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Service Failures in Aircraft Structures Associated with Fatigue, Repeated or Dynamic Loads

By

J. B. B. Owen, D.Sc., M.Sc., A.F.R.Ae.S.

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Summary.—This note gives examples and photographs of several structural defects which have occurred in service and shows that, although many failures may be due to fatigue or the application of excessive static loads, some are probably influenced by the repeated application of loads of high intensity, and by loads of a dynamic character. It is suggested that changes in design aimed at (1) eliminating the loads causing failure, *e.g.*, reducing in one case tab backlash, and (2) alleviating stress concentrations, are ways of reducing the incidence of defects due to repeated loading.

1. *Introduction*.—Fatigue failures, involving usually several million cycles of stress, and failures caused by the application of a slowly but steadily increasing load are familiar in engineering practice. Recently, however, attention has been drawn to failures which may accompany cycles of stress smaller in number, but greater in intensity than those commonly associated with fatigue testing. The present tendency is to describe such failures as ' repeated loading ' failures and this phrase is used in this sense in the text of the present report. During a discussion in a recent Aeronautical Research Council Sub-Committee on repeated loading failures of aircraft structures information was requested on whether repeated load failures occurred in service and this report is an attempt to answer this question.

Repeated loading failures merge on the one hand into failures which would usually be classified as fatigue failures and on the other into failures which occur in sensibly one application of a failing load. The characteristic appearance of fatigue failures is known from the nature of such failures which have been produced under laboratory conditions. Smoothness of fracture surfaces and absence of distortion are characteristic. Similar failures have frequently been observed in practice and some examples are quoted in section 2. The usual type of tensile destruction test on materials and the static loading tests of aircraft components until they break has revealed the nature of the failures to be expected when failure occurs under sensibly one application of high load. Such failures, too, have been encountered in practice and a brief account of a few examples is given in section 3. Considerable distortion, *e.g.*, necking, and fractures at an angle of roughly 45 deg. to the applied load are characteristic in ductile materials. Few repeated loading tests at high load intensity have been made under laboratory conditions. Those which have been done indicate that failures inclined at some 45 deg. to the direction of tensile stress as in tensile tests but with little elongation, and the presence of attrition, are likely characteristics.

* R.A.E. Report S.M.E. 3384—received 27th September, 1946.

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The appearance of service failures and defects, however, frequently show differences from those obtained in laboratory tests. Attention is drawn to these in sections 2, 3 and 4. These differences may be due to the greater complexity of the loading applied in service than in laboratory tests. Repeated loading of greater intensity than that required to cause fatigue, reversal of loading, loading of greater rapidity than is usual in static tests on airframes, etc., may play prominent parts in service failures.

In preparing these notes free use has been made of unpublished reports of the Metallurgy Division of the Materials Department and of the Accident Investigation Section of the Strucural and Mechanical Engineering Department of the Royal Aircraft Establishment. Most of the illustrations are drawn from these sources, the references to which are included at the end of this report.

2. *Fatigue Type Failures.*—2.1 *A Spar Boom Failure.*—The spar booms, in which the failures described here occurred, are made up of a number of lengths of round section tube joined together by plates. Load is transferred from the tubes to the plates mainly through vertical serrations in both. Fig. 1 shows a typical joint in which the boom has failed and pulled off the plugs now seen protruding. A 'close up' of a fractured boom is shown in Fig. 2 from which it will be observed that there are two main areas of failure[1]; the first extending from OO to LL and L, characterised like a fatigue fracture by its smoothness and orientation in the plane normal to the boom length, and the second the rest of the boom, which is more typically tensile in character. The first area, however, shows definite 'tide marks' which are absent from specimens subjected to the usual laboratory fatigue tests, but recent laboratory tests have produced tide marks. The stress cycle employed was $4 \cdot 65 \pm 0 \cdot 7$ tons/sq in. 500,000 times followed by $3 \cdot 25 \pm 1 \cdot 75$ ton/sq in. tension 50,000 times. Failure occurred after this had been repeated 22 times, *i.e.*, a total of 12 million cycles had been reached. In the usual type of fatigue test at $5 \pm 0 \cdot 5$ ton/sq in. cracks were first noted after $2 \cdot 5$ million cycles and failure occurred after 5 million cycles. These stress figures refer to the mean stress in the boom adjacent to the joint and make no allowance for stress concentrations in the joint.

Flight and taxi-ing measurements[2] of boom stresses were made to ascertain the order of the working stresses. When the aircraft is at rest on the ground the boom stress is only about 1 ton/sq in., excluding any 'built in' stresses which may be present. The fluctuations which occurred in a severe taxi-ing run are shown in Fig. 2. In flight the mean steady stress is higher about 4 ton/sq in. and the predominent fluctuations in rough weather about $\pm$ 1 ton/sq in. at $2 \cdot 7$ c.p.s., *i.e.*, about 3 million cycles occur in 320 flying hours in *rough* weather.

This figure of 320 flying hours is roughly the mean life of aircraft which failed in the manner described. A few aircraft failed after about 180 hours flying, but many others have flown considerably longer without boom failure occurring. Usually some cracking was detected at serrations during inspection and restrictions were put on the replacement of booms exhibiting marked cracking.

The area of fatigue appearance in service failures varied considerably. Laboratory tensile tests on cracked booms, however, gave no relation between tensile strength and the area of fatigue appearance but the results indicate that in many cases where booms have failed the mean stress to cause failure would probably have been greater than that associated with steady 1*g* flight.

Cracks occurred in both top and bottom boom. In a few cases the top boom has been found on inspection completely broken; no accident had occurred presumably because in flight this boom is in compression, also final failure might have occurred in the last ground run.

