C.P. No. 666

C.P. No. 666

MINISTRY OF AVIATION

AERONAUTICAL RESEARCH COUNCIL

CURRENT PAPERS

Fatigue Behaviour Under Service and Ground Test Conditions (A Comparison Based on the Dakota Wing)

By

W.J. Winkworth

LONDON: HER MAJESTY'S STATIONERY OFFICE
1964

TEN SHILLINGS NET

U.D.C. No. 539.431 : 533.6.048.5 : 629.13.012.6 :
620.178.3 : [A1] (73) Dakota

C.P. No. 666

November, 1961

FATIGUE BEHAVIOUR UNDER SERVICE AND GROUND TEST CONDITIONS (A COMPARISON BASED ON THE DAKOTA WING)

by

W. J. Winkworth

SUMMARY

Several Dakota wings have been tested in fatigue under various conditions. The objective was to obtain an assessment of full scale testing procedures.

When the ground-to-air loading actions are represented in proper relation to those of atmospheric turbulence, the behaviour in test is found to be similar to that in service.

Replaces R.A.E. Tech. Note No. Structures 306-A.R.C. 23,963.

(82973)

LIST OF CONTENTS

LIST OF APPENDICES

LIST OF TABLES

LIST OF ILLUSTRATIONS

1 INTRODUCTION

Fatigue failures in aircraft primary structures are rare and thus opportunities seldom arise when the results of a fatigue test can be compared with the performance in service. Only from such comparisons is it possible to assess the validity of laboratory procedures for fatigue testing complete structures. The wing of the Dakota aircraft is particularly suitable for such a study as fatigue cracks found in service have been fairly well documented.

Four tests were made in which the loading actions of take-off and landing and of atmospheric turbulence were applied either separately or in combination. It was found that, when both these loading actions were included in the tests, there was fair agreement with service experience on the endurance at which fatigue damage began to appear and good agreement on the type and location of some of the failures.

2 THE DAKOTA WING STRUCTURE

Although the wing was designed some thirty years ago, the structure may still be considered to be representative of many present day aircraft and therefore appropriate for use in the context of this note.

A box type beam with three spars constitutes the primary structure.

From fatigue aspects, it is the bottom or tension face which is important and in the Dakota wing this face is made up of skin, spanwise stringers and the spar tension booms. In the centre section the skin is 16 s.w.g. (0.064 in.) thick, and in the outer wing panels it is 20 s.w.g. (0.036 in.) thick. The spanwise stringers are small bulb-angle extrusions and the spar tension booms are of similar section but considerably larger. The relative cross-sectional areas which constitute the whole are approximately 45% skin, 25% stringers and 30% spar booms. All are manufactured from copper aluminium alloy to the American Specification 24 ST (Appendix 1).

The outer wing is attached to the centre section by a bolted flange type joint which extends completely round the aerofoil and which is referred to as the transport joint. As the plate type webs of the three spars end at the joint, shear as well as bending and torque loads are transmitted by this joint.

Each half of the joint consists of a relatively heavy extruded angle which is riveted to the skin and doubler plate and is positioned outside the normal profile; the vertical flanges of this angle and of that on the other half of the joint receive a large number of $\frac{1}{4}$ in. diameter bolts at close pitch under controlled initial tension to form the final attachment.

On the outer wing the doubler plate extends round the vertical flange of the joint angle, whereas on the centre section the doubler plates terminate on the horizontal flange and the wing skin extends round the vertical flange.

As important fatigue damage occurred at or near the joint between the centre section and the outer wing, the local constructional features are of great interest. Further details are given in paragraph 4.2.

The lower tension skin and stringers in the centre section form detachable tank access panels which are bolted to the spar booms (see Fig.6). At the inboard end of these panels there is a joint with constructional features similar to those of the transport joint and fatigue damage also occurred in this region during the tests.

3 THE FATIGUE TEST LOADS

The test loads were based on operational data obtained from British European Airways and from Transair Ltd.

The following average flight plan was deduced:-

	Height (ft)	Average speed (knots E.A.S.)	Duration
Climb	0 to 8,500	110	15 mins
Cruise	8,500	130	102 mins
Descent, circuit and landing	8,500 to 0	135	18 mins
		Total:	$2\frac{1}{4}$ hours

Average A.U.W. - 27,500 lb (inclusive of 2,880 lb of fuel in the wings).

Aerodynamic data, etc. were extracted from Douglas Report S.M.992[1].

The fatigue loading conditions in flight were simulated by a mean load and a superimposed alternating load. The mean load represented the conditions arising in steady flight and the alternating load those of flight through turbulence which was represented by cycles of up and down gusts of 12 ft/sec velocity. The gust loading was equivalent to an acceleration of $\pm 0.37g$ about 1g, where 1g represents level flight.

Only the gust loading was applied in the first test, but in the second and third tests the loading action incurred during take-off and landing (the ground-to-air load cycle) was interposed between succeeding batches of 15 gust load cycles (2nd Test) and 5 gust load cycles (3rd Test). In the fourth test, ground-to-air load cycles were applied without the gust loading.

The 'on-ground' condition was represented by applying down load to the outer wings, the amount being such that they supported twice their own weight. This was done to obtain compressive stress in the bottom surface of the wing centre-section such as might arise during taxying. The ground-to-air cycle was produced by varying the load between the 1g steady flight and the 'on ground' conditions.

Shears, torques, bending moments and derived test loads are detailed on Figs.1, 2, 3 and 4 respectively.

By means of a hydraulic system, loads were applied through jacks acting in compression between the ground and lever systems attached to the bottom surface of the wing. Reactions were provided by a frame which supported the wing at the fuselage attachments, Fig.5.

The rate of loading was 15 gust load cycles per minute.

4 RESULTS OF TESTS

4.1 These are summarised in Table 1 and detailed in Appendix 2 which also makes reference to stresses measured near the regions where failures occurred.

4.2 Location of fatigue damage

The position of the failures at the inboard ends of the fuel tank access panels and on both sides of the transport joint are shown on Fig.6 and described below:-

Location A

At the inboard ends of the removable fuel tank access panels in the centre-section

Fatigue cracks formed in the skin at the end rivets attaching the chordwise joint angle and, at a later stage when these had become extensive, further cracks occurred in the bulb angle stringers, Fig.7. A few cracks also formed at the $\frac{1}{4}$ in. diameter drain holes near these end rivets and at the large holes which accommodate fuel drain cocks.

Location B

At the transport joint (outer wing side)

Fatigue cracks formed in the vertical flange of the wrap-round doubler; in all cases these were at the spar positions and either tangential to the joint bolt holes, Fig.8, or clear of the bolt holes, Fig.22.

Cracks developed later in a similar manner at the joint bolt holes in the extruded angle.

Location C

At the transport joint (centre-section side)

Fatigue cracks formed in the skin at the end rivets attaching the extruded joint angle, Fig.9. As the skin was sandwiched between the joint angle and the internal doubler plate, these cracks could not be seen.

4.3 Interpretation of test results

From standard gust data[2] and on the basis of the well known Palmgren-Miner cumulative damage hypothesis, it was estimated that fifteen ± 12 ft/sec gust load cycles are equivalent, in terms of fatigue damage, to the loading actions caused by atmospheric turbulence in the $2\frac{1}{4}$ hour average flight defined in paragraph 3.

With this information, the endurances obtained in tests Nos. 2 and 3 may be expressed in hours of flying, test 2 representing service conditions when the aircraft is used on flights of about $2\frac{1}{4}$ hours and test 3 when shorter flights (about $\frac{3}{4}$ hour) apply. It is less realistic to convert test 1 endurances in this way, since no ground-to-air load cycles were applied; this conversion, however, has been done and is included for comparative purposes and because in the past many fatigue tests have been made with such simplified loading.

Test No.4 was made to observe the fatigue behaviour when only ground-to-air loads were applied; in this case the test conditions are unrealistic and conversion of endurances to hours of flight would only be of interest if dead calm atmospheric conditions prevailed and there were no manoeuvres.

The endurances on test in terms of hours of flying have been corrected by adding the previous service flying carried out by the test airframe in each case. This is an approximation which suffices for the comparison to be made.

The following Table indicates the approximate endurances in hours of flight for two conditions, firstly when fatigue damage was first visible (i.e. small cracks had formed) and secondly when complete failure occurred or when damage was extensive and failure imminent.

Test	Location A (Tank access panels)	Location B (Joint - outer wing)	Location C (Joint - C/section)
	Fatigue damage first visible		
1	38,000 hours	60,000 hours	Any early signs of damage were not visible and therefore were undetected.
2	26,000 hours	17,000 hours	
3	18,000 hours	17,000 hours	
	Complete failure or extensive damage		
1	50,000 hours	More than 100,000 hours	50,000 hours
2	36,000 hours	More than 36,000 hours	
3	More than 24,000 hours	24,000 hours	24,000 hours

5 FATIGUE BEHAVIOUR IN SERVICE

From the service experience available, which is possibly small in relation to the total Dakota experience, fatigue damage occurs at the wing transport joint (Locations B and C).

There have been many instances of cracks at B (i.e. in the vertical flange of the outer wing lower skin wrap-round doubler) in the region of the front and centre spars. A statistical survey of these failures which are summarised in Appendix 3 indicates endurances ranging between 2,500 and 24,000 hours, the frequency distribution showing two peaks, the smaller at about 12,000 hours and the larger at about 20,000 hours. Crack lengths were from $\frac{1}{4}$ in. to $6\frac{3}{4}$ in.

A further indication of the fatigue life of these skin doublers is provided by the service inspection and maintenance procedures. Inspections are required at 8,000 hours and then at intervals not exceeding 4,000 hours. The doublers are replaced at 16,000 hours[3].

There is less information on the skin cracks which developed between the transport joint angle and the inside doubler plate on the centre-section side of the joint, i.e. Location C in the tests (see (b) of Appendix 3). Inspection and maintenance requirements[3] call for holes at the front and centre spars to enable visual inspection for cracks at intervals at 1,500 hours, replacements are required to be fitted at not more than 38,000 hours.

There is no information in the available reports concerning fatigue damage at the inboard ends of the tank access panels, Location A, although mention (in (c) of Appendix 3) of cracks on the bottom surface of the centre-section 96 in. outboard of the aircraft centre, on aircraft with lives ranging from 41,200 to 65,600 hours, may refer to similar fatigue damage in the region of the outer extremities of these panels.

6 COMPARISONS

The tests were made to assess the current procedures for fatigue testing complete structures. In making the assessment, the important aspects are whether the fatigue damage in the tests occurred at the locations where it is known to occur under service conditions, whether the development of the cracks followed a similar pattern and whether the life indicated by the full scale tests is in agreement with the known life in service.

A summary of the service experience compared with the fatigue test results is given in Table 2.

6.1 Locations

The available service experience given in reports and letters which are summarised in Appendix 3 quotes many instances of fatigue cracks at Location B, some possibly at Location C and none at Location A.

Thus from the location of damage aspect, it is evident that the fatigue test conditions were realistic at the transport joint but not at the centre of the wing.

The apparently unrepresentative damage which occurred in the tests at A might be explained by differences in the loading conditions or by other influences, for example, the repeated removal and re-assembly of these panels for inspection purposes, not represented in the tests, or the renewal of these panels in service due to damage other than fatigue damage.

6.2 Growth and extent of fatigue damage

The failures which occurred at Location A in the tests provided excellent growth data; unfortunately these cannot be used as no records were found of similar damage occurring on service aircraft.

The service reports do not give information on growth for the failures at Location B, however it is possible to gain some indication from the relevant inspection schedules.

Inspection at B on service aircraft is required at 4,000-hour intervals. This is a relatively long period and as such, suggests a slow rate of growth. In the tests, the growth was also slow, particularly when the damage was confined to the vertical flanges of the wrap-round skin doublers and therefore at a similar stage of development to that indicated in the reports on service aircraft.

Even when the damage became more extensive in the tests and cracks had developed in the main joint angle (no instances on service aircraft were available) growth was still slow and complete failure did not occur at this point in any of the tests.

Growth data for Location C failures are only available from the 4th Test in which the ground-to-air loading action was represented without gust loads. The data suggest a faster rate of growth than that at B; this is in general accord with service experience, since inspections at C are required at 1,500-hour intervals compared with 4,000-hour intervals at B.

6.3 Endurance

As assumptions have to be made in interpreting flight data, in selecting loading actions, and in determining the test loads, and as the conditions of service are quite varied, only a general comparison on endurance can be made between service experience and the test results.

For failures at Location B, Tests Nos.2 and 3 indicate endurances of 17,000 hours to crack initiation and from 24,000 hours to 36,000 hours to extensive failure, the figure of 24,000 hours being associated with short flights as in Test No.3.

The general impression gained from the service reports is that cracks at Location B occur most frequently at about 20,000 hours, although there is wide scatter. Thus, if both major loading actions are properly represented in the tests there is good agreement with service performance. This is not so, however, when the ground-to-air loading action is omitted as in Test No.1 which gave 60,000 hours before failure commenced at Location B.

It is not possible to compare test and service endurances at Location C. Test No.3 indicates the endurance at Location C to be the same as that at Location B but the available service information is meagre. Relevant inspection and maintenance schedules[3] do, however, indicate some measure of agreement, since inspections at both B and C are required to start at the same initial life, i.e. 8,000 hours. The tests further indicate that the fatigue damage at C was of a more serious nature than that at B and this is also supported by the inspection requirements which call for 4,000 hour inspection intervals at B and 1,500 hour intervals at C.

7 CONCLUSIONS

In the region of the wing transport joint (Locations B and C), good agreement was found between service experience and the results of Tests Nos.2 and 3 in which both gusts and ground-to-air loading actions were represented.

Fatigue damage in the same region also occurred in Test No.1 with gust loading only, and in Test No.4 with ground-to-air loading but, as was to be expected, at higher endurances than obtained when these loading actions were combined as in service or as in Tests Nos.2 and 3.

There was no evidence from the service reports of fatigue damage at the inboard end of the centre-section removable panels (Location A). This damage occurred in all the tests and the disparity has not been fully explained.

LIST OF REFERENCES

Ref No.	Title, etc.
1	Wing Analysis - Model D.S.T., Report No. SM.992 7.31.35 Douglas Aircraft Co. Ltd.
2	Royal Aeronautical Society Fatigue Data Sheet L.01.01.
3	Mandatory Notes Applicable to Douglas D.C.3 and Dakota Aircraft (Extracted from the Airworthiness Directive Summary).

APPENDIX 1

THE MATERIAL USED IN THE STRUCTURE

1 24 ST ALCLAD - Clad Aluminium Alloy Sheet

(i) Chemical composition (Nominal)

Copper	4.2 per cent
Manganese	0.5 per cent
Magnesium	1.5 per cent
Aluminium	The remainder

(ii) Heat treatment

Quenched from solution treatment temperature and naturally aged.

(iii) Strength properties (Typical)

(a)	Ultimate Tensile Stress	62,000 lb/sq in.
(b)	Yield Stress (Approximately the 0.2% Proof Stress)	41,000 lb/sq in.
(c)	Elongation	18%

APPENDIX 2

THE FATIGUE TESTS

1 GENERAL

The fatigue test loading was applied by a hydraulic system using jacks in compression which reacted between the ground and lever systems attached to the bottom surface of the wing, Figs.4 and 5.

