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The Fatigue Strength Characteristics of a Single Spar Wing

by

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THE FATIGUE STRENGTH CHARACTERISTICS OF A SINGLE SPAR WING

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SUMMARY

Full scale fatigue tests have been made on the wings of the Vickers Varsity aircraft, in which the conditions of atmospheric turbulence and the transition from ground to air were represented in the fatigue loading programme.

Major failures from fatigue occurred in the spar tension boom in both the inner and outer portions of the wing.

The test results illustrate the fatigue behaviour of single spar wing structures.

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INTRODUCTION

In fatigue testing the Vickers Varsity wing structure, successive repairs and replacements enabled the tests to be continued until failures had occurred in both the inner and outer portions of each wing. Although the tests were made to obtain data for Varsity fatigue life assessment, they also provided an opportunity for observing the fatigue behaviour of a single spar wing structure. The purpose of this Note is to discuss the fatigue behaviour of this particular wing structure and to use the results of the tests to illustrate features of general applicability to single spar wing structures.

The prime importance of the tension boom in such structures was demonstrated by the occurrence of total failure each time this member fractured at a fatigue crack. Visual and radiographic inspection methods were inadequate for finding the fatigue cracks in the tension boom, and the major failures therefore occurred without warning. These are characteristics of single concentrated spar structures which must be taken into account when assessing the permissible fatigue life for service flying.

2 FATIGUE TEST LOADING CONDITIONS

The programme of cyclic loads represented the conditions of atmospheric turbulence met in flight and the transition from ground to air. A simulated take-off was followed by the application of cyclic loads representing eighteen ± 10 ft per sec gusts and finally a simulated landing. This load programme was repeated every $1\frac{1}{2}$ minutes and was considered to represent a 33 minute flight that gave an average of typical usage in service, as shown in Appendix 1.

The bending moments, shears and test loads are given in Figures 2, 3, and 4, and a comparison with measurements made in flight[1] is presented in Figure 5. A brief description of the method of applying the test loads is given in Appendix 2.

3 THE WING STRUCTURE

The Varsity wing spar is located on the 30 per cent chord line. There are also two false spars, at forward and aft positions, but these are relatively weak and transmit only shear loads at the wing root. The spar is composed of tee section booms and a plate web. The booms are aluminium alloy extrusions of material specification D.T.D.363A (Appendix 3) although for one test, replacement booms of material specification B.S.L.65 (Appendix 3) were fitted in the inner wings. The web is aluminium alloy sheet of material specification D.T.D.546 (Appendix 3).

Wing transport joints are located at the aircraft centre and at the outer sides of the engine nacelles. The "inner" wings extend between these joints, the portions beyond the nacelles being termed the "outer" wings.

The wing span is 95 feet.

The wing skins, which are discontinuous at the outer transport joints and at the fuselage side, are conventionally supported by ribs, and are attached to the spar booms by bolts and/or rivets at 12-inch intervals.

Basic features of the structure of the lower wing surface are shown on Figure 1.

4 THE FATIGUE TESTS

Throughout the tests, eight major failures occurred; all of them originated from fatigue cracks in the spar tension boom.

The original wing to be tested, first failed at the port side of the fuselage. The fatigue crack in the boom was considered to have been influenced by local loads diffusing from the skin which terminated at this point. After repairing this failure, the test was resumed and shortly afterwards a similar failure occurred at the starboard side.

At this juncture, the inner wings were fitted with new tension booms which were similar to the original members except for a change in material; a copper aluminium alloy (B.S.L.65) was used in place of the previous zinc aluminium alloy, (D.T.D. 363A). On resuming the fatigue test, failure again occurred at the fuselage side as in the earlier test.

During the third stage of the test similar failures were produced at both port and starboard outer wing joints. On fitting replacement outer wings, steps were taken to prevent the repetition of earlier failures, and in the fourth test, the fractures in the outer wing spar tension boom were in the boom itself and were some distance outboard of the previous joint failures.

These failures and their corresponding endurances are summarised in Table 1 and the tests are fully described in Appendix 4.

5 DISCUSSION

Although the original purpose was to obtain fatigue strength data for Varsity service life assessments, in view of the number and diversity of location of the failures, the tests provided an unusual opportunity for observing the fatigue behaviour of a single concentrated spar wing structure.

5.1 The major failures

The two principal characteristics which were noted from the eight failures were the total collapse of the wing and the apparent suddenness with which this occurred.

The sequence of events which led up to a failure were initiated by the formation of a fatigue crack at a small bolt or rivet hole in the tension boom of the spar. The crack grew until eventually the remainder of the boom section became incapable of reacting the applied loads, and failure in the static loading sense completed the cycle of events which led up to a boom fracture and consequent failure of the wing. Seven of the total of eight major failures were unanticipated because the particular fatigue cracks had not been discovered.

5.2 Inspecting for fatigue cracks

In looking for incipient fatigue damage in structures, the precept is that the earlier it can be found, the better. In this respect, visual scrutiny at frequent intervals has usually been satisfactory for finding fatigue cracks in skins and webs, and generally in all diffuse structural components made from sheet material, provided it is physically possible to see them.

The demonstration that inspection was inadequate in the Varsity fatigue tests may be considered to be typical for structures with concentrated flange spars. The reason why visual scrutiny is unsuitable on such structures is simply a matter of visibility; the vitally important spar booms are covered by skin, fittings, gusset plates, etc., and therefore cannot be seen.

Although X-ray techniques are available for examining hidden parts, these cannot be supplied invariably, nor can they be considered to be completely reliable. Two instances in the tests substantiate this attitude, first, the presence of relatively large steel fittings precluded their use at the outer wing joint, and secondly, an X-ray exposure taken at a root rib position indicated no damage, although a crack must have been present since the boom fractured very soon afterwards.

The experience in these tests thus underlines the accepted attitude that existing inspection methods are inadequate for detecting fatigue damage in concentrated spar structures. For aircraft in service, such fatigue damage would necessarily have to be discovered at a very early stage by methods of unquestioned reliability. It will therefore be seen that the limitations of inspection methods as demonstrated in the Varsity wing fatigue tests are a major influence on decisions concerning the fatigue strength requirements of structures with concentrated flange spars.

5.3 Fatigue damage in secondary structure

Fatigue damage which occurs in the secondary structure may be significant from two aspects, first, in its influence on the primary structure, and secondly, whether it could be used to predict fatigue life.

In the Varsity tests, the earlier major failures were preceded by the formation of fatigue cracks in the wing skin (Figure 6) and possibly also by fatigue damage in the front false spars. The characteristics of the skin cracks with respect to position, size, and rates of growth, indicated that they had no influence on the major structures, but failures in the tension member of the front false spar would introduce some redistribution of load which would be slightly detrimental to the main spar at the outer wing joint.

Although fatigue damage in secondary structure has often been considered in terms of its possible use as a fatigue indicator, the proposal has mostly been discounted on grounds of unreliability. In the Varsity wing tests the proximity of the relative endurances and considerations of scatter support this attitude.

5.4 Scatter of endurances

The scatter of endurances for similar failures on the port and starboard wings was very small in two of the tests and moderate in the remaining two (Figure 7). Taking an overall view of the original wing structure, six major failures occurred between 170,500 gust load cycles and 387,200 gust load cycles. The numerous fatigue cracks - some in an advanced stage of growth - which were found throughout the tension boom indicate that further major failures were imminent. The fatigue strength at various points throughout the structure did not therefore vary to any great extent.

5.5 Location of fatigue crack origins

With the exception of the outer wing joint failures, all of the many fatigue cracks found in the spar tension booms were at relatively small bolt holes ($\frac{1}{4}$" diameter). In addition to the normal geometric stress concentration at these holes, examination of the fractured surfaces (Appendix 4) revealed that corrosion and/or fretting were contributory factors.

5.5.1 In the first test, the cracks originated at the skin attachment bolt holes at the root rib positions where the wing skin terminated, i.e. where the local condition of load diffusion from the skin into the boom via these bolts was relatively severe. The consequent relative movements of the boom, skin and bolts caused fretting on the surface of the boom and in these holes.

All the fatigue cracks in this first test originated at the edges of the holes on the surface of the boom, thus supporting the view that the predominating influence was due to load diffusion and that the presence of fretting was an aggravating influence. There was no evidence of corrosion.

5.5.2 The above remarks apply equally to the second test, with the exception that an anti-fretting strip of plastic material between the boom and the wing skin was effective in preventing fretting on the boom surface, but this was possibly of small benefit, as fretting could still take place inside the bolt holes.

5.5.3 Examination of the outer wing joint fractures (third test) revealed many origins throughout the depth of the bolt holes. Since none were located at the edges of the holes, it was considered that the stress concentration at the bolt hole had not been complicated by bolt bending effects.

The presence of fairly severe corrosion in the holes was an additional important influence, and it was considered that the crack origins were located at corrosion pits. Although the majority of the cracks originated in this manner, there were two further conditions which produced fatigue cracks: these were fretting between the boom and the steel end fitting and changes in cross-section (Appendix 4). Although neither produced primary fatigue cracks, in that the final fracture did not develop from them, they served to indicate alternative potential modes of failure.

5.5.4 In the fourth test, all the fatigue crack origins, except one associated with fretting, were located in the holes, and were associated with corrosion on the surfaces of the holes.

The locations of fatigue origins were thus influenced by a number of factors, and it follows that a significant improvement in the fatigue strength of the existing structure would require major redesign.

6 CONCLUSIONS

From the standpoint of fatigue behaviour, the important findings in these tests were:-

(i) The single spar tension boom was outstandingly the most important member in the structure, and the fatigue cracks which caused the wing to fail were located at bolt and rivet holes in this boom. The local tensile stress concentrations at these holes were the principal factors in the origination of the cracks; the geometric concentrations due to the holes were aggravated by fretting and/or corrosion.

(ii) The significant aspects of the major failures were their suddenness and completeness, the former being accentuated by lack of knowledge of the existence of developing fatigue cracks and the latter to the non-redundant nature of the structure.

(iii) In line with general experience of testing concentrated flange structures, visual and radiographic inspection procedures were inadequate for finding the fatigue cracks during the test.

