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THE DEVELOPMENT OF FULL-SCALE FATIGUE TESTING

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ABSTRACT

Metals can fail through the repeated cyclic application of loads well below their tensile strength. This is of supreme importance in the design of air frames. Structures have to be designed so as to minimize any tendency there might be to accelerate fatigue and, once built, it must be demonstrated that they have adequate endurance against fatigue. Furthermore, should they fail, it must be in such a fashion that its onset will be noticed.

The effect of fatigue on a simple bar specimen shows the influence of repeated cyclic loading on useful tensile strength. The presence of a stress-raiser shows how this further reduces fatigue life. Procedures of local strengthening to counteract stress concentration are compared on details. The rate of fatigue crack propagation in single details shows the need for "fail-safe" structures.

The fatigue properties and fail-safe nature of large aircraft structural components are illustrated by tests on a section typical of a wing box and on a section of a pressure cabin. The need for program loading to simulate service loads is demonstrated in the wing box tests. The fail-safe nature of the pressure cabin structure is shown by starting artificial cracks.

Finally, the complex and sophisticated test equipment needed to insure the adequate testing of a large modern airliner is described in detail. This includes the highly sophisticated electronic gear which programs the fatigue loads as they will occur in service, the hydraulic equipment necessary to apply these loads and the machinery required to provide the power.

Aircraft structures are subjected to both static and fatigue tests. Static-loading tests, intended to demonstrate that the structures can withstand the maximum loads for which they were designed, are relatively easy to perform. Fatigue tests, however, involve rather more complicated procedures. They are intended to show that an economic life can be obtained under the combination of steady loads, oscillatory loads in flight and the loads imposed on takeoff and on loading; in consequence, the testing procedures can be complex. A reasonable forecast of service behavior can, in fact, arise from current structural fatigue testing procedures.

While fatigue as a phenomenon has been well known for over a hundred years, it is in the past twenty years that it has become a problem to the aircraft industry, and in this brief period there have been rapid developments in the technique of fatigue testing.

We began by testing details with typical stress raisers using several specimens in order to get some scatter results. In this work, and with the equipment then available, this testing was confined to the application of a steady load, usually 1 g, in conjunction with a chosen oscillatory load. When testing large structural components such as wings, we employed the same simple loading system, relying on the Palmgren-Miner hypothesis for the interpretation of the test results to allow for the effects of differing stress levels which occur in flight. The Palmgren-Miner hypothesis is that the damage done at any particular stress level is equal to n/N, where n = number of cycles at the stress level and N = the number of cycles to failure at that stress level. Thus:

$$\frac{n_1}{N_1} + \frac{n_2}{N_2} + \frac{n_3}{N_3} \cdots = 1$$

There is some controversy over the accuracy of this cumulative law. However, provided its limitations are appreciated, and in the absence of a better law, it is a valuable aid to the computation of fatigue lives.

A more accurate but more elaborate technique of testing large structural components is now adopted known as program-loading. With this technique the specimen is subjected to a series of differing loads covering the ground to air cycle, gusts of differing magnitude, etc., the number of cycles under each loading being chosen to accord with the statistical data appropriate to the aircraft. By this means one "program" may represent several hundred typical flights.

We have chosen some samples from the work we have done which we hope will be of general interest, and we conclude with a detailed description of the test gear which has been developed by Vickers-Armstrongs (Aircraft) Ltd., for the fatigue testing of the main structural specimen of the VC 10.

FATIGUE TESTS ON DETAILS

The effect of a simple stress raiser is illustrated in Fig. 1. Here is shown the severe reduction in fatigue life caused by drilling one hole diametrically in a bar specimen. There is no load input at the hole and no fretting. The geometric stress concentration factor is 3.16. Using the conventional notation for notch sensitivity,

$$q = \frac{\frac{f_u}{f_n} - 1}{k_t - 1}$$

we get $q = 0.7$ at an endurance limit of 10^8 cycles.

This simple example has been chosen for two reasons: firstly, with an un-notched fatigue limit (at 10^8 cycles) of 9 tons/sq in. and a notched fatigue limit of 3.5 tons/sq in. at the same number of cycles, the notch sensitivity suggests that the practical results obtained might be a little better than the design figures would indicate. Secondly, the fatigue limit for the notched specimen (at 10^8 cycles) is quite close to the loading found to be typical for aircraft structural joints in aluminum alloys (R.Ae.S. Data Sheets—Fatigue) at the same number of cycles and thus illustrates the powerful influence the stress-concentration factor has in determining the life of a component or a structure.

Figure 2 shows the effect of local strengthening on the endurance for a simple countersunk hole. The improvement in endurance is disappointingly small, no

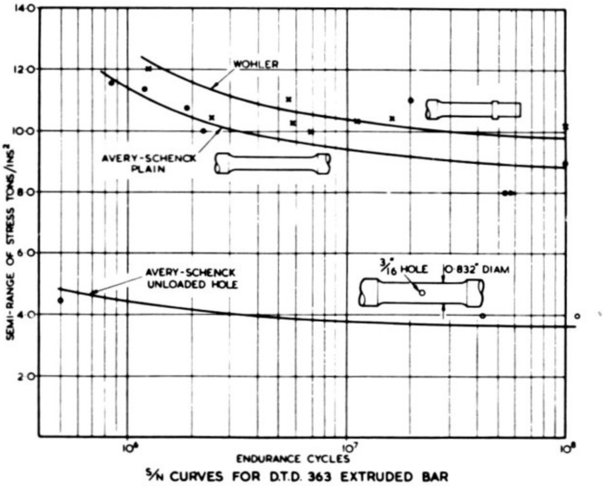


Fig. 1.