The rate of cyclic loading was about 15 load cycles per minute.

Deflection at the wing tips for the ±12 ft/sec gust loads was ±2.6 ins.

Strains were measured at several chordwise positions on the bottom (or tension) surface at 15 ins. from the centre (close to Location A) and at 24 ins. outboard of the wing transport joint. The skin stresses derived from these are given in Table 3.

Inspection of the wing for damage at Location A was straightforward since the cracks appeared on the surface and it was possible to make frequent examinations. At Location B, however, it was necessary to disconnect the transport joint to inspect for possible damage, and at Location C signs of failure were not visible since the cracked skin was sandwiched between the joint angle and an internal doubler plate.

The following notes enlarge on the summary in Table 1.

2 TEST NO.1

Loading:- 1g ±12 ft/sec gust loads only.

Fatigue damage at Location A was more severe than in the subsequent tests; the centre-section was replaced and similar damage in this second centre-section was the primary reason for ending the test.

At this stage, severe damage at Locations B and C had also occurred.

Gust load cycles	
255,000	Fatigue cracks were found at Location A; these cracks grew and others originated as the test proceeded, Fig.10.
300,600	The forward removable tank access panel on the starboard side had fractured at Location A, Fig.11.
313,000	The rear access panel had also fractured, Fig.12. The test was stopped to replace the centre-section as complete failure was imminent at Location A on the starboard side. There were only one or two small cracks on the port side. Fatigue cracks at A first appeared at 209,000 gust load cycles in the replacement centre-section. It sustained 346,000 gust load cycles and, in general, its performance was similar to the original one.

Gust load cycles	
313,000	No damage was visible at Location B when the transport joint was disconnected for centre-section replacement.
446,000	Cracks which possibly began to form at about 400,000 cycles were found at B at the centre spar position on the starboard wing only; these grew fairly slowly, Figs.13 and 14.
623,000	The vertical flange of the joint angle at B had also cracked at the centre spar position on the starboard wing, Fig.15.
659,000	The test was stopped as failure was imminent at Location A on the replacement centre-section.

The crack in the joint angle at B was 5 ins. long and a similar crack 1.5 ins. long was visible at the rear spar position, also on the starboard wing. There was no visible damage at B on the port wing.

NOTE:-

Location C type damage was unsuspected until failure occurred at this point in the third test. By this time the original centre- section had been disposed of and it was therefore not possible to establish whether or not damage had taken place. On removing the joint angle on the replacement centre-section, extensive skin cracks were revealed on the starboard side, Fig.16. No cracks were visible at Location C on the port wing.

3 TEST NO.2

Loading:- 15 gust cycles and 1 ground-to-air cycle per flight of $2\frac{1}{4}$ hours.

Fatigue damage at Location A was again the primary reason for terminating the test, although damage at Location B was also found to be fairly severe. Location C was not examined prior to the disposal of the wing centre-section.

160,000 Gust load cycles
10,666 Ground-to-air load cycles

Fatigue cracks had commenced to form at Location A. Small holes were drilled at the ends of these cracks to retard their growth.

At B there were cracks at the centre-spar positions on both sides, and also at the rear spar position on the port side. The joint angle at this latter position was cracked in its vertical flange over four bolt pitches (about 2 inches). These cracks possibly began to form at about 100,000 gust load cycles.

231,000 Gust load cycles
15,400 Ground-to-air load cycles

The test was stopped as fatigue damage at Location A was extensive.

The extent of cracking at Location B is shown on Fig.17.

Location C was not examined.

4 TEST NO.3

Loading:- 5 gust cycles and 1 ground-to-air cycle per flight of $\frac{3}{4}$ hour.

Somewhat unexpectedly, a complete failure occurred at Location C. The damage at A was relatively minor but at B it was fairly extensive.

80,000 Gust load cycles
16,000 Ground-to-air load cycles

One small fatigue crack was observed at A; a hole was drilled to retard its growth.

Two further similar cracks formed at 125,000 gust load cycles.

90,000 Gust load cycles
18,000 Ground-to-air load cycles

No damage was visible at Location B.

138,000 Gust load cycles
27,600 Ground-to-air load cycles

A complete failure occurred at Location C on the starboard wing, Figs.18, 19 and 20. The fracture was in the wing skin and the developing fatigue cracks had not been found during inspections, because they were between the external joint angle and the internal skin doubler and therefore not visible.

At Location B on the starboard side, there were extensive cracks in both skin doubler and joint angle at the centre spar position, the latter extending some 8 inches. The skin doubler was also cracked at the rear spar position on both port and starboard sides. The cracks probably began to form soon after the previous inspection at 90,000 gust load cycles.

5 TEST NO.4

Loading:- Ground-to-air cycles only.

There was very little fatigue damage at A and only moderate damage at B in this test. The damage at Location C was extensive.

Ground-to-air load cycles	
58,500	A fatigue crack in the skin at Location C starboard side was found by X-ray technique. The position of the crack and its subsequent growth is shown on Fig.21.
107,000	No damage was visible at Location B
116,000	A $2\frac{1}{4}$ inch crack was observed near Location B in the horizontal flange of the transport joint angle at the starboard centre spar position; its subsequent growth was very slow ($3\frac{1}{4}$ inch long at the end of the test).
139,000	Fatigue cracks were found at Location B at the starboard front and rear spar position, Fig.22.

Ground-to-air load cycles	
157,000	The first small crack occurred at Location A.
168,000	A second small crack was found at Location A.
194,000	The fatigue damage at Location C was extensive and the test was stopped for this reason. No visible damage was found on the port side of the wing at Location C when this was stripped for examination.

APPENDIX 3

FATIGUE DATA FROM SERVICE SOURCES

The available service experience is summarised below for each of the sources of information and is compared with the results of the tests in Table 2.

(a) Transair Ltd.

In February 1957, Transair stated that they had 11 aircraft with an average of 8,000 hours flying. There were no visible signs of fatigue damage.

(b) British European Airways

In March 1957, B.E.A. stated that they had been operating 46 Dakota aircraft which had then flown between 8,350 and 18,140 hours. Very few cases of fatigue cracking in the primary structure were found.

Fatigue cracks in the bottom skin of the centre-section occurred on four aircraft. The amount of flying could not be quoted but it was thought to be between 12,000 and 16,000 hours. The precise location was not stated; it may have been in the region of the transport joint (Location C) as the repair scheme involved the replacement of the wrap-round skin.

(c) Federal Aviation Agency

In a letter to the A.R.B., dated 29th June 1960, the following information was given:-

Cracks occurred in the attachment angles and the doublers between 2,500 and 12,400 hours.

The most recently reported occurrences of fatigue cracks were in the bottom skin of the centre-section at 96 inches and 126 inches outboard of the centre line, the aircraft having flown between 41,200 and 65,600 hours; there were nine cases.

In addition, there were three occurrences of fatigue cracks in the upper boom of the front spar in the region of the main landing gear upper truss pivot fittings.

(d) Report No. S.M.11861 - Douglas Aircraft Co. Inc.
Wing attachment angle doubler cracks - 23/7/46

This is a statistical study of the outer wing wrap-round doubler failures (Location B in the tests).

There were 48 instances of fatigue damage in which the cracks were ½ in. to 4 in. long, 25 at the front spar, 16 at the centre spar, 5 at the rear spar and 2 somewhere between spars.

On wings of 11,000 to 16,000 hours, 13 out of 238 doublers inspected were found to be cracked, and on wings of 17,000 to 24,000 hours the corresponding figures were 30 and 167.

(e) Report No. CX-79-United Airlines Inc. - 24/5/45

The outer wing lower doubler failures (Location B in the tests) are described and examined.

Twenty four instances of fatigue damage are listed· all but one of the wings affected had flown between 19,000 and 21,000 hours. The cracks were of about equal frequency of occurrence in the regions of the front and centre spars and very few were found at the rear spar position. The lengths of the cracks ranged between 0.25 in. and 6.75 in., most frequently they were about 1.0 in. long.

(f) Report No. CX-88 - United Airlines Inc. - 10/1/45

Details are given of four instances of fatigue failure in the vertical flange of the bottom skin in the centre-section at the wing transport joint. The lives of the aircraft were between 20,374 and 26,773 hours and the fatigue cracks were in the region of the centre spar. This failure is similar to the outer wing lower doubler failures, i.e. Location B in the wing tests. The method of repair is not detailed, (it might possibly be the steel plate insertion shown in Fig.9 which was incorporated on all the test wings).

Mention is made of the need for uniform tensioning on assembly to prevent fatigue failure of the transport joint bolts.

Other fatigue failures are described, such as cracking of rib flanges and engine mounting members, but these are not of direct interest in the present context.

TABLE 1

Summary of fatigue damage at end of tests

Test No.	Fatigue test loading conditions	Service hours	Gust loads	G-to-A loads	Centre-section tank access panels Location A	Outer wing at transport joint Location B	Centre-section at transport joint Location C
1	Atmospheric turbulence only ±12 ft/sec gust loads	1,567 (First centre-section)	313,000	None	Complete failure imminent on starboard side. Port side also damaged, though not so severe. Damage originated on both wings at approximately 250,000 cycles.		Extent of possible fatigue damage not established as the centre-section was not fully inspected prior to its disposal.
		1,261 (Replacement centre-section)	346,000	None	Extensive damage on port side. Considerable damage on starboard side. Damage originated on both wings at 210,000.		Complete failure imminent on starboard side. No similar cracks were found on the port side.
		1,567	659,000 (Outer wings)	None		Damage originated in region of centre spar on starboard wing at about 400,000. Growth was comparatively slow. Damage extensive on starboard side at 659,000. No similar cracks on port side.	
2	Atmospheric turbulence and ground-to-air 15 ±12 ft/sec gust load cycles per flight.	1,467	231,000	15,400	Both port and starboard sides extensively cracked. Growth was retarded by drilling holes at the ends of the fatigue cracks.	Fatigue cracks in region of centre and rear spars on starboard side and in region of centre spar on port side. Damage possibly originated at about 140,000 gust load cycles. Slow growth of cracks.	Extent of possible fatigue damage not established as the centre-section was not fully inspected prior to its disposal.
3	Atmospheric turbulence and ground-to-air 5 ±12 ft/sec gust load cycles per flight.	3,010	138,000	27,600	Small fatigue cracks on starboard side. None on port side. Cracks first visible at 80,000 gust load cycles.	No visible damage at 90,000. Extensive cracks in region of centre and rear spars on starboard side. Extruded joint angle also cracked on starboard side.	Total failure on starboard side. No similar damage was visible on the port side. (This failure was unexpected. It was at this stage that the second centre-section in Test No.1 was inspected; unfortunately the first centre-section and the one used in Test No.2 had been disposed of.)
4	Ground-to-air only	2,543	None	194,000	One small fatigue crack at 157,000 cycles. Damage and growth were only slight at the end of the test.	Appreciable damage in region of front, centre and rear spars on the starboard side. No visible damage on the port side. Fatigue damage originated at about 130,000 cycles.	Fatigue crack first found at 58,500 cycles. Complete failure at 194,000.

TABLE 2

Service experience compared with fatigue test results

Source	Service experience	Comparison with fatigue test results
Transair	No visible fatigue damage on eleven aircraft with average of 8,000 hours flying.	No damage would be expected.
B.E.A.	Very few cracks on 46 aircraft with between 8,350 and 18,140 hours flying. Four aircraft, possibly with about 12,000 to 16,000 hours flying, had centre-section wrap-round skin replaced.	 Possible agreement as there may have been some cracks at Location C.
F.A.A.	Cracks at Location E between 2,500 and 12,400 hours. Nine cases of cracks in bottom skin of centre-section at 41,200 to 65,600 hours, also three cases of cracks in the upper boom of front spar at the undercarriage position.	Similar damage commenced at about 17,000 hours in Tests 2 and 3. Not reproduced in the tests, probably because the tests were not extended by applying repairs and modifications to the damaged regions.
Douglas Aircraft Ltd.	Many instances of cracks at Location B between 11,000 hours and 24,000 hours.	Good agreement with the results of Tests 2 and 3 if allowance is made for scatter.
United Airlines Inc.	Twenty four instances of Location B failures between 19,000 and 21,000 hours (Report CX-79). Four instances of failure in the vertical flange of the centre-section bottom skin at the transport joint on aircraft with between 20,374 and 26,773 hours. (Report CX-88)	Good agreement as above. Not experienced in the tests; perhaps the test wings were to a later modification standard. (Reports CX-79 and CX-88 were written in 1945)

TABLE 3

Tensile stresses on the lower surface

Spanwise position on wing	Distance aft of front spar position at the transport joint (Stn. 142) in inches	Measured alternating stress in bottom skin in lb/sq in.	Stress at 1g loading in bottom skin in lb/sq in.
Measured at 15 ins. from the ℄ of aircraft, i.e. on the centre-section	1.4	2,700	7,300
	13.75	3,000	7,800
	25.75	3,000	8,000
	42.0	2,800	7,000
	58.0	2,800	7,400
	71.4 (Rear spar)	2,200	6,000
Measured at 24 ins. outboard of the transport joint, i.e. on the outer wing	2.75	2,300	6,400
	12.5	2,300	
	23.75	2,500	6,800
	35.0	2,600	
	40.75	2,600	7,700
	54.0	2,300	
	65.75	2,300	
	76.0 (Rear spar)	2,000	

NOTES:-

The front spar is at the 18% chord position.

The centre spar is at the 39% chord position.

The rear spar is at the 60% chord position.

Strains were measured with a 5.25 inch gauge length mechanical type extensometer.

The 1g stresses are based on the mean of the measured strains from the 'on ground' condition to 1g − 12 ft/sec gust load, and from 'on ground' to 1g + 12 ft/sec gust load. The stresses derived from these strains were reduced by 1,500 lb/sq in. (outer wing) and 200 lb/sq in. (centre-section) to allow for the initial compression in the bottom surface in the 'on ground' condition.