(iv) Only two of the eight major failures occurred at main structural joints although these are obviously places with stress concentrations. This is in agreement with the general experience of such structures in which it has been noted that failures at main joints are of relatively infrequent occurrence, although cracks at the main joints have often been found subsequently when the structure was dismantled.

These findings emphasise two points that apply generally to single spar wings:-

(i) A fatigue test should be made on a complete wing, since tests on details - such as main joints - may give an unreliable measure of the fatigue life of the wing.

(ii) Reliance cannot be placed on inspection to ensure safety in service.

REFERENCE

No.	Author	Title, etc.
1	Anne Burns	Fatigue loadings in flight loads in the wing of a Varsity. A.R.C. Current Paper No.285, May 1956

APPENDIX 1

FATIGUE LOADING CONDITIONS FOR VICKERS VARSITY WING FATIGUE TEST

These were computed from an analysis of the six contemporary roles for R.A.F. Varsity aircraft, which were as follows:-

Role	Average flight duration in hours	Average cruising height in feet	Proportion of total flying time
Bomber	4	20,000	20%
Transport	4	14,000	7%
Coastal reconnaissance	4	2,000	7%
Circuits and landings	0.25	1,000	33%
Instrument flying	2	4,000	18%
Approach aids	2	4,000	15%

The conditions which were considered to represent the overall average of these various types of service use were as follows:-

Duration of flight .... 33 minutes

Cruising speed .... 145 knots E.A.S.

All-up weight .... 33,000 lb

Fuel .... 90 gallons in each outer tank

The fatigue load programme based on these conditions simulated the following actions:

1 A take-off to level flight.

2 Eighteen ± 10 feet/second gust load cycles.

3 A landing from level flight.

Bending moment and shear diagrams for the wing are on Figures 2 and 3.

APPENDIX 2

METHOD OF APPLYING THE FATIGUE TEST LOADS

The fatigue test loads were applied with a hydraulic system, in which pressure switches and counting devices enabled the test to proceed with only intermittent surveillance.

The wing lift loads were simulated by hydraulic jacks which reacted between the ground and the wing, their concentrated loads being distributed by lever systems and chordwise formers.

The engine loads were applied by hydraulic tension jacks attached to anchorage points on the ground.

The applied lift loads were reacted partly by the weight of the test aircraft, and partly by fore and aft anchorages on the longitudinal centre-line.

Since the hydraulic jacks were pin-ended, the aircraft had to be restrained in the horizontal plane by suitably positioned links between the aircraft and anchorage points.

The power for the hydraulic jacks was supplied by an accumulator and by two rams which were motivated sinusoidally by an adjustable stroke reciprocating mechanism.

The minor load (1G - 10 ft/sec gust) was applied by flooding the system from the accumulator, which caused the aircraft to lift until restrained by the vertical anchorage. At this stage, the reciprocating rams were ineffective as certain oil passages were open and oil could pass freely between each side of their pistons. On closing the valves in these passages, the loading jacks were coupled to the rams in a closed hydraulic system and hence they moved sinusoidally as dictated by the stroke of the reciprocating mechanism.

For reasons concerned with the balance of lift and inertia loads the innermost hydraulic jack on each wing remained in the accumulator system, i.e. at constant load.

The minor load was controlled by the accumulator pressure, and the major load by the displacement of the hydraulic rams.

On the completion of each batch of eighteen gust load cycles, the above noted oil passages were again opened followed by isolation of the accumulator, thus allowing the aircraft to land.

The tension jacks simulating the engine loads were always coupled to the accumulator (apart from the gust cycling stages) in order to maintain the engine dead weight load.

The time required for a simulated flight (take-off, eighteen gust load cycles, and landing) was 80 seconds.

The cyclic gust loading rate was 27 per minute.

Photographs of the specimen and test equipment are shown on Figures 13 to 16.

Details of the lever system for the wing loads are given on Figure 4.

APPENDIX 3

THE MATERIALS USED IN THE STRUCTURE

1 D.T.D. 363A - extrusions, for spars, etc.

(i) Chemical composition (nominal)

Copper	1.0 per cent
Zinc	5.3 per cent
Magnesium	2.7 per cent
Manganese	0.5 per cent

(ii) Heat treatment

Solution treatment:	2-4 hours at 455-465°C and quench in water.
Ageing treatment:	4-8 hours at 130-140°C
Annealing treatment:	2 hours at 380°C and cool slowly

A treatment is selected from the above to suit individual requirements.

(iii) Strength properties

(a) 0.1 per cent proof stress:- not less than 33 ton/sq in.
(b) Ultimate tensile stress:- not less than 38 ton/sq in.
(c) Elongation:- not less than 5 per cent for extrusions between $\frac{3}{8}$ in. and 6 in.

2 B.S. L65 - Forgings and extrusions.

(i) Chemical composition (nominal)

Copper	4.4 per cent
Silicon	0.7 per cent
Manganese	0.6 per cent
Magnesium	0.6 per cent

(ii) Heat treatment

Solution treatment:	2-4 hours at 500-510°C and quench in water.
Ageing treatment:	5-20 hours at 155-185°C.
Annealing treatment:	2 hours at 360°C and cool slowly.

A treatment is selected from the above to suit individual requirements.

(iii) Strength properties

(a) 0.1 per cent proof stress:- not less than 26 ton/sq in.
(b) Ultimate tensile stress:- not less than 30 ton/sq in.
(c) Elongation:- not less than 8 per cent for extrusions between $\frac{3}{8}$ in. and 1 in.

3 D.T.D.546B - Clad, high-tensile aluminium alloy sheet (WP condition)

(i) Chemical composition

Copper:	Not less than 3.5, or more than 4.8 per cent.
Iron:	Not more than 1.0 per cent.
Silicon:	Not more than 1.5 per cent.
Magnesium:	Not more than 1.0 per cent.
Manganese:	Not more than 1.2 per cent.
Titanium:	Not more than 0.3 per cent.
Aluminium:	The remainder.

(ii) Heat treatment

Quenched from 500-510°C at 2-4 hours.
Aged at 155-205°C for an appropriate time (WP condition)

(iii) Strength properties

(a) 0.1 per cent proof stress: not less than 20 ton/sq in.
(b) Ultimate tensile stress: not less than 26 ton/sq in.
(c) Elongation: not less than 8 per cent for sheets thicker than 12 SWG.

4 D.T.D. 6103 - Clad aluminium alloy sheet.

(i) Chemical composition: same as for D.T.D. 546B.

(ii) Heat treatment

Quenched from 500 to 510°C at 2 to 4 hours.
Aged - 5 days at room temperature (W condition).

(iii) Strength properties

(a) 0.1 per cent proof stress: not less than 14 ton/sq in.
(b) Ultimate tensile stress: not less than 24 ton/sq in.
(c) Elongation: not less than 12 per cent on sheets up to $\frac{3}{8}$ inches thick.

APPENDIX 4

THE FATIGUE TESTS

The series of tests has, for convenience, been arranged in four parts, i.e. tests Nos.1 to 4 as in Table 1.

The endurances quoted in the following descriptions of the various failures are total values, that is the gust load cycles applied in the test plus the estimated allowances for previous service flying.

1 First test

(Varsity WL 667 - D.T.D. 363A Spar booms).

52,700 Gust load cycles

The first skin crack was found on the bottom surface. During the remainder of the first tests, fourteen similar skin cracks were detected, most of them originating at the corners of cut-outs in the skin. Their rates of propagation were relatively slow. Detailed information is presented in Table 2.

170,500 Gust load cycles

The main spar tension boom in the port side inner wing suddenly fractured (Figure 17). There had been no prior visible indications that failure was imminent; a characteristic which was repeated throughout the tests.

The fracture in the boom was located at the root rib at the side of the fuselage, and was considered to have been influenced by the local diffusion loads from the skin which terminated at the root rib (Figure 1).

The fatigue crack was found to have originated at the outer rear hole of a group of six bolts which provided the most inboard skin to boom attachment (Figure 17). The clearly defined areas of fatigue on the fractured surface of the boom and the points of origin are shown on Figure 18.

The 6 in. × 16 SWG doubler plate between the boom and the wing skin had been fractured by a fatigue crack, which had originated at a sharp reduction in its width (Figure 8). Being sandwiched between the skin and the boom, it had not been possible to visually inspect the doubler plate during the test, but the appearance of the fractured surface and the conditions at the origin gave the impression that this individual failure may have occurred relatively early in the test. The respective cross-sections of the boom and the doubler plate were 5.0 and 0.6 sq in. so that once the doubler had failed the nominal increase in stress in the boom might be some 10%. When regard is paid to load diffusion from the broken doubler plate, however, it is possible that an appreciable increase of load occurred at the next row of bolts, which would significantly affect the fatigue behaviour of the boom. In this connection it is thought significant that the boom actually failed in the next row of bolts, see Fig.8.

In addition to the conditions of stress arising from the load in the boom plus the effect of the skin load diffusing into the boom at this point, a close examination of the boom fracture indicated that the origins of fatigue appeared to have been associated with local areas of fretting on the boom surface and/or just inside the holes. There was no evidence of corrosion.

185,500 Gust load cycles

After repairing the boom and on resuming the test, a similar failure occurred in the starboard inner wing at the central rear bolt in the above noted group of six bolts.

The failure in the doubler plate was also similar. An examination of the fractured surface of the tension boom again indicated the origin of fatigue to have been associated with fretting on the surface and/or just inside the bolt hole (Figure 19). Several other small areas of fatigue (arrows on Figure 19) were associated with fretting.

There was again no evidence of corrosion.

This completed the first fatigue test, and at this juncture the inner wings were re-sparred. In effect, the D.T.D. 363A tension booms were replaced with new members which were dimensionally identical, but were of B.S.L65, a copper aluminium alloy instead of the previous zinc aluminium alloy.

2 Second test

(Inner wings with new B.S. L65 tension booms).