From the data available on laboratory life, flight life and measured boom stresses, it appears that service failures are probably due to either (i) the boom stress raises being more vicious, particularly in service aircraft which failed after comparatively few flying hours, or (ii) occasionally very high built-in stresses were present due to the method of tube manufacture or the technique of assembling the aircraft, plate and joint serrations, say, being forced together.

Considerable improvement in the laboratory life of booms was obtained by arranging that the change of section at the joint was more gradual. The boom material was also changed from one which obtained its strength by cold working, which produced cracks on occasions, to one which obtained its strength properties by heat treatment. These changes seem to have increased aircraft life considerably. The increasing use of concrete runways and the rigorous inspection instituted may also have had considerable influence. The precise nature of the loads causing the failures remains obscure and some tests on the usual type of sleeved joint have indicated that this, unless carefully designed, is not in laboratory test superior to the serrated joint.

2.2 *Web Cracks.*—The centre section web of some wings rises above the spar booms and forms part of a fuselage bulkhead. Fig. 4 is a close-up of such a web. The boom root bolts pick up the large holes seen in the figure. The web frequently cracked, as at F, in the fillet which merges the web proper into the fuselage ring proper. This was probably due to stress concentration at the corner where the web changes in depth.

Buckling of the web has occurred at this corner as seen around the crack F. Such cracks had frequently the appearance of fatigue as in Fig. 5 between F and G but exhibited tide marks as in the spar boom failure in Fig. 2. This was not always the case as in Fig. 6 where only AB and CD are characteristic of fatigue. The intervening portion suggests that high magnitude loads have played a part in the failure. Cracks occurred when the corner F in Fig. 5 was sharp and also when it was well radiused. Fortunately the longest cracks examined ran into adjacent holes and did not continue through them. Thus although such webs, which are continuous in a shallow wing and deep fuselage, are clearly undesirable, no serious trouble occurred.

2.3 *Cracks in Flat Plates.*—These web plate cracks, when the radius is ample, may be considered as due in part to flat plates buckling in compression where not supported, here by the spar flange or the fuselage frame. Cracking was more common where stringers were interrupted at frames or ribs. Such cracking[4] is shown diagrammatically in Fig. 7 and is probably due to severe engine vibration causing panel panting coupled with stress concentration due to the transfer of stringer end load by the flat plate skin.

Cracking has also occurred where hinge bracket bending loads are transferred first into the web as in Fig. 8 with no adequate members to take the loads into the main structure. Shear and bending stress concentration probably play a part in this failure.

2.4 *Rivet Failures or Failure Through Rivet Holes.*—Such failures usually occur where, for example, the wing skin ends and the end load in the skin has to be fed into and transmitted by a continuous boom. Stress concentrations responsible for such failures have been the subject of much theoretical work[6]. In one case of such a tail plane failure, rivets fatigued and the tail plane boom crack extended down into the web. There is also evidence, in tail planes of this type, of such cracks starting from rivet holes in the web near the tail plane root. Similar failures have been reproduced in the laboratory by repeatedly loading this tail plane[7]. The appearance of the fracture surfaces was different, however, from those which occurred in practice and which are shown in Fig. 9[8]. The service failures were more of a fatigue character and calculations of the loads expected on this tail plane suggest that repeated loads of high intensity are probably not the explanation of the service failures. Severe estimates of the tail load are far below its static strength. In a flying accident in which a cracked tail plane was broken the load on the tail plane was in the direction such that the crack was on the compression side !

2.5 *Rudder Failure Due to Excessive Tab Gear Backlash.*—The rudder failure[9] of Fig. 10 is an example of a fatigue type failure although it may also be due to repeated loads of greater intensity than those associated with fatigue. A tab operating rod failure through a screw thread and cracks in a cut-out in the rudder spar which run through holes provided for lacing the fabric are seen in the lower figure. The failure was probably associated with backlash in the tab and when this was restricted no further failure was reported.

3. *High Intensity and Dynamic Load Failures.*—3.1. Frequently the characteristic features of a static strength test failure have been reproduced in flying accidents but in a number of cases, fractures seemed to have occurred almost simultaneously at different places, whereas it is rare for two fractures to occur when a tensile failure occurs in a test specimen. More than one fracture frequently occurs when a structure is broken in the laboratory but these fractures usually occur in some comprehensible sequence. This is also frequently so in service failures but such sequences are sometimes difficult to imagine in the fractures of some aircraft wings, for example the failures described in Refs. 10 and 11. Such failures may be due to high harmonics in the applied loading exciting the wing in its higher modes, in which case repeated loading may again be thought of as playing a part in the failure.

3.2 Bent bolts in wing joints, permanent set at the wing tips, failures in the air which have stopped at the point just before complete collapse occurs, and *V-g* records all indicate that in wartime at least, very high loads are applied to airframes, but in the final collapse unless some special feature is present, there is no means available of knowing whether previous high loading has adversely affected the strength and precipitated the collapse. A special feature indicating that the previous loading history has affected the strength is the presence of old cracks and one such case is discussed in detail in the next section.

4. *A Suspected Repeated Loading Failure of a Fuselage.*—Fig. 11 is a diagram of the aft part of the fuselage concerned and shows the cross-sectional shape. The bottom of the fuselage between frames 23 and 25 has two troughs which converge towards frame 25 and between frames 25 and 26 become one trough. (These troughs are provided to house an arrester hook and its V-shaped attachment structure). Several service defects, which occurred in varying degrees of severity, are indicated in Fig. 11. Many fuselages had buckles in frame 25 between the troughs and had no other defect. The next stage of failure was buckling at both frames 25 and 24 with possibly cracks in frame 25 and rivets starting to pull. In more developed cases the trough walls aft of frame 25 had buckled, frames 25 and 24 had buckled and cracked, and a deep permanent buckle ran diagonally down the fuselage side. Fig. 12 is an internal view of the type of damage at frame 25, Fig. 13 that at frames 24 and 25 and Fig. 14 the external appearance of a fuselage.