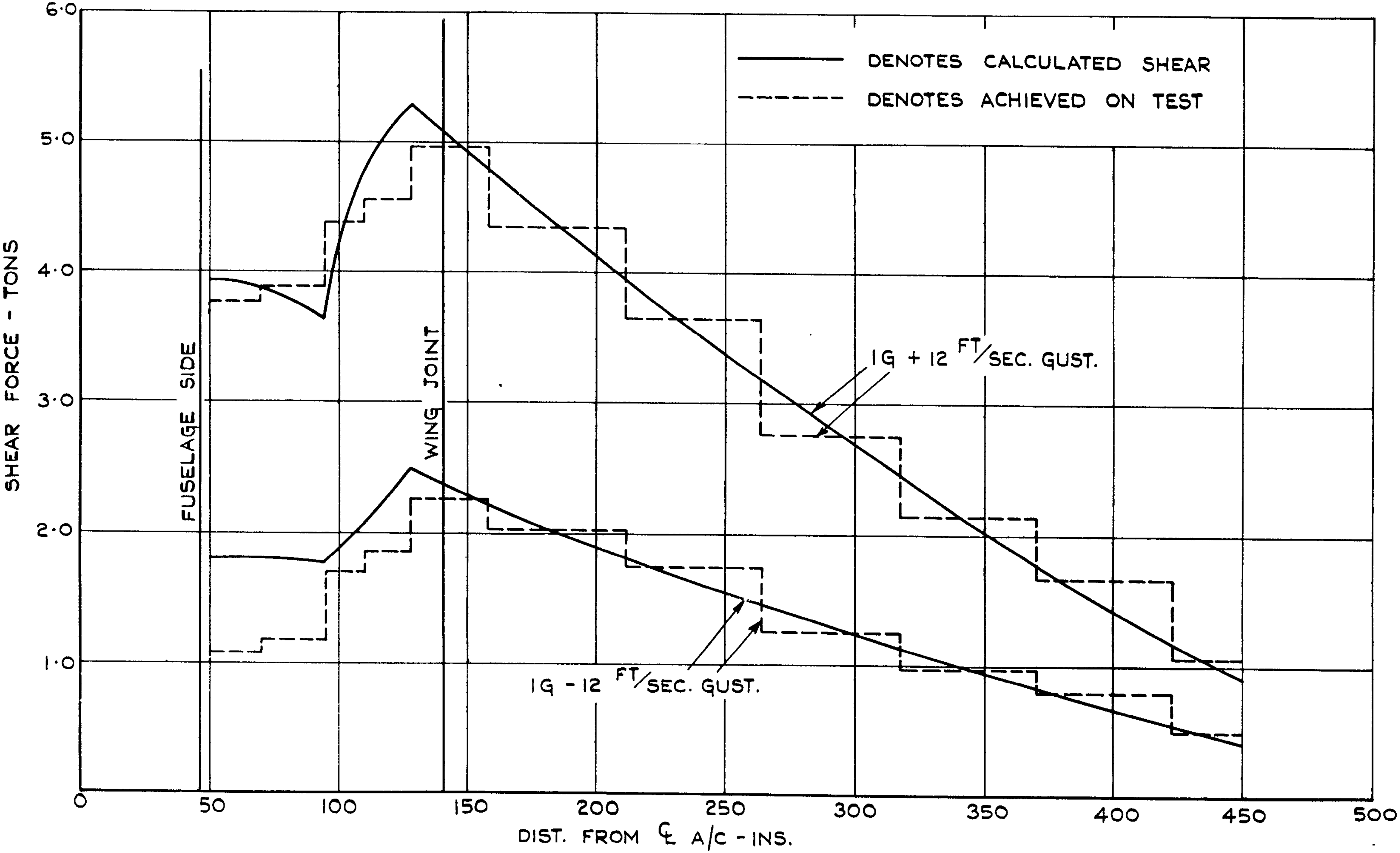


FIG. 1. WING SHEAR DIAGRAM.

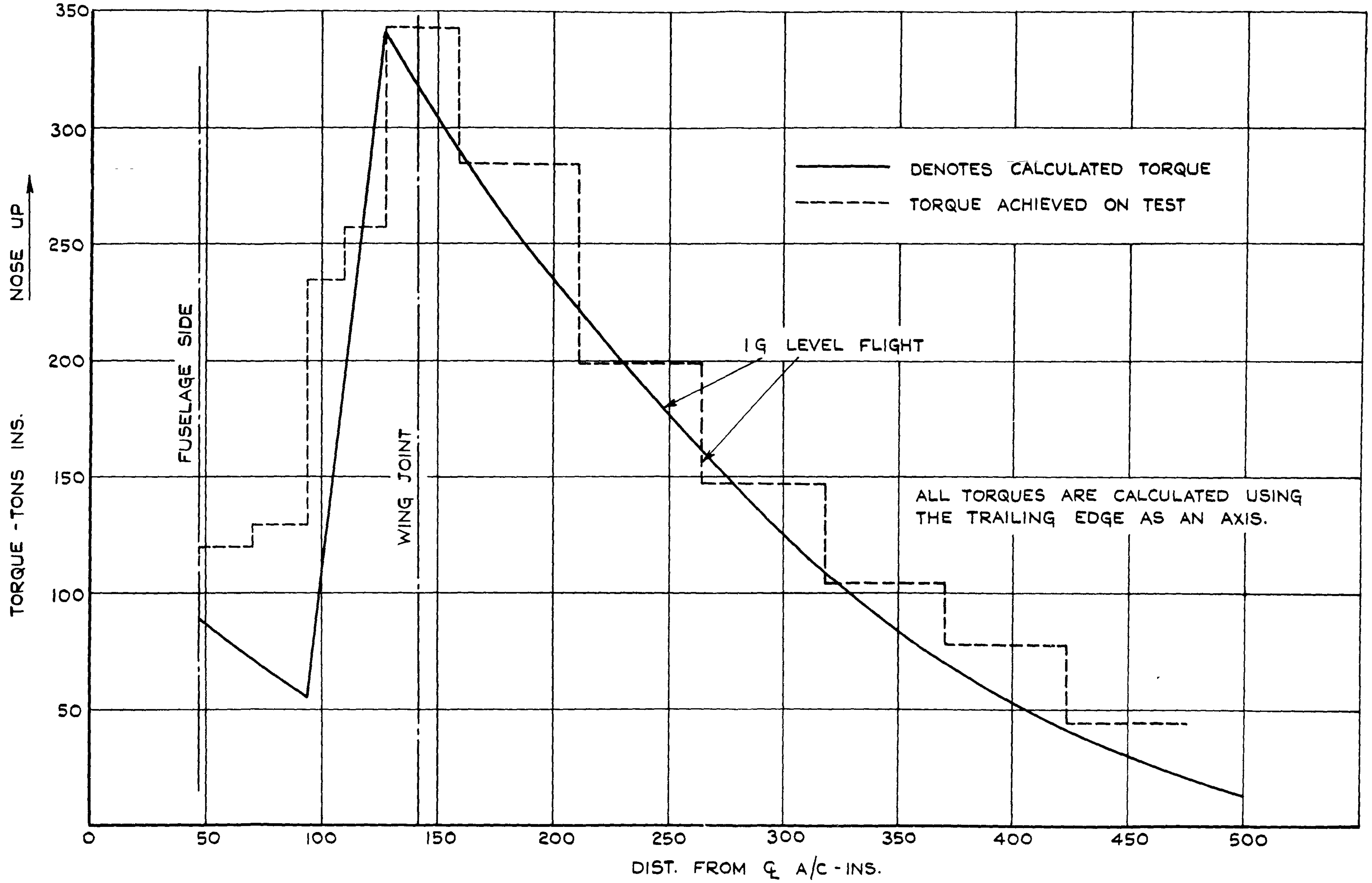
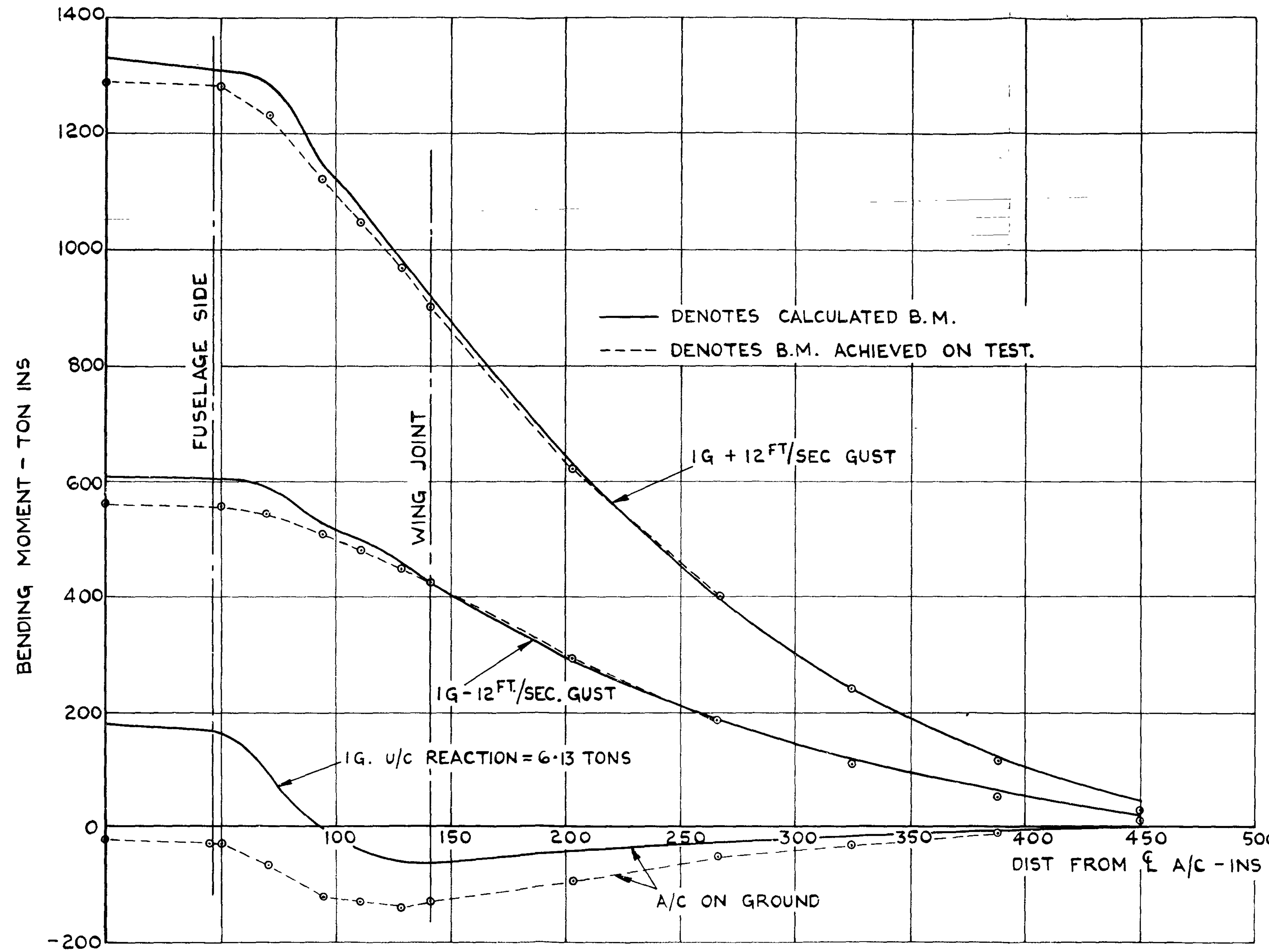
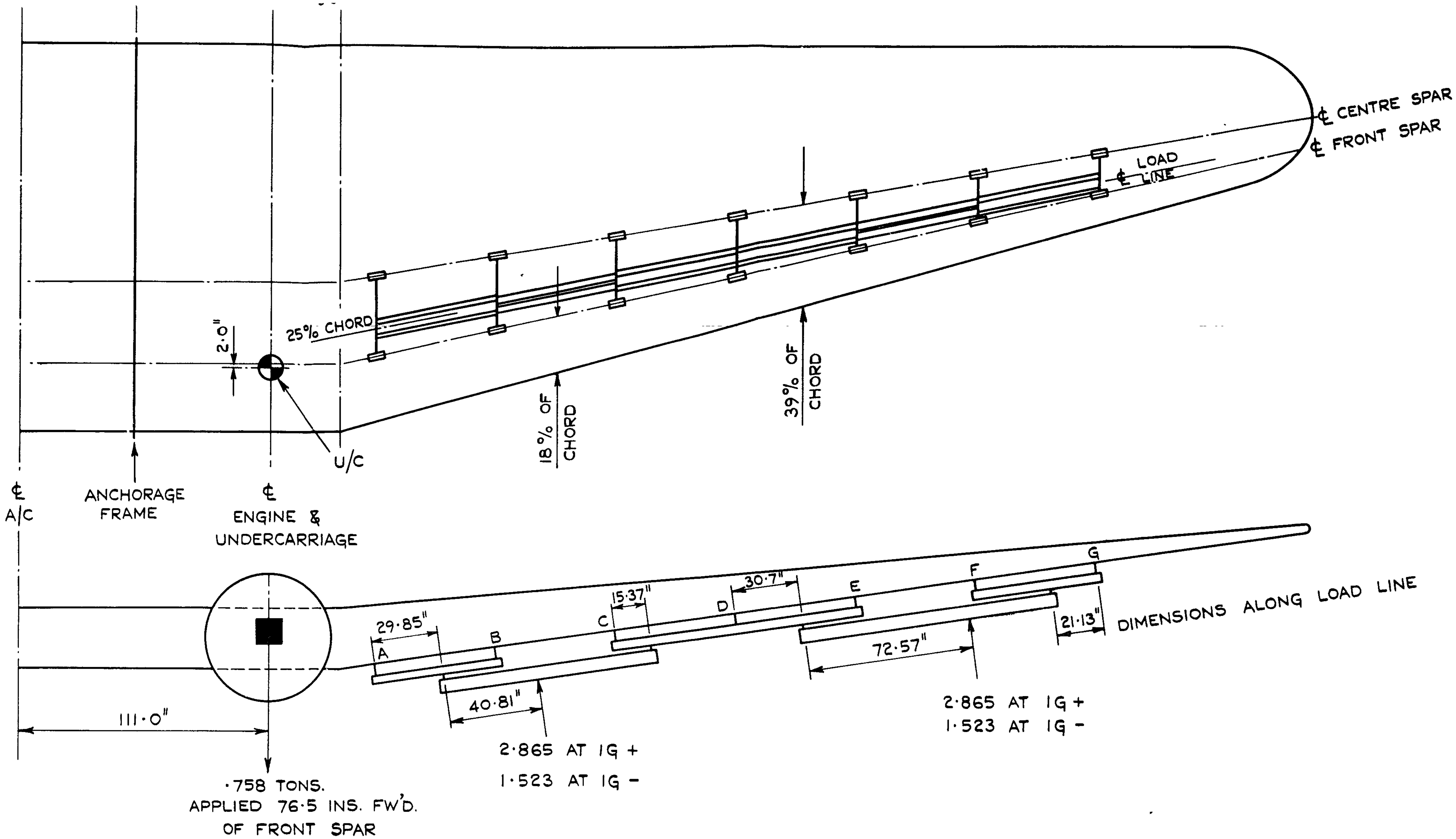


FIG.2. WING TORQUE DIAGRAM.



NOTE:- TO OBTAIN SLIGHT COMPRESSIVE STRESS IN THE BOTTOM OF THE WING CENTRE-SECTION THE 'ON GROUND' TEST LOADING WAS ARRANGED SO THAT THE OUTER WINGS CARRIED TWICE THEIR OWN WEIGHT.

