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Techniques for Tests on Structures Subject to Kinetic Heating

by

P. Talbot

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TECHNIQUES FOR TESTS ON STRUCTURES SUBJECT TO KINETIC HEATING

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P. Talbot

SUMMARY

The main requirements of aircraft static tests are stated. The extent and effects of kinetic heating are described, and various methods of simulating and controlling heating in the laboratory are discussed. The further problem of kinetic cooling leads to the technique of heating and cooling by convection where the heating rate is low. Measurements and test observations are discussed with reference to the safety of the specimen. The application of simulated kinetic heating to static tests is described and reference is made to the special difficulties and needs of the hot fatigue test.

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CONTENTS

1 INTRODUCTION

Before discussing in detail the techniques of hot testing, let us consider in a general way why large scale tests are done and what information they are expected to give.

It is obviously essential that aircraft structures must be able to carry their design loads, yet the constant need to minimise weight means that airframes can only have reserves of strength which are relatively small compared with those of many other kinds of structure. Furthermore, many different loading conditions have to be considered and the resulting structure tends to be complicated and difficult to analyse. Calculations alone cannot generally provide sufficient confidence in the airworthiness of the design to satisfy the Certifying Authority; it is necessary to demonstrate practically that the structure can safely carry the aerodynamic and inertia loads for which it was designed. The purpose of the strength test is to provide this demonstration under safe and controlled conditions.

The various flight loading cases of the aircraft are studied and a series of tests is defined to include the most critical loading condition of each part of the structure. There are normally two main requirements to be satisfied in each test case and these can be stated briefly as follows. Up to a load $1\frac{1}{8}$ times the flight (limit) load there shall be no significant permanent set, and up to a load $1\frac{1}{2}$ times the limit load there shall be no failure which might prevent safe flight. These are referred to as the Proof and Ultimate conditions; $1\frac{1}{8}$ and $1\frac{1}{2}$ are respectively the Proof and Ultimate Factors. For convenience, test loads are usually quoted as a percentage of the Ultimate Load, so that Limit, Proof and Ultimate becomes $66\frac{2}{3}\%$, 75% and 100% respectively. To take every test case up to the ultimate condition would cost a great deal in time and money, since one would expect to inflict some permanent damage at the highest load levels. The usual procedure is to try to cover all cases initially to perhaps 90%. After that the most significant case will be selected and the specimen loaded to Ultimate or possibly to failure.

What has been said so far applies to structures working at near-ambient temperatures, but when the speed of aircraft exceeds the speed of sound, a further design condition becomes important. That is, heating of the structure surface caused by its passage through the air; namely, kinetic heating. In this Report we shall consider briefly how heat transfer takes place in flight and in more detail how it can be simulated in ground tests. Various methods of control are described, and also the needs of both static and fatigue tests. The influence of time-dependent test conditions on data-recording methods is discussed alongside the requirement for automatic monitoring of the test.

2 KINETIC HEATING

The heat source is in the boundary layer, where the kinetic energy of air, initially moving rapidly relative to the aircraft surface, is converted to heat as its relative velocity is rapidly reduced. The exact nature of the heating process is more the concern of the aerodynamicist than the structural engineer, but some understanding of the effects, if not of the causes, is essential if both design and tests are to be on a proper foundation.

The heat transfer process is conveniently described by the equation

$$q = h\,(T_{aw} - T_s)$$

where q = rate of heat transfer to the aircraft surface,

h = heat transfer coefficient,

T_{aw} = adiabatic wall temperature,

and T_s = surface temperature.

The adiabatic wall temperature is simply a convenient forcing temperature and represents the maximum temperature which the boundary layer would achieve under zero heat transfer conditions. It exceeds local ambient temperature by an amount which is proportional to the square of Mach number and has one of two possible values, differing by only a few per cent, according to whether the local boundary layer is laminar or turbulent. The heat transfer coefficient, which is a measure of the efficiency of the heat transfer process through and from the boundary layer, is primarily a function of Reynolds number, but also depends on body shape and to some degree on surface temperature. The heat transfer coefficient also has two possible values depending on whether the local boundary layer is laminar or turbulent, but in this case the turbulent value may be as much as an order of magnitude greater than the laminar value. It is clear, therefore, that while the forcing temperature is closely related to Mach number, the influence of the heat transfer coefficient is such that the rate of heat transfer, or the time taken for a particular thermal mass to reach the forcing temperature, is highly variable. At 50000 ft the transfer coefficient is an order of magnitude less than at sea level, while at 100 000 ft it is a further order of magnitude less. The great significance of this variation has been stressed by Taylor[1].

At this point let us consider the degree of heating which this effect produces. It is important to note that the forcing temperature is linked to Mach number, so that the rate at which it rises depends on the rate of acceleration of the aircraft. While the forcing temperature is increasing, so too is the aircraft surface temperature as it takes heat from the boundary layer. The heating rate which is implied by the difference between the final forcing temperature and the initial surface temperature combined with the final value of transfer coefficient is sometimes quoted. Such a heating rate could only be obtained if the surface were cooled to maintain its initial temperature and the figures are generally misleading. It is more realistic to consider the actual acceleration rate of the aircraft and the rate of change of forcing temperature which this implies. This may be taken as the most rapid possible rate of increase of surface temperature, and the corresponding maximum heating rate is obtained by multiplying by the appropriate value of thermal mass per unit area of surface.