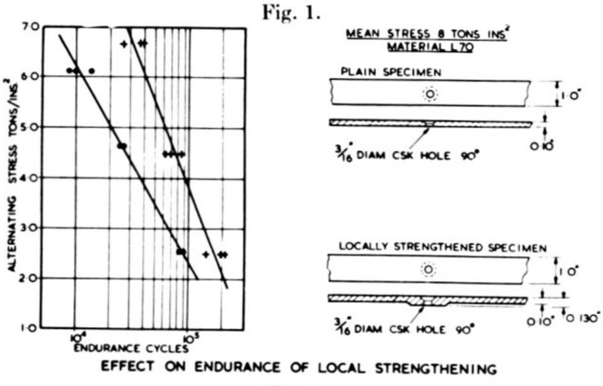


Fig. 2.

doubt due to the one-sided nature of the local strengthening dihtated by practical considerations. Again, the hole was unloaded тnd there was no fretting.

The effect of multiple stress raisers with unloaded holes bushed and unbushed is shown in Fig. 3. The tests were made to find the effect of bushing the holes, following the work of Fisher and Winkworth at the Royal Aircraft Establishment. The 84 test specimens were machined from six production extrusions and every group of six specimens contained one from each of the six extrusions. Bushes were in steel with an interference fit of 0.0021 — 0.0007 in.

The improvement in endurance obtained by the use of interference fit bushes is dramatic. While these tests are with unloaded holes and no fretting, other work with loaded holes shows an even greater relative improvement in mitigating stress raising effects. A point worth noting in Fig. 3 is the remarkably small scatter that we obtained.

The material used for the tests of Fig. 3 was DTD 363A, and part of this program was repeated on material to L65 specification. The results of this series of tests are given in Fig. 4*a*. Only bushed specimens were used and for ease of comparison the curve for DTD 363A bushed specimens in shown dotted. The L65 surve is drawn strictly through the log mean of the results and its shape is

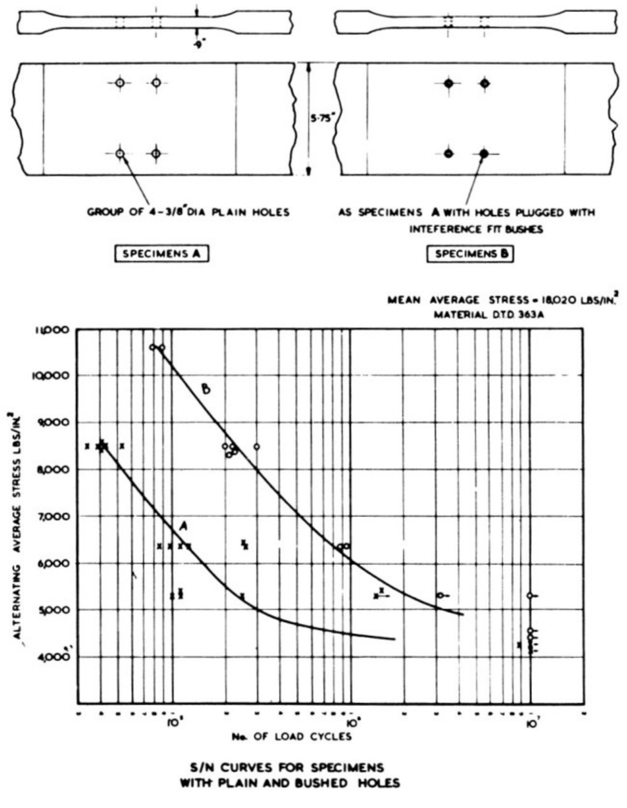


Fig. 3.

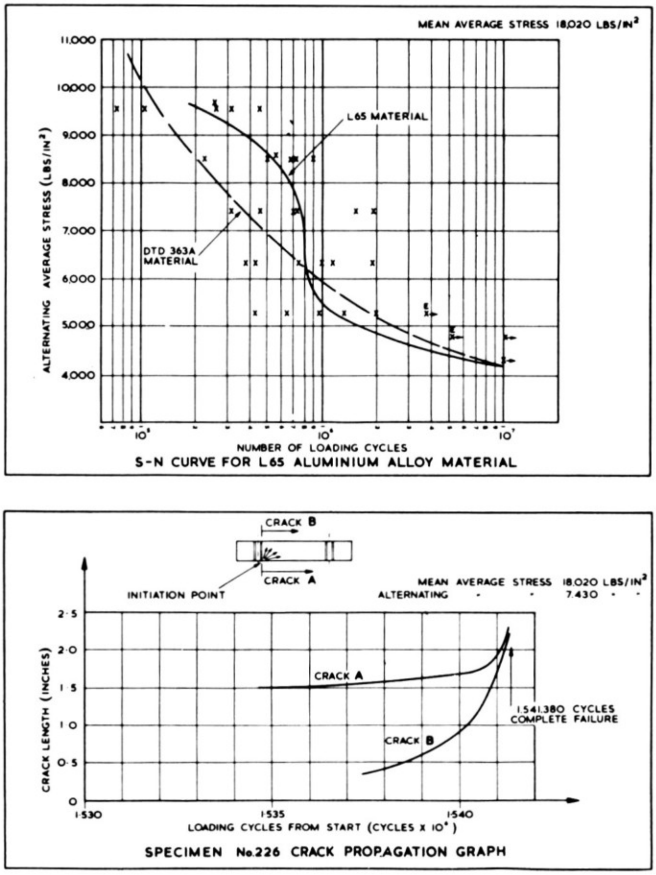


Fig. 4.

untypical. However, the scatter obtained on this series of tests was far greater than that obtained on the previous series in 363A. The reason for this difference is a matter for conjecture.

