Concorde Structural Development

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Development of the supersonic transport is involving a wide variety of activities in the structural field. Although Concorde is designed for Mach 2, a speed at which aluminum alloys can continue to be used for the primary structure, evaluation of the chosen alloys under complex temperature-stress histories involves considerable laboratory effort. Although many aircraft have already operated under these conditions, none has previously had to last for 50,000 hr. The most significant new design considerations are creep and thermal fatigue. Creep affects the choice of basic material, the design of joints, and the design of structures through its interaction with fatigue life. Thermal stresses, arising from differential expansions within the structure, have a more important effect on fatigue design than on static strength. While the experimental techniques involved are novel, practical methods of accelerated thermal testing using convective heating and cooling are described.

Introduction

GREAT Britain and France have been working together on the supersonic transport aircraft project Concorde for many years, and a unique venture is now coming to fruition. Figure 1 shows the state of the prototype in August 1967. Concorde at this time has progressed past the manufacturing phase into the preflight installation and commissioning phase of the aeroplane.

The aircraft is designed to cruise at about Mach 2, a speed at which aluminum alloys can continue to be used as the primary structural materials. Although Mach 2 military aircraft are commonplace nowadays, no type has previously been required to last for 50,000 service hours of which at least 20,000 will be spent at the maximum cruising speed. This requirement has been of prime importance in the choice of materials, the development of manufacturing processes, and the design of the structure itself.

Basic Temperature Conditions

The first structural decision was to choose the appropriate aluminum alloy to meet the new environmental conditions. The most important of these is the effect of kinetic heating from the high-speed boundary layer. The surface temperature depends on the balance of the heat transfer to the skin from the boundary layer, the heat picked up by solar radiation, the heat radiated back from the surface itself to the atmosphere, and the heat transferred from the surface to the internal structure. The latter can occur by means of internal radiation, conduction through the structure, and convection of the air within the structure. It is profoundly affected by such things as the presence of fuel, as in the wing fuel tanks, or the refrigerated air passed through the cabin walls to keep the occupants cool. For the simple case of a part of the wing not affected by fuel tanks or by the proximity of the engines, the heat of the skin is simply a balance between skin friction, solar radiation, and the radiation from the skin internally and externally. Temperatures shown in Fig. 2 apply to such basic structure. Some advantage is obtained by painting the external surface white. A white surface can be made almost as good as a black surface at radiating heat whereas it is much better than a black surface in reflecting solar radiation. Use of such a white surface will result in a cooling of about 8°C at around Mach 2. This cooling is well worth having from the structural and from the air-conditioning point of view. From Fig. 2 it is evident that the design temperature at Mach 2.2 would be 120°C.

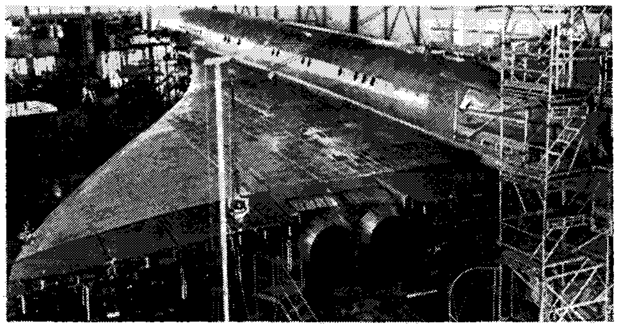

Fig. 1 Concorde prototype, August 1967.

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Material Characteristics

Static Strength

Figure 3 shows the variation of ultimate tensile strength with temperature for a number of well-known aluminium alloys including the material selected for the primary structure, CM.001. The latter material meets a special specification based on the material known in the United Kingdom as RR.58 or in France as AU2GN. The properties of the materials are those obtained after they have been subjected to 20,000 hr of soak at the temperature specified. They would, therefore, be relevant to those structural design cases appropriate to cruising speed or design diving speed. The high-strength aluminum alloys of the 7075 type show a rather rapid deterioration with temperature but the medium strength alloys of the 2024 type still show reasonable properties. L.73 is the British equivalent of the U.S. 2014 material.

The ultimate tensile strength shown in Fig. 4 is for materials tested at room temperature after being soaked at the temperature indicated for 20,000 hr. This property is relevant to those structural design cases that occur at all lower speeds, up to about Mach 1.8. The heavier gust and maneuver loads occur at the lower altitudes and speeds where structural temperatures are lower. There is virtually no

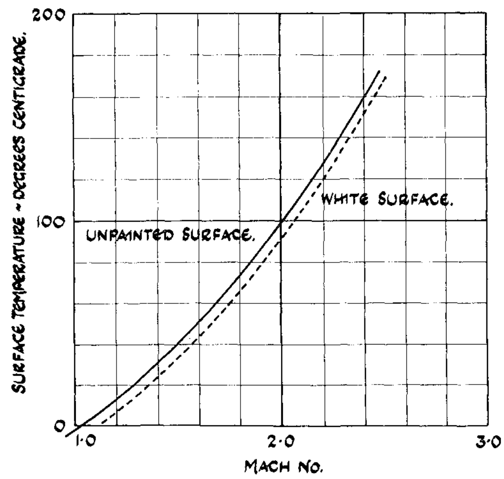


Fig. 2 Equilibrium skin temperature.

permanent deterioration in strength at room temperature of the chosen material CM.001. From this point of view, at least, the structure will have experienced no appreciable change after 50,000 hr in service. Some of the other materials such as L.73 or 7075 show rather more permanent deterioration.

Although the elevated temperature properties of CM.001 fall off slightly to about 92% of the original static strength, the recovery properties are 100% of the original strength. Furthermore, exposure to temperatures of up to 200°C for periods up to 1 hr has virtually no effect on the long-term strength of the material. Thus, no special hazard arises should the aircraft inadvertently exceed the design speeds.