The range of heating rates with which we are concerned is shown, for various classes of aircraft, in Fig.1.

Kinetic heating produces the following important effects:-

(a) Material mechanical properties are reduced, both as a function of temperature and as a result of prolonged exposure to temperature. Also, if the structure temperature remains high under sustained flight load it is possible that significant creep of the material may occur during the life of the aircraft.

(b) The flow of heat into the structure produces non-uniform temperature distributions which are aggravated by uneven mass distribution. Thermal expansion of every element is constrained by neighbouring elements at different temperature, or by different material. This constraint implies a straining of the element which produces a corresponding level of thermal stress.

These effects must be considered both in the design and test of the structure. It is relatively easy in the design stage to allow for a reduction in material properties due to temperature and to do a test at steady elevated temperature by way of demonstration. The transient problem, however, is very difficult both from the theoretical and testing points of view. The major problem in hot strength testing is the correct reproduction of these transient thermal effects.

Before discussing the question of heating simulation it must be made clear that for convenience the range of problems considered has been arbitrarily limited. We shall be concerned only with those cases where it is practicable to adjust continuously either surface temperature or rate of heat input to some required value.

3 CONTROLLED HEATING

The effect of a particular high speed flight plan is to produce a related history of temperature change in the structure. Simulation of the flight heating effects for test purposes can be by whichever mode (convection, conduction or radiation) is most convenient provided only that it produces the correct temperature changes in the structure. For many years it has been generally agreed that radiation is the most versatile method[2], though there may be particular cases which are best treated otherwise. We shall return to this point, but for the moment will discuss only the radiant method.

Following the initial production of a powerful electrical radiant heater in the U.S.A., a range of similar heaters is now widely available, a typical one being rated at 1 kW at 230 volts and 3 kW if overrun at 440 volts. These can be assembled in front of polished metal reflectors to give more than adequate heating intensities (up to 100 kW/ft^2 for short times). The heat output is controlled by variation of the supply voltage, and the response time is only a fraction of a second. Using such a heat source, there remains the problem of matching the heater output to the requirement of a particular flight plan.

Since the heating experience of a point on the surface is defined for a particular flight by surface temperature as a function of time, surface temperature seems to be the obvious variable to control. The block diagram of a temperature control system is shown in Fig.2. However, surface temperature is not defined by the flight plan as are the forcing temperature and transfer coefficient. The surface temperature depends on the local thermal mass and the way in which heat is able to flow within the structure. If surface temperature is to be used as the basis of a control system then it must first be calculated, and this involves making assumptions about the thermal properties of the structure. It was to overcome this difficulty that the "full-equation" method of control[3] was devised (Fig.3).

In this method a computer is used to solve continuously the heat transfer equation. The values of adiabatic wall temperature and heat transfer coefficient are pre-calculated as functions of time for the flight to be simulated and are both fed to the computer in real time. A measured value of the specimen surface temperature is also taken to the computer, and it is thus able to calculate at any instant the required local heating rate. At the same time, the applied radiant heating rate is measured by a radiometer[4]. The difference between heat required and heat supplied is continuously minimised by a servomechanism controlling the heater supply voltage. No assumption of specimen thermal properties is required, for the combination of computer and specimen simulate the flight as accurately as the governing equation represents the physical problem.

In principle, areas of the specimen surface having different heat requirements require independent control by an individual control channel. Every such channel receives the input data, measures surface temperature, calculates the heat required, measures the heat supplied and accordingly regulates the voltage applied to its own group of heaters. Because the local heat requirement depends on local surface temperature and hence on local thermal mass, large numbers of channels are required. In practice, minor irregularities such as stringers are neglected and the area is divided into discrete areas of approximately uniform thermal mass and heat requirement. The heaters covering each area are then controlled independently, and the required temperature measurement is made at a typical central point in each area. Multichannel operation is effected conveniently by a single time-shared digital computer, as described by Horton[3] and Taylor[5].

It should be noted that both the full-equation and temperature control methods have two sources of error in common. First, both rely on the accuracy of prediction of the aerodynamic quantities. Second, both break the specimen down into discrete areas which are controlled according to the requirement of a typical point. The heat input over each area is uniform in the test, whereas in flight it will be determined by local surface temperature which depends on local thermal mass. It is not practicable to divide the structure into very small areas for individual treatment because of the large numbers of control channels which would be needed and the limit set by the physical dimensions of the heaters.

It should not be assumed that temperature control is always inferior, for there are other important considerations. The aircraft in flight loses heat by radiation while gaining heat by convection but during ground simulation the reverse is the case. Because of this it is necessary to adjust the calculated value of "heat required", more heat being needed during simulation than the amount supplied by convection in flight. The table of Fig.4 indicates the various heat flows involved and the origin of the required correction term.