It was found possible to observe the growth of some fatigue cracks during these tests and one L65 specimen was observed completely. The results are shown in Fig. 4*b*. Attention is drawn to the short interval between the first appearance of a crack detectable under laboratory conditions and failure of the specimen.

FATIGUE TESTING OF LARGE STRUCTURAL COMPONENTS

A diagram of a fairly large wing test box and the manner of loading is shown in Fig. 5. Although not indicated in the diagram, the distribution of load was such as to give combined bending and torsion to the specimen. The box, which was 27 ft long, is representative of our current wing design employing machined planks with integral stringers and multishear webs—in this case three.

The material for the wing planks was DTD 5020 for the upper surface and 24ST4 for the lower surface. The two outer portions of the specimen formed fuel tanks. The tanks were filled to capacity during the tests and pressurized with air to 1 psi. The specimen had pump cutouts, access holes, etc., and was in detail fully representative of a typical finished wing structure except for the absence of chordwise joints.

Program loading was used for the tests and Fig. 6 shows diagrammatically the loading sequence. It will be seen that one program represents 460 typical flights. Each flight was in this test equivalent to 1½ hr duration.

The first indication of fatigue occurred at 128 programs (58,880 flights or 88,320 hr) by a small fuel weep at a rivet. This in turn led us to the detection (under laboratory conditions) of a very small fatigue crack at the rivet hole. There had been speculation beforehand whether a very small fatigue crack in the plating of an integral tank would be accompanied by a fuel weep, with a general consensus of opinion that the flexible nature of sealing compound would bridge a minute gap. However, we had fuel weepage and it enabled us to observe crack growth from its early development. The crack growth is shown plotted in Fig. 7*a*. We carried on with our testing for a further 21 programs, interposing occasional proof loads. When the crack developed sufficiently to cause serious fuel leakage we drained the specimen of fuel. Finally, after 154 program loadings, we loaded the specimen to failure which occurred at 76.5 percent of design ultimate.

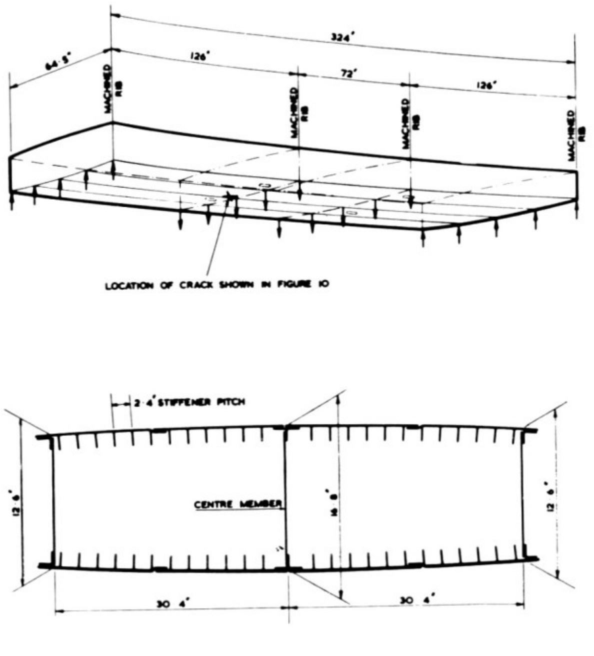


DIAGRAM OF WING TEST BOX
AND LOADING SYSTEM

Fig. 5.

A history of the test program is given in Fig. 7*b*. A close study of this history, in conjunction with the crack growth (Figs. 7*a* and 7*b*) is a fascinating exercise. It will be seen from the photograph included in Fig. 7*b* that the fatigue crack was readily visible (without dismantling) by the time it had reached relatively modest proportions. This feature, taken in conjunction with the slow rate of crack propagation, insures that no undue burden is placed on those responsible for inspection.

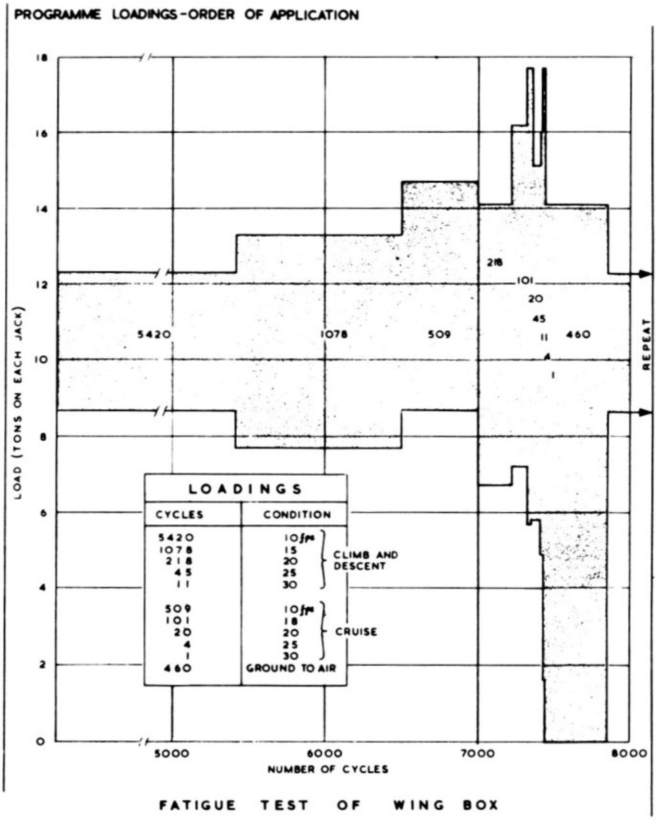


Fig. 6.