In a static test[12] done on the fuselage the external buckling shown in Fig. 14 was reproduced as indicated in Fig. 15 but only after frame 25 had been artificially cut to resemble the service damage shown in Fig. 12. The fuselage had also been previously heavily loaded, the magnitude of the loads exceeding those which were later necessary to cause the heavy fuselage buckles shown in Fig. 15. This heavy loading in the previous test had failed to produce failure at frame 25 as in Fig. 12 although some slight buckling occurred at this frame. In fact the tail wheel unit structure which was used to apply shear and torsion loads to the fuselage failed; a condition which was not common to those service incidents which exhibited only slight defects. With frame 25 artificially damaged the tail unit was strong enough to break the fuselage; but whether this was primarily due to cutting this frame or to previous heavy loading is not established. It is apparent, however, that in service something was happening which differed from what occurred in the first static laboratory test, and resulted in the buckling and cracking of successive fuselage frames. While it is possible that another fuselage may have behaved differently from the test fuselage (insofar as, for example, the frames were initially buckled, or the applying of torsion loads through the fin and tail plane attachments produced a different distribution of loads at frame 25) it seems more probable that under repeated and reversed loading actions a different result from that obtained in the static test would have occurred. Evidence of this is found in the progressive nature of the fractures and defects suggested by the various degrees of damage sustained as already described, in the rubbing together of the fractured faces of frame 25 and in the buckling on both port and starboard edges of the crack that were revealed on close examination.

A possible source of the loads causing failure was tail wheel shimmy caused by the shimmy dampers becoming inoperative. Action was taken to remedy this fault and also to strengthen the fuselage. Reports of trouble then ceased. If is, therefore, not known whether shimmy suppression or extra strength was chiefly responsible for the disappearance of the trouble but it is apparent that it would have been better to have designed the fuselage with a flat stress bearing bottom, and thereby avoid the constructional complication and stress concentration which must have been responsible for the frame failures. A rough calculation assuming Batho shear distribution around the fuselage indicates that the tensile stress in frame 25 where cracking occurred may be several times the shear stress in the fuselage. Strain gauges in the test indicated this also and gave an indication that the frame stresses at the higher loads were non-linear with load.

5. *Discussion of Fatigue and Repeated Loading Failures.*—Of the failures described above the most serious was that involving spar boom failure since it was frequently accompanied by complete loss of crew and aircraft. Only about half the structural failures in the air of aircraft of the type were due, however, to these boom defects. Stability changes with aircraft speed, accompanied by inefficient design of the trim tab control circuit were probably the root cause of many of these other accidents. The total number of flying hours on all aircraft of the type per structural failure in the air compared quite favourably with other aircraft but the writer considers that these failures due to spar boom defects had a profound effect on the morale of those flying the aircraft.

All the fatigue and repeated load type of failures described start at points where stress concentration might be expected. If may be significant that the start of such failures is often characterised by smooth fatigue failures. In tests on structures to ensure their comparative immunity from fatigue or repeated loading failures, it might then be desirable to include some form of fatigue test. Cracking may, however, in some cases afford a relief of stress, *e.g.* the web cracks of the type shown in Fig. 4, which do not pass through the hole into which they eventually run.

The stress concentrations leading to the web crack of Fig. 4 and the frame cracks in Fig. 11 might be classed as stress concentrations due to faulty outline design, whereas the others seem to be due to imperfections in detail design. The magnitude of stress concentration effects is usually difficult to estimate even when the loading is known. Data on load fluctuations is scanty and so too is information on the behaviour of materials under diverse load cycles. Stress lacquers and strain gauges may, however, give the designer information on the intensity of stress concentrations to supplement his experience on designing to avoid stress concentrations.

6. *Conclusions.*—Several further examples could be added to the above but sufficient evidence has already been provided to establish that failures occur in practice which differ from those obtained in static strength tests in the laboratory. Many such failures are of a fatigue character. Others are difficult to classify but their appearance suggests that the repetition of high intensity loads is responsible for some of them. The failures which have occurred in the past have been overcome chiefly by changes in design aimed at reducing the loads causing the failure and reducing stress concentrations. Improved metallurgical technique and material changes have also in some cases reduced the locked up stresses which may have aggravated the stress concentrations already present. On the whole, experience points to the cure of fatigue type or repeated loading type failures lying in the avoidance of stress concentrations in design.

The presence of fractures which appear to have occurred at the same time at different places, in say a wing, suggests that loads of a transcient dynamic character are also met in practice and produce results which differ from those of static tests.

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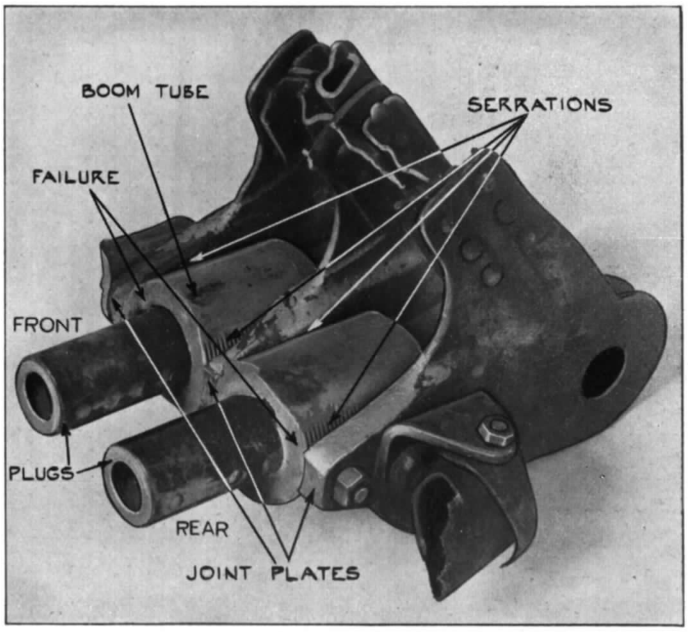

FIG. 1. A spar boom failure.