This aspect of simulation has been investigated by Ham at R.A.E. Although the literature refers to the possibility of continuous calculation of the required test heating rate this has not been done in practice, probably because it involves calculating the fourth power of temperature. What has been done is to calculate a correction on the basis of a calculated skin temperature. The correction is obtained in the form of a required increment of heating rate as a function of time. This heating rate, divided by the current value of transfer coefficient, gives a temperature increment which is added to the current value of adiabatic wall temperature before it is fed to the computer. The computer, in multiplying the transfer coefficient by the augmented adiabatic wall temperature, thus automatically adds to the flight convective heating rate the correction previously determined. By this means a calculated correction is applied without complication of the test equipment. The correction is about 5°C in 120°C and 15°C in 230°C. It is obviously not negligible, and must be taken into account if the full equation method is to realise its potential superiority.

A point in favour of the temperature control method is that it requires no such correction for non-representative losses during the test. It is clearly only necessary to take account of the flight condition when calculating the flight temperature. After that, the control system ensures that this temperature history is reproduced during the test regardless of the various heat transfer processes involved.

In some cases it may be possible to base a test on measurements taken in flight. If this is so, then it is simpler to use skin temperature as the controlled quantity, for this is measured directly in flight. It would be pointless to infer values of heat transfer coefficient from the flight measurements and then use these in a test controlled by the full equation method.

The most usual criticism of temperature control is the one already mentioned, that is of the errors arising in the calculation of transient temperatures through complicated structures. The general problem is indeed difficult. The most obvious snag is the highly variable conductance of structural joints[6], but heat transfer by other modes across open spaces inside the structure is also a significant problem[7]. For example, in a ground test only half of the temperature rise in a particular deep web is caused by conduction, most of the remainder being due to convection. However, it is evident that for temperature control one is really only interested in calculating the temperature at the control points. In many practical cases the control point can be chosen on a piece of free skin, where it is not affected by attached heavy masses. Work by Capey (R.A.E.) has shown that in such cases temperatures can be calculated with adequate accuracy for control purposes. The error involved is of the same order as those which are likely to arise from other causes with the full equation method.

Of course, there remain the cases where even control point calculations are not feasible; for instance on honeycomb structures. Such cases are ideal applications for the full equation method of control. It is then used in the true sense as a simulator and the specimen provides its own answer to the heat transfer problem.

To sum up, it seems most sensible to regard the two methods as complementary. The full equation method is obviously superior philosophically. However it requires careful use and accurate corrections of the kind described above if its potential is to be achieved. The temperature control method[8] is simpler, and therefore preferable for large scale use where there is sufficient confidence in calculated temperatures at the control points or where tests are based on flight measurements.

4 CONTROLLED COOLING

So far we have considered only heat flowing from the boundary layer into the structure. However, where the structure is hotter than the boundary layer, heat will flow in the reverse direction causing cooling of the structure surface. This condition arises on deceleration after high speed flight and is aggravated by a simultaneous loss of altitude which increases the heat transfer coefficient. It is quite possible for the resulting cooling rate to exceed 1 kW/ft^2, and, since the thermal stresses produced are of the opposite sign to those produced by kinetic heating, a new design condition is imposed. Testing for this condition requires a cooling source and a means of control. The only practicable way of achieving the required cooling rate on a large scale is by forced convection using air, though there are limitations to the magnitude of heat transfer coefficient possible with reasonable air speeds, and control of the cooling rate is difficult. A method has been developed at R.A.E., Farnborough, in which control is exercised by adding radiant heat to a cooling effect which always exceeds the actual requirement. The cooling is produced by blowing air which has been cooled to about $-100^{\circ}C$ by the addition of liquid nitrogen over the specimen surface. The cooling stream is contained by the radiant heater reflectors which are extended fore and aft to form ducts from the fans and to a convenient exhaust point. When used with a full equation heat control system, operation is as follows. The radiometer is equally affected by heat transfer due to radiation or convection, so the difference between the convective cooling and radiant heating is always measured. This is compared with the calculated negative heat required, and the difference is minimised by control of the radiant heaters. It is possible to use the same method with a skin temperature control system.

The cooling technique involves the use of additional equipment, fans etc., but this extra cost is comparatively trivial. Much more important, especially for repeated tests, is the cost of liquid nitrogen. In the U.K., where it costs about one shilling and sixpence a gallon, it is about an order of magnitude more expensive than the equivalent quantity of heat. The efficient production and application of cooling is clearly the dominant factor in the economics of thermal cycling tests.

5 CONVECTIVE HEATING AND COOLING

It has been said that radiation is generally favoured for kinetic heating simulation, but that it is necessary to use convection for cooling purposes. The heating rate of the M2.2 transport aircraft is only about $\frac{1}{3}$ kW/ft^2, however, and because of this it is possible to both heat and cool at the appropriate rate by forced convection. The idea is attractive because it eliminates the radiant heaters and reduces the complication of the control system.

For any prescribed flight plan, the heat requirement of a particular point on the specimen surface is a unique function of time. However, the form of the governing equation, where the heating rate is given as the product of a transfer coefficient and a temperature difference, indicates a considerable freedom of choice of forcing temperature or transfer coefficient. Thus, the heating rate can be made correct while using an arbitrary transfer coefficient by adjustment of the forcing temperature. In fact a number of small-scale heating tests have been run on this basis at various places. However, although this is acceptable for the hypothetical point on the surface, it is not good enough for the specimen as a whole because heat transfer rate also depends on local surface temperature and thus varies over the surface according to thermal mass distribution. Hence, to obtain the correct heating rate distribution with an arbitrary transfer coefficient, the forcing temperature would need to be adjusted everywhere to suit local thermal mass. This is quite impracticable, so the method is only strictly applicable where the whole specimen is of uniform thickness or the errors introduced by thickness changes are acceptably small.