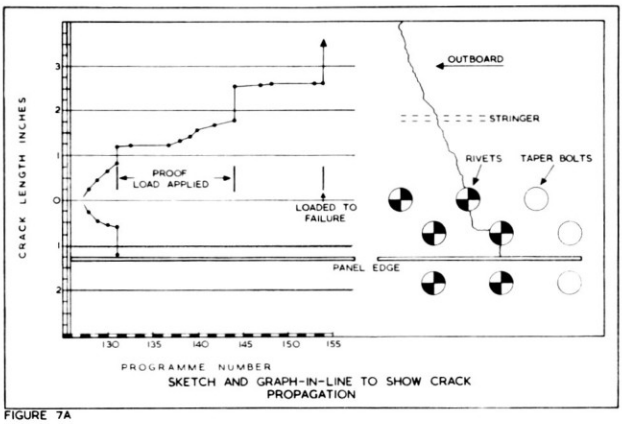


FIGURE 7A

PROGRAMME	OBSERVATIONS
100	MAJOR INSPECTION.
128	CRACK NOTICED. SLIGHT FUEL SEEPAGE.
131	CRACK ELONGATED TO 1¼". FUEL EMPTIED FROM THAT END.
PROOF LOAD (25 TONS)	CRACK EXTENDED TO 2¼" AND TO NEAR EDGE OF PANEL.
NORMAL PROGRAMME 144	CRACK ELONGATED TO 3"
PROOF LOAD (25 TONS)	CRACK EXTENDED TO 3¼".
NORMAL PROGRAMME 154	CRACK EXTENDED TO 3½".

LOADED TO FAILURE :- 28·7 TONS.

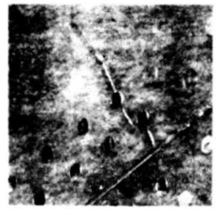


FIGURE 7B

Fig. 7.

PRESSURE CABIN FATIGUE TESTS

A test specimen, representative of the forward 37 ft of a pressure cabin, is shown in Fig. 8. The material of the pressure skin was L72. We used two such specimens for fatigue and fail-safe tests. One had already been tested in the Stratosphere Chamber for 29,589 reversals of pressure (Table 1).

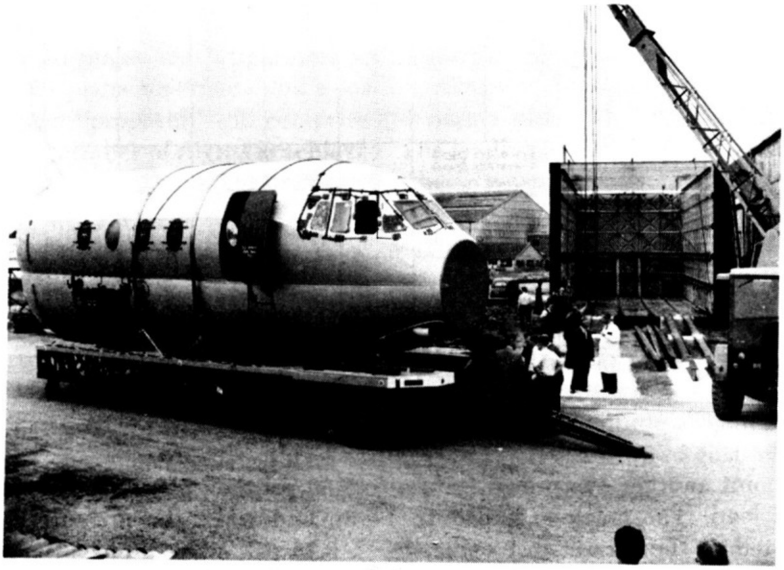
Table 1

Cycles	Cabin temperature, °C	Stratosphere chamber temperature, °C	Pressure range
Proof load, 0–8⅔ psi	+45	+45	
200	+25	−25	0–6½–0
200	+26	−57	½–7½–½
8,500	+26	−26	½–7½–½
1,900	+26	−56	½–7½–½
18,789	+18	+18	½–7½–½

Time for 1 cycle = 1 min 35 sec.

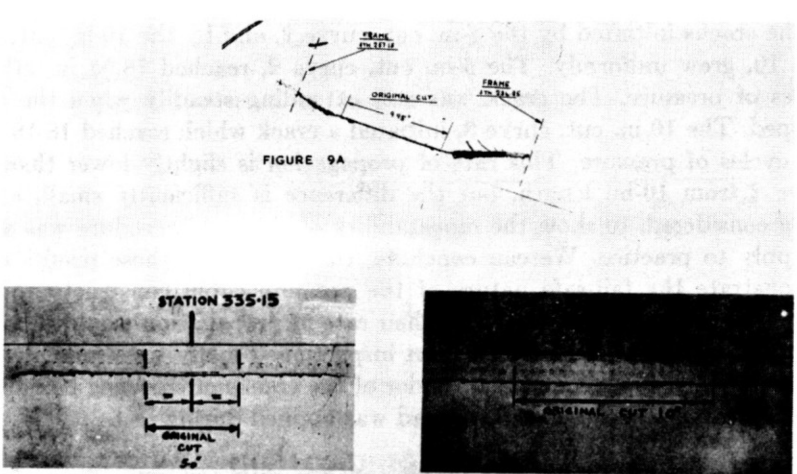
The integrity of the windows and the de-icing system was checked by the variations in temperature.