FIG. 3 WING BENDING MOMENT DIAGRAM



STATION	A	B	C	D	E	F	G
DIST. FROM ℄ A/C (INS)	158	211	264	317	370	423	476
LOAD AT 1G − 12 FT/SEC. (TONS)	·387	·474	·474	·418	·302	·385	·605
LOAD AT 1G + 12 FT/SEC. (TONS)	·727	·892	·892	·788	·568	·725	1·139

DIMENSIONS NORMAL TO ℄ A/C

SCALE : $\frac{1}{4}$" = 1'

FIG. 4. LEVER SYSTEM & APPLIED LOADS.

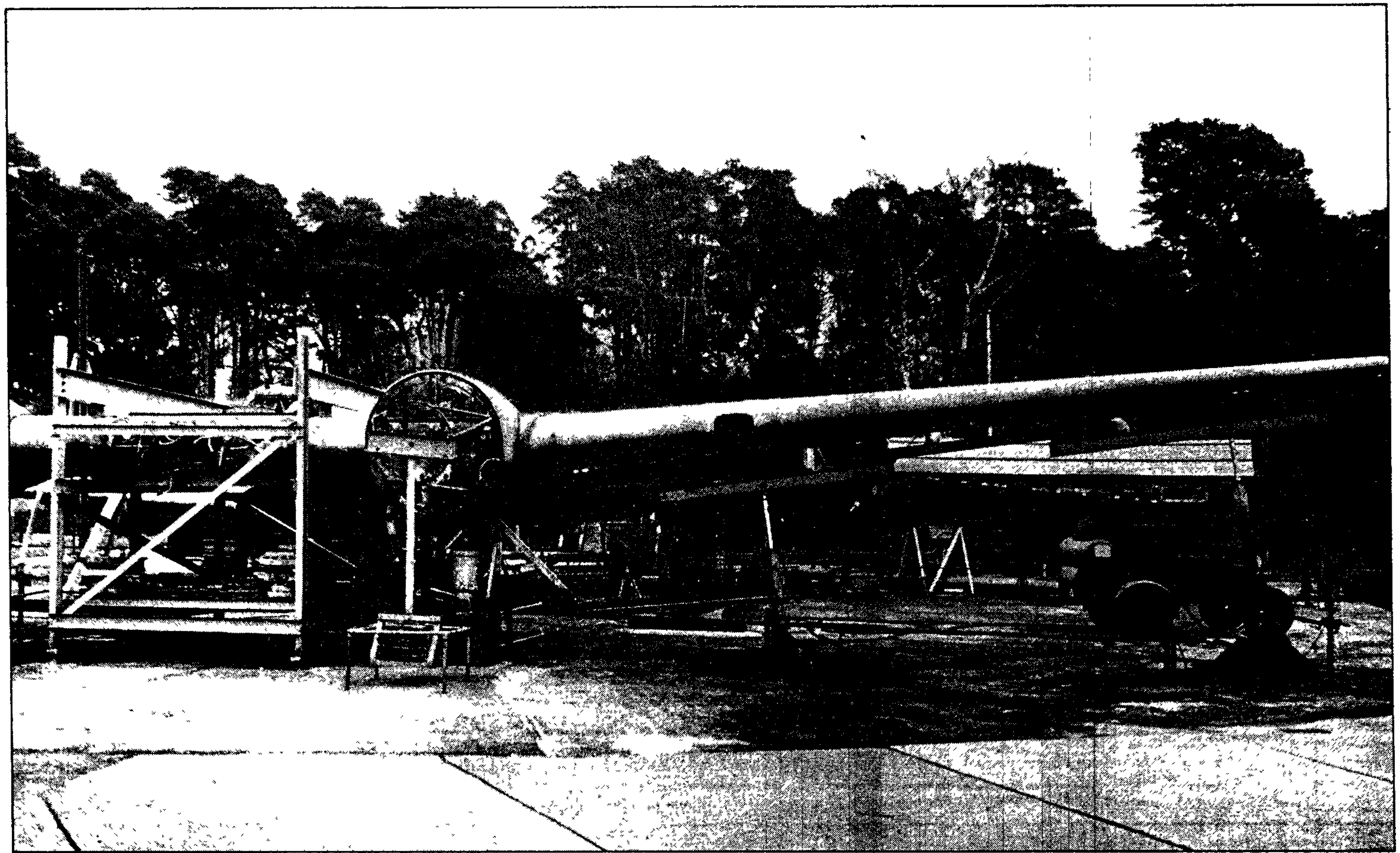
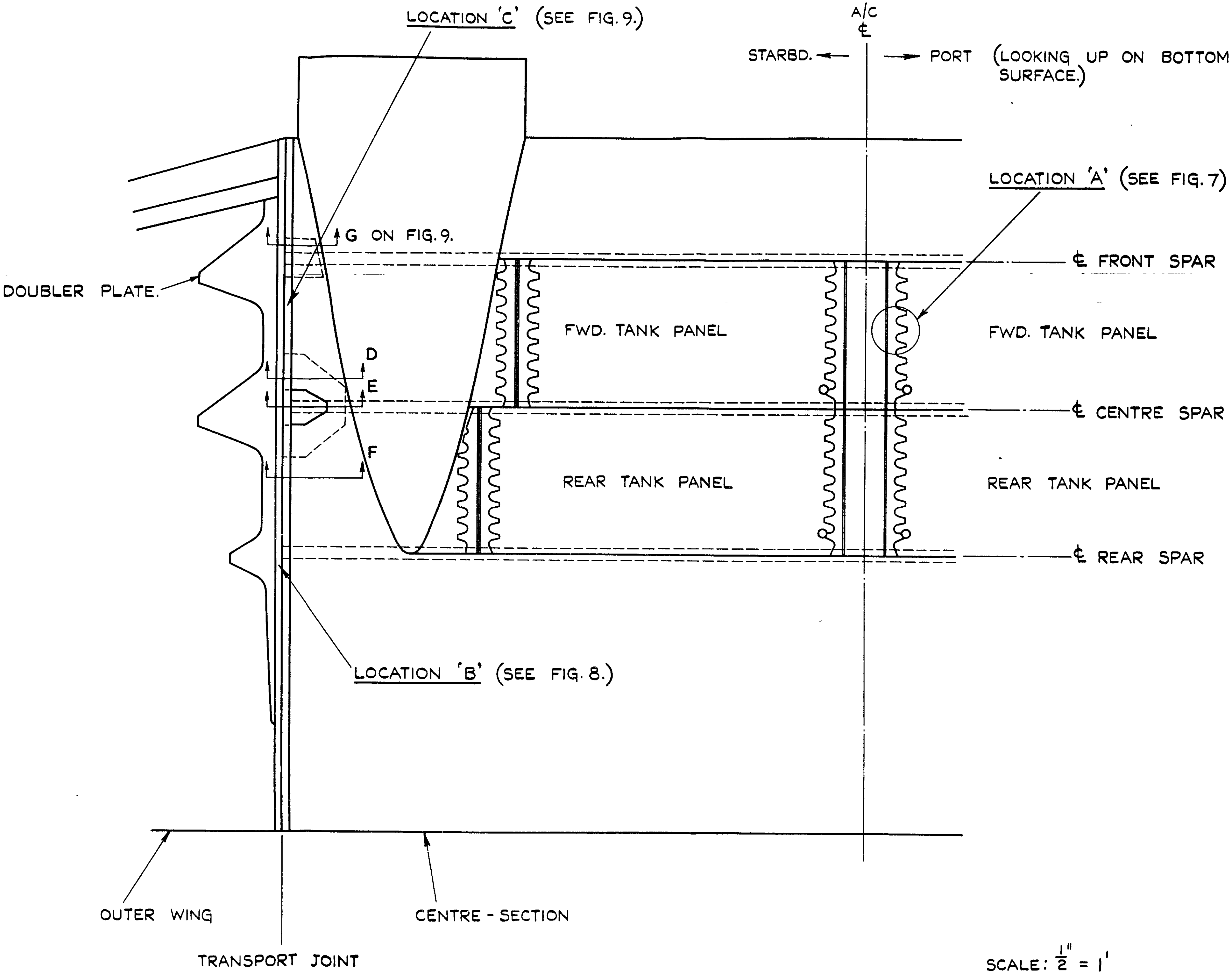


FIG.5. GENERAL VIEW OF DAKOTA WING FATIGUE TEST



SCALE: $\frac{1}{2}'' = 1'$

FIG. 6. THE LOCATIONS OF FATIGUE DAMAGE ON THE BOTTOM SURFACE OF THE WING.

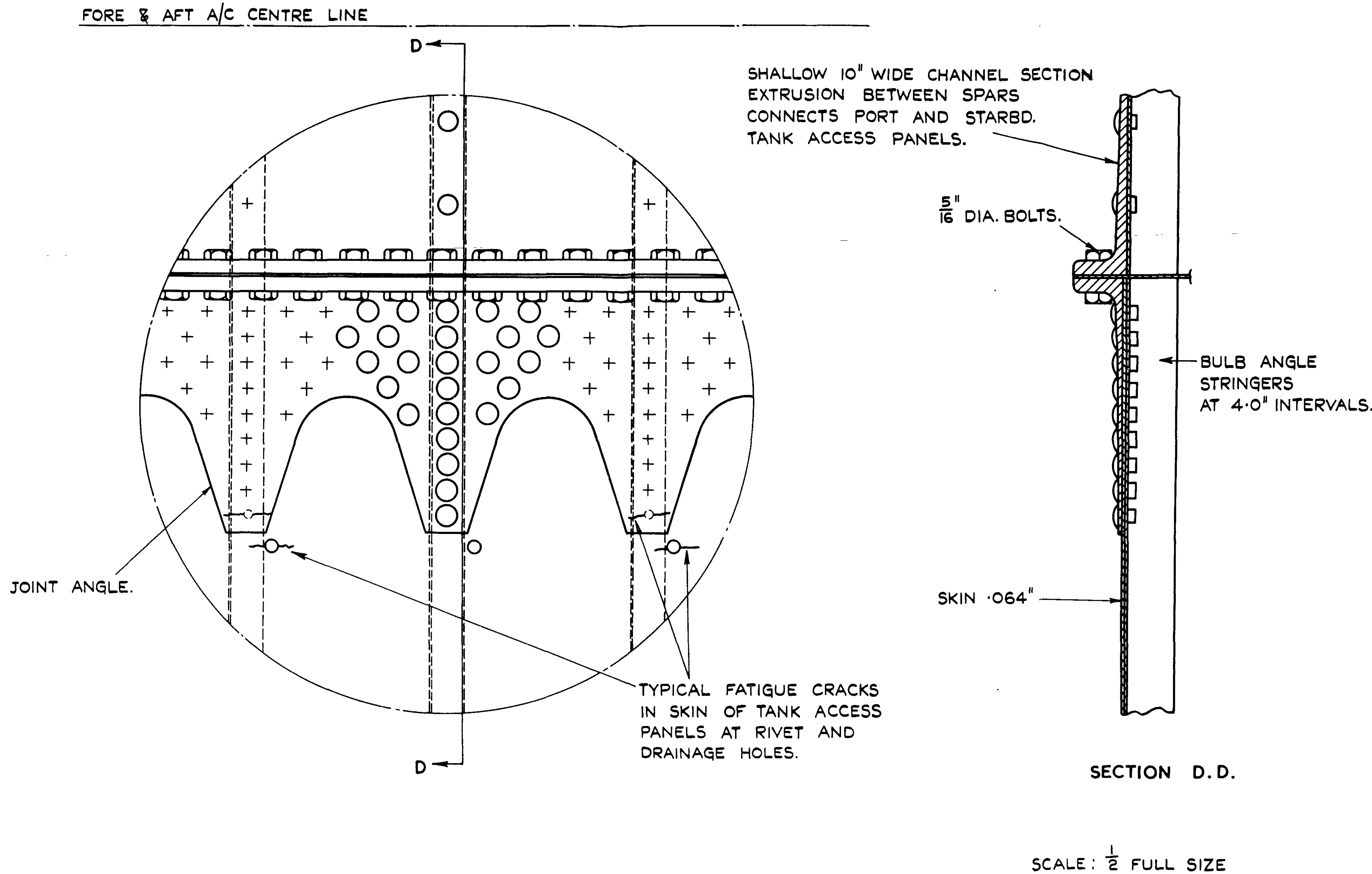
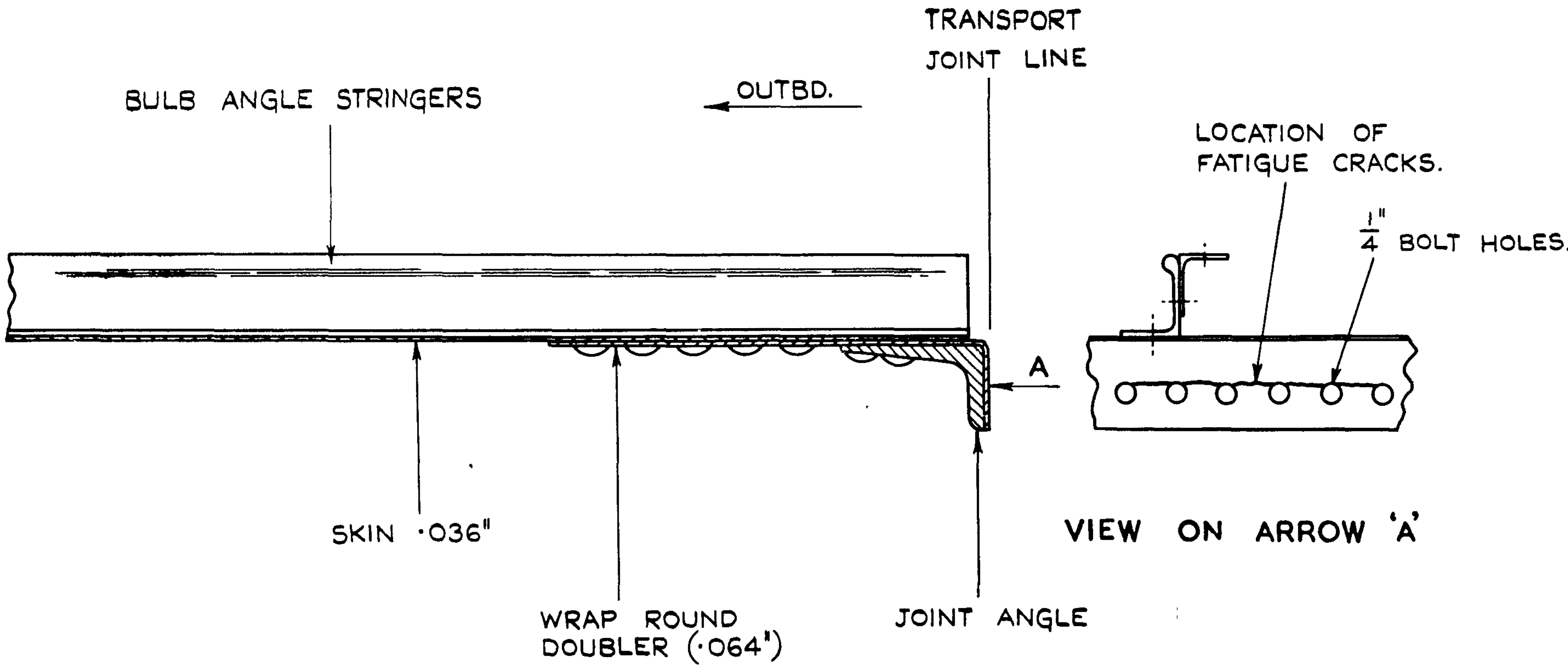


FIG. 7. TYPICAL FATIGUE DAMAGE AT THE INBOARD ENDS OF THE REMOVABLE FUEL TANK ACCESS PANELS IN THE CENTRE SECTION - LOCATION A.