180,500 Gust load cycles

The starboard inner wing tension boom fractured at the root rib, but not at quite the same position as in the first test. The fracture was coincident with two $\frac{1}{4}$ in. diameter bolt holes just inboard of the rib, whereas the previous failures had been located at bolt holes situated 5 in. further outboard. This change in position may have been caused by the addition of an exterior skin doubler plate (Figure 20), which had been incorporated during the re-sparring operation to eliminate local skin buckling at the root rib. This buckling had occurred in the first test and also had been visible on service aircraft subsequently examined. The redistribution of the load diffusing from the skin into the boom, resulting from the addition of this doubler plate was considered to be the most likely cause for the change in position of the fracture. An examination of the fracture indicated that the origins of fatigue were again at the bottom edges of the holes (arrows on Figure 22) and were probably associated with fretting just inside the holes. There was no evidence of fretting on the surface of the boom, which indicated that the plastic interlayer between the boom and the internal skin doubler plate, which had been incorporated during re-sparring, had been effective in preventing fretting on the boom surface.

The horizontal arrow (Figure 22) points to an area on the fractured surface where the propagation of the crack had been relatively fast, possibly just before the final fracture. An examination at high power revealed the characteristic striations first observed by Zappfe and Worden (Photographic Registrations of Fatigue, A.S.T.M., April 1950) and discussed by Forsythe (Aircraft Engineering - 1960, 32, 96). Figure 24 shows a typical area of striations and at higher magnifications it is even possible to distinguish the second order striations caused by individual gust load cycles (Figure 25).

Similar local areas on the previous D.T.D. 363 boom fractures were not marked in this manner.

290,000 Gust load cycles

On resuming the test after repairs, the port inner wing tension boom fractured at 16.5 in. outboard of the root ribs (Figure 21), and was therefore the exception in the four inner wing failures, the previous three having been located near the root rib.

The associated fatigue crack had originated at the furthest inboard hole of a row of bolts which fastened a leading edge inspection panel to the spar tension boom. The bolt hole was relatively near to the forward edge of the horizontal flange and adjacent to a rivet hole (Figure 23).

The origins were again at the bottom edges of the holes (arrows on Figure 23). Although there was again evidence of fretting in the bolt holes, the anti-fretting film had been effective in preventing its formation on the bottom surface of the boom.

There was no evidence of corrosion.

It was noted that the general appearance of the fatigued portions of the fractured surface was relatively rough in contrast with the smoothness of those in the D.T.D. 363 booms in the first test.

This was the end of the second test, and at this stage the tension booms were extracted to facilitate detailed inspection for cracks.

All four inner wing tension booms from the first and second test were found to contain fatigue cracks at bolt or rivet holes. (Figure 9).

Including the fatigue cracks at the fractures, the respective numbers were as follows:-

Test	Inner wing boom	Gust load cycles	Material	No. of fatigue cracks
1	Port	170,500	DTD 363A	8
1	Starboard	185,500	DTD 363A	8
2	Starboard	180,500	B.S. L65	7
2	Port	290,000	B.S. L65	6

Many of these fatigue cracks were visible by eye and therefore could be considered to be in an advanced stage.

The results from these first two tests are discussed in Appendix 7.

3 Third test

(W.L.667 outer wings with D.T.D. 363A spar tension booms).

261,500 gust load cycles

The port outer wing fractured at the root joint. The fracture was near the end of the spar tension boom at the outboard bolt hole (Figure 26).

In consequence of the primary failure, the forward false spar also failed, but examination revealed that the tension member in this spar had also contained a fatigue crack at the end of the joint plate. (Figure 27).

There were no skin cracks in the outer wings.

Examination of the primary fracture in the spar tension boom revealed that the origins of the fatigue cracks in the right hand lug (arrows A and B on Figure 29) were clearly associated with fretting between the lug forming the end of the boom and the steel joint fitting. These were the only examples of fatigue cracks which were clear of the holes and the reason for this was not apparent, as similar fretting was present on the other lugs.

The surfaces of the two separate areas of fatigue were each bounded by a narrow band, which was bright in appearance (arrows A_1 and B_1 on Figure 29), and unusually, was at an angle of 45° to the main fracture surface.

A further small fatigue crack (arrow C on Figure 29) was located in the corner of the section where the lug root radius ended.

On the left hand lug, the origin of the major fatigue crack (arrow D on Figure 29), and those of several other cracks (arrow E) appeared to be associated with the considerable amount of corrosion which was present.

279,500 Gust load cycles

A fatigue crack was found in the forward false spar in the starboard outer wing. It was similar to the previous one in the port outer wing and was repaired at 295,500 gust load cycles.

401,500 Gust load cycles

The starboard outer wing spar tension boom fractured at the root joint in a similar manner to the previous failure in the port outer wing (Figure 28).

This was the only major failure to be anticipated by previous knowledge of the presence of a fatigue crack. As indicated on Figure 28 the lower half of the right hand lug was found, by visual inspection, to be cracked at 392,900 gust load cycles.

There were no skin cracks in the starboard outer wing.

Examination of the primary fracture revealed that the main fatigue crack in the right hand lug had several origins (points between arrows F and G on Figure 30) and other smaller cracks were noted (arrows I and H on Figure 30).

The surface of the hole was heavily corroded, and it was probable that these origins were located at corrosion pits.

These were on cracks at the areas of fretting which were noted to have formed on the lug faces.

There were many origins of fatigue on the left hand lug (arrows J, K and L on Figure 30) with corrosion again in considerable evidence.

The ridge (arrow M on Figure 30) was the boundary between two separate fatigue cracks which had propagated at slightly different levels.

4 Fourth test

(WF 388 Outer wings with bushed joint holes).

The objective of this last test was to find the endurance of the outer wings, excluding the root joints, and in order to prevent a repetition of the joint failures which occurred in the third test, the joint bolt holes in the booms were reamed and bushed after they had completed 354,200 gust load cycles.

319,200 Gust load cycles

The tension member in the port outer wing forward false spar failed as in the third test.

354,200 Gust load cycles

A similar fatigue failure had developed in the forward false spar in the starboard wing.

The test was interrupted and both these failures were repaired, and as previously noted, the root joint holes in the spar tension booms were reamed and bushed.

373,200 Gust load cycles

The starboard outer wing tension boom fractured at two bolt holes located 25 inches outboard of the joint.

There were several origins of fatigue cracks in both bolt holes which appeared to be associated with corrosion.

The alternate light and dark appearance of the surface of the fracture (arrow N on Figure 31) suggests that the final tensile failure had been interrupted, possibly by the combined effects of release of load at the end of a fatigue test flight and an interruption in the test. The dark areas contained within the bands have precisely the same appearance as that outside the bands.

387,200 Gust load cycles

A similar fracture occurred in the port wing spar tension boom at 46 inches outboard of the joint. The fracture was again coincident with two bolt holes.

Several fatigue crack origins were noted in both holes (Figure 32), with corrosion again having had some influence.

One crack (arrow O on Figure 32) may have been caused by a patch of local fretting on the boom surface.

An area of alternate wide dark bands and narrow lighter shaded bands (arrow P on Figure 32), somewhat similar in appearance to those noted in the previous fracture again suggest that the propagation of the crack was alternately rapid and slow during the final stages.

Nine fatigue cracks were found in the spar tension boom from the starboard outer wing and three in the one from the port outer wing. All were located at skin attachment bolt holes. (Figure 10).

APPENDIX 5

MAIN SPAR TENSION BOOM STRESSES

Distance from ℄	Location	Stress - lb/sq in			Derivation
		On ground	1G Flight	10 ft/sec gust	
0	Centre line joint tongue	-2,000	10,000	3,100	Calculated
	" " " fork	-1,700	8,300	2,600	"
22.5"	In fuselage	-1,800	11,000	3,000	Measured
51.0"	Just inboard of root rib	-2,000	12,400	3,400	"
55.0"	At root rib	-1,900	12,300	3,700	Calculated
73.0"	Outboard of root rib	-2,200	12,500	3,800	"
156.0"	Nacelle centre	-2,000	7,800	2,200	Measured
174.0"	Outer nacelle rib	-800	3,000	1,200	"
	Outer wing joint inner end	-2,800	8,600	2,800	Calculated
190.0"	" " " outer end	-3,000	10,900	3,700	"
208.5"	Outer wing	-2,800	10,300	3,560	"
236.0"	" "	-2,800	10,800	3,630	"

As the above calculated stresses were based on the assumption that all the load was reacted by the spar boom, the stresses quoted for stations 73.0 in., 208.5 in. and 236.0 in.must be considered to be higher than actual, since no allowances were made for the effects of the front and rear false spars and the wing skin.

Approximate calculations applicable to the last two stations indicated that the actual stresses would possibly be about 85 per cent of the values quoted in the above table.

Further details of the stresses measured at stations 22.5", 51.0", 156.0" and 174.0" are shown on Figures 11 and 12.

WING DEFLECTIONS

The deflections at the wing tips relative to the centre were:-

On ground to 1G flight .... 13 in.

± 10 ft/sec gust .... ± 3 in.

APPENDIX 6

SERVICE HISTORY OF THE TEST SPECIMENS

One complete Varsity aircraft and a second wing were used for these tests, their respective service histories were as follows:-

	Serial No.	Hrs. flown	Roles	Equivalent fatigue test cycles: Gust	Equivalent fatigue test cycles: Ground to air
Complete aircraft	W.L. 667	831	General duties	9,000	500
Wing	W.F. 388	2,080	Pilot training	64,200	3,570

(The differing fatigue damage rates of these two roles is reflected in the respective ratios of hours flown to equivalent fatigue test load cycles for the two wings.).

The service life of W.F. 388 was terminated by a heavy landing which resulted in severe damage to the fuselage, but only superficial damage to the wing.

Initially, the fatigue tests were made on aircraft W.L. 667, but as the test progressed and failures occurred, the "inner wing" and "outer wing" components of W.F. 388 were utilised as replacements. W.L. 667 fuselage was used throughout the tests.

APPENDIX 7

COMPARISON OF D.T.D. 363A AND B.S. L65 TENSION BOOMS

One of the B.S. L65 tension booms had an endurance which was comparable with those for the D.T.D. 363A booms, the other's endurance was 60 per cent greater. (Table 1).

The log mean endurances were as follows:-

D.T.D. 363A	Tension Boom	177,800 gust load cycles
B.S. L.65	Tension Boom	228,800 " " "

This 29 per cent improvement was considered to be in fair agreement with various other fatigue strength data relating to these two materials.

Although the results were explicable in terms of relative fatigue strengths, two other factors should be considered, which relate to the introduction of anti-fretting films between the booms and the skins, and the addition of small doubler plates at the root ribs when the inner wings were re-sparred with B.S. L65 booms.