We continued cyclic loading in the water tank and suspended this testing at approximately 60,000 cycles in order to conduct a series of fail-safe tests. In three of these we simulated a substantial initial crack by saw cuts. These are illustrated in Fig. 9. The cuts were made at a longitudinal skin lap joint along one of the three lines of rivets. The cut of Fig. 9*a* was made 9.95 in. long and positioned midway between two frames. The 5-in. and 10-in. cuts of Figs. 9*b* and 9*c* were made at frames. Both the plating and the frame were cut, the frame being completely severed at this region. The propagation of the cracks with number of pressure cycles is shown in Fig. 10. The discontinuity in curve 1, Fig. 10, beginning at 240 cycles, requires an explanation. The crack length was measured every 10 cycles and at 250 cycles the length increased from 11.2 to 11.5 in. This was due to an over-pressurization to 7.75 psi which was corrected by the time 260 cycles had been reached, and the small horizontal portion of the curve at 11.7 in. shows the crack did not extend till 270 cycles were reached. At



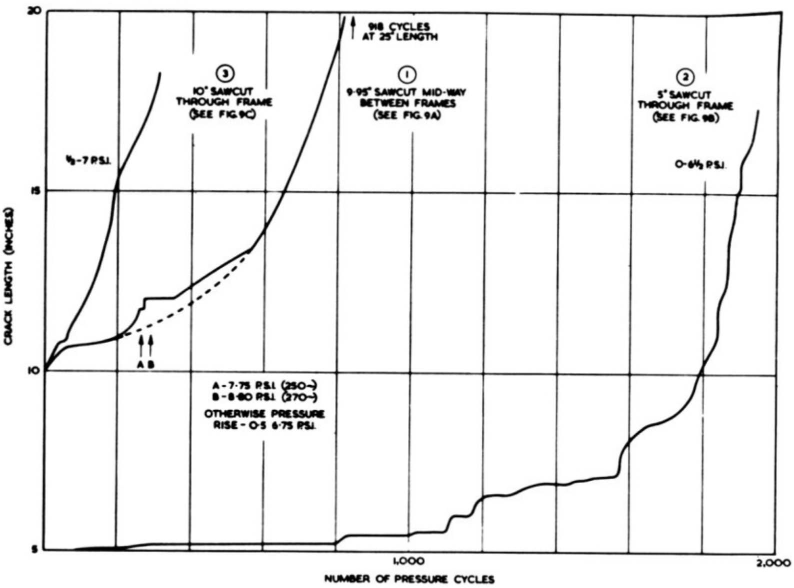
PRESSURE CABIN TEST SPECIMEN

Fig. 8.



SAW CUTS TO INITIATE CRACKS IN PRESSURE CABIN

Fig. 9.



ARTIFICIAL FATIGUE CRACK PROPOGATION IN PRESSURE CABIN

Fig. 10.

this point another overpressurization to 8.80 psi took place, just exceeding the proof load. The crack extended to 12 in. and when the pressure range was adjusted to the correct test value, ½ − 6.75 psi, crack propagation virtually stopped until 350 cycles had been completed. The crack then extended slowly until it rejoined the projection of the original growth curve, at 520 cycles. It was still extending uniformly when the test was stopped at 920 cycles.

The cracks initiated by the 5-in. cut, curve 2, and by the 10-in. cut, curve 3, Fig. 10, grew uniformly. The 5-in. cut, curve 2, reached 18.05 in. after 1,952 cycles of pressure. The crack was still extending steadily when the test was stopped. The 10-in. cut, curve 3, initiated a crack which reached 18.18 in. after 305 cycles of pressure. This rate of propagation is slightly lower than that of curve 2 from 10-in. length, but the difference is sufficiently small, all things being considered, to show the repeatability of the test procedure was adequate to apply to practice. We can conclude, therefore, that these results not only demonstrate the fail-safe nature of the pressure cabin construction, but also show that if cracks were initiated, their rate of propagation would be such that no undue strain would be placed on inspection. Finally we would just like to draw attention to the parallel behavior of the cracks in the wing box and in the pressure cabin tests when an overload was applied during test.

BRIEF DESCRIPTION OF THE VC 10

The V.C. 10, an aircraft with a gross weight of approximately 300,000 lb, is designed expressly to give good take-off performance from relatively short, high-altitude, high-temperature aerodromes. The wings have a gross area of 2,800 sq ft and they are fitted with Fowler flaps and leading-edge slats to provide additional lift for take-off and landing. The four engines are fitted in fuselage mounted nacelles at the rear. The all-moving tailplane is mounted at the top of the fin.

FATIGUE TESTING THE VC 10

In order to make the fatigue test as representative as possible, a study of typical flight plans was made and a loading sequence established on this basis.

A complete "program" will comprise 300 flights which will be repeated until the required number of flights is achieved. The loading system has been arranged to go through the full operational cycle in the following phases:

(*a*) Refuelling
(*b*) Ground-to-air lift—flaps in take-off setting
(*c*) Climb condition—flaps retracted
(*d*) Cruise condition
(*e*) Descent condition including application of air brakes and flaps
(*f*) Touchdown and overswing on landing

During phases (*c*), (*d*), and (*e*), gust loads of varying magnitude are applied. A study of the available statistical data resulted in the breakdown of gust loads in the 300-flight cycle as shown in Table 2.

Table 2

Flight phase	Climb			Cruise			Descent		
Number of flights / Gust velocity, fps	1	19	280	1	19	280	1	19	280
5	195	15	4	286	24	4	300	20	4
10	88	8		112	9		156	11	
15	29	1		39	1		38	1	
20	10			11			11		
25	2			3			3		
30	1			1			1		

THE LOADING SYSTEM

GENERAL ARRANGEMENT

The test specimen is located beneath a group of portal frames as shown in Fig. 11.