SCALE: 1/2 FULL SIZE

FIG. 8. TYPICAL FATIGUE DAMAGE AT THE OUTER WING SIDE OF THE TRANSPORT JOINT - LOCATION B.

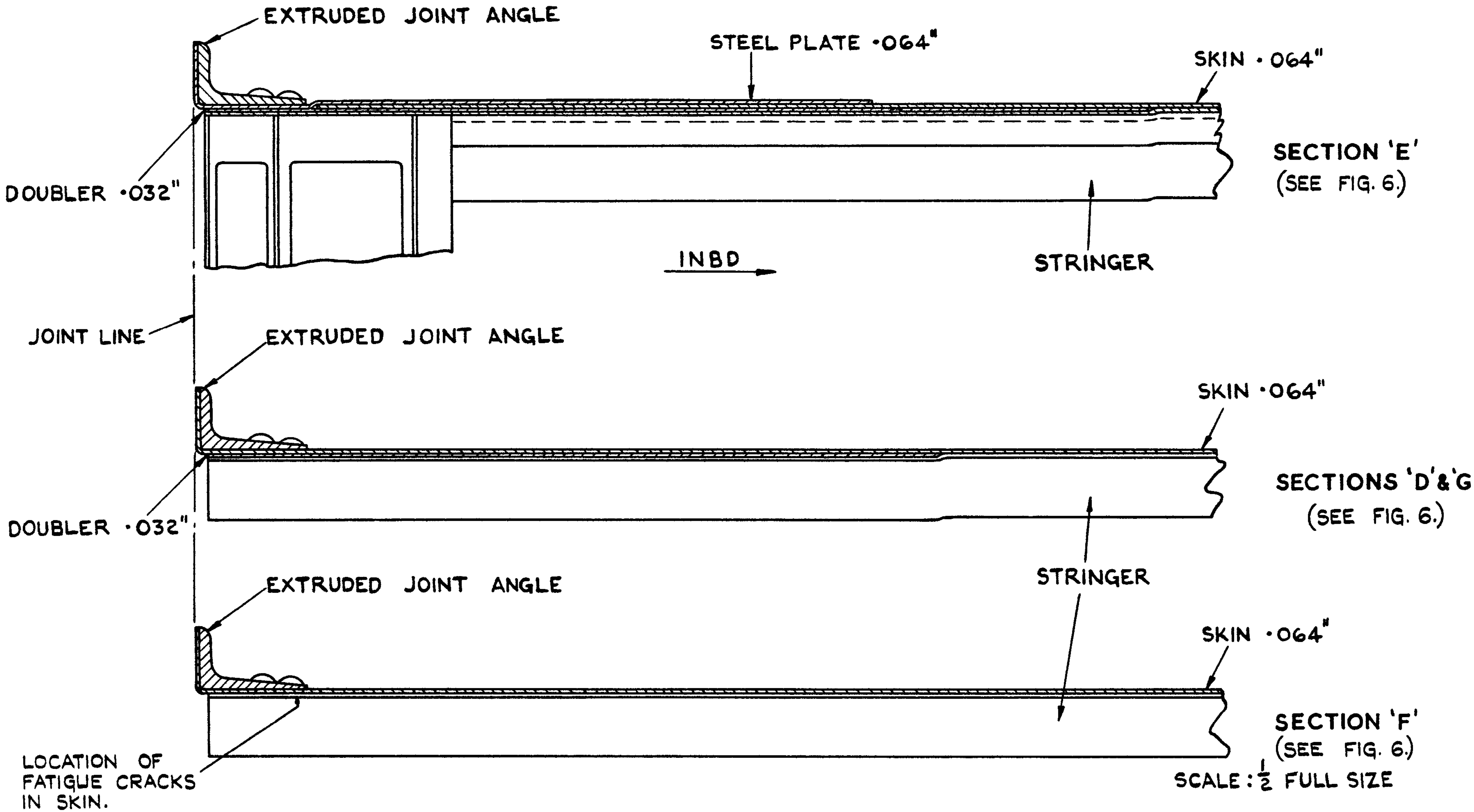


FIG. 9. DETAILS OF WING STRUCTURE AT THE CENTRE-SECTION SIDE OF THE TRANSPORT JOINT - LOCATION C.

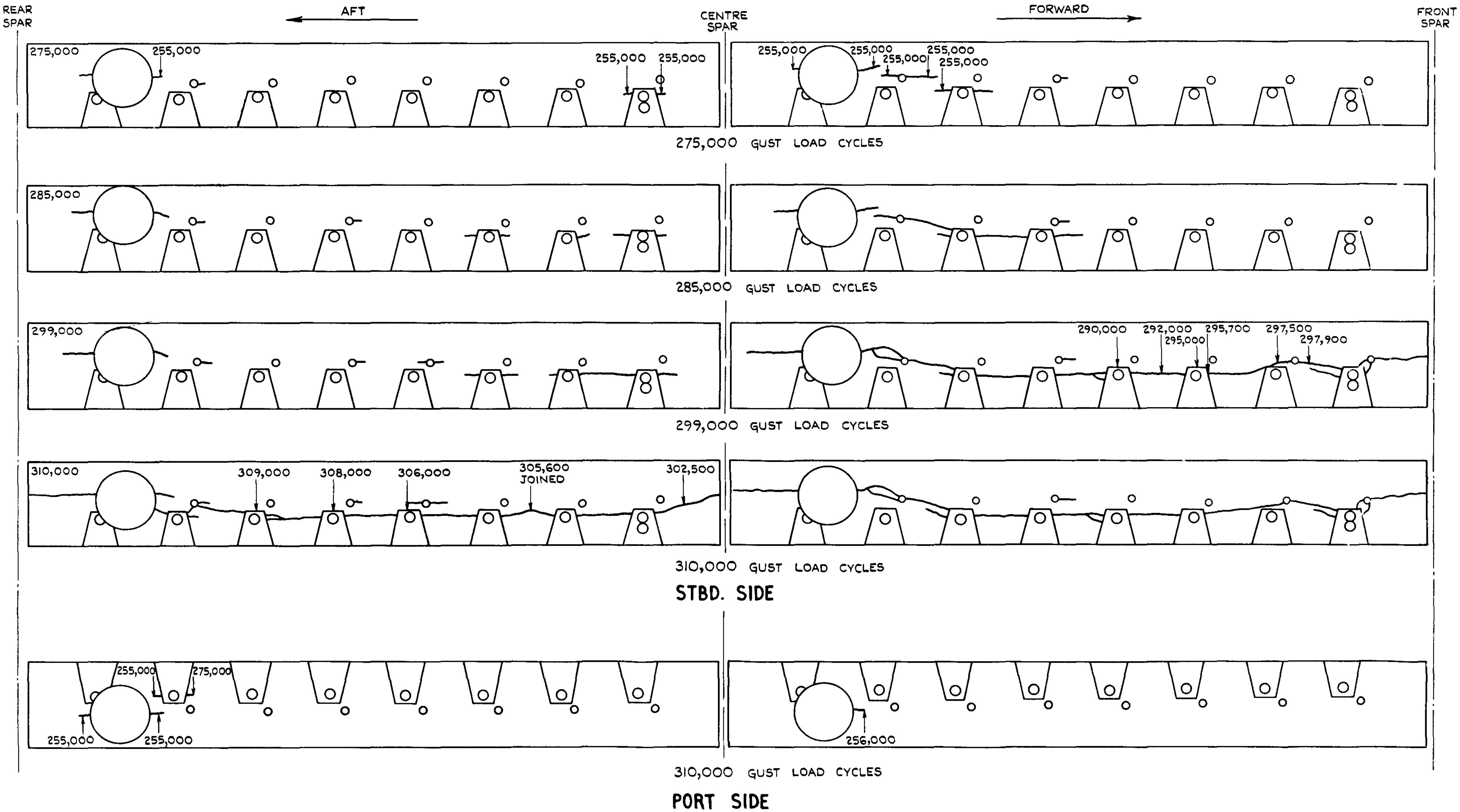
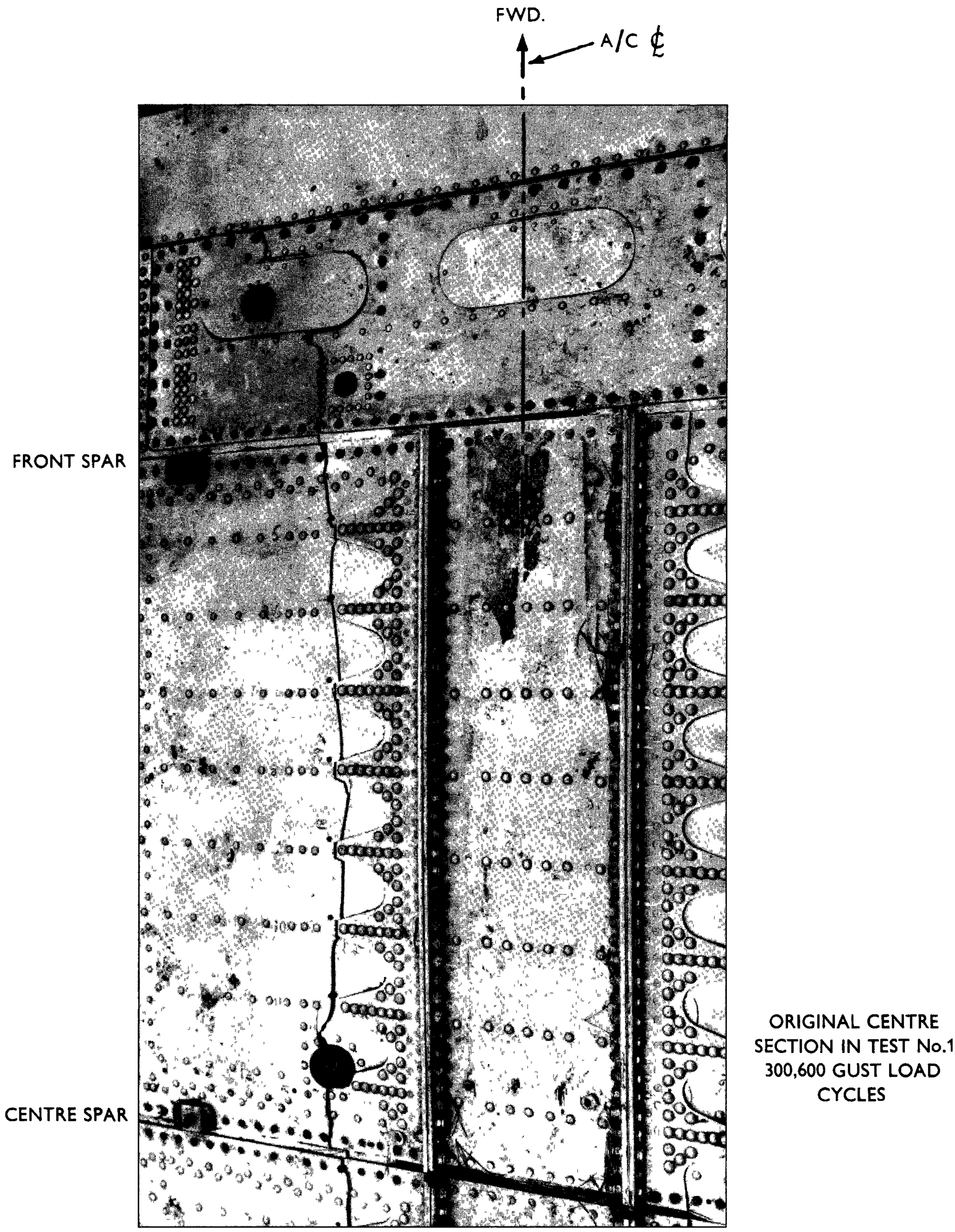


FIG. 10. DETAILS OF SKIN CRACKS AT LOCATION 'A'.



NOTE:-

PHOTO AT END OF TEST WHEN THE INDIVIDUAL CRACKS WHICH HAD ORIGINATED AT RIVET AND DRAIN HOLES HAD FINALLY JOINED TO PRODUCE FRACTURE OF THE ACCESS PANEL

FIG.11. ENLARGED VIEW OF FATIGUE FAILURES IN THE FORWARD ACCESS PANEL IN THE WING CENTRE SECTION

(LOCATION 'A' ON FIGS.6 and 7)

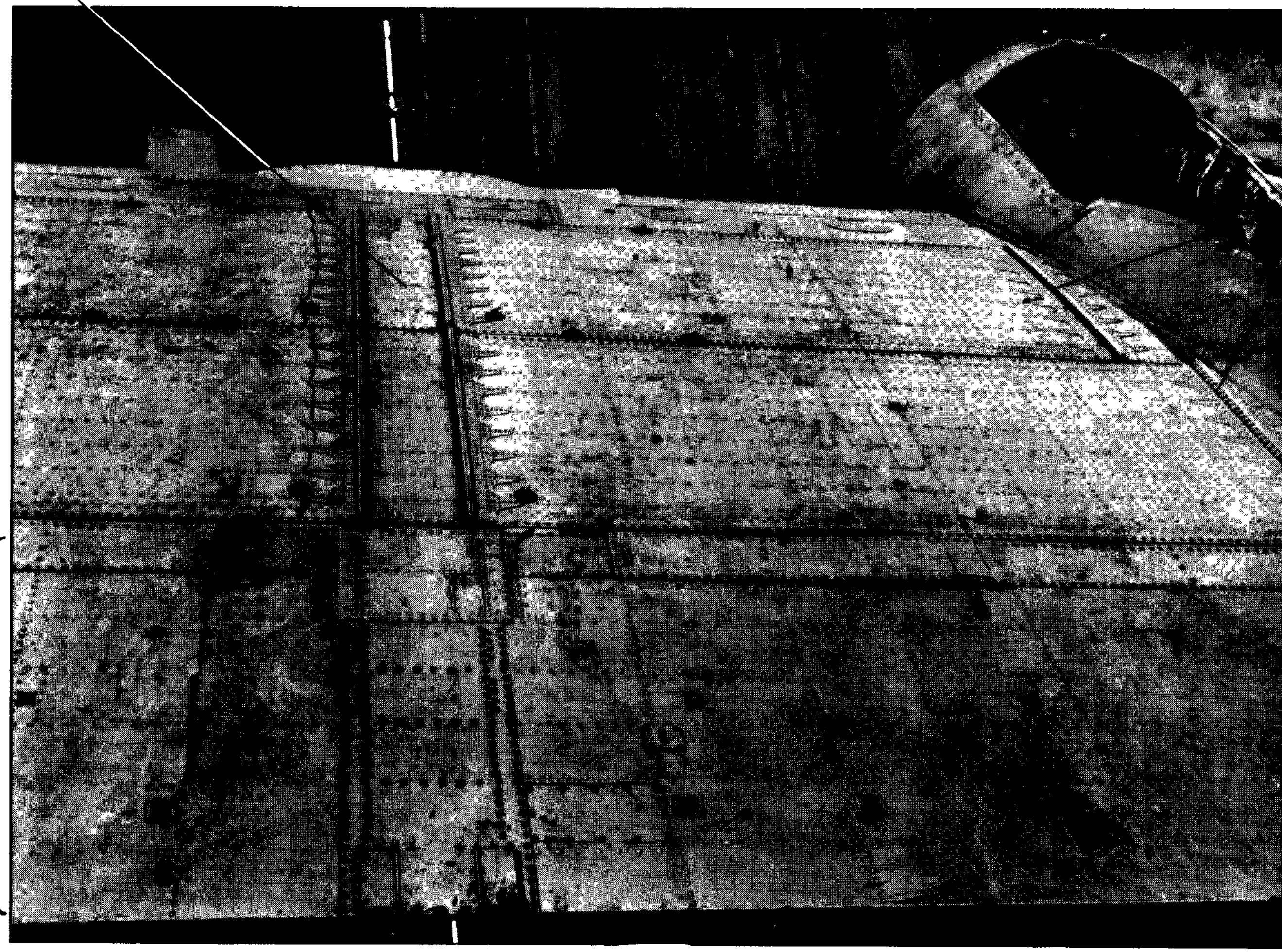


FIG.12. BOTTOM SURFACE OF CENTRE SECTION SHOWING TANK ACCESS PANEL FATIGUE FAILURES

(LOCATION 'A' ON FIGS.6 and 7)