The anti-fretting film prevented fretting on the boom surface, but even so, the resulting benefit may only have been slight, as it was suspected that fretting had taken place in the bolt holes and was associated with the origins of the fatigue cracks which caused final failure.

The skin doubler plates were incorporated to prevent skin buckling, but from the fatigue view point, they tended to intensify the diffusion of load from the skin into the boom, and their influence may therefore have been deleterious. This surmise is justified by only one failure, since the remaining failure did not occur at the root rib.

Although the quantitative effects of these two modifications could not be assessed, it was considered that they had been minor in comparison with the effect of change of material in the booms.

In terms of ultimate tensile strength, B.S.L65 is the weaker by 21 per cent.

The ratio of the 0.1 per cent proof stress to the ultimate tensile stress is 0.87 for both D.T.D.363A and B.S. L65 based on specification values.

TABLE 1

Summary of test results

Test No.	A/C	Boom mat. Specn.	Location of boom fracture: Wing	Location	Side	Dist. from aircraft ℄ in inches	Gust load cycles: Equiva-lent service	Fatigue test	Total	Ground to air load cycles	Equiva-lent hours total	Data obtained
1	WL 667	D.T.D. 363A	Inner	At the root rib	Port	56.8	8,500	162,000	170,500	9,480	5,220	Endurance of inner wings
					Stbd.	57.6		177,000	185,500	10,300	5,670	
2		B.S. L65			Stbd.	53.5	None	180,500		10,030	5,520	Endurance of inner wings if fitted with B.S. L65 Spar tension booms
				Near the root rib	Port	80.0		290,000		16,120	8,870	
3	WL 667	D.T.D. 363A	Outer	At the outer wing joint	Port	196.0	8,500	253,000	261,500	14,530	8,000	Endurance of outer wings
					Stbd.			393,000	401,500	22,300	12,260	
4	WF 388			At skin attachment bolt holes	Stbd.	216.0	64,200	309,000	373,200	20,720	11,400	Endurance of outer wings outboard of their root joints
					Port	237.0		323,000	387,200	21,300	11,800	

NOTE:- All failures were in the spar tension boom

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TABLE 2

Fatigue cracks in the bottom skin in the first test

10 ft/sec gust load cycles	Location			Length of crack when first seen	Origins of cracks	Propagation
	Port or stbd.	Dist. from ℄ aircraft	Dist. from spar + fwd. - aft			
52,700 141,500	Port Stbd.	56.0"	-3.5"	0.2" fore and aft 0.2" fore and 0.4" aft	5/16" diameter bolt hole	2.5" at 162,000 1.0" at 177,000
58,000	Stbd.	73.0"	+22.0	0.2"	Corner of a cut-out	Negligible
68,500 70,500	Stbd. Port	59.0"	+10.0	0.3"	Corner of a cut-out	Nil at 177,000 Nil at 162,000
68,500	Stbd.	208	-34.0	0.2"	Corner of cut-out	Negligible
75,500	Port	60.0	-4	0.75"	Crest of buckle in skin	2.5" long at 162,000 cycles
141,500	Port and Stbd.	215	+3	0.2" 0.4"	Corners of cut-outs	Both 2.1" long
141,500	Port and Stbd.	190	-27.0	0.4"	Cracks in webs of ribs, originating from rivet holes.	Negligible
141,500	Stbd.	209	+11.0"	1.5"	Corner of cut-out	Negligible
198,500	Port	120.0"	-16.0"	2.5"	Corner of cut-out	6.0" long at 243,000 cycles
204,500	Stbd.	101.0"	+1.0	1.0"	Corner of cut-out	Negligible

NOTE: All cracks are in the wing skin, unless stated otherwise (wing skin is 18 SWG thick)

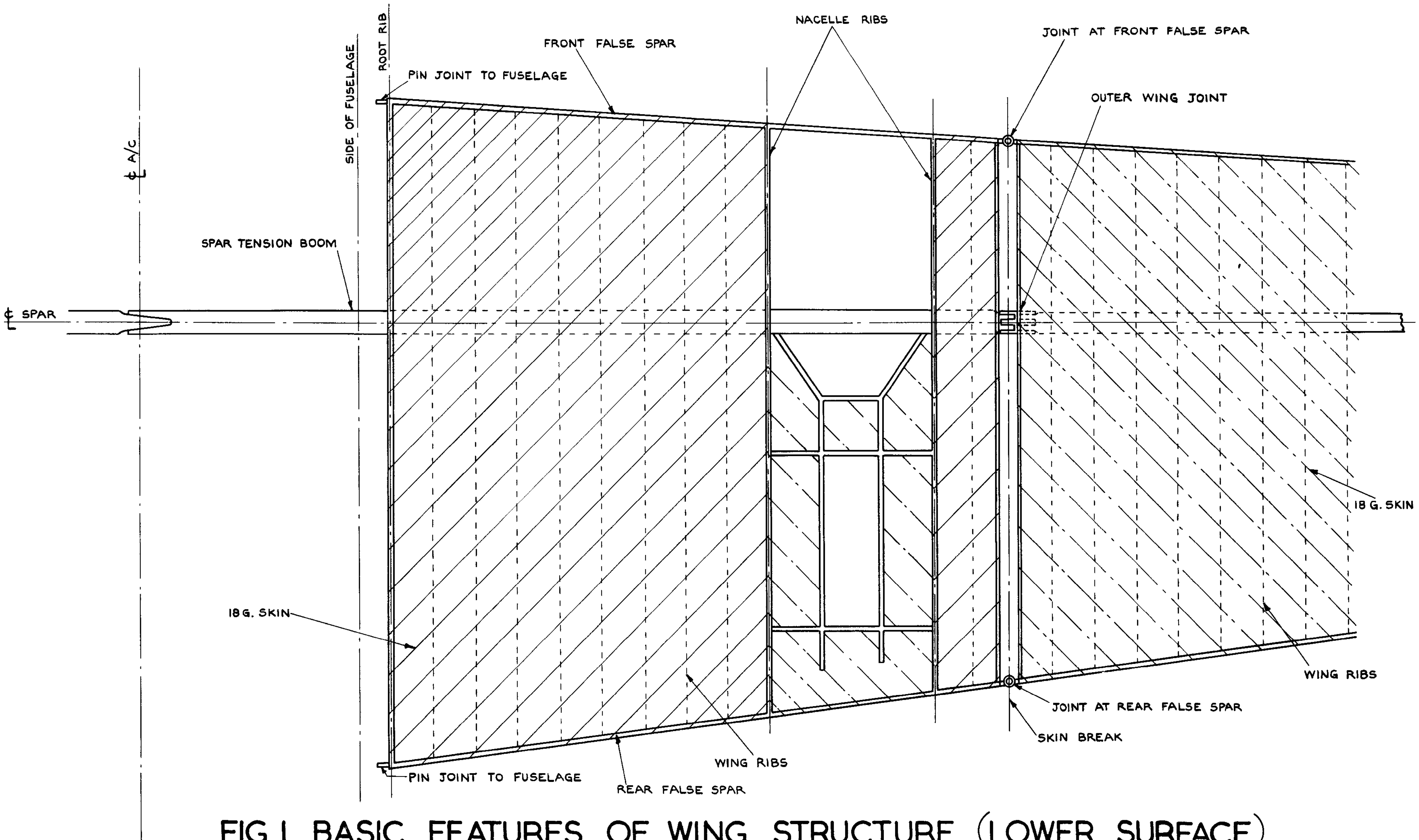


FIG. 1. BASIC FEATURES OF WING STRUCTURE (LOWER SURFACE).

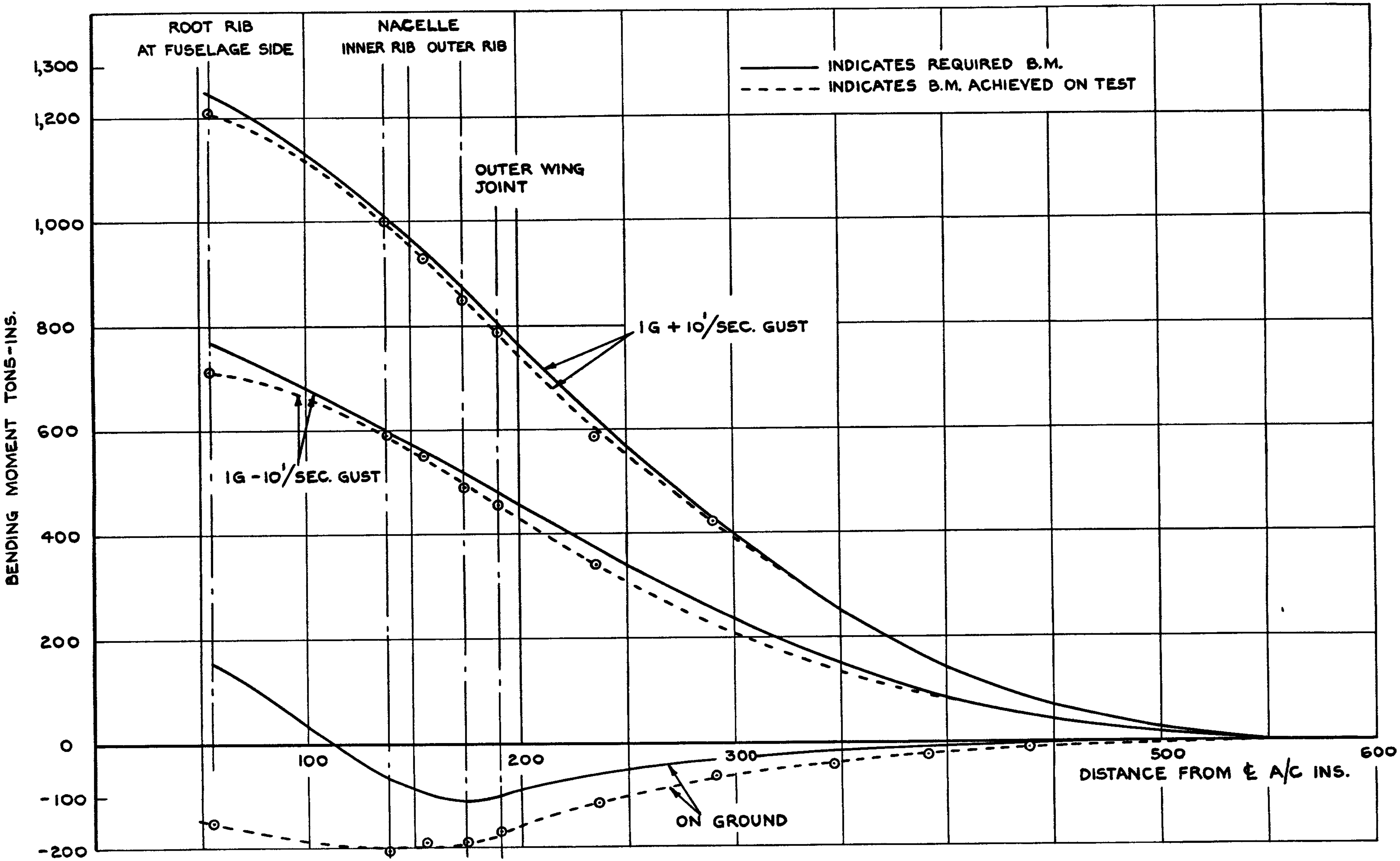


FIG. 2. VARSITY WING BENDING MOMENTS.