Fatigue Strength

Static strength is perhaps less important for a civil aircraft than fatigue strength. To achieve the required fatigue life the steady load on the material is limited to about 25% of its ultimate tensile strength. Figure 5a shows the characteristics of some of the candidate alloys when fatigue-tested at 120°C with a constant mean stress of about 25% ultimate. There is little to choose between the materials under these conditions. Figure 5b shows the fatigue characteristics of notched specimens with a theoretical stress concentration factor of 2.61. CM.001 is superior to the others throughout the range. However, as will be seen later, the fatigue problem on the airframe is more complex than is indicated by such simple test results.

Creep Strength

Creep strength of the material used in the primary structure is another property that is most important in terms of the

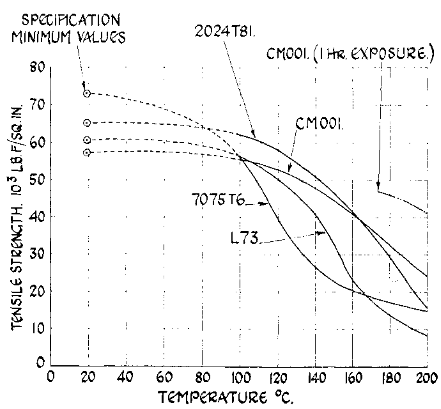


Fig. 3 Ultimate strength at elevated temperature after 20,000 hr.

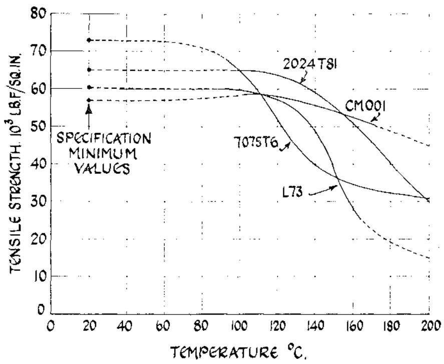
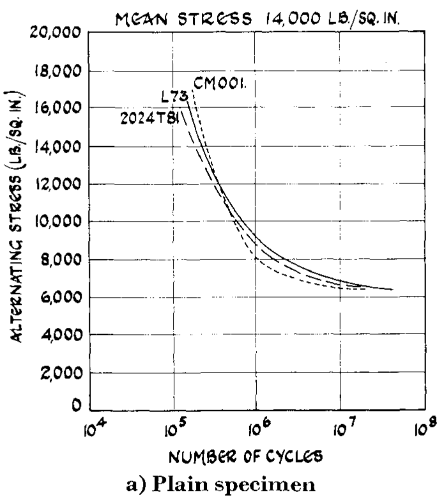
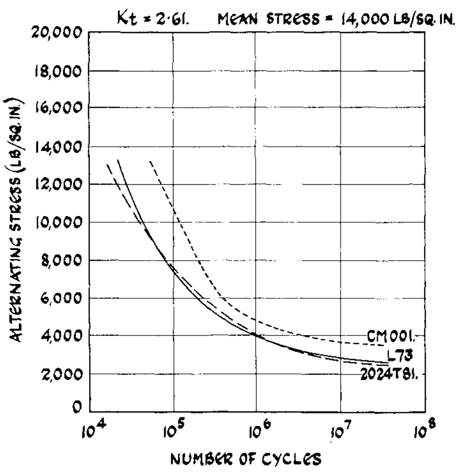


Fig. 4 Ultimate strength at room temperature after 20,000 hr at elevated temperature.

long-term structural life. Although creep is a familiar problem to the engine designer it is a relatively new consideration for the airframe designer. Most creep data available at the early stages of the Concorde project were in the form of creep rupture information. It is desirable to restrict the amount of creep deformation to an amount well below this rupture condition. It should be limited to a small value equal to that obtained by one application of limit design load, i.e., 0.1% of the original length. Creep data available at the early stages of the program were inadequate for measuring the relative performances of various alloys against this criterion. Considerable test technique development was necessary to get adequate accuracy within this range of small deformations. Figure 6 shows the variation of creep strain with time for several aluminium alloys when subjected to a



a) Plain specimen



b) Notched specimens

Fig. 5 Fatigue life at 120°C.

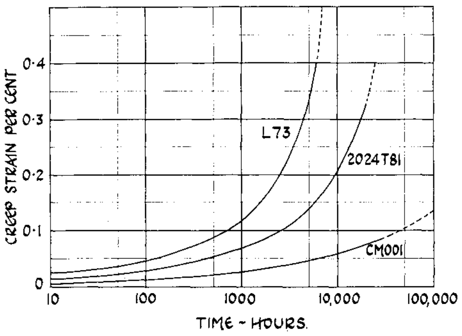


Fig. 6 Creep strain of plain material.

steady stress of 25,600 lb/sq. in. at a temperature of 120°C. CM.001 remains within its 0.1% limit for up to about 50,000 hr. Under the same conditions, L.73 is unusable over this length of time.

It is in creep strength characteristics that CM.001 achieves its greatest advantage over the other aluminum alloys. The load levels required to produce 0.1% of strain are far above those that would normally be allowed as working stresses because of fatigue considerations. Only in more elevated temperature conditions does the 0.1% strain creep stress fall near that required for fatigue strength considerations. Such conditions only occur in areas such as the engine installation and here lower stress levels can be accepted.