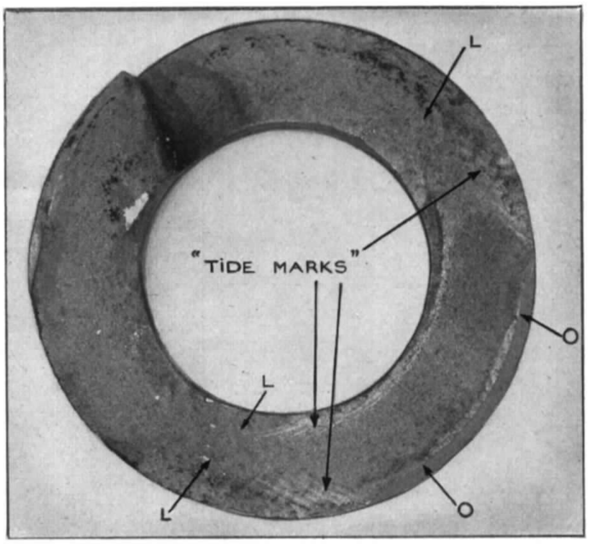

FIG. 2. Enlarged view of fractured boom.

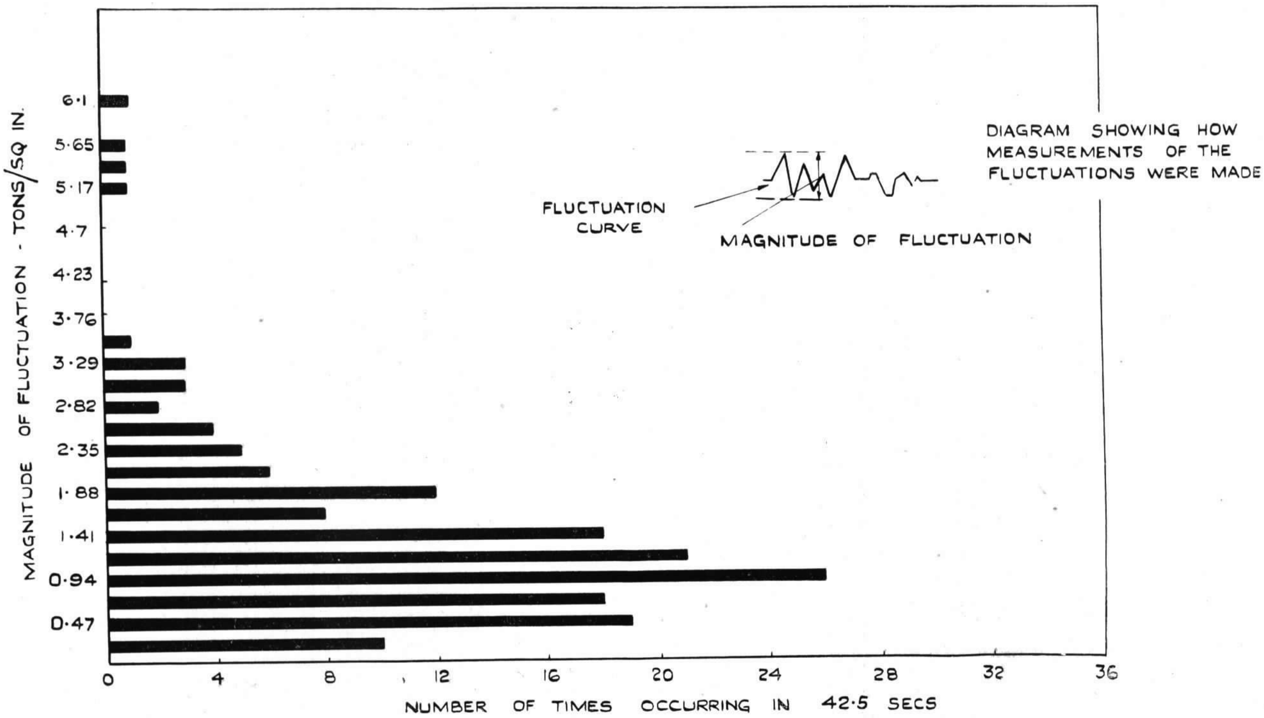

FIG. 3. Fluctuation frequency diagram—taxi-ing at Upper Heyford.

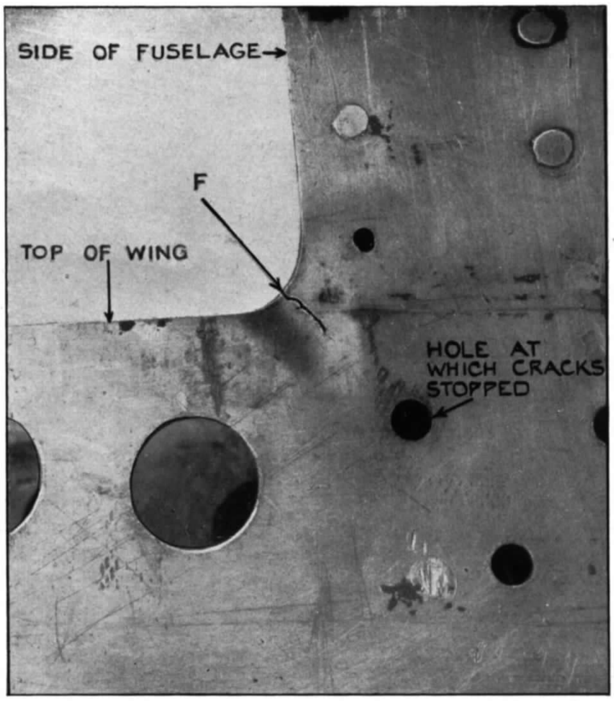

FIG. 4. Cracked web.

