THE TESTING OF SUPERSONIC TRANSPORT STRUCTURES IN FATIGUE

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ABSTRACT

An examination is made of what has to be represented in a fatigue test on a supersonic transport constructed in aluminium alloy. The simulation of the effects of heat is shown to present the greatest challenge and requires more research. The preliminary results of such research indicate that it should be possible to complete the test in about five years.

INTRODUCTION

It is possible to complete the fatigue tests on a subsonic transport in an acceptable time; it is sufficient to represent only the stress variations caused by external loads and the pressure in the cabin, and a flight can be simulated in five minutes [1]. Admittedly, time-dependent phenomena, such as corrosion, are not properly represented, and due allowance has to be made in the interpretation of the test results. In the case of supersonic transports, however, there are three additional factors that have to be considered [2], each associated with temperature and time—namely, the strength of the material after soaking at temperature, the effect of creep, and the changes in stress caused by the variation of temperature. The simulation of the actual temperature and time as in service makes the total time of testing prohibitive; to realize a life of 20,000 flights, it may be necessary to simulate 40,000 to 60,000 flights each averaging one hour and 20 minutes. This requires six to nine years in testing time alone, plus another two to three years for inspection, repair and maintenance of the test specimen and test equipment. A test that lasts eight to 12 years cannot be contemplated with equanimity; unless information is available from the test well before aircraft in service reach a comparable life, there may be severe economic penalties from repair and maintenance, and the probability that safety may not be preserved is too high for comfort. So, clearly, means to accelerate the test are vital.

The objective of this paper is to examine how the test might be accelerated in the particular case of a transport constructed in aluminium alloy. The loading throughout the flight is first examined, and it appears probable that the heating part of the test can be separated from the mechanical loading that represents ground loads, turbulence in flight, and pressure in the cabin. Investigation of the mechanical loading concludes that for the aircraft as a whole it may be sufficient to represent the loads in transition from ground to air (including allowance for a bump while taxying) and in turbulence in flight, as well as those caused by pressure in the cabin. It is next shown that the three effects of heating must be simulated; that the thermal stress cycle causes significant fatigue damage; that soaking at elevated temperature (which degrades the strength of the material) and creep would both appear to be represented by a similar exponential law linking temperature and time so that the test might possibly be shortened by increasing the temperature by some 10°C to 20°C. The interaction of fatigue and creep is also shown to be important. Mechanical loads and thermal effects are then considered together to indicate to what extent the test might be speeded up, and an estimate made of the time required for a practical test, including allowance for inspections, maintenance, and repair.

THE ENVIRONMENT IN A TYPICAL FLIGHT

The behaviour of the structure in fatigue depends on the cumulative effects of stress variation throughout the life of the aircraft with due allowance for the effects of creep and overaging in the material. In practice, there will be short flights and long flights and some flights when supersonic speeds are not reached. Although eventually it may be found a desirable refinement to represent on test the different types of flight, it is sufficient at this stage to consider a typical flight with a view to the discovery of those simplifications that will make the test practicable.

A typical flight is shown in Fig. 1 for an aircraft designed to cruise at Mach 2.2. The aircraft taxies out, takes off, and climbs at subsonic speed to about 36,000 feet, while the temperature of the skin varies from 15°C to −5°C. At the same time, an acceleration to Mach 2.2 in 15 minutes sees the skin temperature rise from −5°C to 120°C. It stays constant at 120°C throughout the cruise and then drops from 120°C to −20°C when the

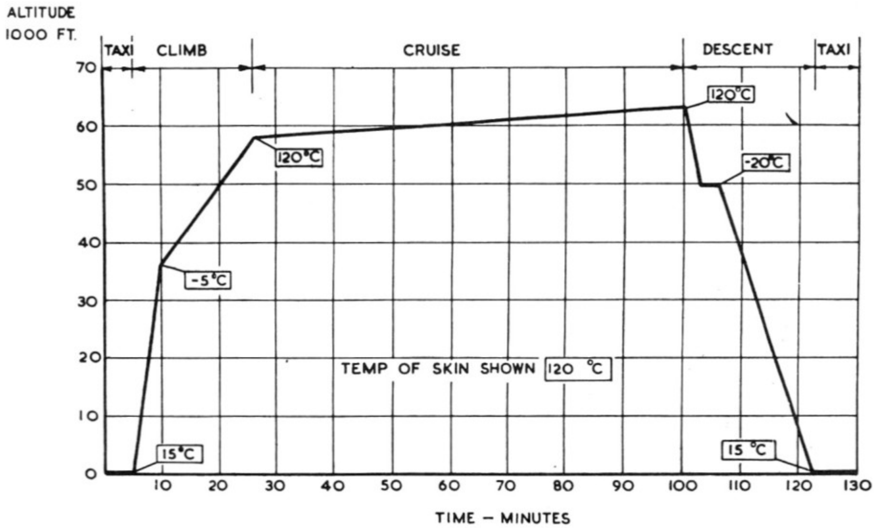

Figure 1. A typical flight.

aircraft decelerates for seven minutes during the descent. The descent continues at subsonic speeds to ground level where the skin temperature again reaches 15°C; finally, the aircraft taxies back to base. Clearly the only significant difference from a subsonic aircraft is the one of temperature at altitude. Therefore, it might be sufficient to represent the taxying and subsonic parts of the flight at essentially room temperature, interspersed with the simulation of kinetic heating effects at altitude.

A further simplification comes from consideration of the loads on the structure. The loads on the ground, takeoff, and landing occur at room temperature. Since turbulence decreases with altitude, it is found that nearly all the gust loads occur at roughly room temperature and that few gusts occur when kinetic heating is important. As regards pressure in the cabin, this too is applied essentially at room temperature. It thus appears sufficient to:

1. Apply at room temperature loads on the ground, loads due to turbulence, and pressure in the cabin.
2. Apply the effects of heating with the cabin already under pressure and with the loads on the aircraft appropriate to steady $1g$ level flight conditions.

