Phillips. (See ref. 2.) The use of these formulae is described subsequently.

(3) The lives for each role and the coefficients of the fatigue meter formulae are then divided by the factor to allow for scatter in performance. In general, only one specimen will have been tested so that according to the recommended factors in AvP 970 a value of 5 should be used, but in practice, a factor of $3\frac{1}{3}$ has been used for all lives based on major tests. Although this procedure is difficult to justify theoretically, it was considered reasonable in view of the greater certainty obtained from this type of test. As there has been no regular shortfall in achieved service life that could be attributed to this cause, the practice has been allowed to stand. It should nevertheless be recognised that it is extremely difficult to obtain feedback of service data on which to base a reliable correlation analysis.

(4) The lives for each role are divided by a further factor of 1.5 to allow for variations of load spectrum experienced by individual aircraft flying the same role. This factor is not applied to the fatigue meter coefficients as the meter registers the individual variations.

It should be emphasized that the utilisation pattern originally laid down should represent as nearly as possible the anticipated usage in service because the fatigue test is based on this pattern and although estimates can be made for other patterns, as shown above, the accuracy of prediction is likely to fall when the new patterns deviate markedly from that used on test.

FAIL-SAFE STRUCTURES

The procedure described is aimed primarily at preserving the safety of safe-life type structures which can fail without prior warning. A similar procedure is also necessary for fail-safe structures, which are defined as those in which fatigue cracks or component failures can be found before the strength falls to an unacceptable level. As the whole concept of fail-safe stands or falls by the ability to detect cracks early, the importance of ensuring that all cracks can be found cannot be overstressed. Hence, it is essential to obtain as much information as possible from the full-scale test on the probable location of cracks. Thus, the full-scale test is as important for fail-safe structures as for the safe-life type although the emphasis is different.

The test should first demonstrate that the structure really is fail-safe, that is, that at no time during the service life is there likely to be an undetectable major failure. The main dangers are design errors leading to an early unexpected catastrophic failure or the accumulation of many small failures late in the life which are insignificant and difficult to detect individually but which may suddenly join to give a catastrophic failure. The long riveted joints of pressure cabins are particularly vulnerable to this latter type of failure as the skin experiences similar stress cycles at all rivet locations. Hence, small cracks, which will almost certainly escape detection, are likely to form at about the same time along the rivet line, and these may suddenly join into one long, possibly catastrophic crack.

The second purpose of the test is to show which are the likely areas of cracking, when the cracks are likely to occur and how fast they will propagate. This information will enable inspections to be started early enough and to take place frequently enough to ensure safety. In addition, the actual inspection techniques can be developed on the complete built-up structure.

In order to show that no catastrophic failures will occur during the required life, a fail-safe structure is required to be tested to the same factors on life as a safe-life

structure. In order to demonstrate that cracks are fail-safe, a crack must be allowed to propagate on test for three inspection periods after it has reached the shortest length that can be found with certainty under the inspection method to be used. At the end of that time it must sustain 80 percent of the ultimate load.

The demonstration of the residual strength characteristics poses a practical problem. The 80 percent ultimate load cannot be applied at the end of the crack propagation phase if the test has not reached the factored required life because if it does not survive the application of the load, the specimen is lost or severely damaged and if it does survive, the rest of the test will be invalidated because of the unrepresentative residual stress pattern generated by this exceptionally high load. The usual technique is to run the crack for three inspection periods or until the crack is considered long enough just to sustain the test load. (In this case a shorter inspection period will be imposed in service.) The crack is then repaired with a patch and the test continued. At the conclusion of the test the patches are removed one at a time and the 80 percent ultimate load applied. This is clearly not entirely satisfactory but no completely satisfactory solution has been found. In some cases it may be possible to simulate the relevant cracks on the static test specimen if this is still available and apply the test load to that, but care must be exercised to ensure a crack tip that is typical of fatigue.

These requirements ensure safety but it is also necessary to ensure a reasonably economic aircraft. It is therefore a requirement that the first crack shall not appear on the weakest aircraft from a fatigue point of view before half the specified life has been achieved and that the amount of repair work shall not become uneconomic on the weakest aircraft before the whole specified life is achieved.