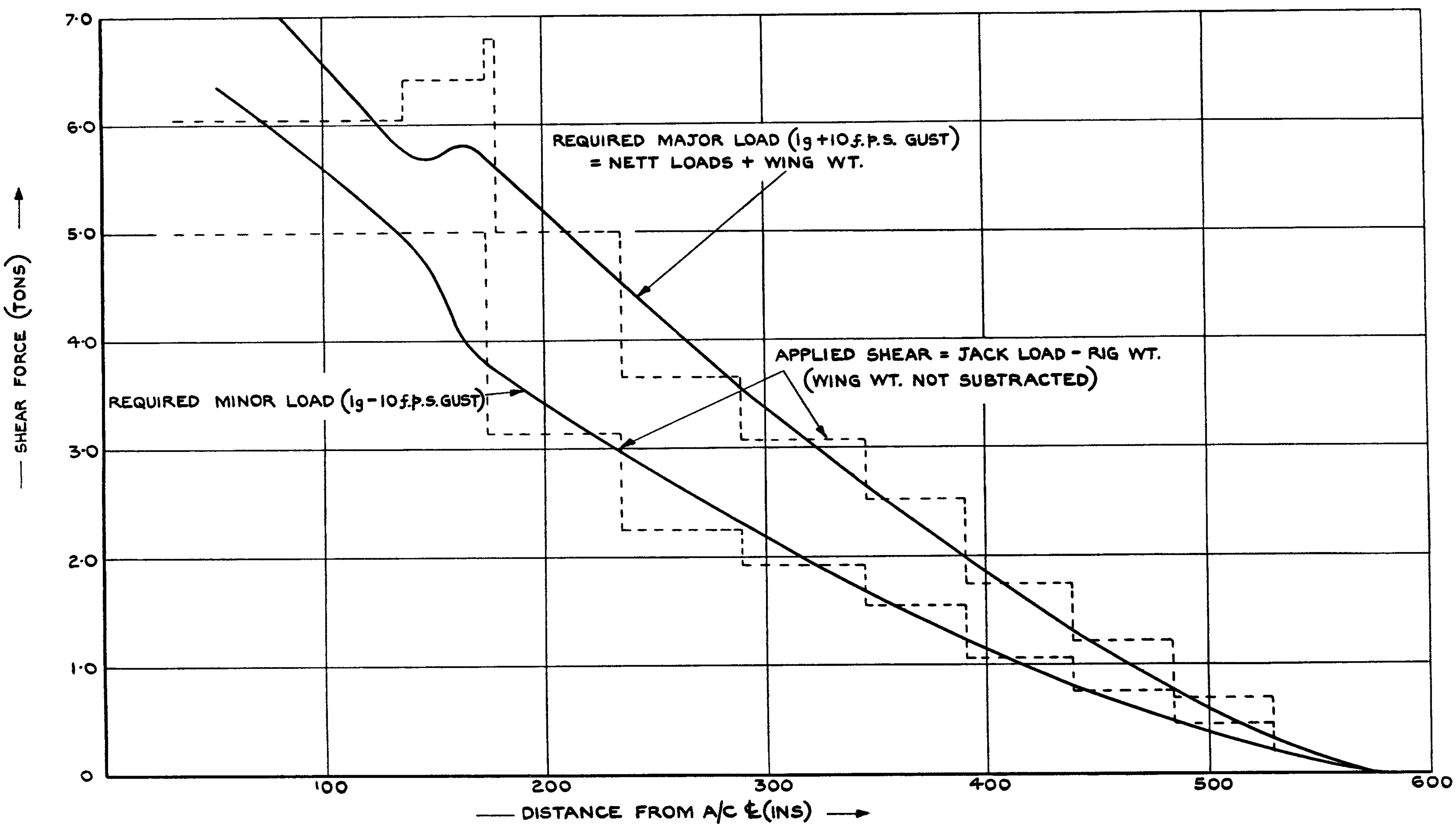
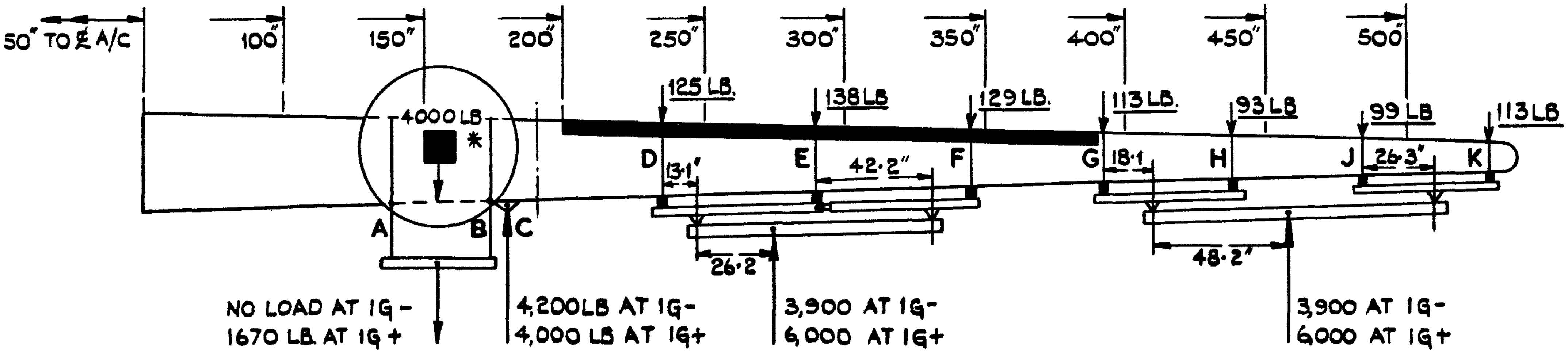


FIG. 3. VARSITY WING SHEARS.



ALL STATIONS EXCEPT A & B ARE ALONG THE SPAR. A & B ARE ON ENGINE C.G. 109" FWD OF SPAR.

STATION	A	B	C	D	E	F	G	H	J	K
DIST FROM ℄ A/C (INS)	138	174	180	236	291	346	392	438	484	530
(TONS) LOAD AT 1G-10'/SEC.	0	0	1·88	·914	·412	·417	·548	·356	·359	·48
(TONS) LOAD AT 1G+10'/SEC.	-·37	-·37	1·78	1·406	·634	·64	·843	·547	·552	·738

NOTE :-

* 4,000 LB LOAD AT NACELLE. REPRESENTED ENGINE WT. IT CONSISTED OF 3,000 LB. JACK LOAD +450 LB. RIG WT. + 550 LB. LEAD. DOWN LOADS UNDERLINED ALONG TOP OF WING DENOTE RIG WT. SHADED AREA ALONG TOP OF WING INDICATES 650 LB. LIQUID BALLAST REPRESENTING 90 GALLS OF FUEL WING JOINT IS 190" FROM ℄ A/C.

FIG. 4. LEVER SYSTEM AND LOADING DIAGRAM VARSITY WING FATIGUE TEST.