Basic Material

CM.001 material is based on RR.58. The manufacturing processes were modified to improve creep resistance. Static strength, fatigue strength, and creep strength were major factors in the selection of CM.001 as the primary structure material for Concorde. However, similarity to RR.58, which is not a new material and with which there was considerable previous experience, was also a factor. RR.58 has been used in engines and on airframes in engine nacelle ares. It is a good material for forgings and has been used for undercarriage parts as, for example, on the Caravelle. RR.58, and, similarly, CM.001, can be treated in manufacture as any normal aluminum alloy available in all conventional forms such as sheet, plate, bar, forgings, and extrusions. It can be manipulated using all conventional processes. It can be fastened together by rivets, bolts, spot welds, or adhesives. Its corrosion and stress corrosion resistance is similar to that of conventional aluminum alloy materials. (Although the whole of the airframe will be painted, this is mainly to reduce surface temperatures.)

Joint Design

Riveted Joints

A considerable amount of development work has gone into finding the best design of riveted joints for the various combinations of stress and temperature which occur on the aircraft. This effort involved consideration of rivet material, rivet geometry, spacing and size of rivets, as well as the type of sealant between the mating surfaces of the sheets to be joined. The design criterion adopted was to restrict creep deformation in a joint than that would normally be permitted under the limit loading conditions in static tests, i.e., 2% of rivet diameter.

It was believed in the early stage of the program that the creep of riveted joints would in fact follow the same extrapolation laws as those for the basic materials themselves. Initial deductions on the behavior of the riveted joints were based on short-term testing at temperatures higher than would occur in operation (e.g., 175°C or 150°C). It was thought that aluminium rivets would be acceptable from the point of view of creep, provided that the shear stress was limited to give a strain less than 1% of rivet diameter to avoid cracking of the heads. It was not until results became available from the 120°C real time tests that it became obvious that the initial selections were not the best (Fig. 7). These tests showed that head cracking occurred at lower strains than had been previously assumed. Monel rivets are now used in all areas where sustained stresses are high and where the temperatures are 120°C or more. Monel riveted joints have the same creep strains as aluminum but do not suffer from head cracking. No difficulty has been found in setting this type of rivet or obtaining a satisfactory flush finish.

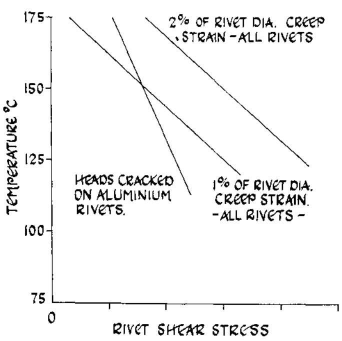


Fig. 7 Allowable creep stresses for riveted joints.

Bonded and Brazed Structures

A significant amount of honeycomb is used in the airframe structure. Adhesives are used with aluminum honeycomb and brazing in the case of stainless-steel honeycomb. In the choice of materials for aluminum honeycomb, creep has played a significant part. Whereas the usual aluminium core materials suffer quite substantial creep strains at 120°C, 2024 core material shows much lower strains. Development of core material in RR.58 should result in even better creep performance. The choice of the adhesive has been based not only on the conventional properties of shear strength or peel strength, but also on its creep properties. Extrapolation from short time data is more difficult for adhesives since increased testing temperatures may result in significant chemical changes to the adhesive, thus degrading the results. Again there is no fully satisfactory substitute for long-term testing at the correct temperatures. Adhesive manufacturers' long-term tests are usually confined to evaluating the static properties of the material. Creep data are inadequate.

Brazed stainless-steel honeycomb has been used in the hotter areas of the airframe, namely the engine nacelles, where engine temperatures are added to the normal kinetic heating temperatures. For this aircraft the application of brazed honeycomb has been limited by creep. At temperatures above 250°C creep becomes unacceptably large. For hotter areas either welded honeycomb or fabricated structure employing welding or mechanical jointing methods must be employed.

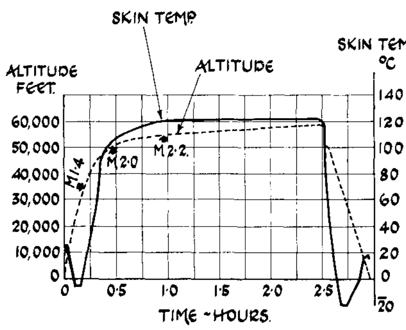


Fig. 8 Typical flight profile.

Thermal Environment

All aircraft are subjected to a random variation of stresses, and similarly all aircraft are subjected to variations of temperature. The difference in the case of the supersonic transport is the much greater significance of the temperature variation, not only in relation to material properties, but also because of its effect on the stress distribution in the structure. It is useful to consider the basic mission of the aircraft, which is a long-range flight profile involving sustained cruising at over Mach 2 (Fig. 8). Altitude is plotted against time of flight indicating the speeds and the equilibrium temperature of a typical part of the skin. Initially, the skin temperature drops as the aircraft climbs. As the aircraft speed increases above Mach 1, kinetic heating causes the skin temperature to rise. It reaches its maximum temperature of 120°C when the aircraft speed has achieved Mach 2.2. The reverse process occurs during the deceleration and descent phase when the temperature again drops rapidly in the below-zero ambient temperature at subsonic speeds at altitudes of around 30,000 ft. Skin temperature will then rise again as the aircraft descends to sea level. There is thus a once per flight cycle of temperature from −20° to 120°C. This cycle of temperature is associated with other once per flight cycles such as the ground to air cycle and the cycle of pressurization in the cabin. Superimposed upon these basic load cycles are a multitude of smaller additional cycles due to maneuvering and gust loads. The supersonic transport spends a relatively small amount of its life at the low altitudes, where turbulence is greater. Therefore, these additional loadings are not as significant as for the subsonic aircraft. Most of the short-term variations of flight loads occur, in fact, while the structure is relatively cool. During the cruise, when the structure is hot, there is very little variation of flight load.