Hence it can be seen that the test requirements are similar for both fail-safe and safe-life aircraft. Therefore, although the designer is encouraged to design fail-safe (if he believes his design to be fail-safe he is at liberty to use lower factors in the design to allow for scatter than he would for safe-life design), the structure is judged on its performance in the test, such failures that occur being judged on their merits. Failures which can be considered fail-safe will require inspection in service starting at the factored life, followed by repair or replacement only if they occur, whereas safe-life failures require either modification of the failed item or retirement of the whole structure at the factored life.

MONITORING IN SERVICE

The object of monitoring in service is to relate the load spectra experienced by individual aircraft to the failures that occur on the fatigue test. In order to assess the service load spectra, each aircraft is equipped with a fatigue (load) meter, which is a

counting accelerometer recording the number of times each of eight levels of centre-of-gravity acceleration is exceeded. The actual levels recorded depend upon the type of aircraft, there being a number of standard instruments, but usually there are five levels above 1g and three below. These instruments are read after every flight and the counts recorded together with information on the type of sortie, take-off and landing weight, stores carried, number of pressurizations and any other details considered relevant to the consumption of fatigue life.

There are then three main methods of using this information: the fatigue meter formula, role lives, and total number of occurrences of a particular event, each of which tells the operator to initiate some action. Each method relates a failure under the known test loading to the loads experienced in service with the appropriate factors. If the failure on test is safe-life, reaching the factored life means either that the item must be replaced or that the complete structure must be retired. If the failure is fail-safe, reaching the factored life means that inspection must start. These inspections continue at a frequency determined by the same methods; that is, the individual load spectra are related to the test loadings during the crack-propagation phase so that inspection periods may fluctuate in time depending upon usage. In practice, inspections are called for either at fixed time intervals to coincide with normal scheduled maintenance or when the monitoring system indicates an inspection to be due; it is usually possible to ensure that most inspections occur at scheduled maintenance periods.

Of the three methods of assessing the fatigue life, the fatigue meter formula is considered to be the most accurate and is used when the stress levels at the monitored stations can be related to centre-of-gravity accelerations. This usage usually covers wing stations and fuselage stations affected by longitudinal bending. The operator is supplied with a formula consisting of coefficients by which to multiply the counts recorded on each flight at each level of g, together with overall factors depending upon type of sortie, take-off and landing weight, stores carried, etc. He thus calculates flight by flight a steadily increasing number which is a measure of fatigue damage. When the number reaches a certain value, he initiates the appropriate action, either retirement or start of inspection. In the early days of fatigue meter formulae, one simple set of coefficients was used to monitor the one safe-life failure that determined the ultimate life of the structure. Today, with fail-safe structures it may be necessary to monitor a series of possible failures, inspections for which will start at different times. In addition, with the large variety and weight of stores that can be carried, it has become necessary to allow for relatively large variations in the relationship between stress at the station to be monitored and the centre-of-gravity acceleration recorded. Hence it is sometimes necessary to have a series of formulae and correction factors for one aircraft.

The second method of monitoring, role lives, is used to cover periods of flying in which the fatigue meter is unserviceable or to monitor parts for which the fatigue meter counts have no relevance but for which it is known that the load spectrum varies with the type of sortie, or role flown. In this method, the load spectra are first estimated for each type of flight (in the case of flying with an unserviceable meter these are obtained by analysis of other aircraft records on similar sorties). The factored lives are then calculated by using Miner's hypothesis and the adjusted S-N curve derived from the test. It should be noted that if average load spectra are used, the factor of 1.5 for variation within the same sortie must be included. Each hour's flying is then divided by the factored life in the role to give the fraction of damage done. When these fractions add up to 1, the appropriate action is taken. When used to cover periods of meter unserviceability, the operator is given a coefficient based on this fraction by which to multiply the number of hours flown in each role and this number can be added to the number obtained from the fatigue meter formulae.

The third method of monitoring is the simplest and can be used when the fatigue damage in a part is due entirely to one operation, say pressurization, when the life to "action" is given in terms of the numbers of occurrences of that operation. Again the time to action by the operator is based on the number of such cycles to failure in the fatigue test with the appropriate factor. In the case of pressurization, if the test is carried out by using maximum pressure differentials for every cycle and all pressurizations recorded in service are assumed to be to maximum differential, the factor used is $3\frac{1}{3}$.

One byproduct of the recording of fatigue meter readings after every flight together with the type of sortie flown is that the load spectra are analysed on a sortie basis and used in estimating load spectra for future aircraft.

