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Determination of the Life Remaining in the Model HU-16E Airplane Wing

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DEPARTMENT OF THE NAVY
U. S. NAVAL AIR DEVELOPMENT CENTER
JOHNSVILLE
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DETERMINATION OF THE LIFE REMAINING IN THE MODEL HU-16E AIRPLANE WING

FINAL REPORT

A laboratory fatigue test was performed to determine the remaining service life of the HU-16 airplane. A Coast Guard HU-16E airplane, considered to be representative of those in service and having a total of 7,216 flight hours, was withdrawn from service and used as the test vehicle. During the fatigue test, a total of 8,200 test hours were accumulated on the test article prior to catastrophic wing failure. Post-failure examination revealed the presence of exfoliation (corrosion) in the wing main beam lower spar cap. The program results indicate that the presence, but not necessarily the amount, of exfoliation was a determining factor in the wing fatigue failure.

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SUMMARY

A model HU-16E aircraft was fatigue tested to determine its remaining service life. This test was required due to the discovery of exfoliation in the 7075-T6 aluminum alloy spar caps of the wing center section of some HU-16's. This program was jointly sponsored by the U. S. Air Force, U. S. Coast Guard, U. S. Navy, and the Canadian Armed Forces.

A Coast Guard HU-16E airplane, considered to be representative of those in service and having a total of 7,216 flight hours, was withdrawn from service and used as the test vehicle. During the fatigue test, a total of 8,200 test hours were accumulated on the test article prior to catastrophic wing failure. Post-failure examination revealed that the failure originated in the main beam lower spar cap in the area of the jack pad holes. The presence of exfoliation (corrosion) was also revealed during this examination. The program results indicate that the presence, but not necessarily the amount, of exfoliation was a determining factor in the wing fatigue failure.

It is recommended that a wing reinforcement, in the form of a steel strap doubler on the spar cap, be installed on U. S. Coast Guard aircraft no later than 9,500 flight hours and on other user agencies aircraft no later than 6,500 flight hours. The difference in flight hours being due to a difference in fatigue damage accumulation as a result of service usage.

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INTRODUCTION

The presence of damage due to corrosion in extruded 7075-T6 spar caps in the wing center section of some HU-16 aircraft was revealed several years ago. This damage was in the form of fine cracks emanating from rivet holes and running parallel to the extrusion direction.

An investigation was performed, reference (a), to determine the cause of the cracking and its effect on the fatigue life of the part. This investigation determined that the primary cause of the cracks was exfoliation which started at the rivet holes. The investigation also indicated, based on specimen tests, that the presence of this internal exfoliation reduced the fatigue strength 40% to 60% from that of uncorroded spar caps.

On the basis of these determinations it was recommended that a full scale fatigue test of the entire wing assembly of an HU-16E airplane be performed in order to determine the life remaining in this model aircraft. The recommendation was subsequently accepted and a follow-on full scale fatigue test was jointly funded by the U. S. Air Force, U. S. Coast Guard, U. S. Navy, and the Canadian Government. The Coast Guard supplied the test airplane used in the program.

DESCRIPTION OF THE TEST SPECIMEN

The service airplane made available for the test was the HU-16E airplane, Coast Guard Number 1264. The history of the airplane, as of October 1966 when it was made available for test, is as follows:

Tour No.	Date	Calendar Months	Flight Hours	Water Landings	Field Landings
1	6/52-7/54	25	1116.2	?	?
-	7/54-2/55	7	Overhaul & Modification		
2	2/55-5/57	27	1059.8	?	?
-	5/57-9/57	4	Overhaul & Modification		
3	9/57-9/60	36	1457.7	?	?
-	9/60-2/61	5	Conversion (UF-1G to UF-2G (HU-16E)) & Overhaul		

Tour No.	Date	Calendar Months	Flight Hours	Water Landings	Field Landings
4	2/61-5/63	27	1702.8	?	?
-	5/63-8/63	3	Standard Rework		
5	8/63-8/65	24	1255.4	145	598
-	8/65-12/65	4	Standard Rework		
6	12/65-10/66	11	690.4	26	274
-			Fatigue Test		
Totals		173	7,282.3	171+	872+

The wing consisted of a center section permanently bolted and riveted to the hull and two outer panels. The wing center section extended to W. S. (wing station) 169 and was symmetrical about the airplane centerline. The center section was of the typical two spar construction.