When the heat transfer coefficient in flight is fairly small, however, it is possible to achieve the same value on the ground by blowing air at ambient pressure. The change in density balances the change in speed so that the high altitude supersonic cruise values are obtained by air speeds of about 150 ft/sec. Under these conditions, if the bulk air temperature is made equal to the forcing temperature in flight, the correct distribution of heat transfer rate is obtained. The air stream is made to flow over the specimen by an enveloping surface a few inches away, and vertical walls running fore and aft keep the flow straight and allow the pressure drop over varying chordwise lengths to be balanced. The mean height of the ducts so formed determines the mass flow required to produce the speed necessary to give the required transfer coefficient. For economy the mass flow is the minimum consistent with an acceptable change in temperature of the air while traversing the specimen. By increasing the duct depth with distance from entry, it is theoretically possible to reproduce the flight variation of transfer coefficient with Reynolds number due to streamwise length. The required variation in depth is from about 6 inches at a point 5 feet from the leading edge to about 9 inches at a point 100 feet further aft. The ducts are continued aft of the specimen and formed into a return circuit containing the fan and heating and cooling apparatus; the whole arrangement resembles a closely-fitting wind tunnel. Fig.5 shows a schematic layout of such a system. The object of the return circuit is to improve thermal efficiency, since only a small temperature change is acceptable in the air stream to minimise longitudinal errors in heating rate. Simulation of a particular flight is achieved by controlling the air temperature and speed as functions of time. It is not practicable to take account of sudden variations in transfer coefficient, such as those caused by transition from laminar to turbulent flow, but within this limitation and those imposed by acceptable air speeds and fan power the method seems promising. The method described is being studied at the R.A.E. Farnborough for possible application to Concord fatigue tests. Theoretical investigations by Finklaire (R.A.E.) have led to various refinements of the basic system for which he made the original suggestion.

6 TEST OBSERVATIONS

The bare minimum requirements of an approval test can be met without taking any observations during the test other than those needed to show that the required test condition is being applied. No one has ever suggested that this should be the practice for a number of good reasons. First, the measurement and recording of deflection, strain, temperature etc. are useful to the designer in confirming his assumptions and calculations. Second, by confirming or modifying theory, the measurements improve design practice for subsequent aircraft. Third, observations of the specimen's behaviour as load is increased can lead to a test being stopped before it is heavily and needlessly damaged. Fourth, if the specimen does break, previous measurements are sometimes useful in locating the source of weakness and explaining the failure. In this sense, observations are taken to include looking and listening around the specimen and also noting the magnitude and rate of change of a limited number of deflections and strains. The reliability and accuracy of test measurements is of fundamental importance. The difficulty is not so much the accuracy required in a single measurement as that of making a large number of such measurements under conditions which are far from ideal.

With these limitations in mind, it is relatively easy to take measurements at each level of load during a static test at room temperature. Deflections can be read directly from dial gauges or hanging scales and there is plenty of time to read numbers of strain gauges in turn using simple equipment. In transient heating tests, however, the situation is quite different. Strain, deflection and temperature are all changing with time and one is forced to record a large amount of data automatically. Furthermore, the measurements themselves are physically more difficult and prone to error. Deflections can no longer be measured directly, some kind of transducer is necessary to give a remote indication. These are usually electrical and require calibration according to the measuring equipment which is used.

The measurement of transient temperature requires understanding and care. The measuring device must not perturb the specimen temperature merely by its presence, and it must make intimate contact so that its temperature is always that of the specimen. Neither of these conditions is easy to satisfy but probably the best solution is to use a fine-wire thermocouple in which each wire is separately welded to the specimen and lies close to the surface for a short distance from the weld[9].

The measurement of transient thermal strain, especially of skins exposed to radiant heat, is even more difficult. There are a number of types of high temperature strain gauges available but many are unsuitable for transient measurements. Their construction and method of attachment inevitably produce significant temperature differences between the gauge and the specimen material, and the presence of the gauge considerably perturbs the local temperature and strain pattern. The application of a temperature correction under these conditions is of doubtful value. The difficulties are aggravated by the magnitude of the scatter in the correction term for nominally identical gauges which can be serious even for measurements at steady temperatures.

This particular difficulty has been overcome in selected-melt gauges by careful heat-treatment of the gauge material. As a result its properties are modified so that resistance changes not due to thermal strain largely cancel out over a particular temperature range when the gauge is fixed to a particular material. The problems of minimising the thermal mass of the gauge and ensuring good thermal contact with the specimen remain. However, this appears to be the most profitable line of development and is the one now being pursued in the U.K.