In general, it is felt that although the fatigue meter has provided and is still providing an extremely valuable method of monitoring fatigue life in service, more elaborate methods are required to cope with the changes in the stress and centre-of-gravity acceleration relationship that now occur on most aircraft. Moreover, some monitoring system must be developed for areas such as the tailplane, fin and undercarriage for which methods of monitoring are still in the exploratory stage and for which there is little or no operational data on load spectra.

THE GENERAL PROBLEM OF THE HELICOPTER

In general, the approach to the fatigue problem in the helicopter is based on the same concepts as those used for fixed-wing aircraft; that is, the load spectrum and fatigue performance for each component must be determined and its life estimated and monitored in service. In the helicopter, however, most of the critical items are

contained in the rotating parts and their controls and these parts are subjected to fluctuating loads even under steady flight conditions. Therefore, large numbers of cycles are accumulated in a short time, and there is a consequent shift of emphasis to the fatigue behaviour at the low stress end of the S-N curve. This shift of emphasis results in one of the main differences between fixed-wing aircraft and helicopter requirements; all the factors are on stress instead of on life as factors on life become meaningless when the S-N curve is nearly horizontal. The fact that stress cycles are generated even during steady flight has its impact on the estimation of load spectra. It is clear that in order to have any reasonable life at all, stress cycles in steady-flight conditions must be below the fatigue limit. Therefore, the life is determined principally by occasional excursions of the stress-cycle magnitude above the fatigue limit which are usually found to occur during a few transitory manoeuvres and short periods in a few flight conditions such as at high speed. Determination of load spectra becomes a process of defining these manoeuvres and flight conditions, estimating the frequency with which they will occur, and estimating the magnitudes and numbers of cycles occurring in each of the manoeuvres or flight conditions specified.

THE DESIGN-DEVELOPMENT PHASE FOR THE HELICOPTER

Essentially, the same information needs to be written into the customer's specification for the helicopter as for the fixed-wing aircraft, that is, life required, types of sortie to be flown, operating weights, and stores to be carried. However, the estimation of load spectra from this requirement is in terms of frequencies of occurrence of the various critical manoeuvres and flight conditions. Owing to the lack of measured operational data, these values have to be estimated from a consideration of how the helicopter is going to be used. However, with the present state of knowledge it is virtually impossible to calculate the stresses arising in the many components associated with the rotating parts and their controls during these critical manoeuvres and flight conditions. Consequently, at the design stage, the stresses calculated for steady cruise condition are multiplied by 1.5 and maintained below the fatigue limit of the factored S-N curve. Past experience has shown that this method provides a reasonable design starting point. The S-N curve used is either a relevant one from tests on similar components from a previous helicopter or a material curve with allowance for stress concentrations, etc. The factor at this stage is 2 on stress.

During development, the loads and stress spectra are steadily acquired by progressive flight measurement on an extensively strain gaged prototype helicopter. Simple manoeuvres are flown, and stresses are measured, related to the appropriate S-N data, and assessed for safety. The helicopter is then cleared for the next more complex manoeuvre. At the same time S-N data are built up by constant-amplitude tests on the more critical items.

HELICOPTER FATIGUE SUBSTANTIATION

The final life substantiation is based on flight measurements of stress, S-N curves obtained either by constant-amplitude tests or programme-load tests factored to allow for scatter, and calculations using Miner's hypothesis.

In order to obtain the stress spectrum for each component, each of the manoeuvres or flight conditions considered likely to produce fatigue damaging cycles is flown at least three times. In those conditions where the three flights give widely different results, more measurements are made. In the first analysis only the maximum stress cycle is associated with each manoeuvre or flight condition and it is conservatively assumed that this cycle occurs at the typical frequency of the stress cycle in that component for as long as the manoeuvre exists. For those components and flight conditions where the subsequent fatigue analysis shows this analysis to give unacceptably low lives, a more elaborate analysis takes place which provides a spectrum of stress amplitudes to be associated with that flight condition. The total stress spectrum can then be obtained for each component by using the frequencies of occurrence of each manoeuvre or times spent in each flight condition estimated from the specification together with the measured stress amplitudes for these manoeuvres and conditions. To allow for variations in stress from helicopter to helicopter when flying the same manoeuvres, the measured stresses are usually multiplied by a factor of 1.2.