The front spar (10% chord line) and main spar (35% chord line) were of conventional design with sheet aluminum webs and with extruded angles of 7075-T6 alloy forming the caps. Riveted web stiffeners added rigidity. The box is completed by riveted aluminum alloy (2014-T6) top and bottom skin. Ribs are of sheet metal and truss construction. A D-shaped leading edge riveted to the front spar, a flap closure beam, flaps, and riveted trailing edge skin complete the construction. The engines and nacelles are part of the wing center section. Bladder type fuel cells are also enclosed in part of the area between the spars. A view of the wing center section is shown in figure 1.

This wing is an in service conversion/modification of the basic HU-16E wing. Part of this modification was accomplished by adding 70 inches to the outboard end of each original center section as shown in figure 2. This increase in span required strengthening of the existing center section spar caps. This was accomplished by nesting laminated SAE 4130 steel straps inside the horizontal leg of three of the spar caps and on both sides of the horizontal leg of the lower main spar cap as shown in figure 3. These straps are fastened to the cap angles by the skin attachment fasteners. Figure 4 shows a typical strengthened spar cap.

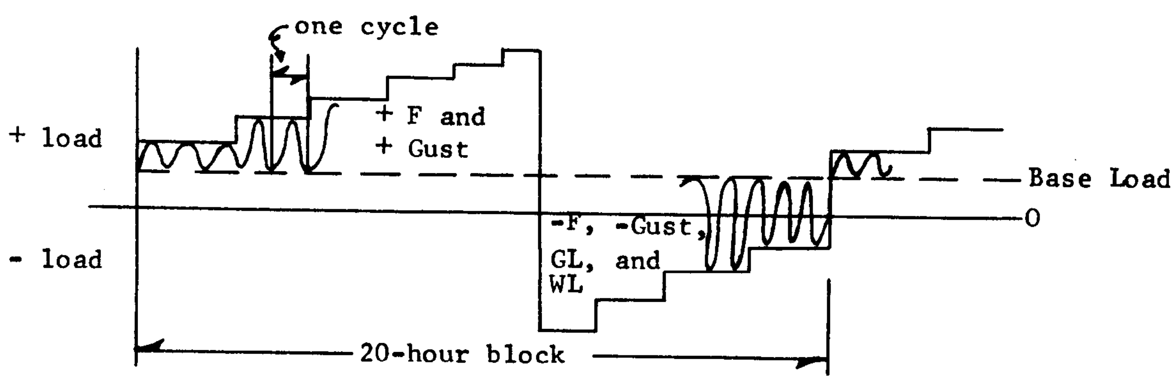
The outer panel, similar in construction to the center section, extends from W. S. 229 to W. S. 580, the wing tip. The outer panel-center section splice or gap band is from W. S. 219 to W. S. 229.

The extended wing model of the HU-16 airplane has the following designation: HU-16B -- U.S.A.F., HU-16D -- U.S.N., HU-16E -- U.S.C.G.

TEST PROGRAM

The test program consisted of one wing fatigue test, combining flight loads and landing loads. The test loading conditions, test loads, and point of application of the test loads are included in Appendix A. The loading was symmetrical, involving both port and starboard wings.

The loading spectrum was a combined spectrum consisting of the positive and negative flight maneuver spectrum C of MIL-A-8866 (reference (b)), the ground landing spectrum of MIL-A-8866, a water landing spectrum, and a gust spectrum. The latter two spectra were developed by the Naval Air Development Center and the Grumman Aerospace Corporation and were partially based on actual measured data on similar type aircraft. The spectrum was applied in a lo-hi sequence with a 20-hour block size and a 1g base load. A graphical description of the sequence is shown below:



Where, +F = positive flight maneuver
-F = negative flight maneuver
GL = ground landing
WL = water landing

The maneuver spectrum, gust spectrum, and landing spectra are shown in tables 1, 2, and 3, respectively, in terms of frequency per 1,000 hours. The combined positive loading condition spectrum and negative loading condition spectrum are shown on table 4 in terms of frequency per 1,000 hours. The spectra are shown in tables 5 and 6 for the block to block frequency used during the test, where one block is equivalent to 20 flight hours.