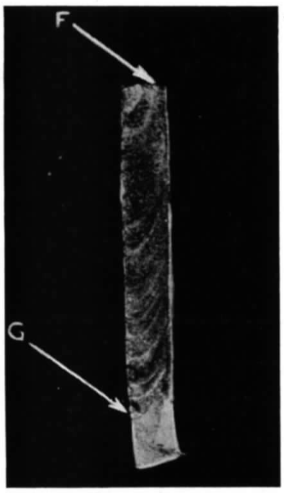

FIG. 5. Web fatigue crack.

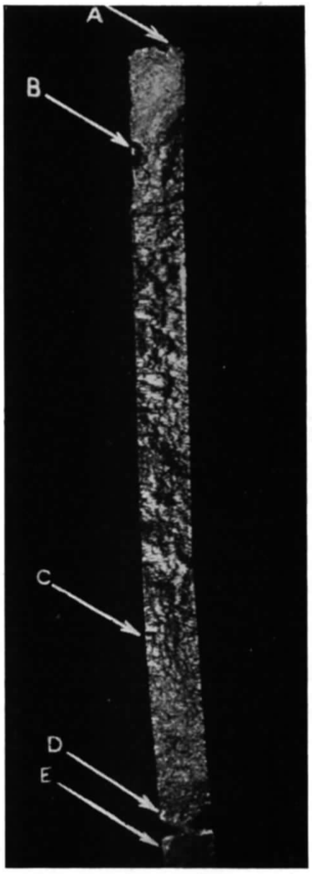

FIG. 6. Web crack.

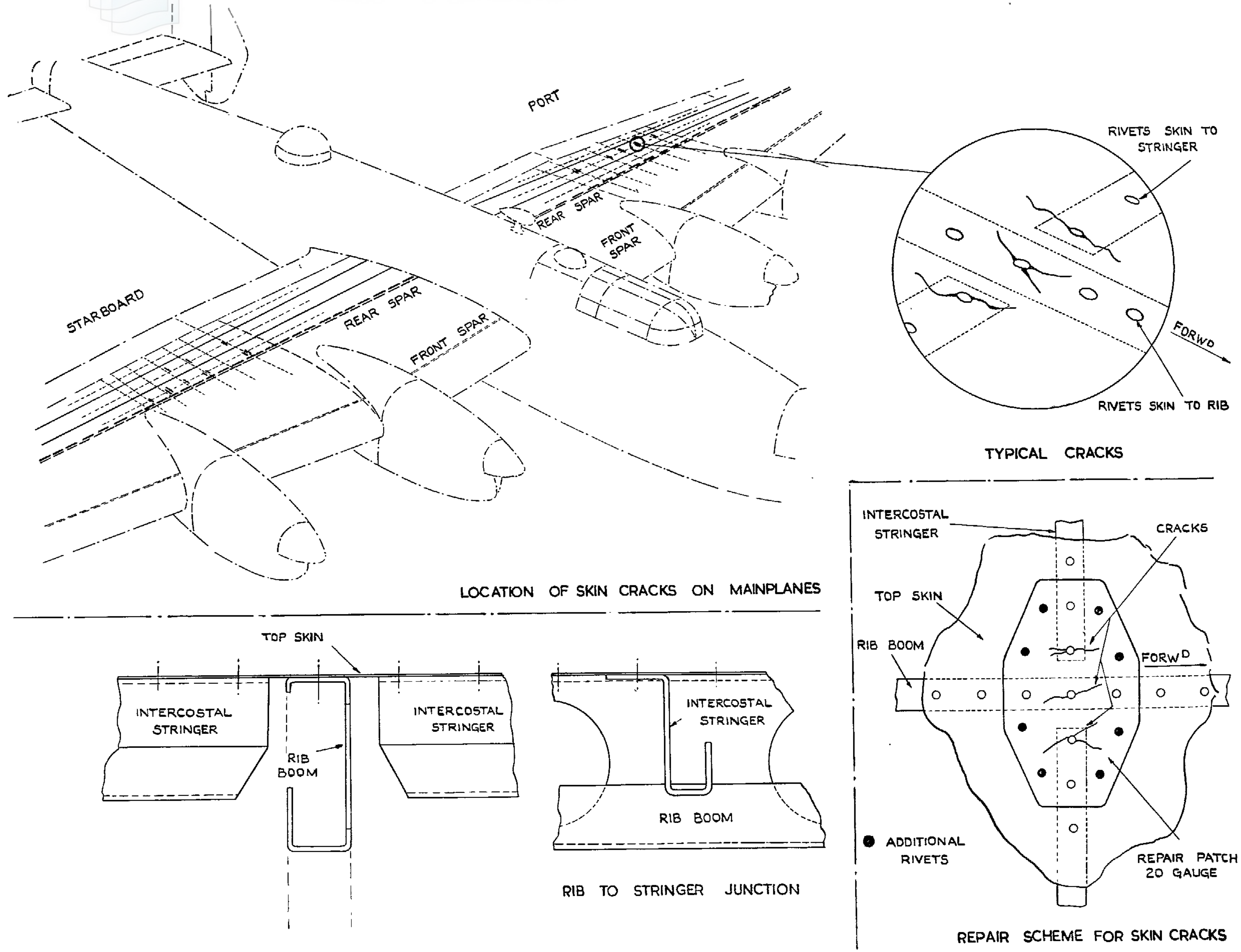

FIG. 7. Wing skin cracks.