There remain two caveats to this happy deduction. The first has to do with the effect of fluctuating load on the creep that occurs with temperature. Load fluctuations are due to gusts, and gusts are met in clusters [3] so that if representation is needed, it might be adequate to represent a rough flight once every 100 flights, for example. The second point has to do with acoustic fatigue, both from the noise from the jet engines and from the boundary layer; the noise from the jets affects defined areas at the rear of the aircraft and it is assumed that these will have been tested separately in the appropriate noise conditions. The effects of boundary-layer noise, too, have to be investigated separately from the main test.

REPRESENTATION OF LOADS

In tests on subsonic aircraft it has become established practice to represent the transition from ground to air (often referred to as *ground-to-air cycle*), the turbulence met in flight, and the pressure in the cabin with the addition of any other loads (for example, loads on the fin and rudder) if the particular layout of the structure seems to warrant it. In the case of the supersonic transport, however, something more complicated may have to be done. In a slender Delta, for example, stress in the wing in flight may well be of similar order spanwise and chordwise, and under ground loads, significant bending stresses are produced, particularly in the fuselage. It seems at first sight, therefore, that loads during taxying, takeoff, and landing may need to be applied in some detail. This question is resolved by examination of the order of stress variation and the number of cycles likely to arise at typical sections of the wing and fuselage.

In practice, loads of different intensities occur in a random order and less frequently with increasing load. For ease of presentation and to gain some insight into the relative importance of the loads in the different phases of flight, it is convenient to present the stress variations as an equivalent number of cycles at some selected constant amplitude. The equivalent number is deduced using the Palmgren-Miner Cumulative Damage Hypothesis [4]. It is appreciated that this hypothesis does not make sufficient allowance for the large number of cycles of small stress amplitude [5], nor does it take into account the sequence of different loads and the associated effect of the fretting actions [6] that are common in a structure. Nevertheless, the hypothesis permits a first sifting that indicates the more important loads in each phase of the flight and also gives an indication of the relative importance of each phase.

Figure 2 gives the stress history at a typical station on the underside of the wing away from the undercarriage. It is seen that the stress cycles on the ground are not comparable with those caused by turbulence in the climb and descent and, moreover, are essentially compressive. It would appear,

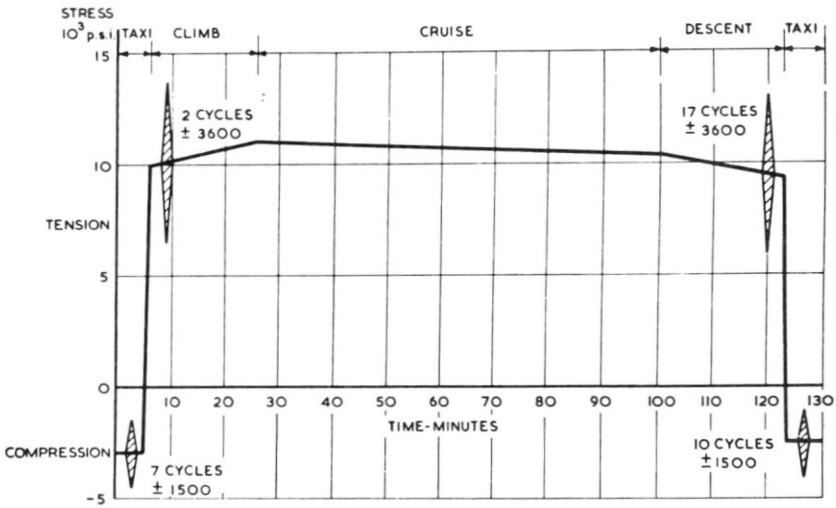

Figure 2. Stress history—wing bottom skin (no kinetic heating).

therefore, that the cycles during taxying might be omitted. In relation to the stress history portrayed in Fig. 2, such a decision is justifiable, but for other parts of the wing the decision is not yet obvious. For instance, on the top of the wing, creep in compression occurs when the wing is hot and this may accentuate the damaging effect of the subsequent cycles in tension during taxying. (This matter is considered in more detail under thermal effects.) In the region of the undercarriage itself, the loads from the undercarriage diffuse into the main wing structure, so that, locally, the stress variations during taxying may be important. Consideration of this region is likely to determine how complicated the representation of undercarriage loads has to be.

Figure 3 gives a similar history for a position on top of the fuselage. The situation here is that there is always a tensile stress and the fluctuations during taxying are small. After takeoff, the pressure increases in the cabin and increases the mean stress, but since turbulence is met mostly at low altitude, there would appear to be considerable latitude about the precise moment when pressure in the cabin needs to be applied. The nominal hoop stress due to pressure is perhaps 11,000 lb/in.^{2} over a considerable extent of the cabin. These stresses are typical of the general run of the structure; around openings, such as doors and windows, they are greater to a degree that depends on the success of the design in the reduction of the stress

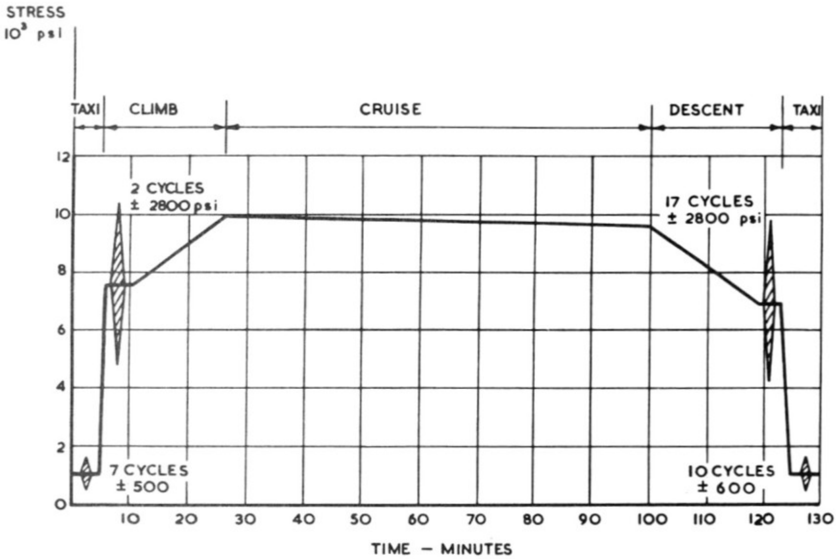

Figure 3. Longitudinal stress history—top of fuselage skin (no kinetic heating).

concentration caused by the structural discontinuity of the opening. It is suggested that the assessment of the loads that need to be included in the test should be on the assumption that around openings the stresses might well be doubled. For the fuselage, then, it seems necessary to represent the ground-to-air loads, the loads due to gusts in flight, and the loads due to pressures.