METHOD

The test airplane was supported in a test jig at two points; the forward support point being the bulkhead at hull station 200 and the aft support being the bulkhead at hull station 568. The design of the supports was such that the fuselage was free to roll, but restrained against pitching and yawing. The airplane was mounted in the jig so that the wing reference line was parallel to the floor and the plane of symmetry was perpendicular to the floor. An overall view of the setup is shown in photos 1 and 2.

Loading of the wing was accomplished by the use of formers and hydraulic jacks. Loads were applied to both wing semispans simultaneously. External loads were applied to the hull and cycled simultaneously with the wing loads to relieve hull bending. All loads were applied by hydraulic actuators which were part of an electro-hydraulic, closed loop, servo controlled loading system. Each actuator had independent control by means of individual servo valves and servo controllers with load cells to generate feedback information. The load direction and phase relationship for the multiple actuators were programmed by a four channel, punched tape, digital programmer which generated the command signal for each actuator. The loads were monitored on strip chart recorders which had been modified to provide overload protection. Two additional and independent overload protection systems were provided by trimmable error detectors on each servo control unit and stroke limit switches on each actuator. Triggering any overload system would immediately dump hydraulic pressure at all actuators and at the hydraulic power supply. Photograph 3 shows the programmer/controller system.

The tare weight of the test specimen and load apparatus was counterbalanced by the use of dead weight.

Strain and deflection measurements were taken in blocks 2, 15, 25, 50, 200, 410, 435 and 610. Detailed visual inspections for fatigue damage were also performed during these blocks. Minor visual inspections were performed every day of test.

After final failure of the right-hand center section a GAC (Grumman Aerospace Corp.) designed steel strap reinforcement (reference (c)) was incorporated on the left-hand center section as shown in photo 4 and the loading system and jig structure were modified to permit continued testing of this semispan. The GAC steel strap was approximately 24 inches long extending from W. S. 144 5/32 to W.S. 168 27/32, 3 inches wide at the inboard end, and 1.4 inches wide at the outboard end. The strap thickness was stepped from .07 inches inboard to .312 inches at the jack pads to .09 inches outboard. It was fastened to the main beam lower spar cap by the use of steel fasteners and was designed to function as a spar cap stress reducer rather than as a wing fix.

RESULTS

The test results, catalogued as to the block in which they occurred, are given below:

Block 1 (20 test hours): In the 55% 1.1. (load level) an oil can effect on the upper wing skin was discovered. This effect was occurring on both semispans just inboard and outboard of the nacelles (W. S. 87.5 and 152.5) and 75 inches forward of the trailing edge. The oil can effect is a common condition for the type of wing construction being tested and continued, with no ill effects, throughout the entire test.

Block 25 (500 test hours): At the 125% 1.1. the lower skin rivets (type AD-4's) at the aft-cover main-beam connection between W. S. 169 and 190 failed. The failure was a shear failure at the skin inner surface and was identical on both semispans. Approximately 80 rivets (40 per side) failed. The rivet failure for the right-hand wing is shown in photos 5 and 6. A detailed inspection was performed and indicated no other damage. The rivets were replaced with identical rivets and testing continued

Block 30 (600 test hours): At the 105% 1.1. the new rivets on the left-hand semispan (those installed in block 25) failed. The rivets on the right-hand semispan were undamaged. A detailed inspection again indicated no other damage. The rivets on the left-hand wing were again replaced, but with the next larger size rivets (AD-5's). The rivets in the right-hand wing were not changed.

The U. S. Coast Guard did not feel that the 115% and 125% load levels were realistic in so far as representing Coast Guard service usage and hence were penalizing the test results. The rivet failures which had not occurred in service, gave credence to their argument and as such they requested that the 115% and 125% load levels be eliminated from the test spectrum. This request was complied with beginning in block 31.

Block 89 (1,780 test hours): At the 95% 1.1. one rivet on the right-hand semispan (W. S. 169 at the same location as the block 25 failures) was found with its head sheared off. The next three rivets outboard were observed to have cocked heads. All the rivets along this line, from W. S. 169 to W. S. 180, were removed and replaced with oversize (0.135 inch diameter) cherry-monel rivets and testing continued.