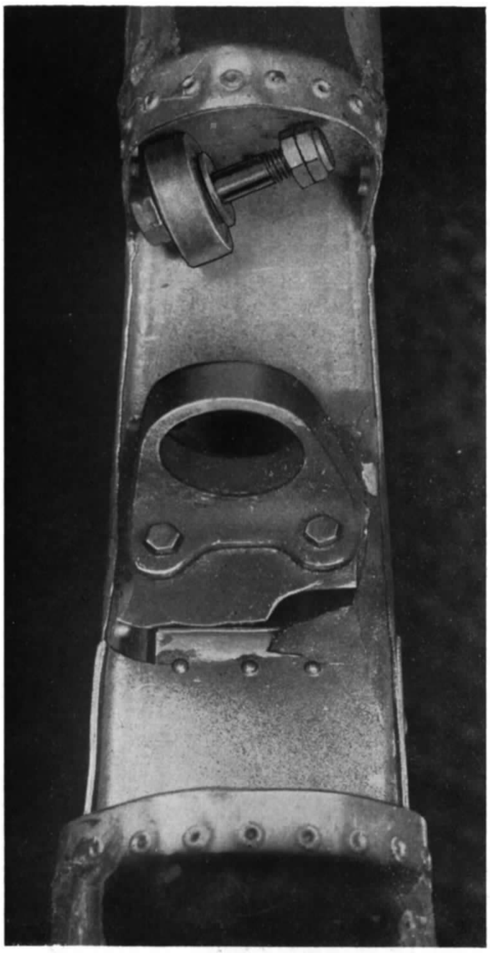

Fig. 8. Rudder hinge failure.

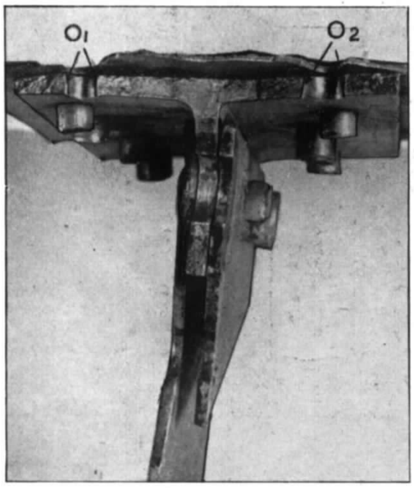

Fig. 9. Tailplane crack.

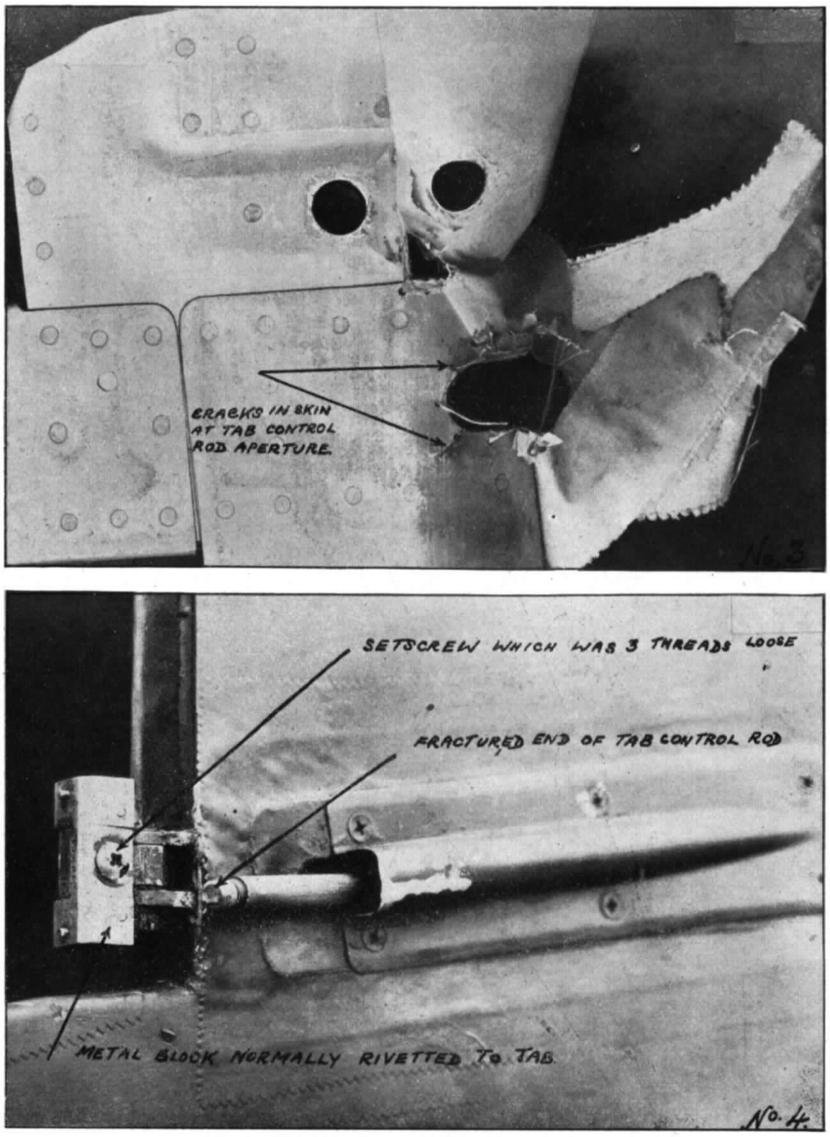

Fig. 10. Rudder failure.