Thus, for the aircraft as a whole, it is concluded that it may be sufficient to represent the loads in transit from ground to air, in turbulence in flight, and due to pressure in the cabin. The omission of taxying loads seems justified for most of the structure, although it would be prudent to increase the ground-to-air cycle to make some allowance for the bumps during taxying, but in the region of the undercarriage the representation may have to be more complex.

THERMAL EFFECTS

In this section is examined the possibility that the thermal environment may affect the endurance of the structure. It is convenient to consider the thermal effects as those arising from transient heating and cooling—i.e., thermal stresses—and those due to the accumulated time at temperature and stress.

It is proposed to deal with these in turn, establish what needs to be represented on present evidence, and underline where that evidence is weak and further work is required.

THE THERMAL STRESS CYCLE

Thermal stress cycle, as used here, denotes the variation in stress that occurs in the structure under the transient conditions when the aircraft accelerates or decelerates to give changes in Mach number. The amplitude of the stress cycle depends on the range of temperature, the rate of change of temperature, and the structural form.

Typical situations in the structure where significant thermal stress cycles may be produced are:

1. In the fuselage structure in the way of the wing.
2. In the webs and booms of spars and ribs in the wing.
3. Where structural elements of different heat capacities are joined together—e.g., differing thicknesses of skin, robust members to take large concentrated loads from the undercarriage, etc.
4. Areas contiguous with large heat sinks such as are provided by the fuel in the fuel tanks.

On a section about 9 in. deep (Fig. 4), Harpur and Sellars [2] calculate the thermal stress cycle in the spar boom to be $\pm 6{,}000$ psi and note the considerable benefit to be gained from design features that reduce thermal stresses. The temperature distribution at the end of the climb is shown in Fig. 5, while Fig. 6 shows a history of stress in the spar boom throughout a flight with the thermal stress cycle added to the steady-flight loads. Note that these values do not necessarily apply to the flight of Fig. 1.

The fluctuating stress of order $\pm 6{,}000$ psi, which might occur some 17,000 times during the life of the aeroplane, is clearly important from the fatigue standpoint. The more so since the associated mean stress may be of order 10,000 psi. A similar calculation on a section near the root of the wing indicates a thermal stress cycle of $\pm 20{,}000$ psi unless appropriate design measures are taken. Such measures include the use of corrugated webs, dished webs, and braced webs. Unfortunately, elimination of the thermal stress cycle by design means is impractical, since the structure has to perform a number of functions (such as providing fuel tanks), and the final design is therefore one of compromise. The success of the compromise varies from aircraft to aircraft, but preliminary investigations suggest that somewhere in the structure there may exist a thermal stress cycle of some $\pm 8{,}000$ psi and that this cycle may be associated with appreciable mean

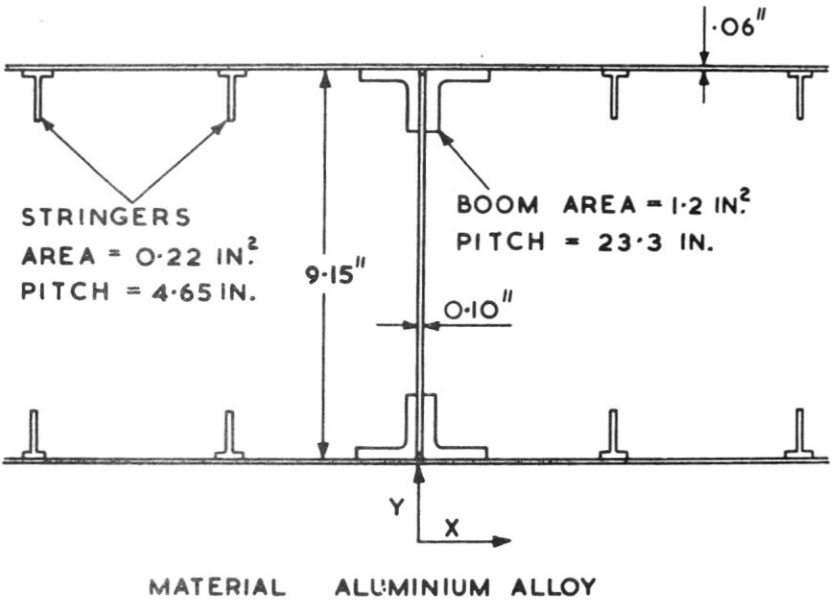

Figure 4. Section considered.

stress. These are calculated values and there remain doubts on their accuracy in complex regions such as the wing-fuselage junction.

OVERAGING OF THE MATERIAL

Aluminium alloy can be heat-treated to give optimum strength properties as judged by a simple tensile test. If heat treatment—that is, soaking at elevated temperature—is continued beyond this optimum stage, the tensile strength deteriorates. This deterioration is called overaging. The extent of the deterioration is shown in Fig. 7, where it is seen that if the specimen is soaked for 20,000 hours at 120°C, the ultimate strength at room temperature falls by some 1 per cent compared with the original strength of the material.