Tests to determine the effect of environment on the structure could be simplified by combining various parts of a typical flight plan into blocks of a large number of flights. All the hot steady load part of the flights could be carried out in one single part of the test. Subsequently, a fatigue test could be made with the structure cold until failure occurs. However, the results of such simplification can be misleading since the period of creep conditions may, in fact, affect the resulting fatigue behavior considerably. Alternatively, by introducing a large number of interspersed periods of creep, in between the periods of fatigue cycling, one arrives at a more meaningful result. In general the higher the creep strain the lower is the fatigue life, and if a part is working up to the limiting allowable strain of 0.1% its fatigue life may be only half the room-temperature value (Fig. 9).

Some parts of the structure, such as the fin, are not loaded during the hot conditions. Instead of creep during each flight there is a period of hot soaking. It is necessary to introduce a large number of interspersed periods of hot soak during the course of the fatigue test. Figure 10 shows the effect of heat "damage" on fatigue life. Heat damage of unity corresponds to one aircraft life and is calculated for different test conditions using an assumed relationship between temperature and time. It can be seen that the results are somewhat less favorable than those obtained when the structure was subjected to periods of creep.

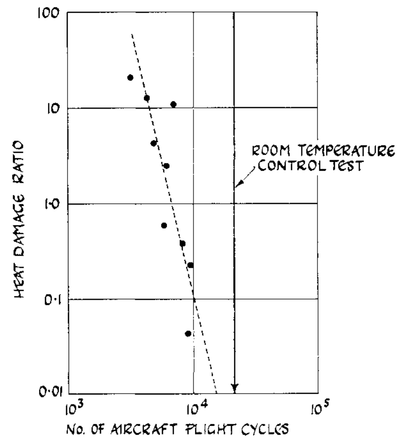


Fig. 10 Effect of unloaded temperature cycles on fatigue life.

A rig has been devised to apply representative loading to a number of small specimens simultaneously. Figure 11 shows the type of test cycle that can be applied, using this facility at Hawker Siddeley Ltd., Manchester, on both simple specimens and joints. Tests made with such representative conditions must inevitably involve cycling at a speed consistent with the time taken to raise and lower the temperature. In the case of tests on larger pieces of structure, or on complete airframes, the frequency of testing is further reduced since the time of carrying out one flight cycle must depend entirely on the time required to get all the heat into the structure and then to get it out again and return the structure to its original datum condition. In the case of complete structures this may take up to 1 hr per cycle. The effect of fatigue test frequency has long been a problem but it is now complicated by temperature considerations.

Thermal Stresses

So far this paper has been concerned largely with the effect of variations of temperature and stress on the characteristics of the material. Some of the most significant stress conditions in a supersonic transport are caused by the thermal effects themselves. Because of the differences of temperature throughout the structure, different parts of the structure expand by different amounts. If not relieved by thermal expansion joints, these give rise to stresses in the structure, additive to any others arising from the normal flight loads on the aircraft. If sustained during cruise conditions, an increase in the amount of creep can occur.

This effect is not particularly onerous in terms of ultimate static strength. If these thermal strains plus the strains arising from mechanically applied loads, such as gust loads, are such that the total strain goes beyond the yield, then, with a ductile material, the thermally induced stresses will be

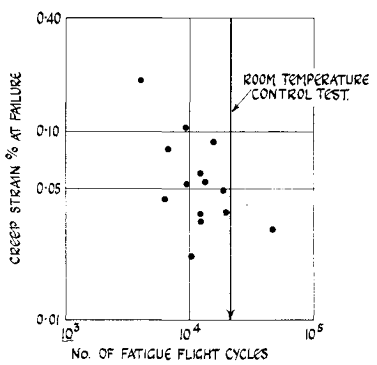


Fig. 9 Effect of creep on fatigue life.

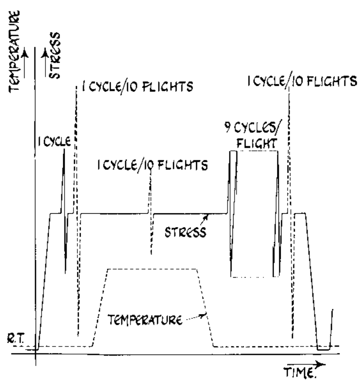


Fig. 11 Typical fatigue test cycle.

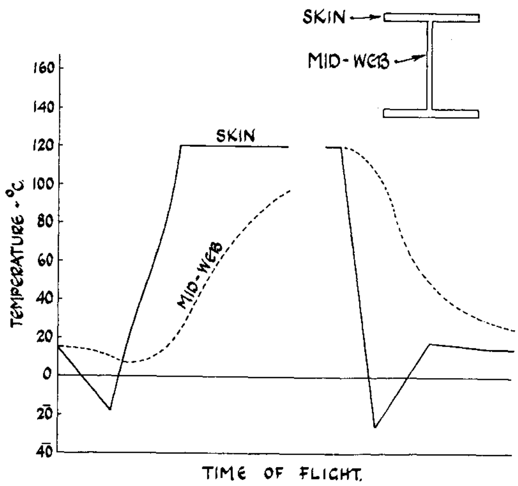


Fig. 12 Temperature variations in a spar.

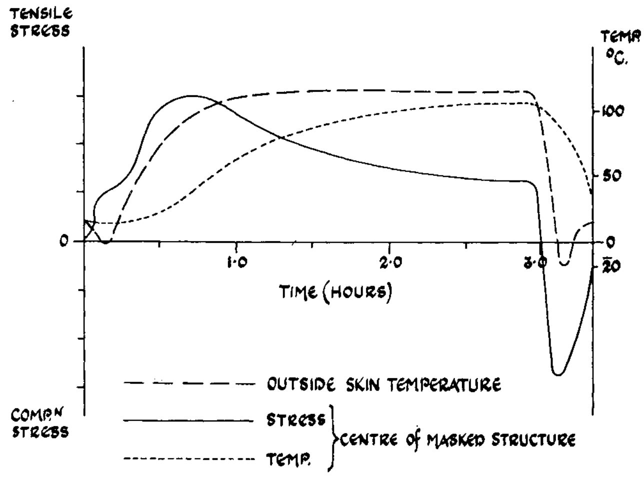


Fig. 14 Flight cycle of temperature and stress in wing masked area.

automatically relieved and will not appreciably affect the ultimate strength. They are, however, an important consideration for proof design conditions.