WING LOADS

Ten overwing hydraulic jacks operating through a conventional beam-and-link system are used to apply the loads to the top surface of the wing. A further two jacks apply load through a compression distribution linkage from beneath the wing, making a total of six loading points per side on the wing (see Fig. 12).

FUSELAGE LOADS

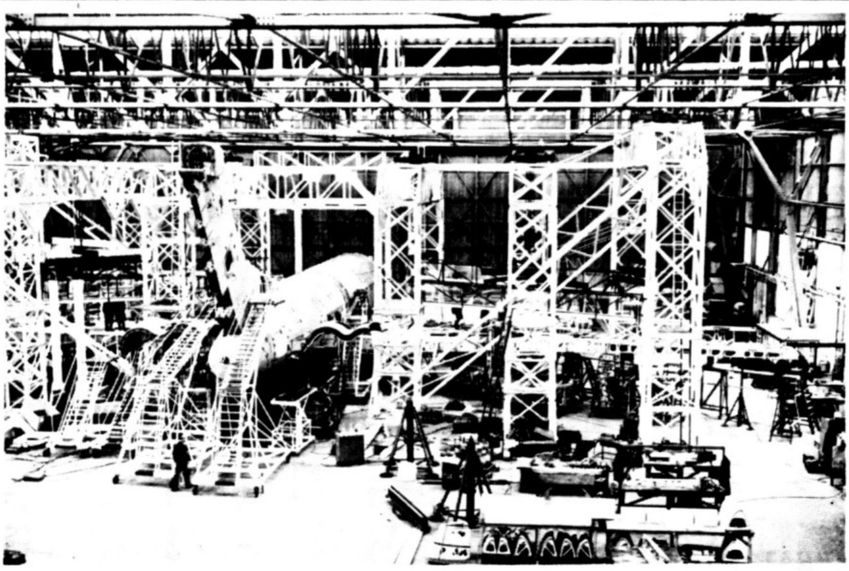
The fuselage is loaded at nine points requiring eight hydraulic jacks, the ninth loading point being a fixed pivot which provides freedom in pitch, roll and yaw, but restraint against translation in all directions.

During flight load simulation the landing gear is lifted clear of the ground by pivoting the airframe about the fixed reactor. Four of the eight hydraulic jacks are used for applying loads which are external as far as the fuselage test component is concerned, *viz.*:

1. Front fuselage loads
2. Landing gear loads on the gear uplocks
3. Engine nacelle loads applied through dummy engines
4. Tailplane loads applied through dummy tailplanes

The other four hydraulic jacks apply the appropriate inertia loads through the passenger and freight floor structures.

In addition, two small jacks are used to balance the weight of the dummy extended landing gear when the aircraft is in the flight condition. These are also used for roll stabilization.



VC 10 AIRCRAFT IN TEST FRAME

Fig. 11.

HYDRAULIC CIRCUITS

The principle of the hydraulic circuits used is shown in Fig. 13. An amplifier demand signal is fed to the electrohydraulic servo valve, the output of which operates the external delivery control of a variable delivery pump. The pump lever is spring-centered and at zero output from the servo valve the pump is

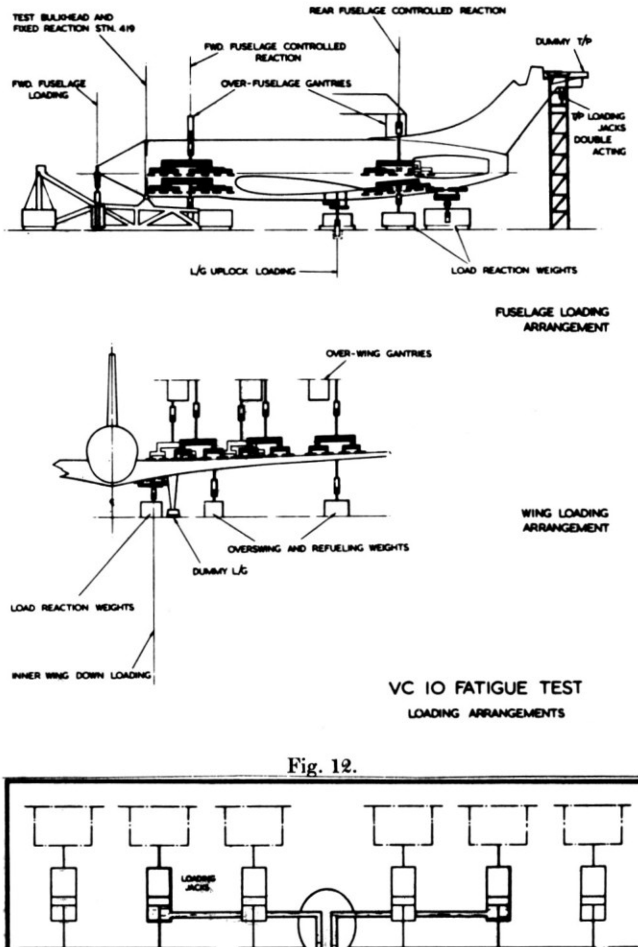


Fig. 12.

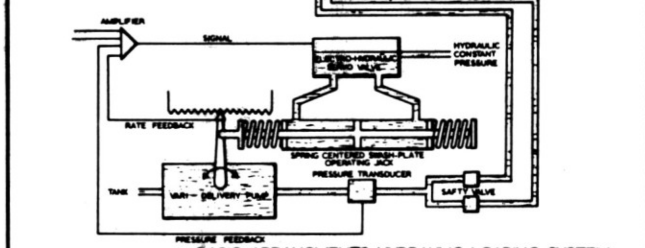


Fig. 13.

stable in the non-delivery position. A signal to the servo valve creates a differential pressure across the jack piston resulting in movement of the pump control lever.