Fisher [8] suggests that the strength for various times and temperature conditions can be plotted against the parameter:

$$\phi = \log t - \frac{A}{T}$$

where A = a constant for the particular material
t = time in hours
T = absolute temperature, °K

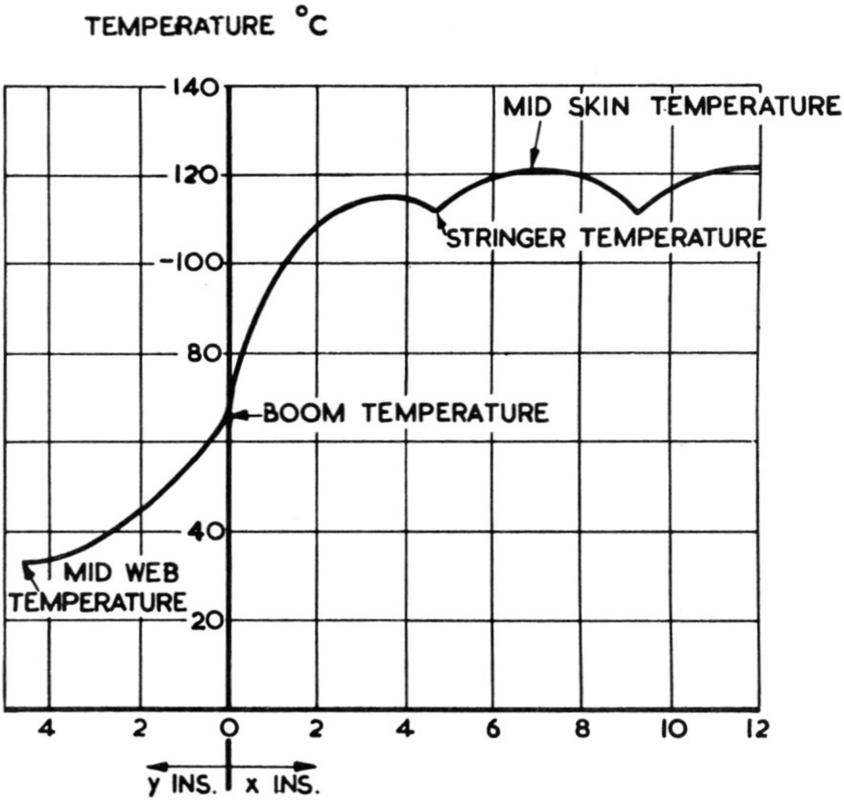

Figure 5. Temperature distribution at end of climb.

Figure 8 shows test results.

The fatigue strength of structural elements at room temperature in the higher-strength aluminium alloys appears to be largely independent of the ultimate strength of the material. The possibility therefore exists that the effect of overaging might also be small. Some confirmation of this possibility is provided by fatigue tests on simple specimens with a hole tested before and after overaging (see Table 1). More realistic data would result from tests that include periods of overaging interspersed with fatigue.

These arguments are academic, since in practice overaging and creep occur together. The interaction between fatigue and creep is complex and, as shall be seen, cannot be ignored. Overaging is thus bound to be included.

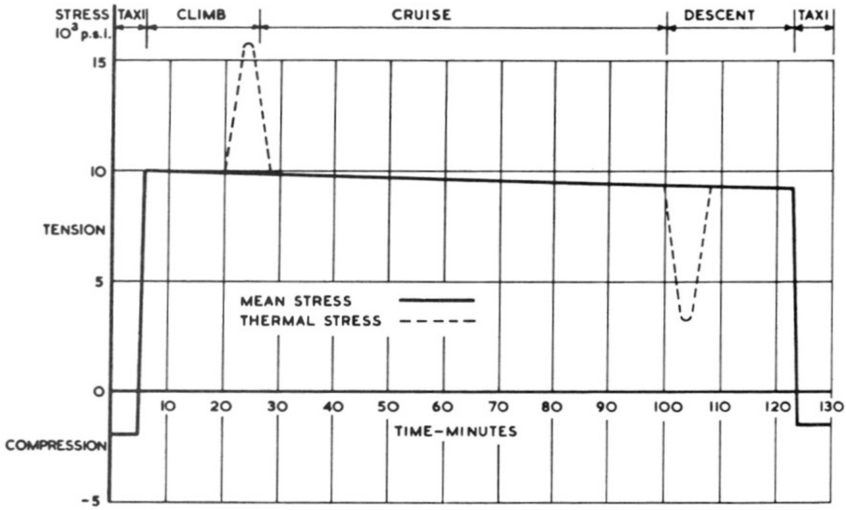

Figure 6. Mean stress and thermal stress cycle in spar boom.

TABLE 1

EFFECT ON FATIGUE STRENGTH AT 10^6 CYCLES OF THE PRESOAK FOR 1,000 HOURS AT 150°C TRANSVERSE SPECIMENS HOLED (K_T = 2.7)

Material	Type of stressing	Type of specimen	Condition	
			As received	After 1,000 hr at 150°C
DTD.5070A	Fixed minimum stress of 1,000 lb/in.²	Holed	9,100 ± 8,100	8,800 ± 7,800
	Fixed mean stress of 12,000 lb/in.²	Holed	12,000 ± 7,400	12,000 ± 6,000
2024-T.81	Fixed minimum stress of 1,000 lb/in.²	Holed	8,500 ± 7,500	8,800 ± 7,800
	Fixed mean stress of 12,000 lb/in.²	Holed	12,000 ± 6,800	12,000 ± 6,300
BSS L.73	Fixed minimum stress of 1,000 lb/in.²	Holed	6,900 ± 5,900	6,400 ± 5,400
	Fixed mean stress of 12,000 lb/in.²	Holed	12,000 ± 6,100	12,000 ± 6,100

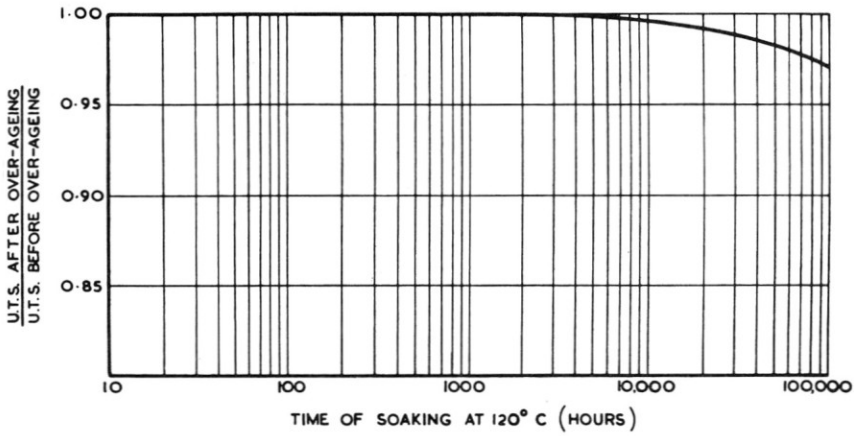

Figure 7. Ultimate tensile strength at room temperature after over-ageing at 120°C.