The S-N curve for each component is obtained in most cases by testing at least six specimens under constant-amplitude loading; normally three specimens are tested at each of two stress amplitudes. A curve of predetermined shape based on past experience is then drawn through the mean values of life obtained in each of the two groups and this curve is factored on stress values to allow for scatter. When six or more specimens have been tested, a factor of 1.6 is used for light-alloy components, and 1.4 steel and titanium. (The figure for titanium is provisional, being based on limited data.) Where less specimens have been tested, higher factors are used. It is considered that gear boxes show less scatter than other components; therefore, a factor of 1.4 is used if one gear box is tested and 1.3 if four or more specimens are tested. These factors are appreciably bigger than those quoted in AvP 970, but are based on the latest information on scatter and current practice.

The fatigue life is then determined by using Miner's hypothesis except that a value for $\sum \frac{n}{N}$ of 0.75 is used. When the maximum stress amplitude in the whole stress spectrum is below the fatigue limit of the factored S-N curve for that component, the item is considered to have a virtually infinite life. The fatigue limit for light-alloy specimens is taken as that stress amplitude giving a life of 10^9 cycles and for low and medium strength steels, that giving 5×10^6 cycles. Where testing of light-alloy

components has only been taken to 5×10^6 cycles, a factor of 1.35 on stress is used to estimate the fatigue limit.

For components experiencing a complex load history, it is considered advisable to test under a mixed load level to simulate more nearly the actual conditions, although the loads will be increased to allow for scatter and to obtain failures in a reasonable time. (This procedure is in contrast to fixed-wing practice where tests are conducted under real loads for factored times.) The results are used to locate the mean S-N curve for the component in the same way as for fixed-wing aircraft; that is, a predetermined shape of S-N curve is factored in the stress direction until the cumulative damage calculation gives the mean life achieved on test under the known loads.

MONITORING HELICOPTER LIFE

At present there are no monitoring instruments for the helicopter analogous to the fatigue meter for the fixed-wing aircraft. Therefore, all components are assigned safe lives in flying hours. Although the helicopter is used in many roles, there has been no attempt as yet to define different lives for each role or record times spent in each role. Consequently, lives have been assessed in whichever role is considered to be most severe for the component under consideration and those lives considered to be the retirement lives irrespective of the subsequent usage.

FAIL-SAFE FOR THE HELICOPTER

It is clear from the previous two sections that in many ways there are greater difficulties in estimating safe lives for helicopters than for fixed-wing aircraft. The lives are very dependent upon a few transitory loads occurring during certain flight conditions and manoeuvres. The flight conditions themselves are not easy to define accurately and the magnitudes of the loads within those conditions are likely to vary considerably depending upon pilot technique and state of maintenance of the helicopter. Moreover, helicopters of the same type are used for a wide variety of jobs; hence, variations in life of similar components are liable to be very large. In addition, minor damage such as a small score can result in a drastic reduction in life as the large number of cycles of stress otherwise below the fatigue limit are thus raised to a level where they add to the damage. In the circumstances, designs to fail-safe principles are highly desirable from a safety point of view.

It is often thought that this concept with its implication of redundancy can only be obtained at the cost of extra weight. It has been found in fixed-wing aircraft that this is not necessarily the case and, in fact, once the principles of design detail have been

mastered, there may actually be a saving of weight in those areas of the structure designed by fatigue because lower factors to allow for scatter can be used in the design of fail-safe parts than could be used if those parts were safe-life. This condition occurs because it becomes no longer necessary to ensure that fatigue initiation probability approaches zero, the only criterion on frequency of fatigue failure being the economic ones of maintenance and repair costs. The fact that so many of the helicopter rotating parts are fatigue designed and the variations of loading from aircraft to aircraft are so great and yet there are so few fatigue failures in service suggests that there may be appreciable overdesign and therefore significant weight saving to be gained by fail-safe design as well as the added safety.

At present, there are no requirements for fail-safe for helicopters in AvP 970, but there should be no basic problem in writing such requirements in general terms. Indeed, the approach would be identical to that used for fixed-wing aircraft; namely, that any failure shall be found before the residual strength falls below an acceptable value. However, the real problem, once the principles of fail-safe as defined by the requirements are fully understood, is one of detail design and it is here that the main attack must be made if the advantages of fail-safe design are to be realised. In addition, since the early detection of failures or cracks is vital to fail-safe, it would be worth putting more effort into the development of inspection techniques. This effort may involve special systems for particular parts, such as the blade inspection method developed by Sikorsky in which the blade is inflated and cracks detected by loss of pressure. However, it must be remembered that helicopters frequently operate in relatively primitive conditions so that simple techniques are required.