In addition to taking measurements which show how the specimen is responding to the test condition, it is also necessary to take measurements to confirm that the correct test condition is being applied. In a transient test the latter operation should certainly be performed automatically. A continuous comparison should be made between measured values of load and temperature and the corresponding required values, and differences greater than a prescribed maximum should automatically stop the test. There is no question of human operators doing this necessary monitoring. Even if a sufficient number of observers could be found and they could remain always sufficiently alert, they would have scant time for decision-making if faced with a crisis. It is necessary for this "auto-stop" system to be separate from the control system, and it is accepted that it will be of the same order of complexity. Although intended primarily to ensure that the loading system is working properly, it is worth considering extending its function to stopping the test if the specimen is reacting badly to the correct test conditions. Questions such as which parameters should be measured by the "auto-stop" and their limits of acceptability are themselves very interesting but are beyond the scope of this Report.

When all of the data from a test has been recorded there remains the problem of assessing its significance. This involves processing by a computer and a print-out or plot of the results followed by a good deal of examination by qualified staff. There is much to be gained by transfer, as far as possible, of the assessment function to the computer. This would mean defining criteria with which the test data could be compared and only data which differed significantly from predictions would be printed out. This deliberate selection of data is especially important during fatigue tests, where otherwise a large amount of unnecessary data is likely to accumulate.

An extension of the ideas put forward so far is to use a digital computer on-line during the test. This would control and check the test condition as well as process the data for immediate examination. It would also enable data which requires prior processing to be surveyed as part of the "auto-stop" facility. The "auto-stop" device and the use of on-line computation are now being studied in the U.K.

7 STATIC TESTS

The classical method of doing a static test is to apply a steady load, which is periodically increased by a fixed amount, taking observations and measurements at each load level. Steady flight loads are applied directly and the dynamic effects of a manoeuvre case are accounted for by the steady application of loads due to inertia forces. The whole problem is then essentially static, and it is assumed that the test result is independent of time.

However, for transient thermal loading there is nothing analogous to the concept of inertia forces to reduce the problem to the steady state. Unless the whole structure is at uniform temperature, it is evidently experiencing a flow of heat as shown by the temperature gradients. To reproduce the flight transient temperature condition it is necessary to perform a ground cycle in real time. The problem then is how to apply the mechanical loads due to the assumed manoeuvre in a way which properly combines the mechanical and thermal effects and also allows the severity of the case to be graded in a predetermined way. Given a sufficiently powerful and responsive loading system, there is no reason why the manoeuvre load should not be applied in real time. However, one would then be faced with the problem of phasing the loading and heating cycles so as to produce the required combination for test purposes. In practice the problem may be insuperable because the critical condition may occur at different times in different parts of the structure.

At the other end of the scale, a slowly reacting load system could be used to apply a particular level of load to the unheated specimen. With this constant level of mechanical load maintained, the thermal cycle could then be applied in real time. A particular test case would be covered by a number of such tests, the level of load being raised incrementally from test to test while the thermal cycle remained the same. The advantage of this method is the separation of mechanical and thermal effects. On the other hand, it does require a good deal of testing time and in some cases frequent thermal cycling of the specimen may be undesirable. It is this technique which has been used at the R.A.E. Farnborough, using a loading system designed for room temperature static tests on the principles described by Walker[10].

An alternative is to use a fast loading system to vary the load in an arbitrary way during the thermal cycle. This also would allow one to distinguish between thermal and mechanical effects in the strain gauge records, and it may also reduce testing time. However, the critical moments for various parts of the structure due to thermal effects will occur at different times. If the load is also varying with time the whole pattern of superposition is complicated and one cannot be sure that every worst combination has been achieved. In short, the varying load technique is likely to produce useful design information, but the steady load technique is necessary for strength approval testing.

It is worth noting, however, that in certain cooling cases, where the critical period is preceded by a fairly long heating and soaking phase, it is advantageous and may be necessary to be able to apply the required load quickly.

So far it has been assumed that the applied thermal cycle would be an actual flight case. However, as in the case of mechanical loading, it is necessary to increase the severity of the test condition by some numerical factor in order to establish the required level of confidence in the result. The magnitude of the factors which are applied to mechanical loads and the way in which they cover the uncertainties involved is a subject in itself. Once having chosen the factors, however, the implications for the structure will be fairly clear. The case of thermal loading, however, is more complicated, and the general objective of making the condition more severe can be approached in different ways. One could apply a factor on speed, on atmospheric temperature, on forcing temperature, on transfer coefficient, on temperature range or a combination of these. The difficulty is to define a factor which is philosophically acceptable, can be used in the design process and can be demonstrated by test.

No uniform philosophy has been adopted in tests done so far at the R.A.E. In one case the heating cycle corresponded to a uniform acceleration slightly higher than the aircraft capability. In another, a factor of $1\frac{1}{8}$ was applied to the range of forcing temperature calculated for the flight, the time scale being unchanged. Limitations are imposed on this technique by an upper temperature above which material mechanical properties are reduced unrepresentatively, while a lower limit is set by practical and financial implications of a high degree of cooling.

8 FATIGUE TESTS

The objects of a full scale fatigue test are to indicate those parts of the structure which are prone to failure and to demonstrate a satisfactory life under representative loading conditions. Such tests at room temperature have become commonplace, although problems remain both in their planning and interpretation. The general principle is that the loads due to landing, pressurisation and gusts can be applied during the test much faster than they occur in service. It is therefore possible to compress a number of lifetime's experience into a reasonably short testing time.