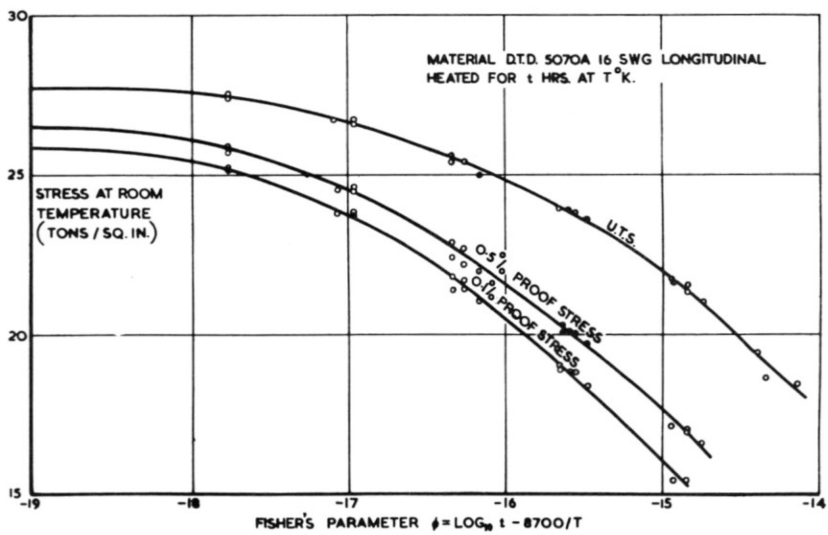

Figure 8. Static tensile properties of D.T.D. 5070A after over-ageing under zero stress.

CREEP

As regards creep alone, it should be noted that design is based on working stresses that should produce little creep deformation, something less than 0.1 per cent creep extension at the end of the life of the aeroplane. Most of this extension occurs in the primary creep stage and is realized in the early life of the structure as shown in Fig. 9, which is based on tests on plain specimens and a stress of 22,400 psi—roughly twice the working stress likely to be used in design.

In the structure there are a multitude of stress concentrations—for example, around rivet holes and bolt holes. At these places the local creep strain may be some three to four times the average. Such concentrations are important locally but do not greatly affect the gross distortion of the structure, such as the deflection of the wing tip, as is demonstrated in Fig. 10, which compares creep tests on specimens with and without hole under the same gross stress conditions.

Tests on specimens after creep has been imposed (Fig. 11) indicate that the strength falls steadily, but slowly with the amount of prior creep. Thus far, few data are available and as yet none applies to creep strains of order 0.5 per cent or less. Moreover, in reality, the temperature occurs in cycles and the loads are not necessarily steady while creep is occurring (mention has already been made of gusts at altitude). Creep data on plain specimens under fluctuating loads and on structural elements at representative working stresses are sparse, but what there is suggests that creep strain increases under varying load conditions. From the standpoint of a proving test, the possibility must also be recognised that in the design stage there may be underestimates of stress and temperature conditions. All these arguments support the representation of creep in the test.

It is generally accepted that there is appreciable recovery of creep strain when the load is removed while the temperature is maintained; in these

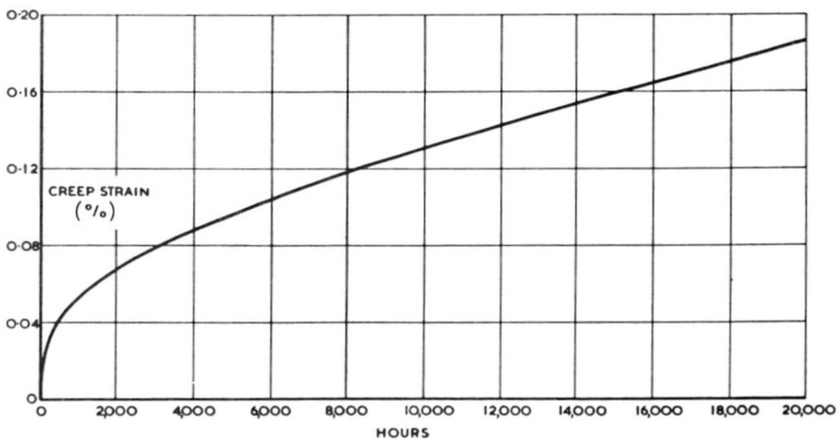

Figure 9. Creep of D.T.D. 5070A at 120° C, stress 22,400 lb/in.²

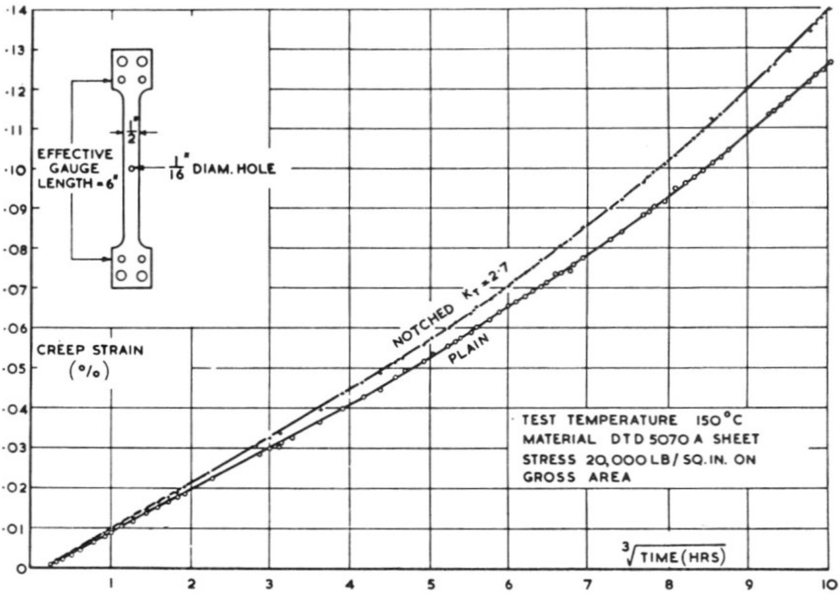

Figure 10. Comparison of creep of plain and notched specimens.