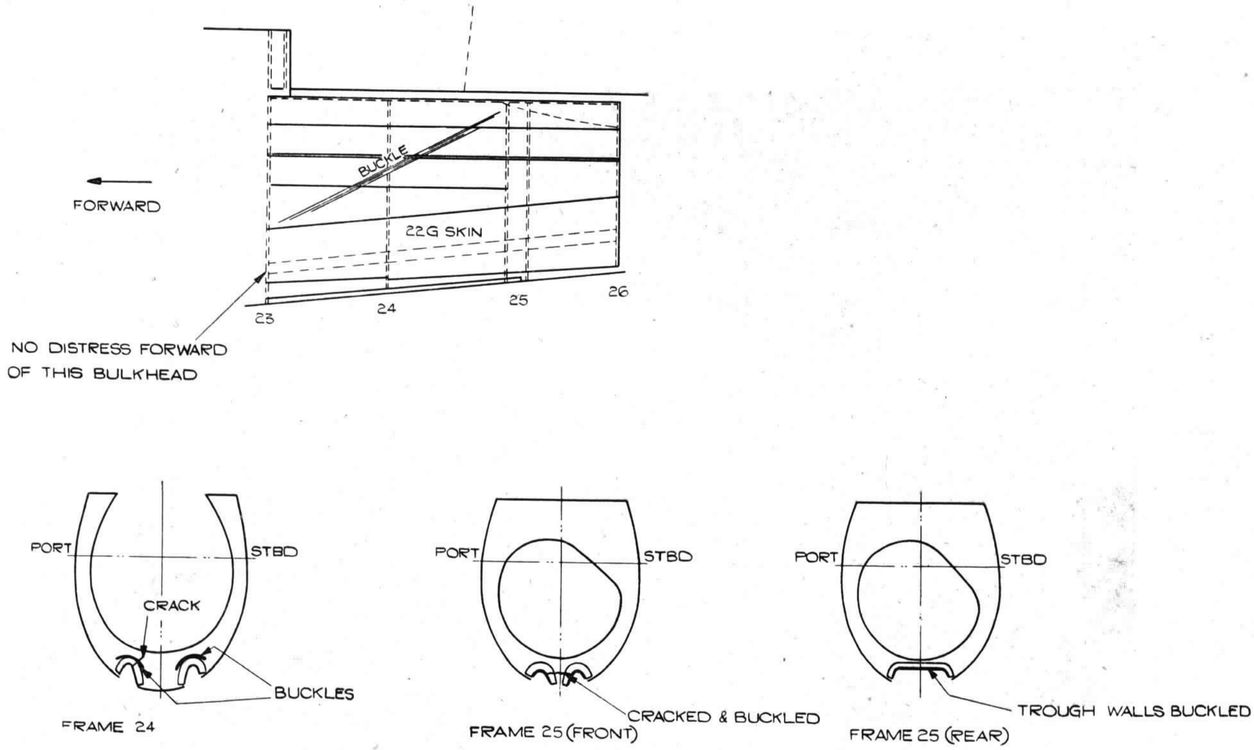

FIG. 11. Aft fuselage defects.

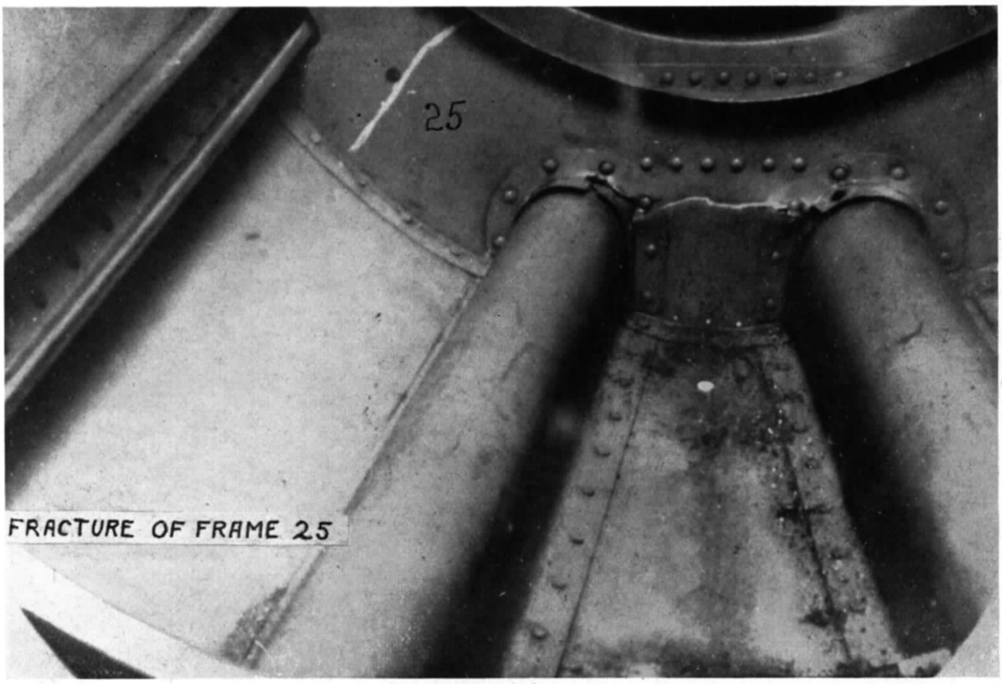

FIG. 12. View showing damage at frame 25.