In the case of a supersonic transport aircraft there are additional complicating factors. A significant proportion of fatigue damage occurs as a result of thermal stresses, so these must be represented. Also, the flight time spent at high temperature produces some creep of the structure. Both the creep itself and its interaction with the fatigue process require to be taken into account. The conditions which must be represented therefore include a typical thermal cycle to produce the required levels of thermal stress during heating and cooling, and the required amount of creep during the cruise phase. This is an expensive complication, but a more fundamental difficulty is the compression of the time required by the tests in a way analogous to room temperature fatigue testing. Much thought is being given to this problem, which is discussed in detail by Van Leeuwen[11] who also comments on some of the solutions which have been proposed[12]. The difficulty is how to increase by a certain factor the damage caused by a single thermal cycle. Modifying the heating cycle produces different factors in different parts of the specimen depending on the local structure. Changing the cruise temperature level puts a factor on creep which must be compatible with the factor on thermal stress. In the absence of a simple factor applying to the whole specimen it is essential that the local test environment is accurately defined and that the local factor can be determined.

This Report is primarily concerned with the technique of producing the test environment. The fatigue test calls for application of a particular thermal cycle which involves heating, holding at temperature and cooling the structure. Any of the methods described are able in principle to do what is required, and the choice of the most suitable one in a particular case is not easy. There are many things to consider, some of which are unique to this kind of testing. First, the cycle must be applied many thousand times, so the equipment must be robust and long lived. Second, the equipment must be reliable both in the sense that it produces the required result and that it produces it on demand so that no testing time is needlessly lost. Third, its maintenance requirement must be low enough to fit in with the normal periodic shut-downs of the test for specimen inspection. Fourth, it must be flexible enough to allow reasonable variation of the thermal cycle if this becomes necessary possibly as a result of flight measurements on the prototype. Fifth, it must be as economical as possible, especially in the inherently expensive cooling phase.

Whatever method of heating and cooling is used, the total physical and intellectual effort required to carry out a large scale hot fatigue test is very much greater than has previously been expended on any other kind of aircraft strength testing. The achievement of supersonic flight brings to test engineers a challenge as great as any which it has offered over the whole range of aeronautical science.

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CLASS OF VEHICLE	HEAT INPUT kW/ft^2 ($\doteqdot W/cm^2$)
M 2·2 TRANSPORT AIRCRAFT	0·3
M 2·2 MILITARY STRIKE AIRCRAFT	1·5
M 3·0 RESEARCH AIRCRAFT	3·0
SOUNDING ROCKET (ASCENT)	5 → 15
GROUND-TO-AIR MISSILE	> 50
MISSILE RE-ENTRY HEAD	> 500