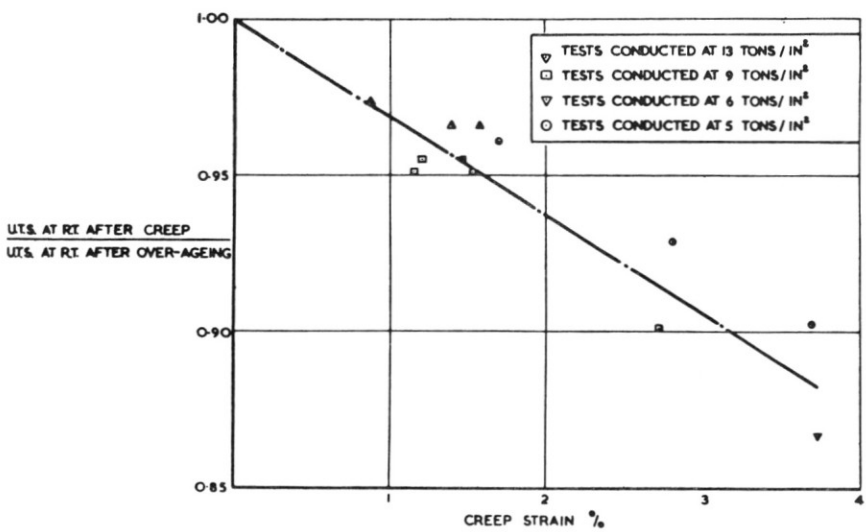

Figure 11. The effect of creep strain on the R.T. ultimate tensile strength.

circumstances a recovery of 0.05 per cent creep strain may result after the imposition of 0.3 per cent creep strain. In an aircraft the load is maintained while the temperature falls; some preliminary tests under these conditions show a recovery of order 0.005 per cent creep strain after an original creep strain of 0.3 per cent. More evidence is required, but the first indications are that there need be no allowance of time for creep recovery in a major test.

The necessity to factor the apparent life in a fatigue test is acknowledged by requirements [10], but has the time at creep also to be factored? At the average level of creep extension quoted above, it is the secondary stage which is of concern and a doubling of the time there gives some 30 per cent increase in creep extension. In the concentrations at holes, local creep extension becomes a relaxation in terms of stress distribution and there is a decrease in the rate of creep extension with time. A similar redistribution of load occurs from the more highly stressed members to the less highly stressed. These are arguments which suggest that the factor on creep should not be large. Until further work is done in the laboratory, the value of this factor is one of engineering judgement.

Under given stress conditions, times and temperatures can be correlated with the same parameter ϕ as has been mentioned in relation to overaging, namely

$$\phi = \log t - \frac{A}{T}$$

For aluminium alloy, an increase in temperature by 10°C shortens the time needed to produce the same strain by a factor of 3.5 and a rise of 20°C shortens the time by a factor of 12.5. Increase in stress also shortens the time needed to produce a given strain, but is nothing like so powerful; furthermore, an increase in stress may alter the gross distribution of stress in the structure with a consequent unrepresentative effect on the test result. Any factor required on the time at creep can be realised most simply by a small increase in temperature, with the added advantage that the test is speeded up. Since creep is so sensitive to temperature, the accuracy of control of temperature in a test cannot be overemphasized. In a major test, control to within ±3°C may prove extremely difficult and hence the test engineer may have to contemplate the application of a temperature which has been increased to allow for the accuracy of temperature measurement and control. If control could be made consistent, then individual accurate calibration of the points of measurement at the start of the test would be worth while.

CREEP-FATIGUE INTERACTION

It has been shown above that the fatigue loads occur essentially at room temperature, so that in service there is a sequence of fatigue, creep, fatigue, creep, and so on. It is thus necessary to establish the extent of the interaction. Simple reasoning based on the local relaxation under creep—at the edge of a hole, for example—suggests that some degree of creep might give improved endurance in fatigue. Preliminary work by Heath-Smith [9] on L65 copper alloy (Fig. 12) gives some support to this hypothesis, even

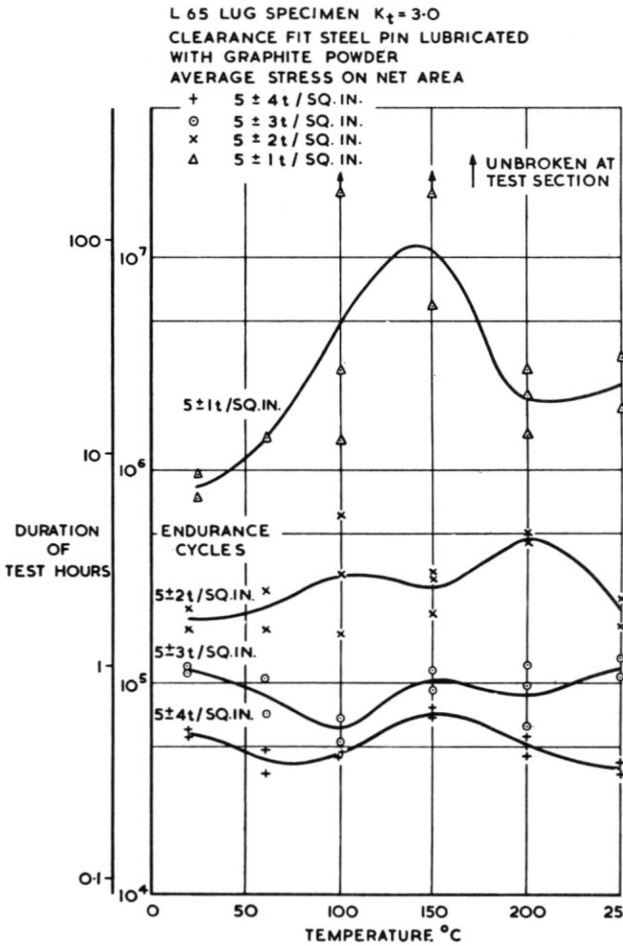

Figure 12. Effect of temperature on log mean endurance of lug specimens at constant stress levels.