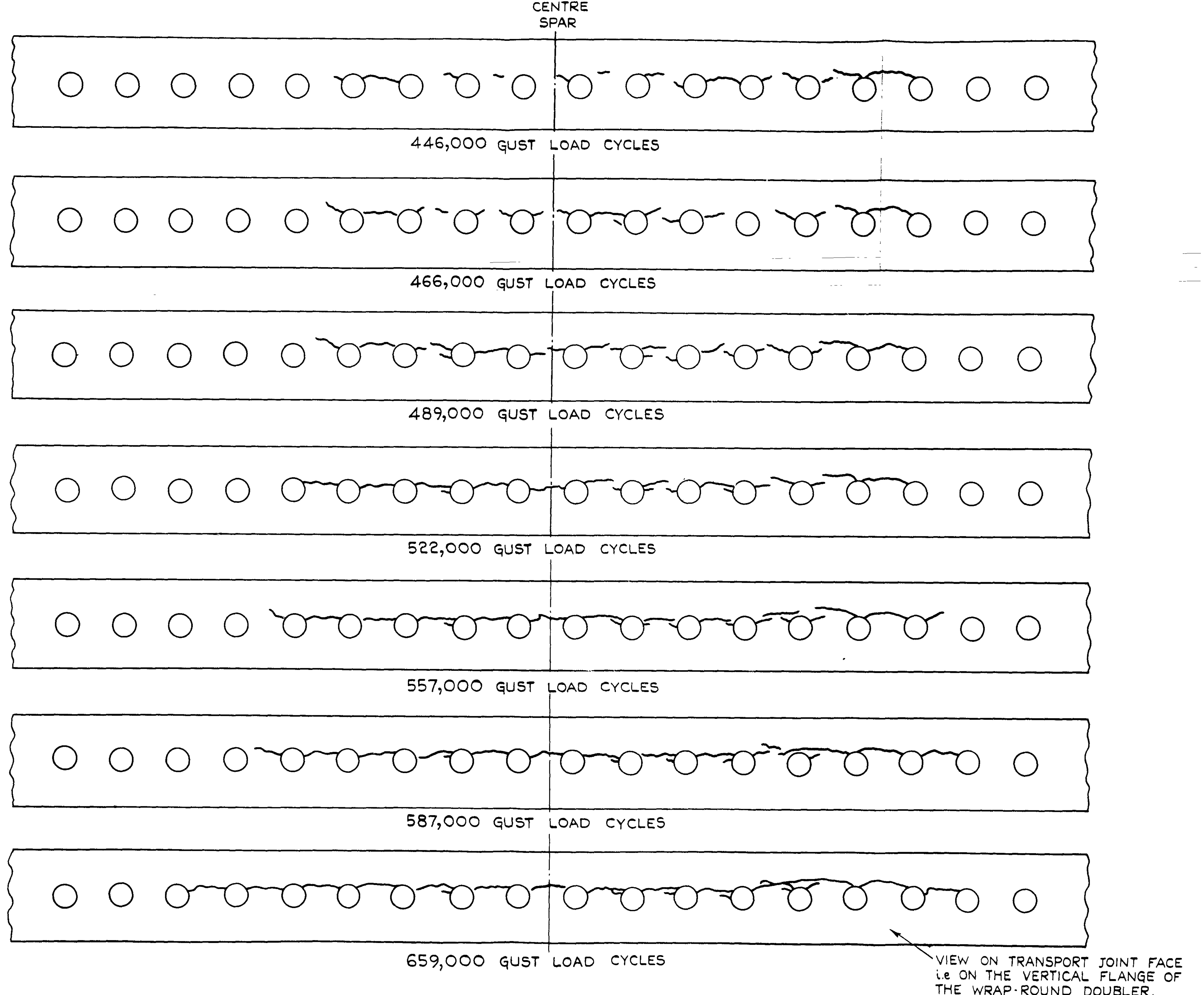
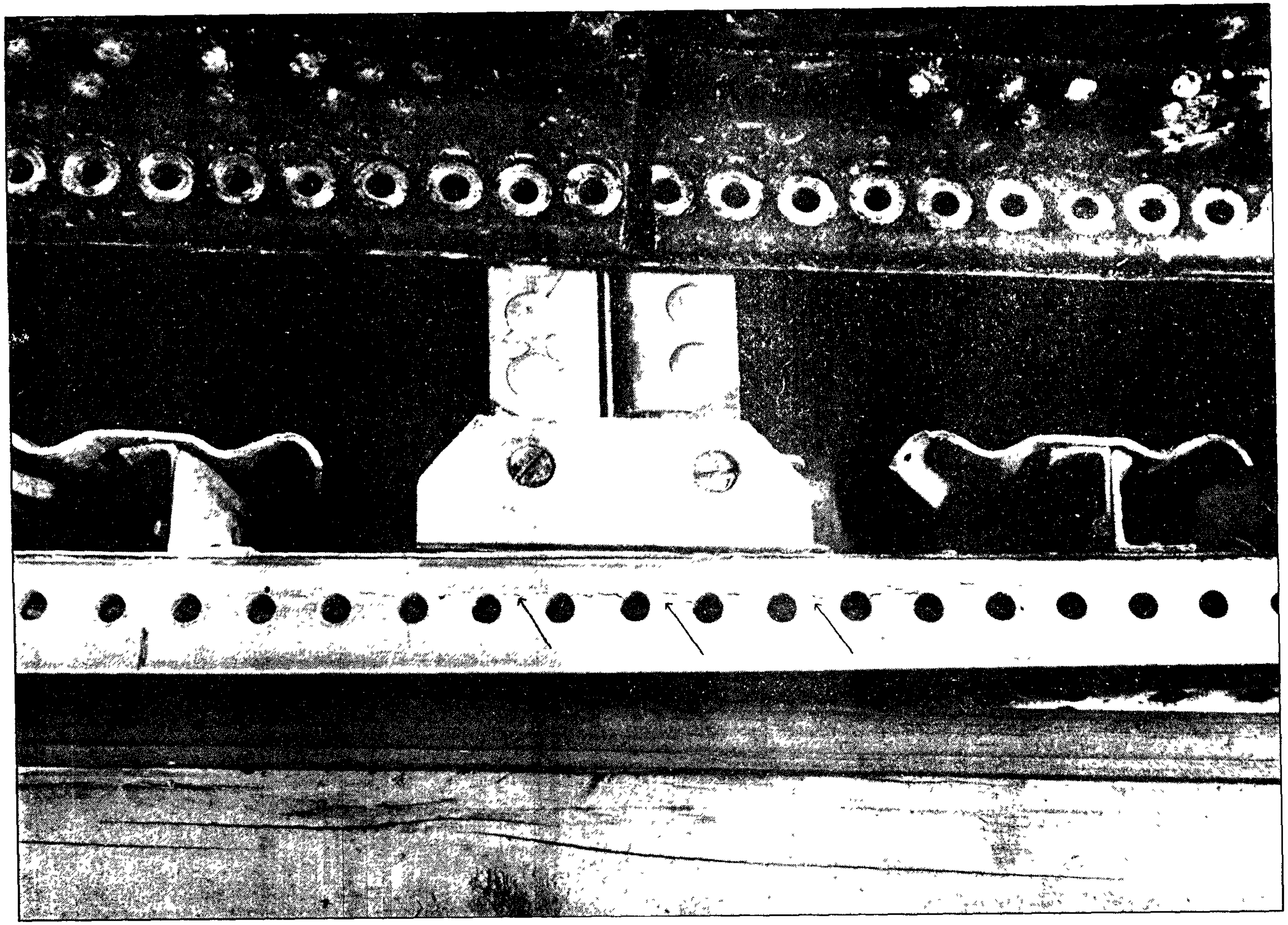
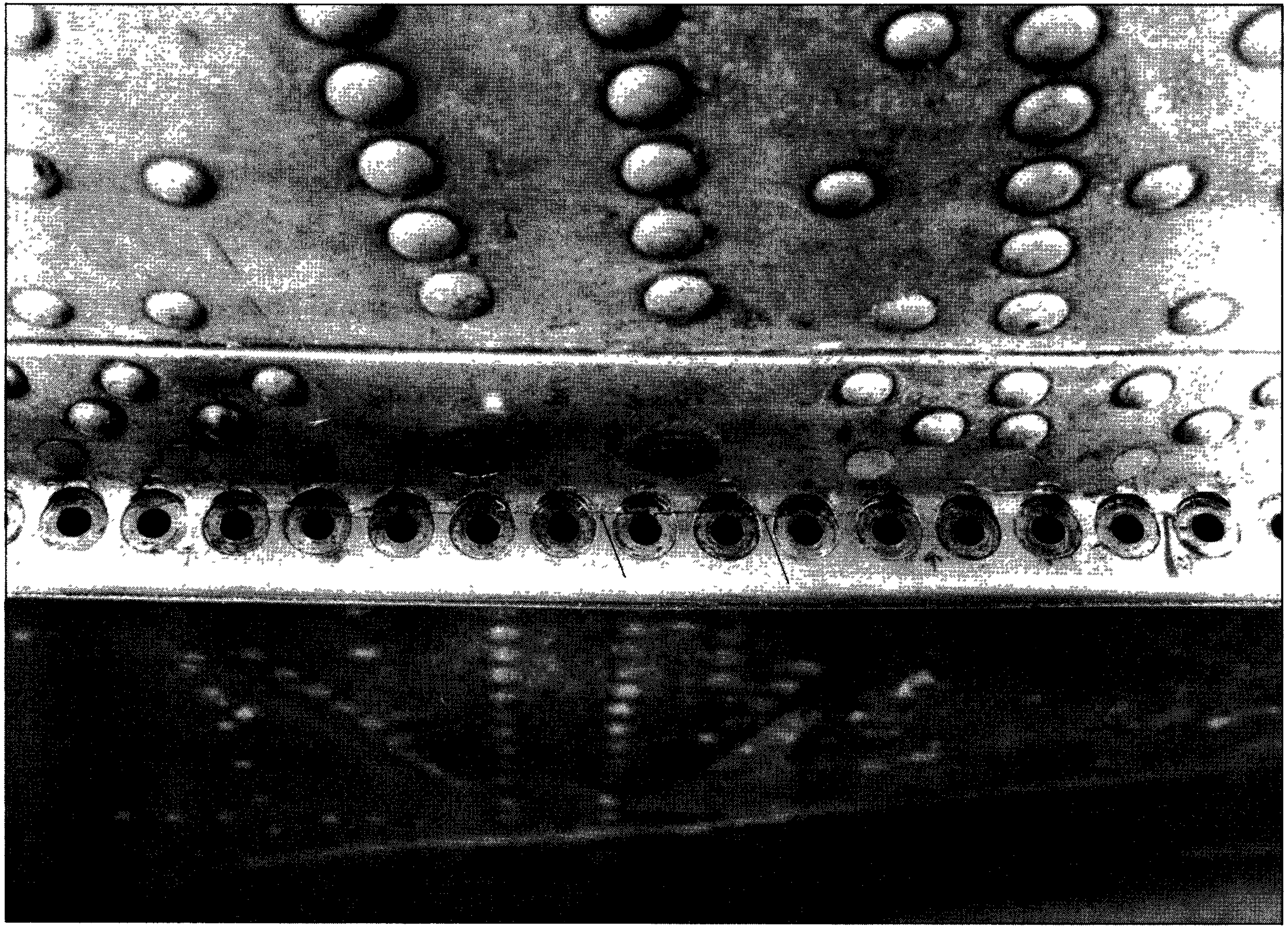


FIG. 13. GROWTH OF FATIGUE CRACKS AT LOCATION 'B'
STARBOARD OUTER WING IN TEST No. 1.