Thermal Fatigue

The severity of the stresses induced by thermal strain is indicated by the simple case of two equal-area aluminum alloy members joined together, one hot and the other cold. The stress in pounds per square inch is 118 times the temperature difference in degrees Centigrade. Thus with temperature differences of typically 100°C, stresses of 11,800 lb/sq. in. will be induced. These stresses, added to other flight stresses, can give once per flight ranges of total stress which can be sufficiently high to cause fatigue problems. They cannot necessarily be dealt with simply by increasing the amount of material put into the structure, since they are determined rather by the relative areas of the two parts joined together.

It is instructive to look at examples of such thermal stresses. Figure 12 shows the temperature variation during a typical flight of the outside skin, and also the corresponding temperature variation of the center of a spar or rib web. There is a cycle of temperature difference reaching a maximum towards the end of the acceleration phase and later a maximum of the opposite sign at the end of the deceleration phase. Increasing or decreasing the rate of acceleration or deceleration has a corresponding effect on these temperature differences. However, the large temperature differences, which cause the highest thermal strains, are those associated with relatively deep structure, and here the variation of the internal temperature depends much more on the rate of heat soak through the structure. The more severe thermal stresses are not susceptible to a very great amount of modification by altering the flight procedures on the aircraft.

Another important example of the thermal stress problem occurs in the area of the intersection of the fuselage and wing shown in Fig. 13. The supersonic transport configuration dictates that a fairly large proportion of the fuselage structure is typically of this section. There is an area of the fuselage below the floor which is intersected and therefore thermally shielded by the wing. The top portion of the fuselage during cruise is maintained at a temperature of about 120°C. That section of the fuselage, which is masked by the wing, is much cooler and picks up heat only due to conduction through the structure itself plus a certain amount of internal radiation and convection. The bottom of the fuselage is also subjected to the full temperature of 120°C. During acceleration the top and extreme bottom of the fuselage are trying to expand in the fore and aft direction, while the center of the fuselage below the floor will remain cool and will try to retain its original dimensions. The top and bottom of the fuselage will be in compression while the center portion, underneath the wing, will be strained in tension. In deceleration the reverse will occur. By this time the portion masked by the wing will have warmed up but the portion above the wing and at the extreme bottom of the fuselage will cool down more rapidly as the aircraft decelerates. The top portion of the fuselage and the extreme bottom portion will try to contract while the portion masked by the wing will try to remain at its extended length. The combined effect is to produce a once per flight

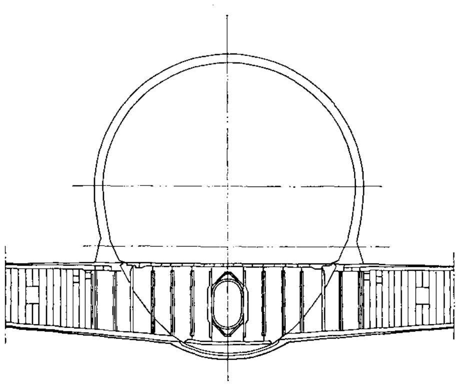


Fig. 13 Fuselage section.

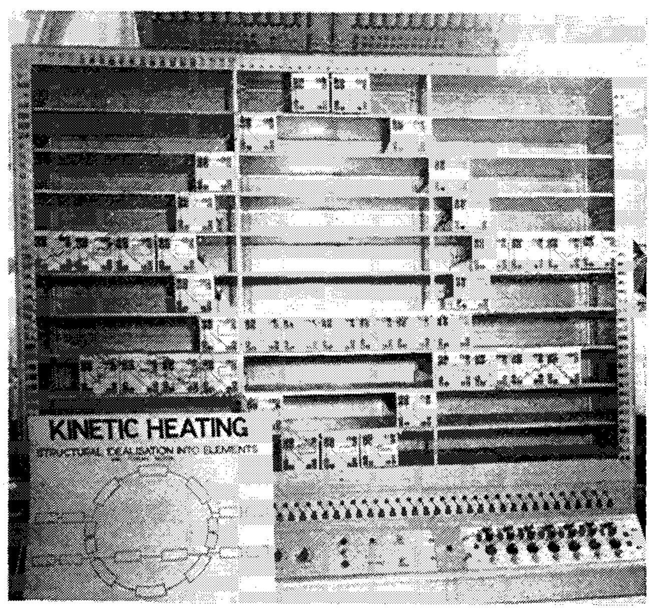


Fig. 15 Kinetic heating analogue computer.

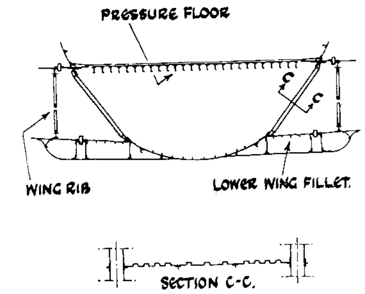


Fig. 16 Strain relievers in fuselage.

cycle of stress in the masked area which is tensile in the acceleration phase with the material relatively cold and compressive in the deceleration phase with the material relatively hot (Fig. 14).