Block 128 (2,560 test hours): The rivets on the left-hand semispan, in the same location as the block 25 incident, were observed to be cocked. Since the rivets were not sheared, they were not removed and testing was continued. At this time the -90% and -100% load levels, which had not as yet been called for in the test spectrum and therefore had not as yet been applied, were eliminated from the test spectrum, using the same reasoning as that used in eliminating the 115% and 125% load levels.

Block 260 (5,200 test hours): At the 105% 1.1. a loud noise was heard. A detailed visual inspection was performed and no damage was found. Testing was therefore continued.

Block 410 (8,200 test hours): At the 105% 1.1. a catastrophic failure of the right-hand semispan center section occurred. External visible damage consisted of skin tear out on the lower skin aft of the main beam between W. S. 159 and W. S. 169 as shown on photo 7. The area between these two wing stations is considered the inboard splice joint for the wing extension. Internal damage consisted of the following: Failure of the trailing edge between W.S. 159 and 169 as shown in photo 8. Failure of the front beam web and lower spar cap at W. S. 139, shown in photo 9. Failure of the lower skin front beam connection from W. S. 139 to the rib at W. S. 159 as shown in photo 10. This photo also shows the failure of the W. S. 159 rib-lower wing skin connection. Failure of the main beam web at W. S. 153, the lower spar cap at W. S. 155 15/32, and of the beam-lower skin connection between W. S. 153 and W. S. 159, all of which is shown in photo 11. A close up view of the main beam lower spar cap failure is shown in photo 12. The failure occurred at the outboard wing jack fitting hole in the main beam lower spar cap at W. S. 155 15/32, also shown in photo 12. Each semispan has two 21/64 inch diameter wing jack fitting holes. The holes are in the main beam lower spar cap and are centered about W. S. 154 11/32 and are 2 1/4 inches between centers. The hole at station 155 15/32 is called the outboard jack hole while the one at station 153 7/32 is called the inboard jack hole. These holes are drilled completely through the lower spar cap and receive the jack pads which are used when jacking the wing for replacement or testing of the main landing gear. The failure at the outboard jack hole was the main failure, all other failures (e.g. front beam, rib, etc.) were secondary. Inspection also revealed that the internal steel straps on the horizontal leg of the lower main beam spar cap besides having failed at the outboard jack hole were cracked at the inboard jack hole. The upper steel strap was cracked from the edge of the jack hole forward to the edge of the steel strap, as shown in photo 13, while the lower strap besides having the same crack as the upper strap was also cracked from the jack hole aft about 1/16 inch. The visual examination also revealed the presence of two 1/8 inch diameter rivet holes, one forward and one aft of each jack hole. Examination of the inboard jack hole area showed that these rivet holes had been plugged. This can be seen in photos 13, 14 and 15. A sketch of the fracture area indicating the plugged holes, is shown in figure 5. An examination of the fracture surface of the lower main spar cap, shown in photos 16 and 17, reveals that there was a considerable amount of crack growth. Counting in toward the failure initiation point, at least 15 striations or beach marks (representing 75 blocks of loading or 1,500 test hours) predominately emanating from the plugged rivet holes, are visable to the eye prior to beach mark wash out. A metallographic and fractographic examination of the fracture of the 7075-T6 aluminum alloy lower main beam spar cap and the two attached SAE 4130 steel straps was performed by the AMD (Aero Materials Department). This examination, reported on in reference (d), exposed the presence of severe intergranular corrosion (exfoliation) perpendicular to the fracture surface. This is shown in photos 18 and 19. Photo 18 shows the fracture surface, a sketch indicating the orientation of samples used for the metallographic examination, and a metallographic section showing intergranular corrosion. Secondary fatigue cracks can be seen emanating from the corrosion laminations. Photo 19 shows the metallographic section of photo 18 after further polishing of the surface to a depth of .010 inches. The fractographic examination disclosed that the predominant topographic feature in the vicinity of the jack and rivet holes was fatigue striations, shown in parts a and b of photo 20. Beyond the rivet hole areas, ductile rupture was the main fracture mechanism.

Fractographic examination of the steel strap failure surface showed a dimpled structure (part C of photo 20) indicative of a ductile fracture caused by a simple overload.