FIG 1 APPROXIMATE KINETIC HEATING RATE OF VARIOUS VEHICLES

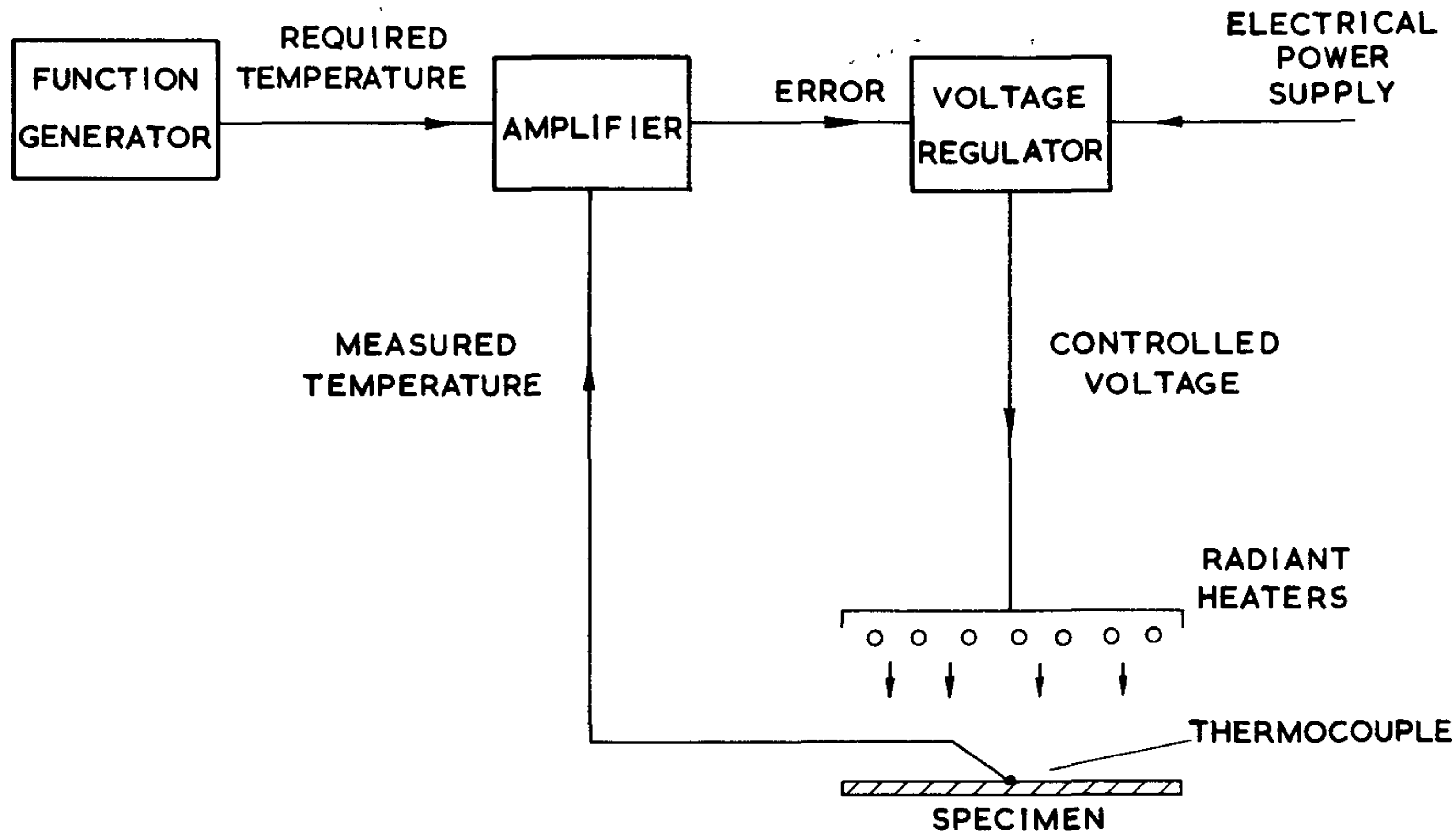

FIG 2 TEMPERATURE CONTROL SYSTEM

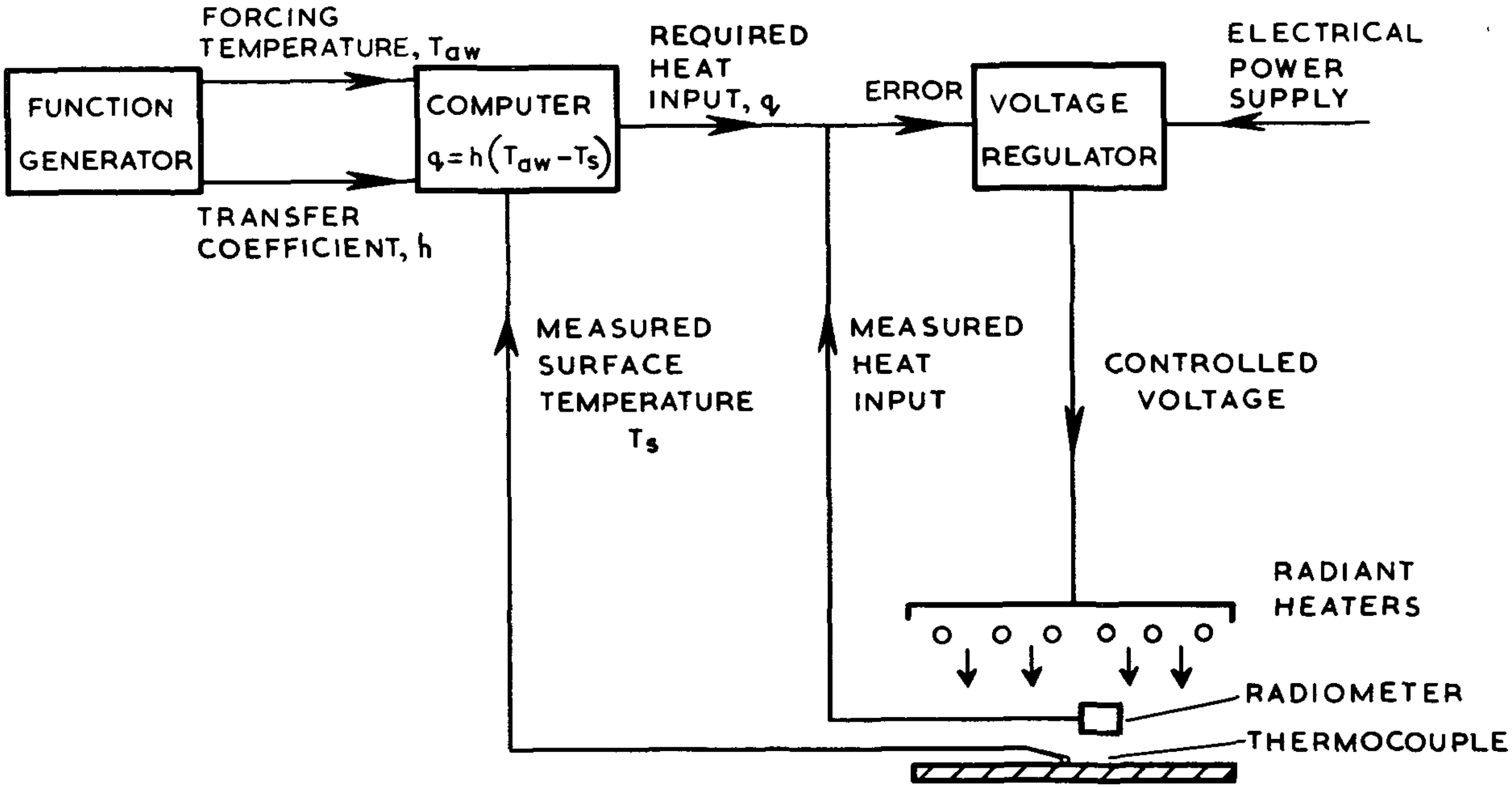

FIG. 3 FULL - EQUATION CONTROL SYSTEM

DIRECTION OF HEAT FLOW	MODE OF TRANSFER	
	FLIGHT	TEST
IN	CONVECTION (B) *	RADIATION FROM HEATERS (A)
OUT	RADIATION (D)	RADIATION CONVECTION (C)