Assessment of these problems has been greatly assisted by the use of the special-purpose computer (Figure 15), which is a direct analogue of the thermal characteristics of the structure. Each of the important thermal parameters is represented directly by its electrical analogue. It is possible to study rapidly the temperature distributions throughout the structure which result from various thermal environments. Temperature and stress time histories of the type shown in Fig. 14 are automatically plotted for all the elements of the structure. The ease with which the structural parameters can be altered has greatly facilitated the devising of methods of alleviating these thermal stresses.

In the foregoing case, it is possible to reduce these stresses considerably by means of strain relievers in the sides of the fuselage masked by the wing (Figure 16). These consist of channels in the skin of the fuselage which permit it to deflect in a fore and aft direction but at the same time preserve the fuselage intact as far as the normal hoop tensions due to pressure loads are concerned. They eliminate the longitudinal stress cycles at the center of the fuselage skin. There remains a stress cycle at the top and bottom of the strain reliever but of an acceptable reduced magnitude.

Thermal Fatigue Testing

Early development work was carried out using specimens of the type shown in Fig. 17 which was used to study the fuselage/wing masking problem. The center portion of the specimen represents the part of the fuselage masked by the wing while the outer portions represent the top and bottom of the fuselage and the wing structure itself, each part having the correct thermal characteristics. The outer portions are in turn heated by radiant lamps and cooled by passing air over them. This type of test facility is relatively cheap and can test quite rapidly since the total time for one cycle may be as little as 6 min.

The next stage involved two large test specimens each consisting of a section of a fuselage with part of the wing simulated. One specimen (Fig. 18) has been statically tested in Toulouse under mechanical loads at temperature to determine both the temperature distributions through the structure and the resulting stress distributions.

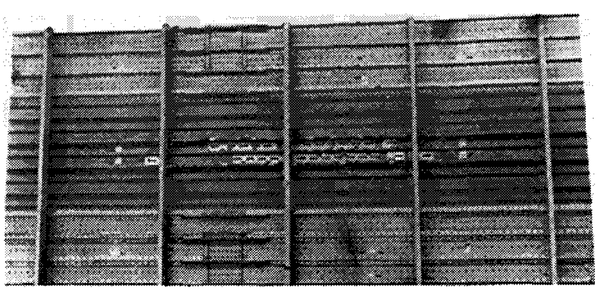


Fig. 17 Fuselage panel for thermal fatigue test.

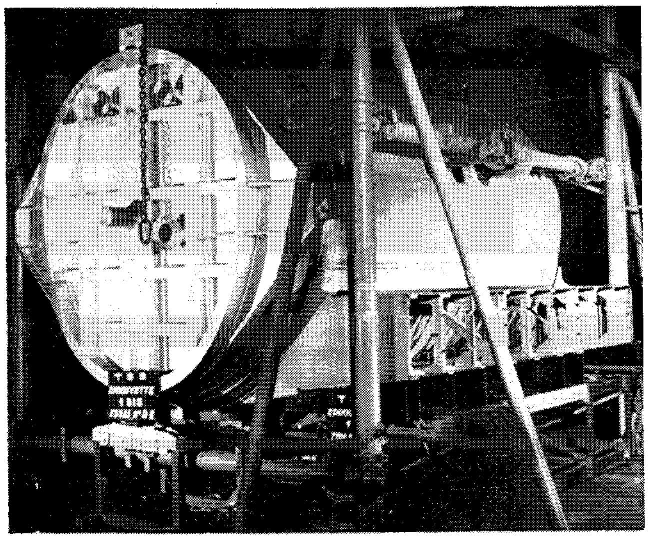


Fig. 18 Test specimen 2.2, section of fuselage.

Fatigue tests have been carried out on another similar specimen, subjected to thermal cycles in a special test facility (Fig. 19). In most kinetic heating tests, the heat is applied by means of radiation from infrared lamps, which requires individual control of each local area by monitoring the surface temperatures to regulate the heat input. It is necessary to have a large number of separate control areas, e.g., about a dozen in the case of the specimen shown in Fig. 18. To cool the specimen, it must be subjected either to cold air produced by a refrigeration system or by injecting a coolant such as liquid nitrogen. Either system implies ducting air over the specimen and it is evident that economies can be achieved if the same method is used for the heating phase. The main advantage of such a convective heating system is that it is necessary to control only the inlet temperature and mass flow of the hot or cool air over the specimen in order to get correct representation of the temperature conditions, since the correct heat-transfer conditions are simulated. Such a simplified system is even more attractive on much larger specimens such as a complete airframe, which may require one or two additional controllers for air temperature and mass flow but still be considerably simpler than the corresponding radiant heat system, which would require many hundreds of control points.

This particular facility has used effectively a closed-circuit wind tunnel around the specimen with two bypass systems which can be alternately connected to this circuit (Fig. 19). Air, heated by gas heaters to 160°C, is passed into the circuit during the acceleration phase of the flight cycle. In the cool-

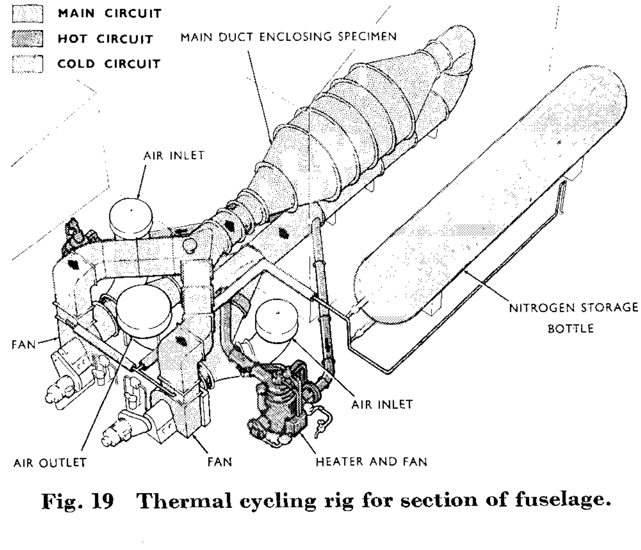


Fig. 19 Thermal cycling rig for section of fuselage.