when fretting must have been present. These tests also indicate, however, that if a certain degree of creep is exceeded then a drop in fatigue performance is to be expected. All of this suggests that in the case of fatigue there might be a critical amount of creep that could be accepted. Tests on wings of 2024 material (Fig. 13) suggest that sequences of fatigue and creep have small effect; the apparent slight reduction of endurance at this position in the structure is balanced by an apparent increase for cracks at other positions. Further data on DTD.5070A material (Fig. 14) where sequences of fatigue and creep were applied to small specimens are disturbing and their precise implication not known. In the first place, the alternating stress is high so that the endurance is only some 10^4 cycles; results in the 10^5 to 10^6 region are needed. In the second place, although nominal mean stress levels in these tests are perhaps a bit high, they are counteracted by the stress concentration at the hole in the specimen which is lower than that expected in a structure. These counteracting effects persist until a crack is initiated, but after that stage the experiments are perhaps pessimistic. On the grounds that the fretting actions in the structure tend to initiate cracks early, it seems that the interaction of fatigue and creep may well reduce the endurance of the structure by speeding up the rate of crack propagation.

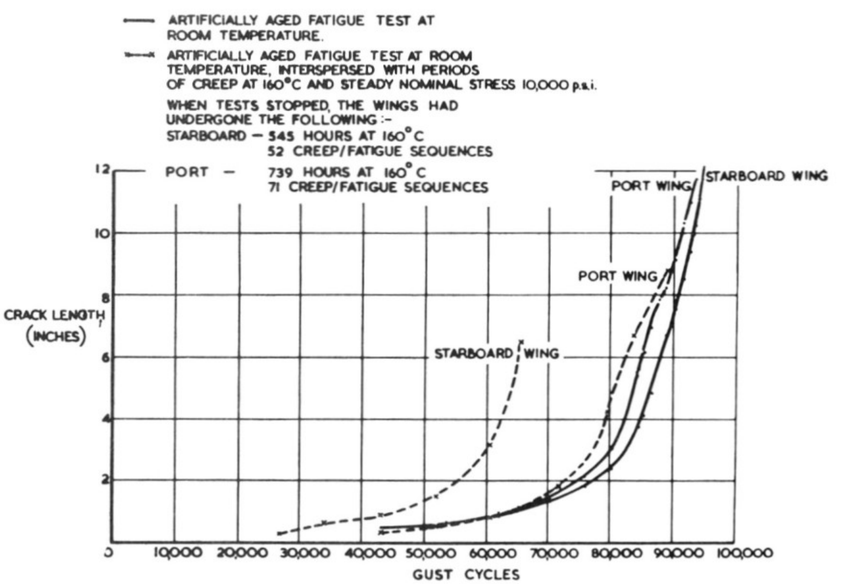

Figure 13. Wing fatigue tests—the effect of interspersed creep (cracks at end of Stringer 7).

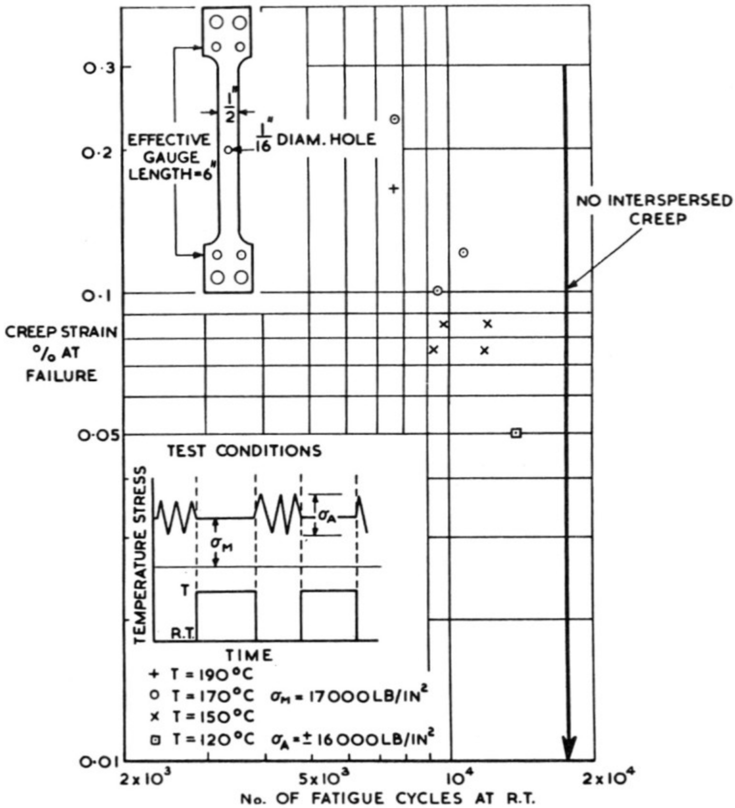

Figure 14. The endurance at room temperature of D.T.D. 5070A notched specimens under interspersed creep and fatigue conditions.

Such an increase in the rate of crack propagation needs to be considered in conjunction with the results of tests on cracked sheets under stress and with acoustic loading superimposed, which also show an increase in the rate of crack propagation. The relevance of these two findings to fail-safe design is of utmost importance. Not only must these speed-ups be remembered when interpreting particular test results, but since discovery is fundamental to a fail-safe design, research on inspection methods and procedures needs to be pursued energetically, so that there will be certainty that cracks in service will be found in time.

Attention has been concentrated on the tension surface. The compression surface of the wing will undergo creep in compression, so that if in tension a certain amount of creep is beneficial, creep in compression with subsequent tension loading during taxying may be harmful. Fortunately, the stress amplitudes during taxying are fairly small and the net effect may be negligible, but this needs to be demonstrated in the laboratory.