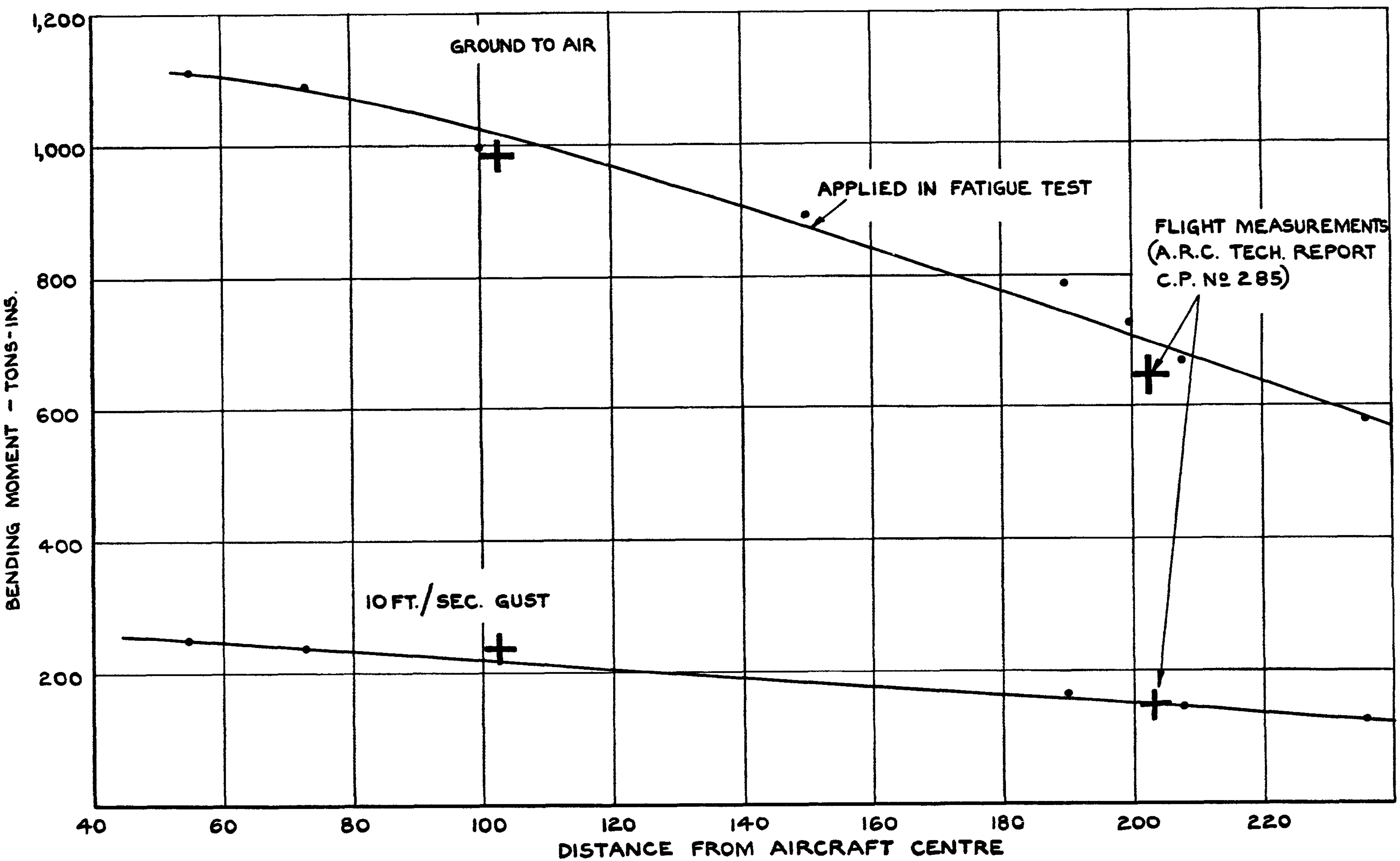


FIG.5. COMPARISON OF FLIGHT AND TEST WING BENDING MOMENTS.

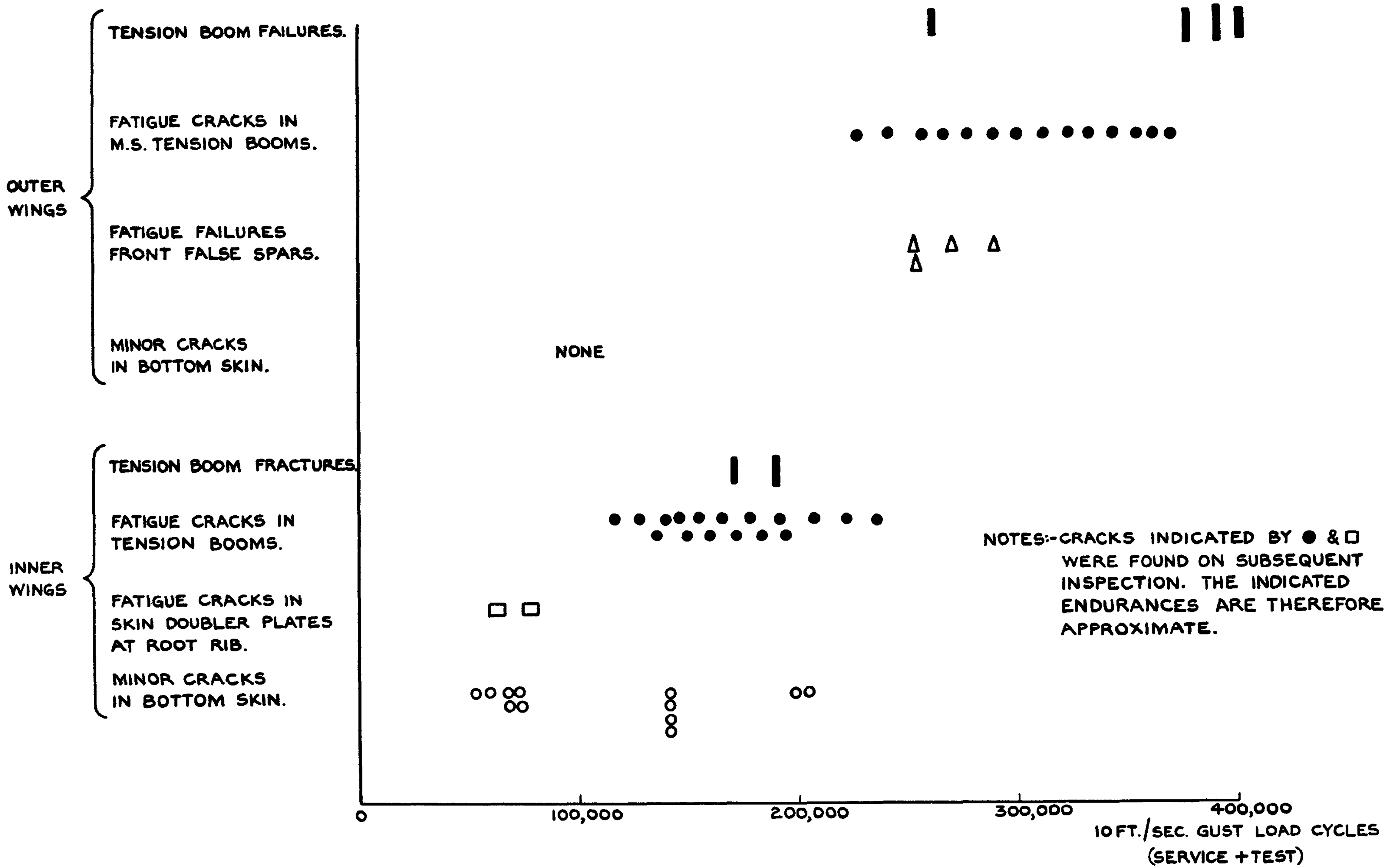


FIG.6. SEQUENCE OF FATIGUE DAMAGE.