Fig. 20 Box specimen for accelerated thermal fatigue research.

ing phase this hot air is replaced by air, cooled by injection of liquid nitrogen.

To carry out one heating and cooling cycle a minimum time of about 1 hr is necessary, even if the steady cruise period of the flight is foreshortened. The acceleration phase of the flight, and the subsequent time required for the internal parts of the structure to rise to the equilibrium value, is at least 30 min. The deceleration phase can then be commenced but again about 30 min elapses before the inside part of the specimen drops to the datum temperature at which the new flight cycle can be started. Increasing the rate of heating or cooling the outside surface does not have much effect on these times since they are largely determined by the heat flow through the structure and its thermal inertia.

The only feasible way to reduce total testing time is to increase the damage put in by each of the test cycles. This implies increasing the magnitude of the stress cycle, which implies increasing the temperature range. The simplest way of increasing the temperature range would be to increase the maximum temperature in the cycle. Since the creep of the material is dependent upon the maximum temperature, however, this could cause excessive creep deformation, thereby affecting the actual fatigue behavior. It similarly affects the static strength of the material, but this may not be so important.

Increase in the maximum temperature, therefore, is limited and leaves the alternative of a reduction of the minimum temperature, i.e., cooling the specimen to lower temperatures than occur in flight. Unfortunately, the cost per test cycle is affected far more by reductions in the cooling temperature than it is by increases in the heating temperature. However, since the effect of increasing the temperature range is itself to reduce the number of cycles appropriate to an equivalent aircraft life, the over-all cost increase is not considered excessive.

There is a limit to which this process can be carried even if a high degree of refrigeration is applied to the system. Being concerned not only with initiation of fatigue cracks but also with the subsequent rate of propagation of the cracks and the residual strength of the structure in the presence of these cracks, it is not desirable to increase the stress ranges applied during the fatigue cycle to an excessive degree. Furthermore, achieving correct combinations of flight or pressure stresses with thermal stresses requires that these be increased in step with the thermally induced stresses. To halve the testing time would require an increase of about 25% in the thermally induced stresses and, to be consistent, a similar increase in the cabin pressure. This is about the maximum increase that could be permitted and still avoid invalidating the results during fatigue crack propagation and residual strength testing.

A number of small specimens representing various typical compontents of the aircraft structure (Fig. 20) are being tested at the Royal Aircraft Establishment at Farnborough, under a variety of different test conditions to study the nature of the fatigue defects that occur. The RAE tests will assist in providing rules for interpreting the accelerated thermal fatigue tests and will check their validity. Interactions between creep, hot soaking, and fatigue may result in false conclusions being obtained from accelerated tests and make simulation of thermal effects by applying equivalent mechanical loadings questionable. Results of fatigue life tests made by reproducing the thermal stress cycle using mechanical loading alone have not shown good correlation with the fatigue life experienced when the stress was induced by the correct thermal conditions. A possible reason may be that under thermal stress conditions, the maximum tensile stress occurs when the structure is cold, while the maximum compressive stress occurs when the structure is hot. The correct phasing of stress cycle and temperature cycle could be all important, particularly if the material buckles during the compressive part of the cycle.

Concorde Component Test Program

It is evident that, in the majority of cases, the thermal effects on Concorde components will have to be produced directly by cycling the temperature on the specimens. Figure 21 gives an impression of the scope of the development program on structures for Concorde. All these specimens will be tested statically under thermal conditions. Some, in addition, will be subjected to thermal cycling in order to assess fatigue life and fail-safe characteristics. The majority of these specimens have now been completed and are intended to provide design information as well as a basis for certification clearance. An example is the large forward fuselage specimen (Fig. 22). It has already been tested statically under cold conditions and will later be statically tested under hot conditions, followed by fatigue testing consisting of thermal cycles, pressurization cycles, ground to air loads, etc. Finally it will be subjected to fail-safe testing, involving thermal cycling to investigate the propagation of cracks.

The test facility, a development of the one shown in Fig. 19, is being assembled at this moment at the RAE. The ducting, which completely encloses the specimen, must itself be flexible to allow the bending deformations of the structure to occur without either providing restraint to the airframe or significantly affecting the air flow across the specimen. A further complication is that air must be used for pressurization. This implies filling the specimen with polystyrene, or similar material, to reduce the volume of air both for safety and to reduce the pumping capacity required. However, the "filling" itself will tend to absorb heat during the thermal cycles and its contribution to the thermal inertia of the system

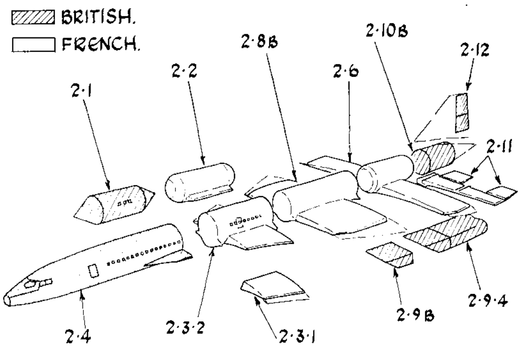


Fig. 21 Component test specimens.

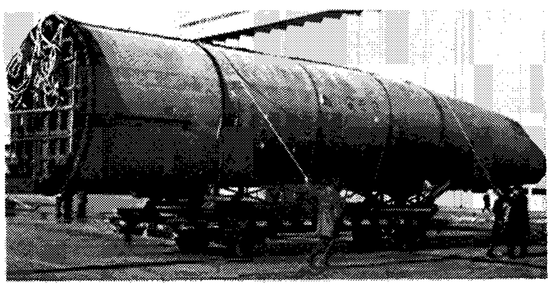


Fig. 22 Test specimen 2.4, nose and forward fuselage.