THE TIME REQUIRED TO TEST

In service the aircraft will be used over a variety of routes and on a proportion of the flights, estimated [2] to be 16 per cent; supersonic speeds will not be realised. For civil transports the number of flights is a better measure of the fatigue life than is the number of flying hours. Annual utilization of about 2,000 flights is envisaged which, if 10 years is judged to be an economic life, gives a life in service of 20,000 flights. To prove the fail-safe features of the design it is necessary to realise on test some 40,000 flights and more than 60,000 flights for those parts of the structure in the "safe life" category. The typical flight shown in Fig. 1 lasts a little over two hours, whereas the average length of flight in service is likely to be [2] of order one hour and 20 minutes. If the test could not be accelerated the actual testing time alone would range from six to nine years. To this there would have to be added some two to three years for maintenance, repair, and inspection of the test specimen and test equipment. Such a time scale cannot be tolerated, since the results from the test would not be available sufficiently in advance of service experience to ensure safety and to gain guidance on inspection and repair procedures. Furthermore, if the design should prove to have a poor endurance, the economic penalties might range from costly modifications to aircraft in service to early withdrawal of the fleet. It is obvious that the test must be completed in a reasonable time; it is suggested that a maximum of five years should be the aim.

Completion of the test in five years requires the fullest use of technical knowledge to simplify the test conditions. The compromises adopted will be particular to the individual problems presented by each design. Procedures should be devised so that the maximum time is spent in actual testing. Inspection of the test specimen must be planned to give the minimum interference in the testing programme. Testing equipment has to be highly reliable. In short, a balance has to be struck between the possibility of occurrence of defects in the early stages of the test and the need to keep the test sufficiently advanced on aircraft in service. It might be assumed, for instance, that serious defects are unlikely to be found before 5,000 flights are achieved on test or, in other words, that the design has some competence.

As has been noted it should be practicable to apply the mechanical loads in five minutes. In this connection, with skin temperatures of 120°C or more, pressurization of the cabin will be with air, not water, which implies the use of a filling medium in the cabin to reduce the volume of air that is required. The fullest use has to be made of prior tests of partial specimens of the cabin, using first water and then air, to provide assurance that the final test will be safe from explosive fracture and that reliable methods have been devised to detect cracks before they become dangerous.

The heat cycle may be considered in two phases—the transient phase of heating and cooling and the steady phase during cruise. Heating and cooling each occupy about 15 minutes in reality and there seems little prospect of much shortening of this time. An increase in the rate of heating leads to an increase in thermal stress, but this cannot be pushed too far or the maximum stress—the sum of stress due to mechanical loads and thermal load—gets high enough to influence unduly the creep and fatigue behaviour. In so far as fatigue is concerned, the thermal stress cycle and the stress cycles from the mechanical loads can be considered together; selective increase of a particular cycle or group of cycles leads to difficulties in the interpretation of the test result. As a first shot, therefore, it is proposed to assume that the thermal stress cycle is applied in the same time as in service—that is, 15 minutes to heat and 15 minutes to cool.

The steady conditions occupy 40 minutes of the average supersonic flight when the skin temperature is 120°C. This time can be shortened radically by raising the temperature. The equivalent creep and overaging at 130°C requires about 12 minutes and the comparable time at 140°C is 3.5 minutes. Unfortunately, these times and temperatures relate strictly to those parts of the structure at and close to the outer cover. The fact that heat has to flow into the depths of the structure, where there are variable heat sinks in fuel tanks, etc., means that there is a very real limit set to the amount of speed-up that can be tolerated. Calculations indicate that something of the order of 10 minutes might be achievable at the expense of increased difficulty in testing since it would necessitate special heating and cooling of the internal structure.

There remains the question of the time required for inspection, maintenance and repair. This time is related to the number of flights realised in the test and the experience of fatigue tests on subsonic aircraft is relevant, provided allowance is made for the greater complexity of the test. Estimates at this stage can but be approximate but they suggest that about two years may be associated with 40,000 and three years with 60,000 flights on test.

The times deduced above are judged to be achievable without too great an extrapolation of experience and knowledge. It is estimated that a subsonic flight can be represented in five minutes and a supersonic flight in 45 minutes. This means that if 16 per cent of flights are subsonic, the time to realise 40,000 flights on test totals roughly five years, and 60,000 flights require 7.5 years.

It is at once apparent that further reduction in these times is highly desirable. It is also obvious that further reduction has to come from a decrease in the time required to simulate the total heat cycle. In turn, this will complicate the interpretation of the test result and so research to aid interpretation must be continued energetically.

Thought is being given to the possibility of representing two flights at once, as follows:

1. Double up the number of mechanical load cycles.
2. Increase the thermal stress cycle by approximately 25 per cent so that the fatigue damage per test cycle is doubled.
3. Choose the temperature in the cruise phase so that the factored amount of creep will have been represented by the time 40,000 flights have been simulated on test.

The objective is to represent two supersonic flights in 55 minutes. The application of twice the number of mechanical loads in five minutes appears straightforward, but the representation of the heat effects in 50 minutes presents problems. If it can be done, however, then 40,000 flights should be realised on tests in 3.8 years and 60,000 when five years are up, since little heat may need to be applied in the last 20,000 flights. The test engineer is greatly exercised at the moment to evolve an adequate cooling system and there remain doubts whether with a practical system the internal structure can be cooled properly in the time. It may be necessary to extend the time between each heating cycle by perhaps half an hour, in which case 40,000 flights on test would take about 4.75 years.

CONCLUDING REMARKS

It is concluded that completion of the test within five years should be practicable in the case of aluminium alloy aircraft. It is emphasized that completion in this time involves test conditions that complicate the interpretation of the test result and that research must be directed to aid such interpretation. It is tacitly assumed that the design will be on fail-safe principles wherever practical. Where fail-safe design is not practical, detailed tests should be made where possible so that the amount of testing of the complete aircraft can be curtailed.

The task is a great challenge to the test engineer. He must take advantage of every feature of the particular design in order to reduce the complexity of his test equipment and test procedures to a minimum. For example, there may be appreciable areas of structure where thermal effects are negligible, and in such areas considerable latitude should be possible in the way in which heat is applied in the test. In extreme cases it might even be possible to omit heat altogether.

Although attention has been confined to the fatigue test on the complete aircraft, components such as the undercarriage and control surfaces may also be needed to be tested in fatigue. In many cases it should be possible to test these separately and this may have to be done to simplify the major test.

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