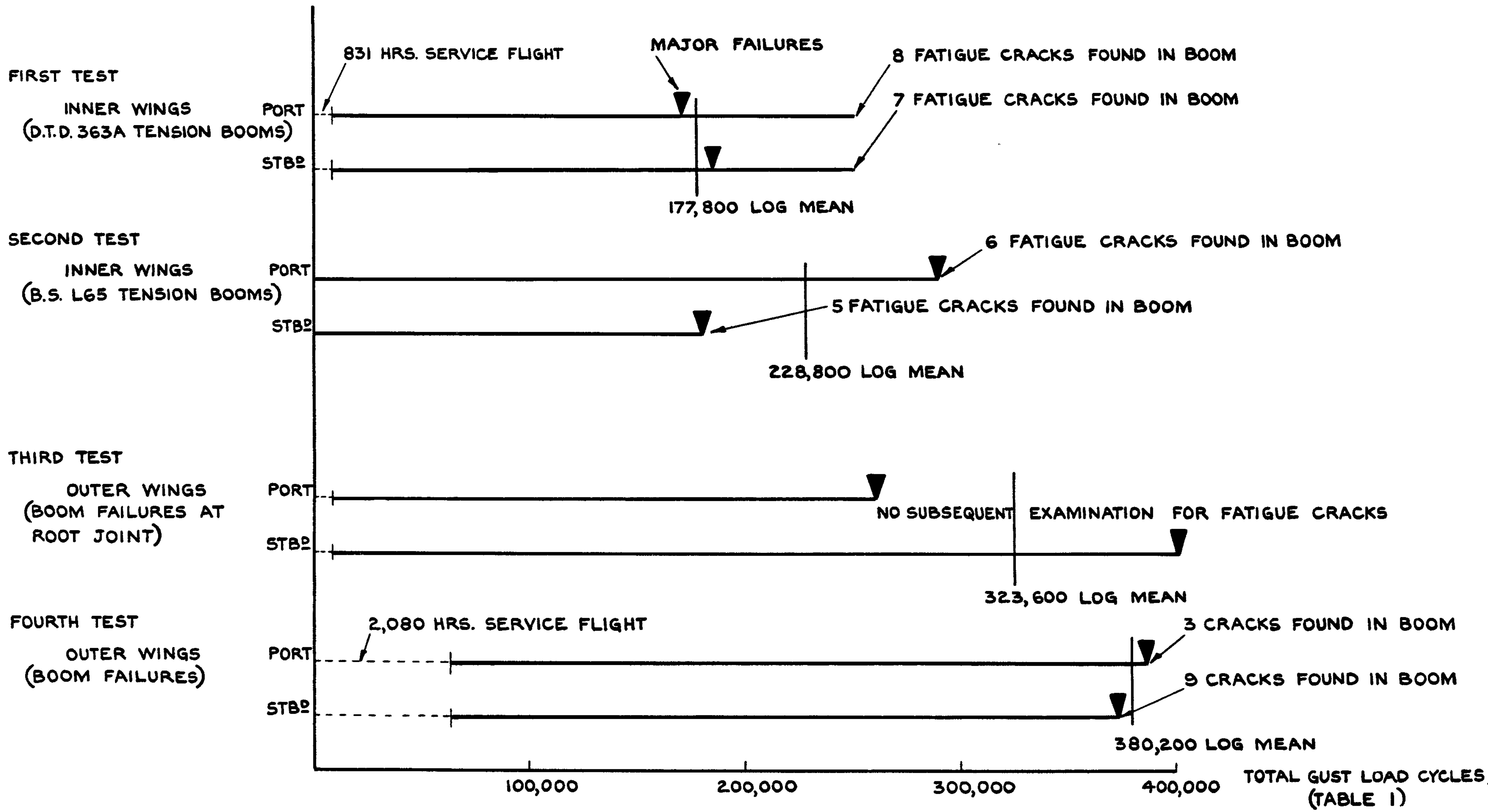


FIG. 7. SCATTER OF ENDURANCE

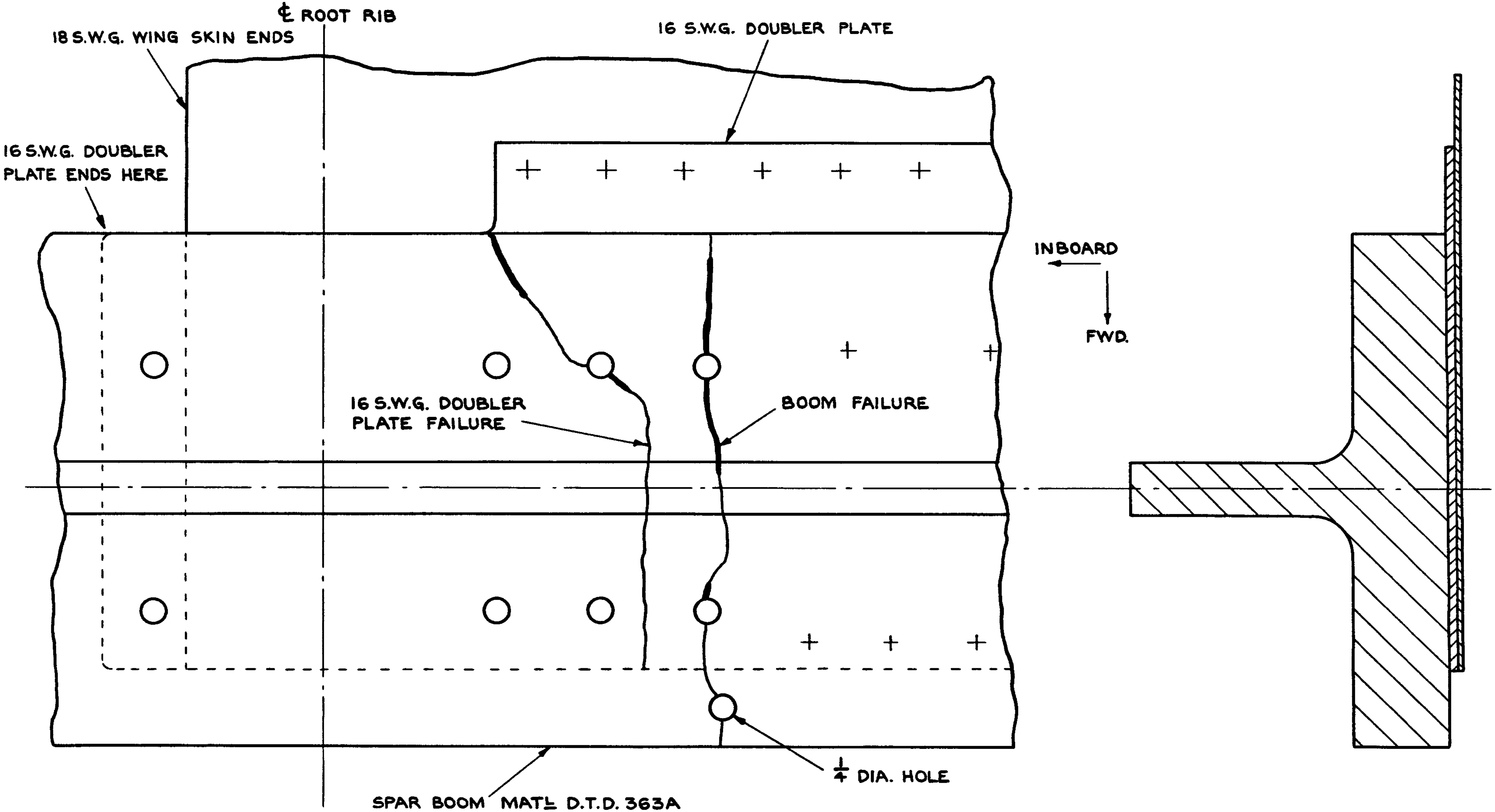


FIG.8. SPAR TENSION BOOM FRACTURE (PORT INNER WING) AND ASSOCIATED FATIGUE CRACK IN DOUBLER PLATE 170,500 GUST LOAD CYCLES FIRST TEST.

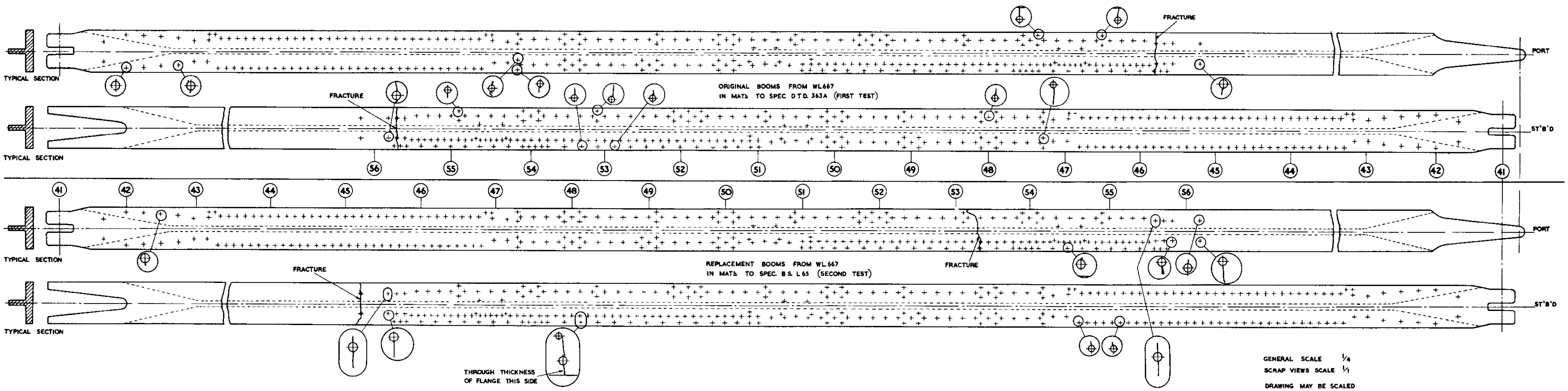


FIG 9 LOCATION OF FATIGUE CRACKS IN THE INNER WING TENSION BOOMS.

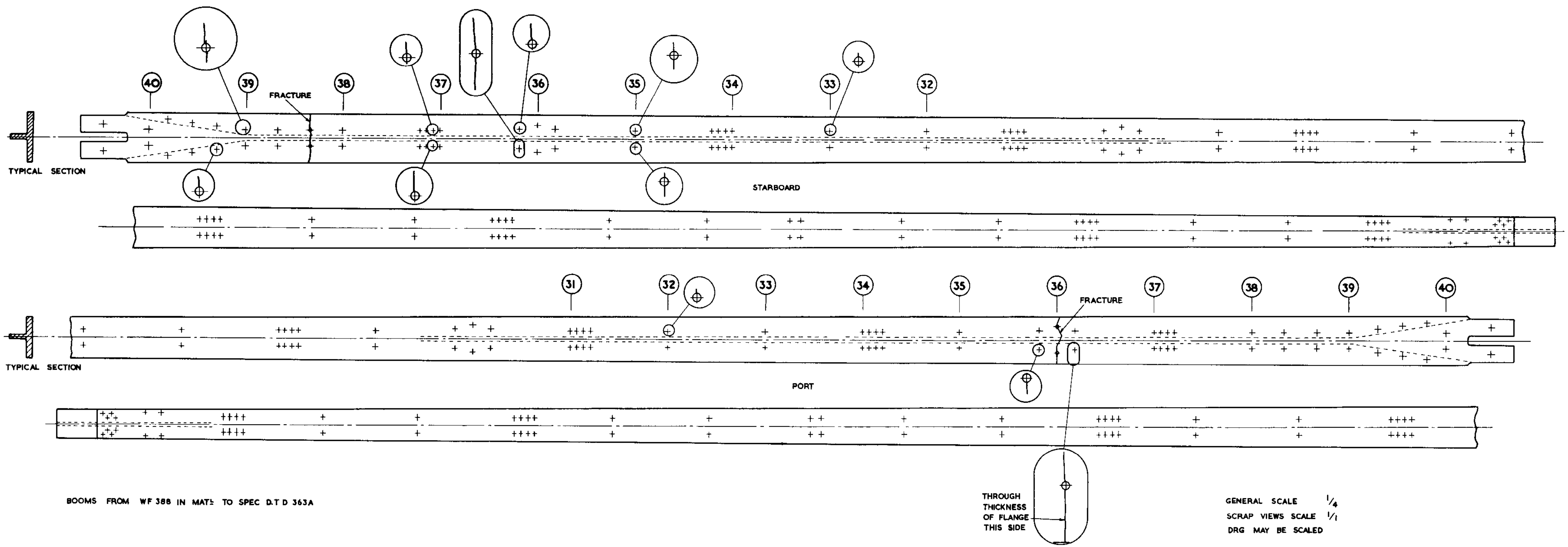


FIG. 10. LOCATION OF FATIGUE CRACKS IN THE OUTER WING TENSION BOOMS. (FOURTH TEST)