FUTURE WORK

The procedure described in this paper for coping with the problems of fatigue in aircraft structures which has evolved over the years has maintained an acceptable standard of safety. Nevertheless, every step in that process contains problems that could lead to inaccuracies. As the customer demands longer lives for his expensive aircraft, the need for better life estimation is of paramount importance, both for the safety of safe-life aircraft and the economy of fail-safe aircraft.

The areas in which effort is still needed can be considered in two main groups: those associated with defining the load-stress spectra and those concerned with the determination of fatigue performance. If the load-spectrum problems are considered first, wing loads and fuselage bending loads are reasonably served by the fatigue meter; this meter monitors loads on individual aircraft and provides operational data. However, with the wide variation in the stress and centre-of-gravity acceleration relationship

possible in modern aircraft because of the high rates of fuel usage, and the large range of stores carried, some more direct method of obtaining stress spectra is required. If such methods should be developed for operational use, they would be invaluable in monitoring fatigue life consumption of fins, tailplanes, undercarriages, and possibly even helicopter components, although in the latter case there is an additional practical problem of recording outputs through rotating machinery. In the event of monitoring by direct stress measurement being developed, it may be found that the process of feeding back to the design stage will be more difficult than that for current monitoring methods using counting accelerometers, bearing in mind that for both systems allowance must be made for the response characteristics of the aircraft on which the measurements were made before these measurements can be applied to the new aircraft. This procedure is already used to a large extent for response to turbulence, and the power spectral approach used in this connection is being applied to estimating undercarriage loads. However, more work needs to be done in relating the theoretical work in this field to measurements in flight and during ground operations.

The problems associated with fatigue performance will now be considered. The outstanding need is for a new cumulative damage hypothesis that takes sequence effects and fretting into account. With the greater understanding of the effects of residual stresses around stress concentrations, it is hoped that methods of accounting reliably for the former will not be too long delayed. In view of the increasing tendency to design fail-safe, there is a need for more work on methods of predicting crack propagation rates in complex structures under variable loading and the residual strengths of the cracked structures.

It is unlikely that even improved methods of estimating initiation time, crack propagation rates, and residual strengths will enable us to dispense with the major fatigue test. However, such improvements may help in the simplification and interpretation of this test. There are a number of questions in this connection that still require further attention. Firstly, to what extent can the time-consuming low-level stresses be omitted? Secondly, what should be the magnitude of the biggest load applied in test? What precisely is the effect of a load equal to or greater than proof load on the subsequent behaviour and can this effect be counteracted in any way? This consideration is important in solving the problem of proving the residual strength of a cracked structure.

To summarize, it is considered that work will be required in the following areas:

- (1) Theoretical work on dynamic response giving load and stress distributions
- (2) Development of flight measurement and analysis techniques to check and modify the theoretical assessments

(3) The development of operational monitoring devices measuring stress directly. These devices may be expensive and consequently limited to use on a few aircraft.

(4) The development of monitoring devices that can be used on every aircraft to measure parameters that can be related to the stresses measured on the more elaborate instruments. It is considered essential on military aircraft that some monitoring device is used on every aircraft as the variations in load spectra on aircraft in the same role can be very large.

(5) Development of new cumulative damage theories to account for sequence effects and fretting

(6) Development of methods of predicting crack-propagation rates in complex structures under variable loading

(7) Development of methods of predicting residual strengths of cracked structures

(8) Assessment of what stress levels should be included in fatigue tests under realistic loads

(9) The development of aircraft capable of sustained supersonic flight means that more work will be needed in the fields of estimating, measuring, and monitoring stresses due to thermal effects, and interpreting their influence on the fatigue problem.

REFERENCES

1. Raithby, K. D.: A Comparison of Predicted and Achieved Lives of Aircraft Structures. Tech. Note No. Struct. 301, Brit. R.A.E., 1961. (Paper presented at I.C.A.F. Symposium on Fatigue (Paris), May 1961.)
2. Phillips, J.: Formulae for Use With the Fatigue Load Meter in the Assessment of Wing Fatigue Life. Tech. Note No. Struct. 279, Brit. R.A.E., 1960.