Also shown is the safety valve positioned between the feedback transducer and the loading jack which enables the servo system to be divorced from the airframe in the event of failure.

Six of the ten applied loading systems are as shown in the diagram. The remaining four take a direct hydraulic supply from the servo valve. These four are used where oil demand is low because of small deflections.

LOAD CONTROL

The basic concept of the load control system is shown in Fig. 14. The required program of loads is derived from a program unit which comprises two sets of multiway switches and a tape reader. The first set of switches is called the *counting unit*. This dictates the applied gust load levels and mean load changes including take-off and landing.

The second set of switches is called the *pattern unit*. This selects the appropriate variable potentiometers giving the required jack loading.

The tape reader controls the number of loads applied at a particular level corresponding to the given position of the counter unit.

Metallized punched tape is prepared by a digital computer; a particular character detected by a phototransistor dictates a change in the applied load level. Each load application is arranged to traverse the tape on to the next position.

The gust speed control shown as an offshoot of the program unit serves to minimize testing time. Low gust-load levels can be applied more rapidly than high load levels and the counter unit is used to control test speeds in relation to the load levels being applied.

The required servo signal can be divided into two components, a steady or mean level signal and an alternating signal, corresponding to gust loading. In order to maintain independent control of the components, the required voltage output is taken from two potentiometers. The alternating voltage is taken from a sine/cosine potentiometer, the wiper of which is rotated by an electric motor through a variable-speed gearbox and magnetic clutch. Remote control of gust

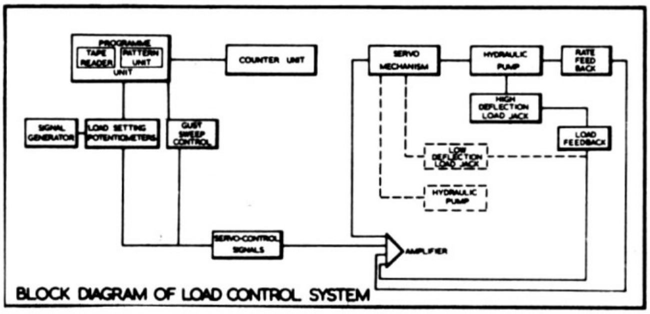


Fig. 14.

application and frequency is achieved by electrical operation of the clutch and variable speed gear with the motor running continuously.

The resulting servo output signals after modification of the variable load potentiometers are fed to an amplifier, the output of which controls the electro hydraulic servo valve.

Two feed-back loops are used, a rate feedback and an applied load feed-back (Fig. 13). The load feedback is taken from a pressure transducer in the loading jack line and the rate feed-back is proportional to the pump hydraulic output. Pump control level movement is used to supply the rate feed signal to the amplifier by directly moving the wiper on a potentiometer. A steady state occurs when the load feed-back equals the demand signal with the rate signal zero.

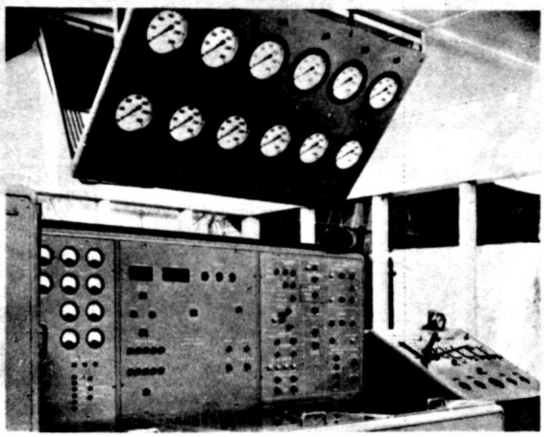
CONTROL CONSOLE

The control console is shown in Figs. 15 and 16. The basic operating controls and electrical monitoring are mounted on the three vertical panels. The dials on the left monitor applies loads directly by a parallel circuit to the load feedback. The central panel carries the flight records and manual override switches to the program unit; the third panel operates as described in the preceding paragraph.

The banks of variable potentiometers determining the various applied loads are mounted on the horizontal panels beneath the Perspex covers. These are arranged in rows of ten pairs, corresponding to the number of independent loading circuits and to gust and mean loads. The number of rows corresponds to the number of different applied loads.

The large panel above the console carries the pressure gages in each applied load circuit and in the two reaction galleries.

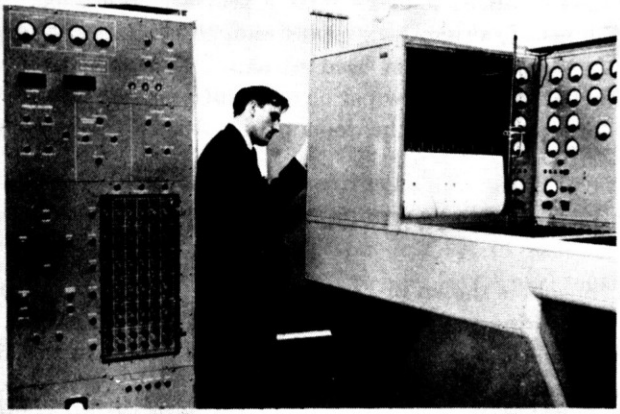
The smaller console to the right of the main console carries the diesel engine controls and instruments and remote diesel clutch control.