* SOLAR RADIATION IS NEGLECTED

DURING A TEST A = B BY ARRANGEMENT;

FOR CORRECT NET HEAT FLOW, ONE MUST
ADD QUANTITY (C - D)

FIG 4 THE ORIGIN OF THE LOSS-CORRECTION TERM IN THE FULL EQUATION SYSTEM OF CONTROL.

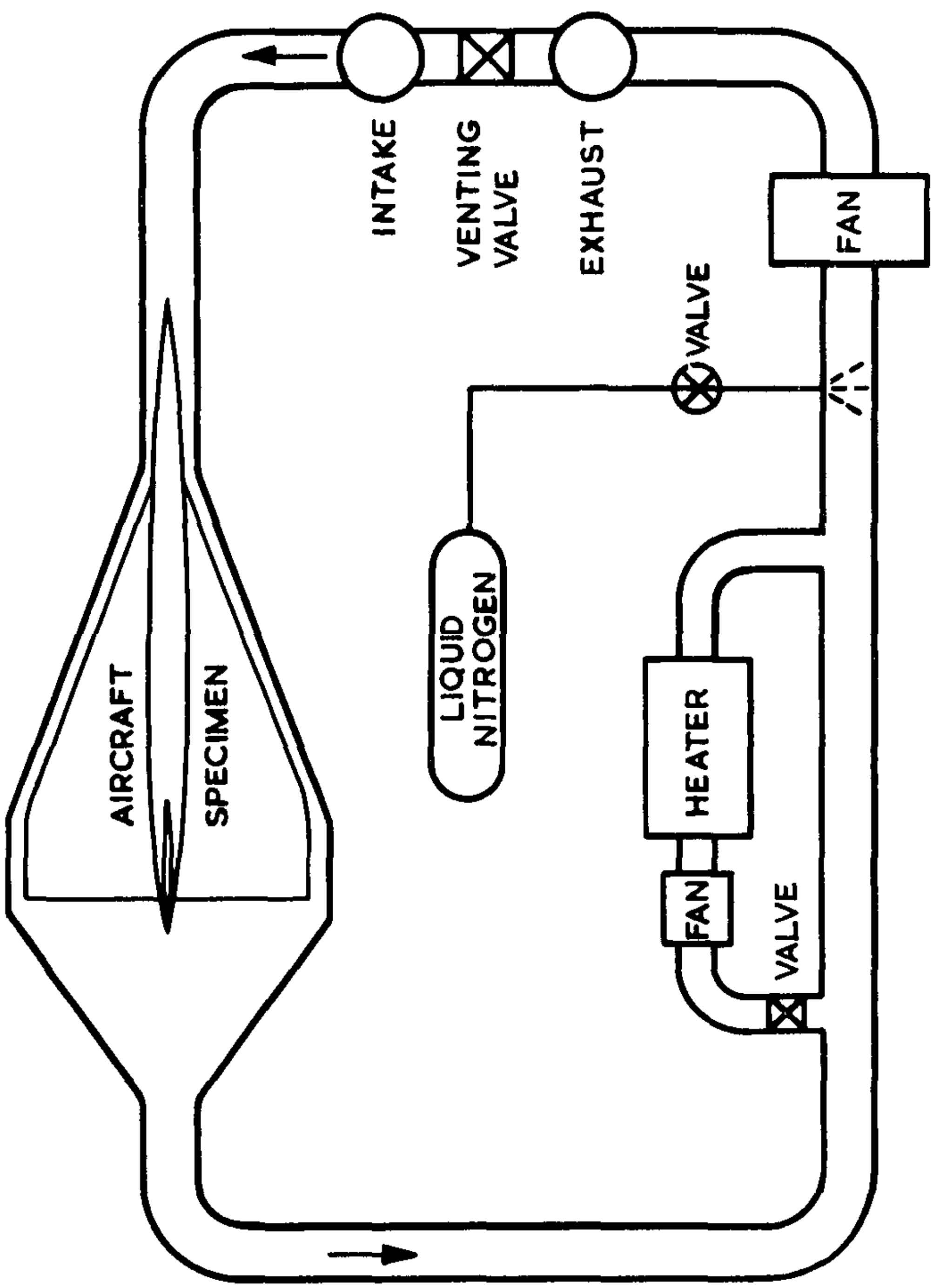

FIG 5 CONVECTIVE HEATING AND COOLING SYSTEM

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