STARBOARD OUTER WING
IN TEST No.1
CENTRE SPAR POSITION
PHOTO TAKEN AT 587,000
GUST LOAD CYCLES

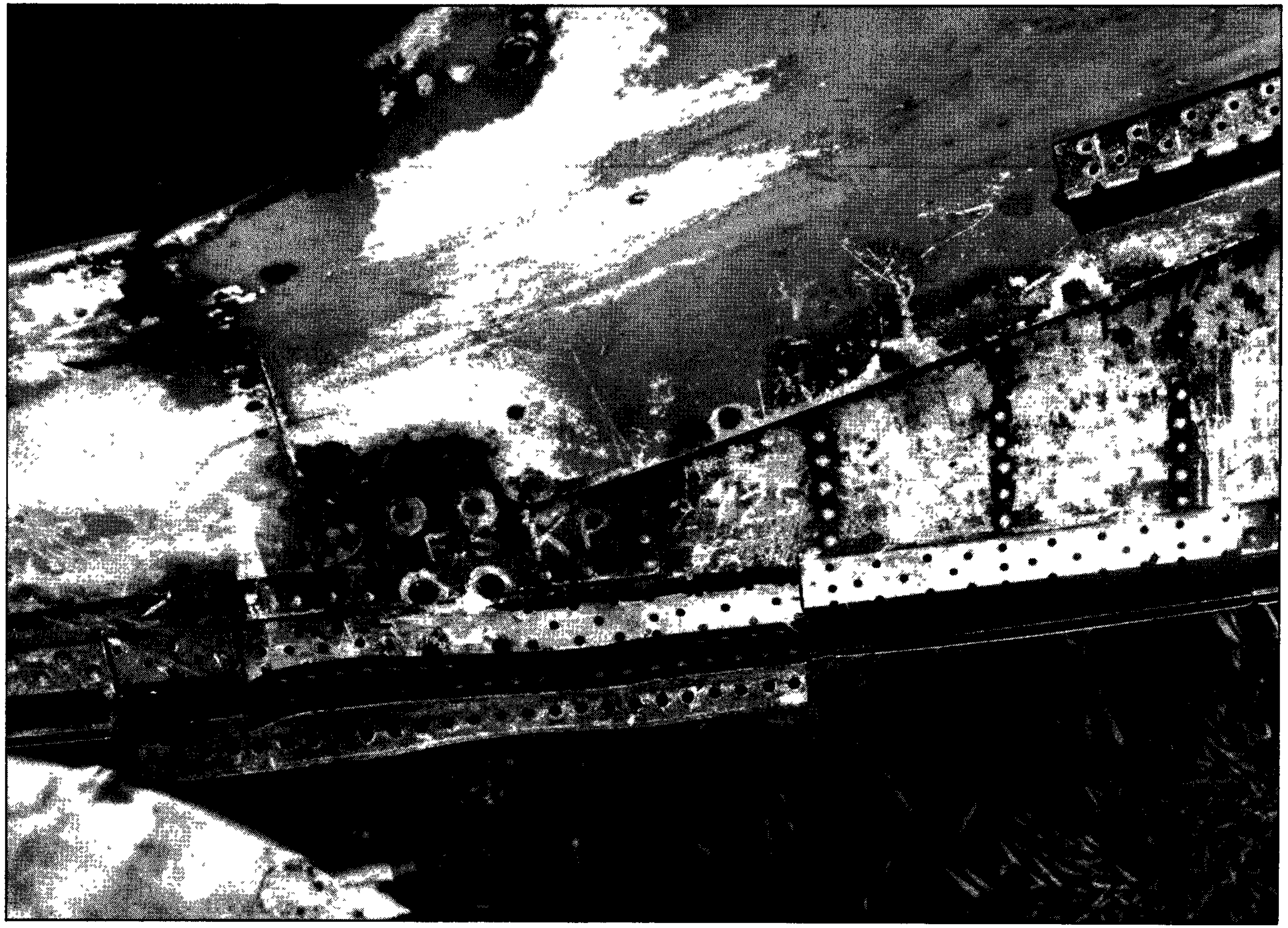
FIG.14. FATIGUE DAMAGE IN THE 16 S.W.G. WRAP-ROUND DOUBLER AT THE TRANSPORT JOINT (OUTER WING SIDE)
(*LOCATION 'B' ON FIGS.6 and 8*)



STARBOARD OUTER WING
IN TEST No.1
659,000 GUST LOAD CYCLES
AT CENTRE SPAR POSITION

FIG.15. FATIGUE CRACK IN THE VERTICAL FLANGE OF THE EXTRUDED JOINT ANGLE AT THE TRANSPORT JOINT (OUTER WING SIDE)

(LOCATION 'B' ON FIG.6)



REPLACEMENT CENTRE SECTION
IN THE FIRST TEST.
346,000 GUST LOAD CYCLES
SKIN CRACKS UNDER THE TRANSPORT
JOINT ANGLE IN THE REGION OF
THE FRONT SPAR

FIG.16. FATIGUE DAMAGE IN THE CENTRE SECTION AT THE TRANSPORT JOINT

EXTRUDED JOINT ANGLE REMOVED TO SHOW FATIGUE CRACK IN SKIN (LOCATION "C" ON FIG.6)

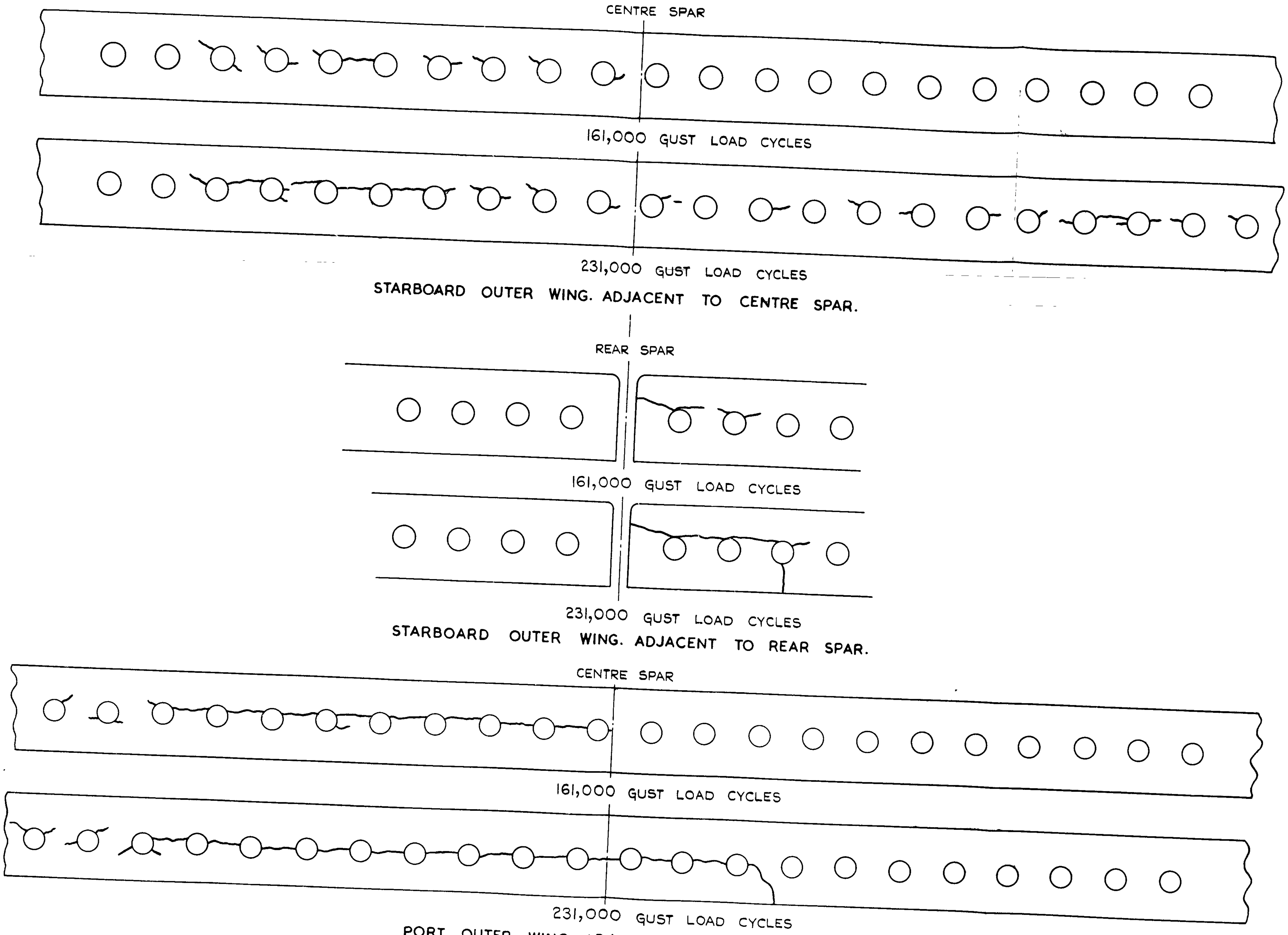


FIG. 17. GROWTH OF FATIGUE CRACKS AT LOCATION 'B' IN TEST No. 2.

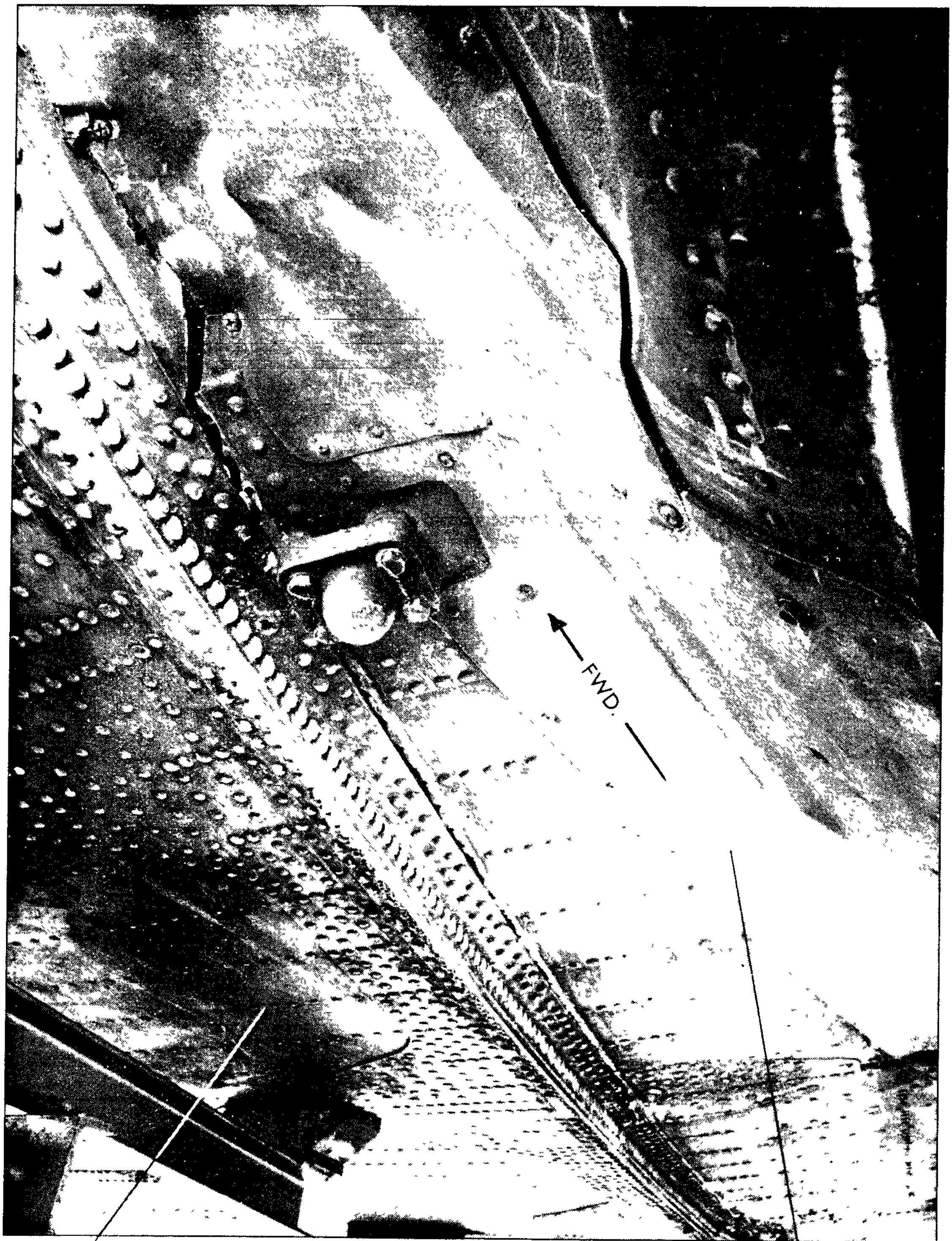


FIG. 18. FAILURE IN BOTTOM SKIN OF CENTRE-SECTION AT THE TRANSPORT JOINT

(LOCATION 'C' - TEST No. 3)

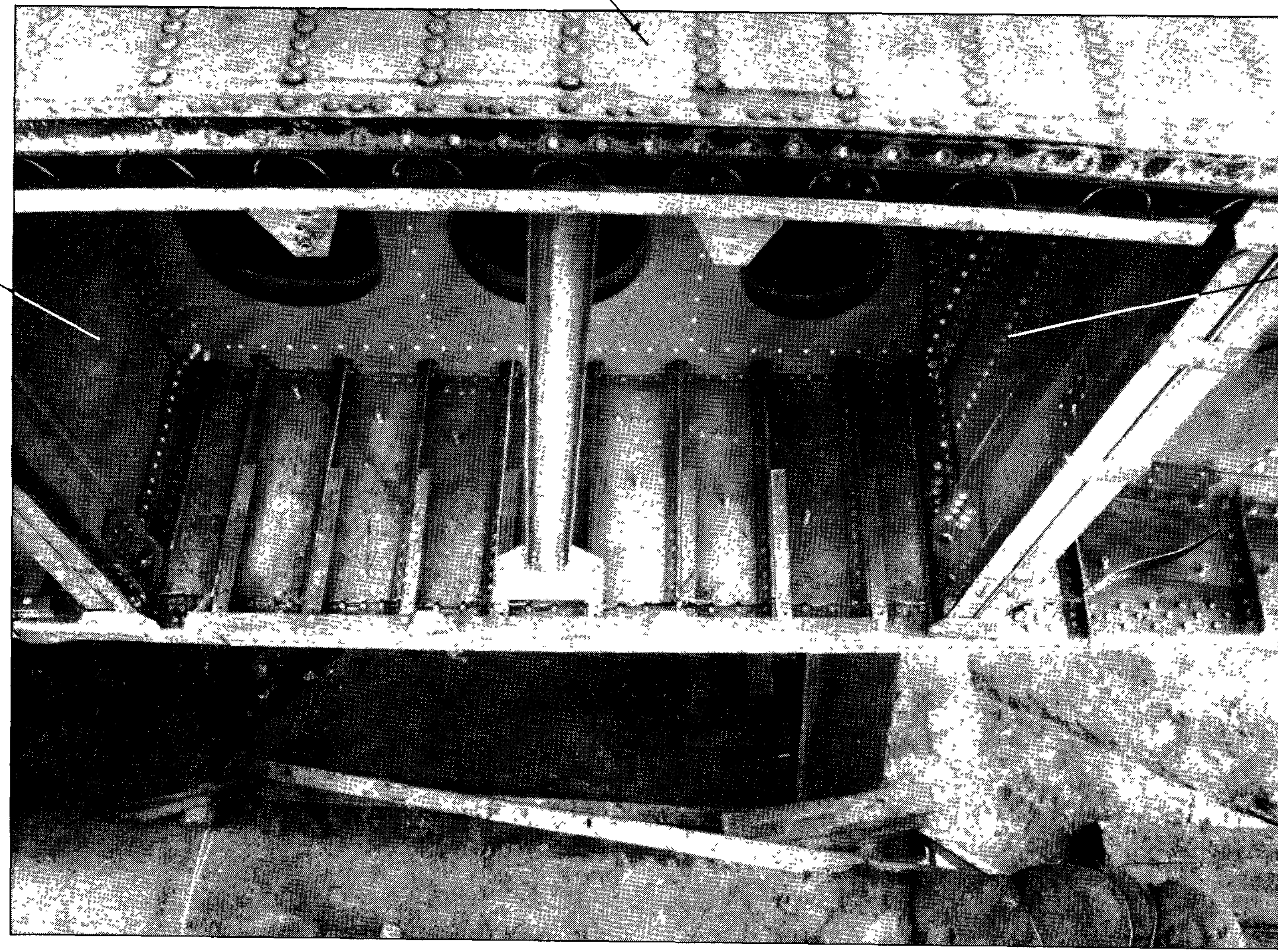


FIG.19. FAILURE OF BOTTOM SKIN OF CENTRE-SECTION AT THE TRANSPORT JOINT

(*LOCATION 'C' - TEST No.3*)