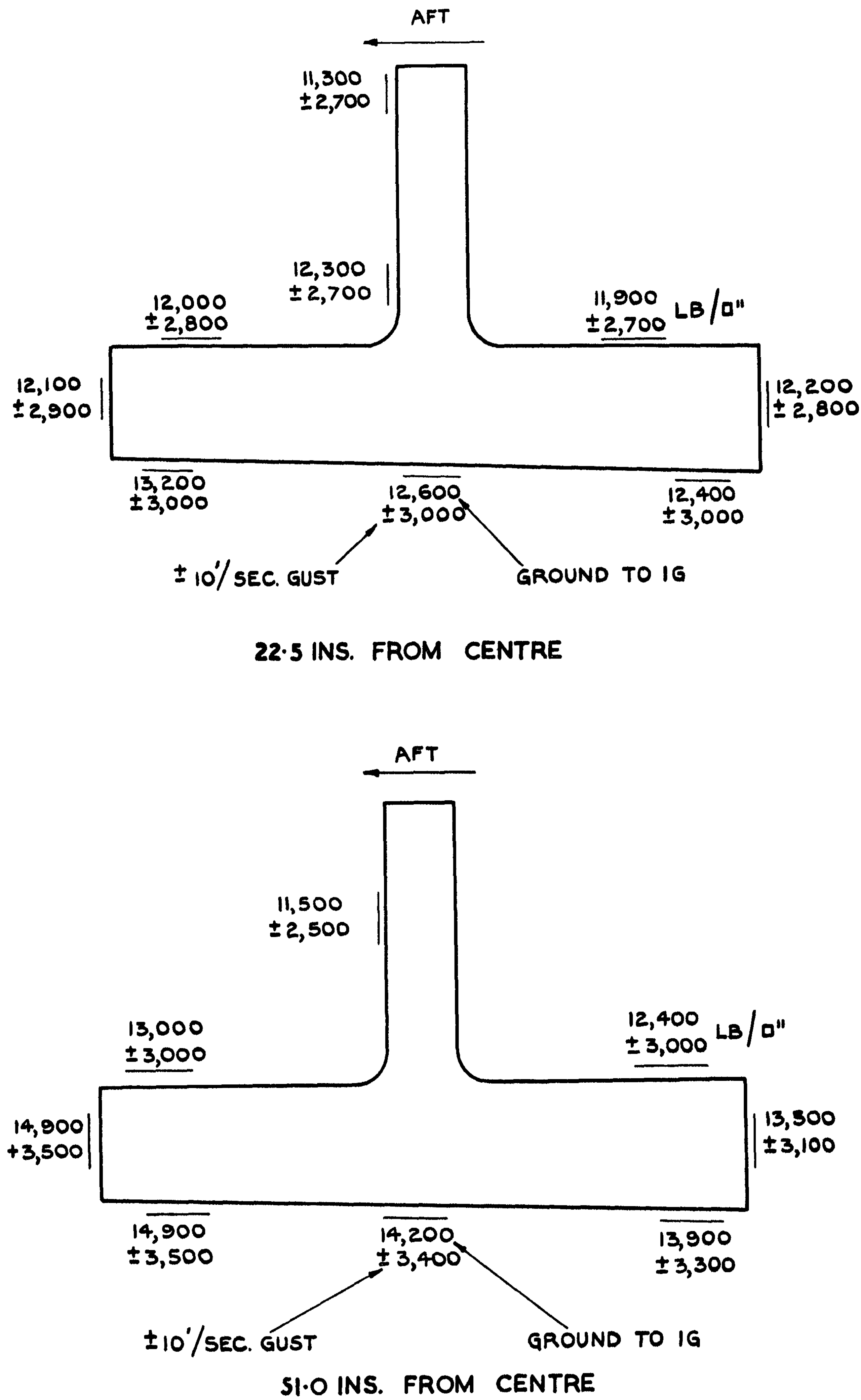
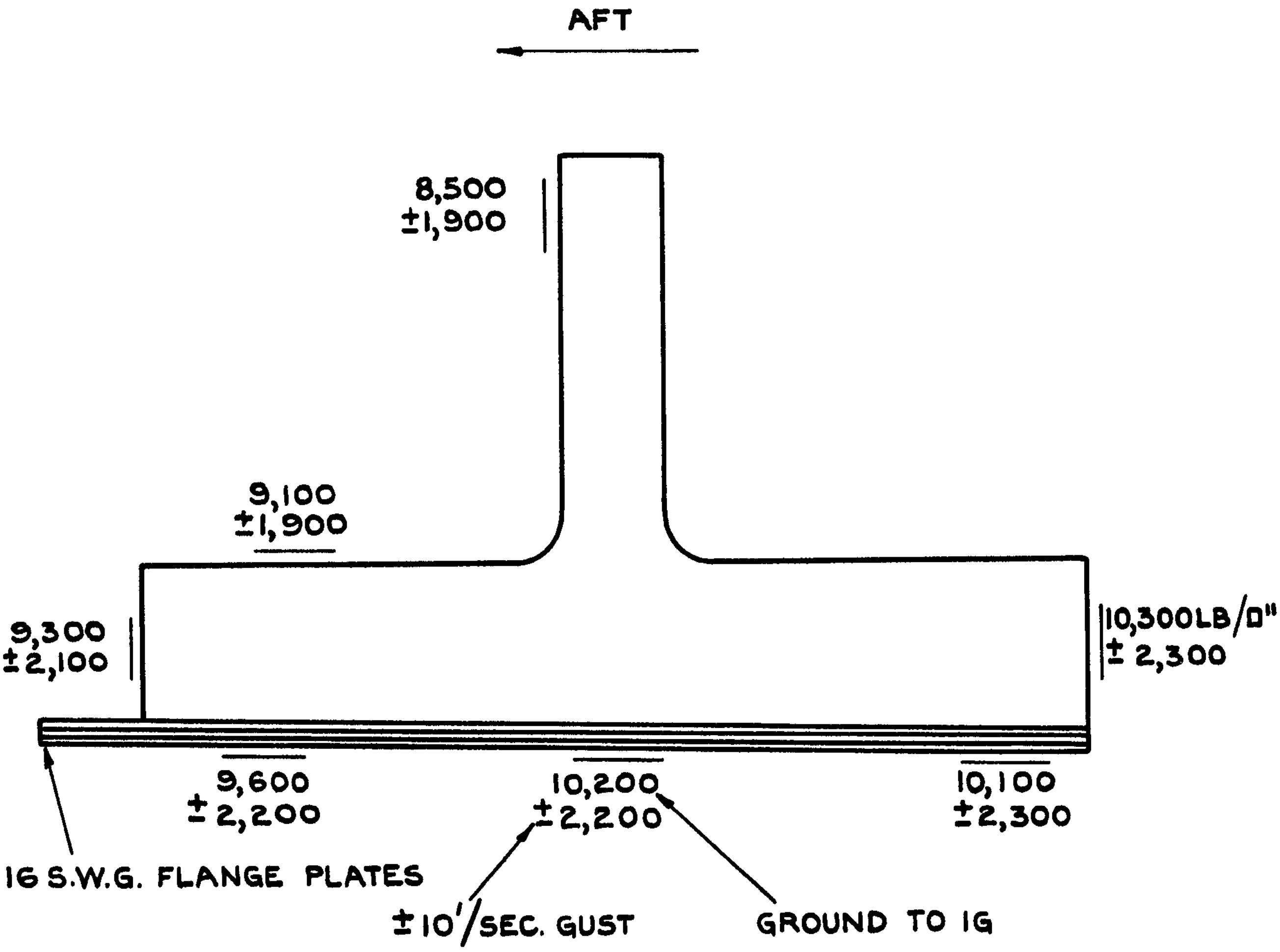
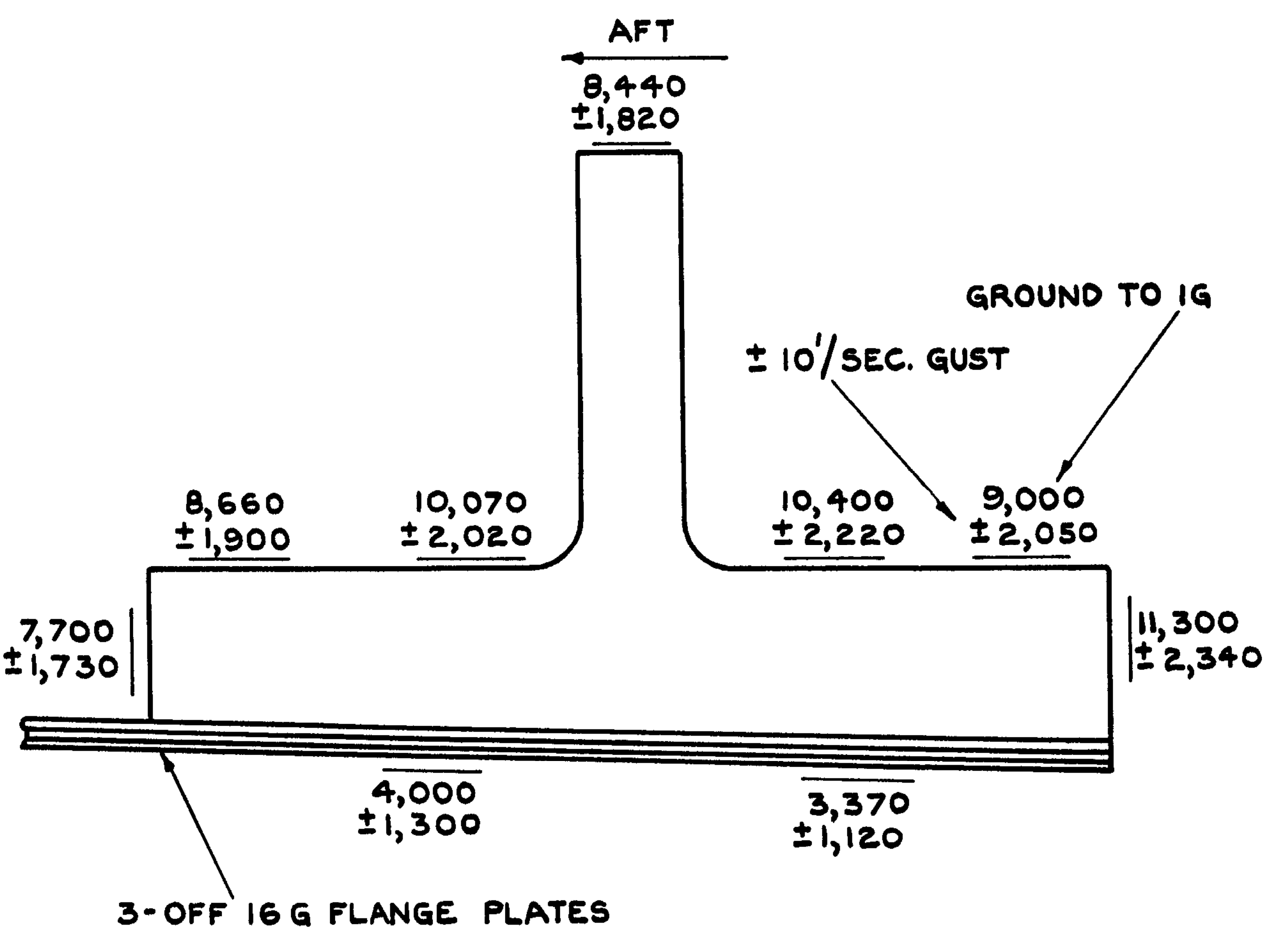


FIG.11. MEASURED STRESSES IN INNER WING SPAR TENSION BOOM.

(BASED ON AN E OF $10 \cdot 5 \times 10^6$ LB/□")



150 INS. FROM CENTRE (NACELLE ℄)



170 INS. FROM CENTRE

FIG.12. MEASURED STRESSES IN INNER WING SPAR TENSION BOOM.

(BASED ON AN E OF $10{\cdot}5 \times 10^6$ LB □")

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Winkworth, W.J. October, 1959.

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