FATIGUE TEST CONSOLE

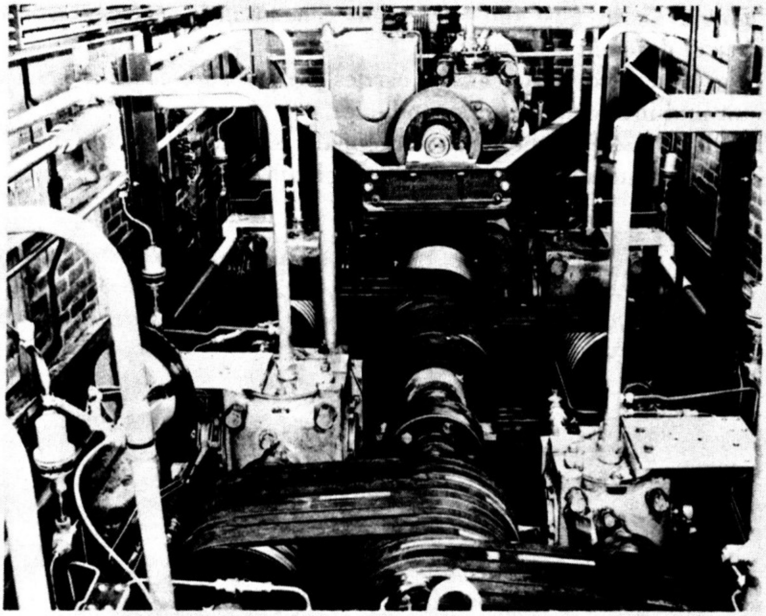
Fig. 15.

Figure 16 also shows the bank of "pen-recorders." These consist of a small jack tapped into the applied load lines loading against a spring. A pen held against moving paper swings on an arm as circuit pressure varies; a continuous record of pressure variation is therefore obtained. This is not intended as an accurate record of actual applied loads but as a continuous check on the program.



TAILPLANE FATIGUE CONTROL CONSOLE

Fig. 16.



MAIN POWER SUPPLY MOTOR AND PUMP

Fig. 17.

POWER SUPPLY

Figure 17 shows the layout on the powerplant. In all, seven pumps are driven via multiple fan belts from a 350-hp diesel engine. Six of the pumps, varying in size in relation to the oil demand of the loading position they supply, are controlled by the servo mechanism described above. The seventh pump works as an auto pump maintaining the pressure in the accumulator supplying the remaining four systems. The same accumulator also provides all pressures to the pilot operated valves and servo valves and for the overswing and refuelling cycle.

DISCUSSIONS

Authors: B. Stephenson and D. James
Discussor: F. J. Plantema, National Aero and Astronautical Research Institute

Referring to Fig. 7 of the paper, the most striking feature is the large effect of the application of the proof loads. They give an instantaneous crack extension but nearly stop further crack propagation for some time and so result in a fairly large retardation of crack growth. I suppose that the proof loads were considerably larger than the maximum load according to the program of Fig. 6. Hence, the omission of the proof loads would probably have given a much more rapid crack propagation. I also wonder whether proof loads were applied earlier during the test.

Authors' reply to discussion:

Dr. Plantema correctly points out that the application of proof (or limit) loads after the initiation of the fatigue crack produced a significant increase in crack length and a retardation in the immediately subsequent rate of crack propagation under the more normal loads of the program. His query is whether, on average, a higher rate of propagation would have been shown had the proof loads not been applied. This may or may not have been the case. However, the primary object of a test of this nature is to determine the crack length at which a critical extension of the crack occurs under the application of the proof load. The important comparison is between the growth which might occur due to the high load infrequently experienced and the average growth over a period between inspections.

Authors: B. Stephenson and D. James
Discussor: Dr. Ing. Ernst Gassner, Laboratorium fuer Betriebsfestigkeit

Program-fatigue test including "ground-to-air cycles" on spars with holes with and without unloaded interference bushes (LBF/1). From fatigue tests of the R.A.E. and Vickers-Armstrongs, Ltd. it is known that the fatigue curve for joints is raised by employing interference bushes on bolts, and *that the finite life part is raised more than the fatigue limit.* To calculate the improvement of the fatigue life under spectrum loading program-fatigue tests seemed to be necessary to arrive at reliable conclusions.

The specimens for the program-fatigue tests were cut from spars of the Viscount 810. Tensile strength and yield point were considerably higher than specified for L65 material. For the evaluation of the load spectrum an assumed flight plan was used. The relative gust distribution was derived from R.A.E. Rep. Structures 216. Gusts between 2 ft/sec and 10 ft/sec were also included.

According to a figure with the highest stress of the program due to a gust of 40 ft/sec plotted as a function of the total number of gusts ($\geq$ 2 ft/sec) and the number of flights of one hour respectively, the *fatigue life is increased seven times* by the introduction of interference bushes. *Three specimens with interference bushes were only tested up to fatigue life for the specimen without interference bushes and did not show any damage.*

The possible scatter has been accounted for in these tests by providing the specimens with six holes which will not interfere with each other. For the specimens with interference bushes only the fatigue life until the first visible crack has been employed.

Finally, it should be said that the use of L65 material did not give the expected lower crack propagation. Obviously this must be explained by the previously mentioned high tensile strength, which was far beyond the specification requirement. (Ref.: E. Gassner, "Review of investigations on aeronautical fatigue in Western Germany," LBF-Bericht S-34.)

Authors' reply to discussion:

I thank Dr. Gassner for his valuable contribution. The tests he has made with programmed loadings confirm the value of controlled interference fits to give an extended fatigue life.

The sevenfold increase in life that these tests show cannot necessarily be fully exploited as other features of the design may become the limiting factor for fatigue strength—for example, riveted attachments where the close control of interference is dubious or impractical.