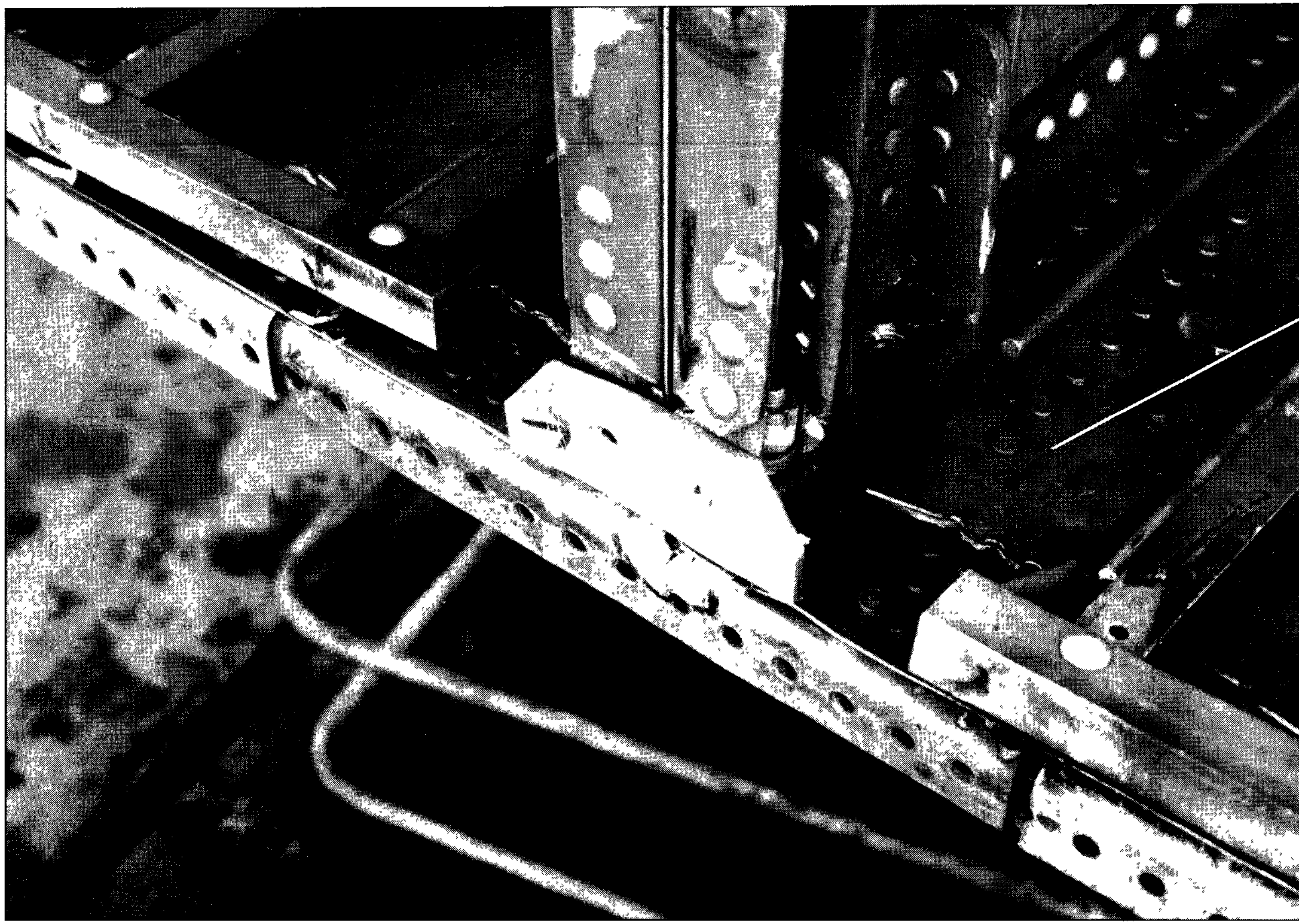
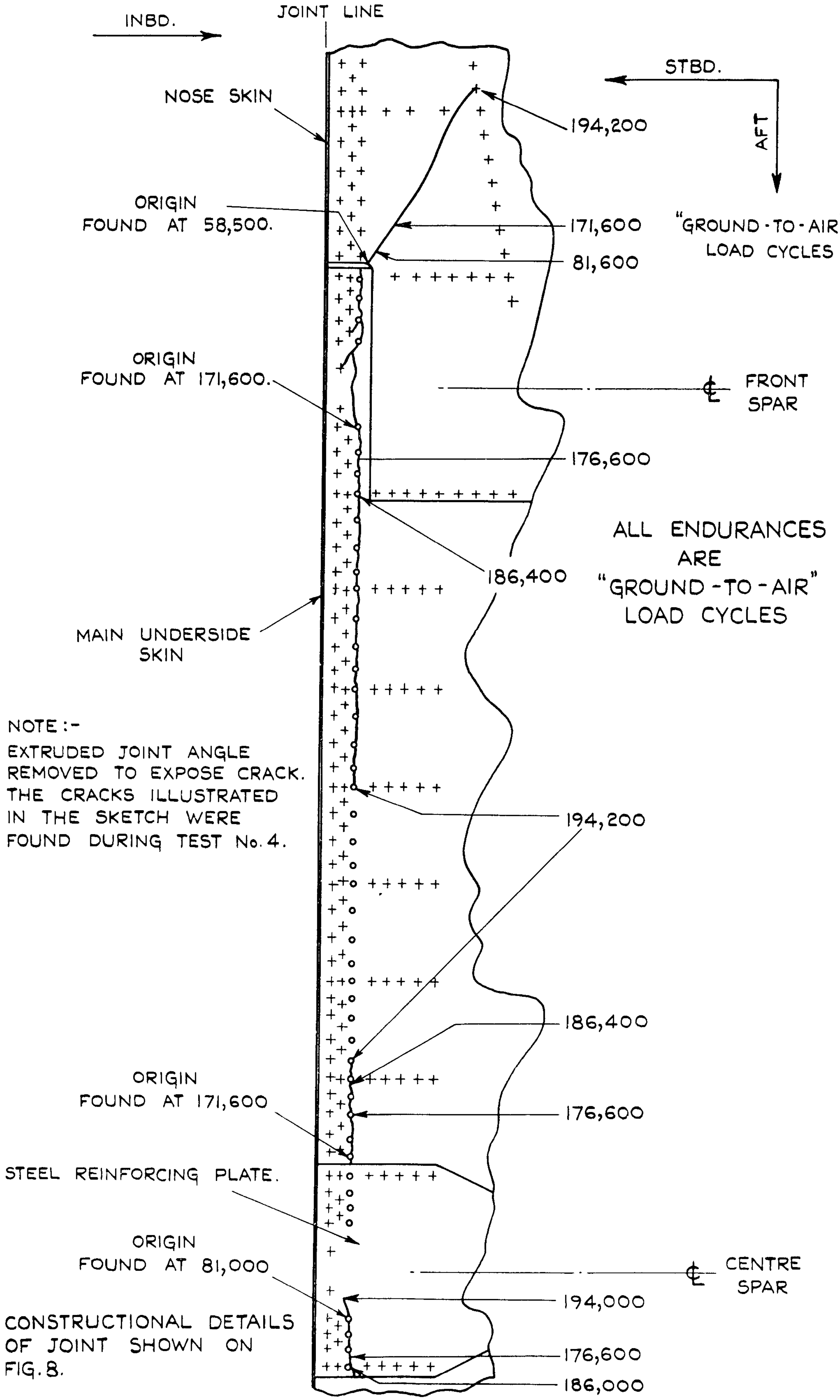


FIG.20. FAILURE OF BOTTOM SKIN OF CENTRE-SECTION AT THE TRANSPORT JOINT

(LOCATION 'C' - TEST No.3)



INBD.
JOINT LINE
STBD.
AFT
NOSE SKIN
194,200
ORIGIN FOUND AT 58,500.
171,600
"GROUND-TO-AIR" LOAD CYCLES
81,600
ORIGIN FOUND AT 171,600.
FRONT SPAR
176,600
ALL ENDURANCES ARE "GROUND-TO-AIR" LOAD CYCLES
186,400
MAIN UNDERSIDE SKIN
NOTE:- EXTRUDED JOINT ANGLE REMOVED TO EXPOSE CRACK. THE CRACKS ILLUSTRATED IN THE SKETCH WERE FOUND DURING TEST No. 4.
194,200
186,400
ORIGIN FOUND AT 171,600
176,600
STEEL REINFORCING PLATE.
ORIGIN FOUND AT 81,000
CENTRE SPAR
194,000
CONSTRUCTIONAL DETAILS OF JOINT SHOWN ON FIG. 8.
176,600
186,000

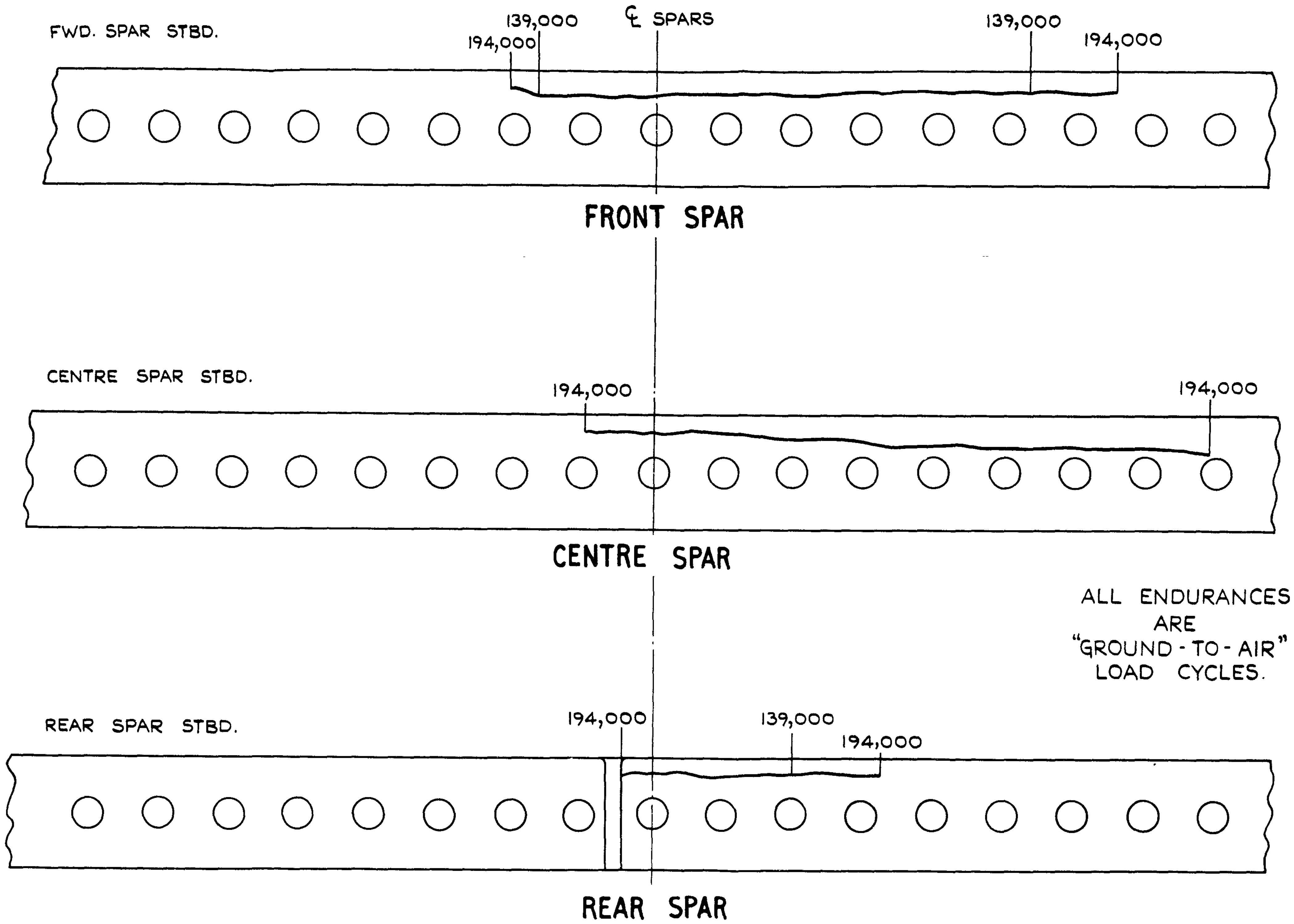


FIG. 22. GROWTH OF FATIGUE CRACKS AT LOCATION 'B' IN TEST No. 4.

A.R.C. C.P. No. 666

539.431 :
533.6.048.5 :
629.13.012.6 :
620.178.3 :
[A1] (73) Dakota

FATIGUE BEHAVIOUR UNDER SERVICE AND GROUND TEST CONDITIONS (A COMPARISON BASED ON THE DAKOTA WING) Winkworth, W.J. November, 1961

Several Dakota wings have been tested in fatigue under various conditions. The objective was to obtain an assessment of full scale testing procedures.

When the ground-to-air loading actions are represented in proper relation to those of atmospheric turbulence, the behaviour in test is found to be similar to that in service.

A.R.C. C.P. No. 666

539.431 :
533.6.048.5 :
629.13.012.6 :
620.178.3 :
[A1] (73) Dakota

FATIGUE BEHAVIOUR UNDER SERVICE AND GROUND TEST CONDITIONS (A COMPARISON BASED ON THE DAKOTA WING) Winkworth, W.J. November, 1961

Several Dakota wings have been tested in fatigue under various conditions. The objective was to obtain an assessment of full scale testing procedures.

When the ground-to-air loading actions are represented in proper relation to those of atmospheric turbulence, the behaviour in test is found to be similar to that in service.

A.R.C. C.P. No. 666

539.431 :
533.6.048.5 :
629.13.012.6 :
620.178.3. :
[A1] (73) Dakota

FATIGUE BEHAVIOUR UNDER SERVICE AND GROUND TEST CONDITIONS (A COMPARISON BASED ON THE DAKOTA WING) Winkworth, W.J. November, 1961

Several Dakota wings have been tested in fatigue under various conditions. The objective was to obtain an assessment of full scale testing procedures.

When the ground-to-air loading actions are represented in proper relation to those of atmospheric turbulence, the behaviour in test is found to be similar to that in service.

D 82973/1/Wt 59 K4 11/63 R

C.P. No. 666

Printed and published by
HER MAJESTY'S STATIONERY OFFICE
To be purchased from
York House, Kingsway, London w.c.2
423 Oxford Street, London w.1
13A Castle Street, Edinburgh 2
109 St. Mary Street, Cardiff
39 King Street, Manchester 2
50 Fairfax Street, Bristol 1
35 Smallbrook, Ringway, Birmingham 5
80 Chichester Street, Belfast 1
or through any bookseller

Printed in England

C.P. No. 666

S.O. Code No. 23-9013-66