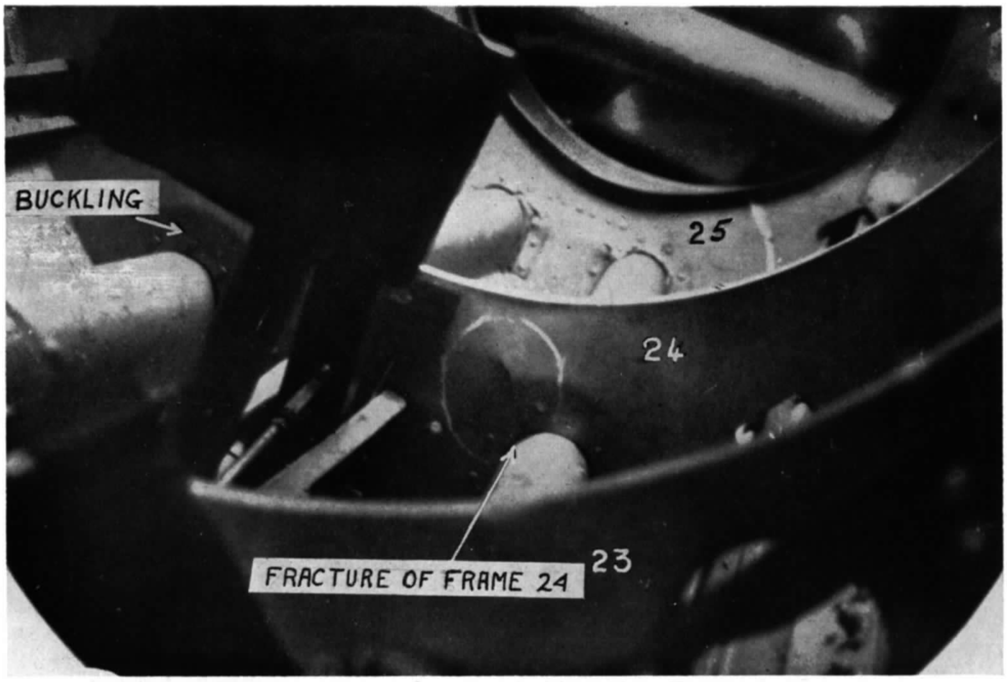

Fig. 13. View showing damage at frames 24 and 25.

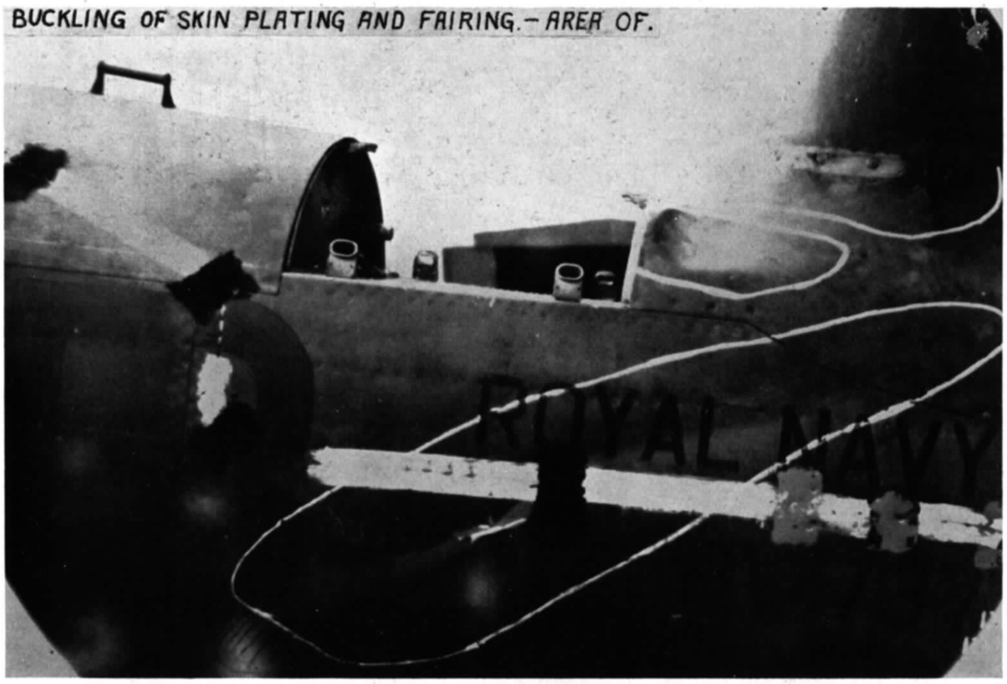

Fig. 14. External buckle on fuselage.

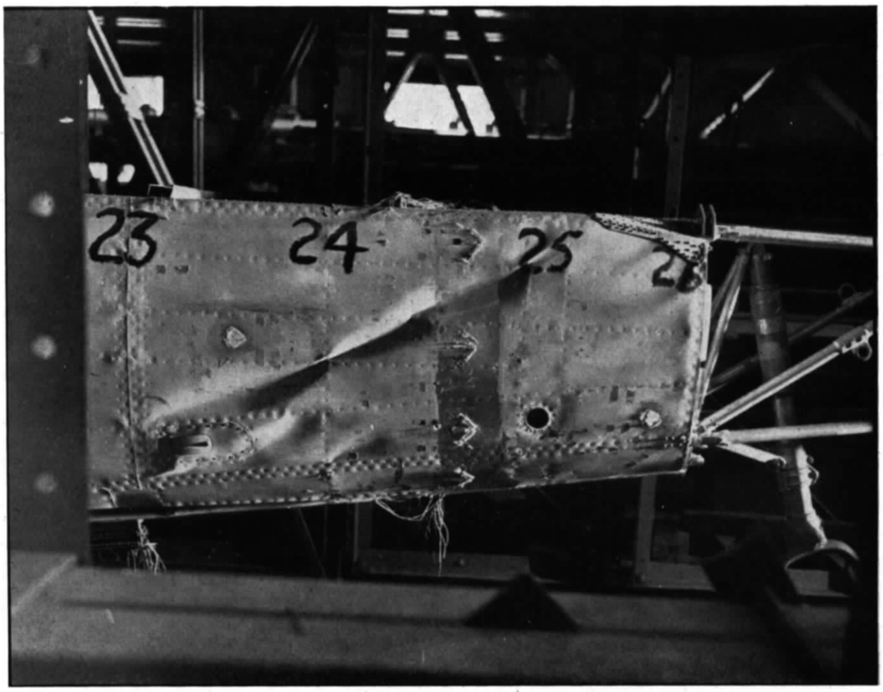

FIG. 15. - Fuselage failure in test.

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