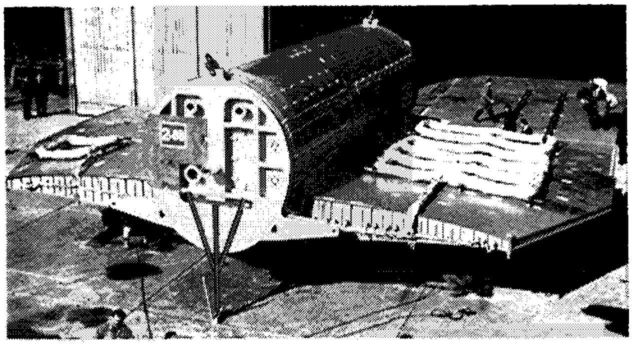


Fig. 23 Test specimen 2.8B, center wing and fuselage.

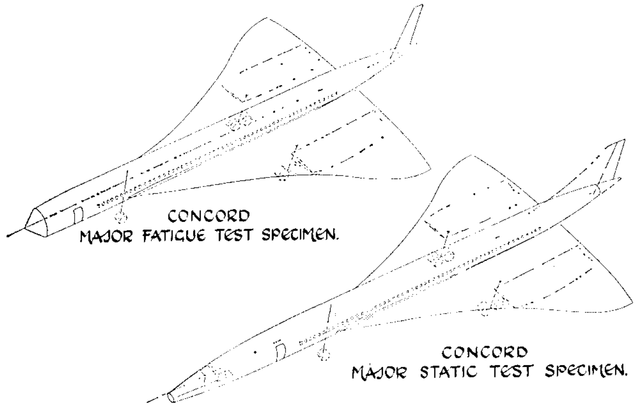


Fig. 24 Major static and fatigue test specimens.

is significant. It is necessary to introduce a rudimentary air conditioning system into the specimen both to insure correct temperature distributions on the skin and to remove heat from the filling in each cycle. Naturally the presence of the filling and also the presence of the ducting covering the whole specimen makes inspection during testing more difficult than usual. To achieve a high utilization, necessary if the total testing time is to be kept within resonable bounds, automation of the control equipment has been provided by the use of on-line digital computers for control with continuous monitoring of load levels, pressure levels, temperature conditions, and strain measurements throughout the structure.

The largest of the other specimens which already exist is the wing center section specimen (Fig. 23) which is well on through its test phase in the new structures facility at Centre d'Essais Aeronautique de Toulouse (CEAT) in France. As can be seen from Fig. 21 the program in France is every bit as extensive as that in the U.K.

Major Airframe Tests

The activity on structural testing will culminate in two major tests, consisting of two virtually complete airframes, one tested statically and the other tested for fatigue (Fig. 24). The static test will be in the new structures facility at CEAT, and the fatigue test in the new structures facility at the RAE.

The static test aircraft will be subjected to flight and ground loads including thermal effects where appropriate. It will be necessary to simulate correctly the effect of air conditioning on the temperature distributions through the fuselage and also the heat-sink effect of the fuel by filling the tanks with an appropriate fluid. The test techniques for this large specimen are being based on those being developed now for the smaller specimens.

Likewise the fatigue test facility at the RAE will be based on the experience gained with the rig for fatigue testing the section of fuselage (Fig. 19) and then subsequently on experience with the forward fuselage specimen (Fig. 22). Major attention is being given to studying the most efficient method of carrying out the test.

It is fairly well accepted on a world-wide basis that it is desirable to carry out some large-scale fatigue testing at least on the first of a new generation of aircraft. Most manufacturers will consider including such a major fatigue test in their development programs unless the type is closely related to a previous successfully tested design. Although such tests undoubtedly provide evidence for certification, their greater value to the manufacturer, and subsequently to the operator, probably lies in detecting fatigue weaknesses within the structure at a relatively early stage. The design can then be modified in sufficient time to avoid such weaknesses causing unscheduled delays during operation. Airworthiness may well be ensured by carrying out a substantial amount of fail-safe testing but this will give no indication of the extent to which crack initiation might occur in service and will not give assurance to the manufacturer and the operator that the aircraft is going to have a reasonable structural maintenance cost. As the design cruising speed is increased the effects of thermal stresses become more important. Subsequent change of primary structure material to titanium or stainless steel, may do little to reduce the effect of thermal stresses, if it is accompanied by a corresponding increase in maximum speed. If the cruising speeds chosen are as high as the material properties themselves will allow, then the problems of creep and fatigue interaction once more become important.

It seems probable, therefore, that, for supersonic transports designed to cruise above Mach 2, a test facility will be required capable of matching the thermal cycling. Such a facility is more expensive to provision, and may be up to six times more expensive to run, than the equivalent cold test facility. As the speed and size of the aircraft increase the costs of such a test add considerably to the investment involved in development of the aircraft and may be a factor in limiting the choice of design speeds in the future.

Concluding Remarks

This paper has concentrated on some of the important structural design aspects of the Concorde. Within the limitations of one paper it is impossible to cover all the structural development of this adequately. The structure of the aircraft itself has not been described in great detail. In fact it has a fairly conventional appearance and a full description would be better included in another paper or perhaps better still it should be seen for itself. This can be done any day at the factories of Sud Aviation at Toulouse or British Aircraft Corporation at Bristol. It is hoped however, that this paper will have provided some insight into the facilities and techniques which are being applied to the structural design of the Concorde in an effort to insure safe and practical, as well as a comfortable and economic, means of supersonic transportation in the 1970's.