An ultrasonic inspection of the lower cap strips of the right-hand wing, outboard of the failure, was performed. For this inspection the lower skin covering the main and front beams was peeled back from W. S. 169 to W. S. 280. The inspection indicated slight exfoliation in the beam lower cap strip between stations 229 and 239 (the outboard wing splice joint). The exfoliation was observed around the single row of fastner holes.

A detailed examination of the left-hand wing in the area of the jack holes was performed after the right-hand wing failure and prior to test continuation. This examination performed by X-ray, indicated that one of the internal steel straps was cracked from the forward edge of the outboard jack hole to the forward edge of the strap. The inspection also seemed to indicate two very small cracks emanating from the forward edge of the plugged rivet hole which is located forward of the inboard jack hole. These indications of cracks from the plugged rivet hole were not evident in **later** inspections.

At this time the right-hand wing was dummied-up by fabricating a fake wing from steel beams. This dummy wing was tied down and testing of the left-hand wing continued.

Block 615 (12,300 test hours): At the 105% 1.1. a crack developed in the left-hand wing, aft-cover skin as shown in photo 21. This crack originated at the aft-cover main-beam skin fastening rivet at W. S. 184 1/2 and extended approximately 7 inches. This crack was stop-drilled and testing resumed. Prior to test resumption, a visual inspection of the interior of the wing at the jack pad holes was performed. This inspection which was hampered by the presence of sealant did not reveal any structural damage.

Block 625 (12,500 test hours): At the 105% 1.1. a catastrophic failure of the left-hand semispan center section occurred. This failure, shown in photo 22, was almost identical to the right-hand semispan failure. The major difference was that the failure originated at the inboard jack pad hole rather than at the outboard hole as in the right-hand wing failure. Examination of the fracture surface indicates a large amount of exfoliation, as evident in photo 23, and a large number of beach marks. Counting in toward the failure initiation point, 43 striations (representing 215 blocks of load or 4,300 test hours) predominately emanating from the plugged rivet holes, are visible to the eye prior to beach mark wash out. This wash out area is about six times larger than that of the right-hand wing fracture surface. Fatigue markings are also evident in the lower internal steel strap, also shown in photo 23. Secondary failure of the GAC steel strap also occurred. The failure mode was shear tear out at the single fastner at W. S. 160 1/2.

Strain gage readings were taken at various times during the test program. The results of these readings are listed in table 7 and plots of some of the readings are shown in figures 6 and 7. As is shown, there was a significant change in strain after the installation of the GAC steel strap. However, there were no changes in strain prior to failure which might have indicated impending failure. Wing deflection measurements were also made during the program. A listing of the results of these measurements can be found in table 8.

DISCUSSION

The failures of the lower wing skin rivets early in the test program were due to a local phenomenon and have no bearing on the final catastrophic wing failure. This is evident because more than 6,000 test hours were endured between the occurrences of the two failures, the rivet failures were outboard of the wing failure, and the rivet failures can be accounted for by a design deficiency. This deficiency is shown in figure 8 and takes the form of a skin doubler splicing across the wing station 169 skin-breather joint. When high loads are encountered, the breather joint tries to breathe, the doubler tries to prevent the movement and, therefore, the rivets outboard of the breather joint fail. These rivet failures are not critical provided the airplane is not loaded beyond 105% of the design limit load. These rivets can be used as a load indicator in that if they are found to be failed on a service airplane it would indicate that the airplane had exceeded 105% of limit load (2.91g).

Examination of the fracture surface of the wing catastrophic failure showed the presence of two 1/8 inch diameter rivet holes, one forward and one aft of each jack hole. These can be seen in figure 5 and photos 13, 14, 15, 16, 17, and 23. Investigation revealed that these two rivet holes are called out in the GAEC drawing 85011 for the short wing model airplane as being for AD-4 rivets to hold the jack fitting retaining clip. During the conversion from the short wing to the long wing version of the airplane, these rivet holes were plugged, steel straps installed over the holes and new retaining clip fastener holes were drilled at a spanwise distance from the jack holes (outboard of the outboard jack hole and inboard of the inboard jack hole). As shown in figure 5, there is only 1/8 inch of material between the jack hole and each plugged rivet hole. It is also evident that the jack hole and each plugged rivet hole is countersunk and that these countersinks extended into each other. Also, as shown in the photos listed above, the forward plugged hole is very close to the radius between the horizontal and vertical legs of the spar cap. As can be seen in photo 24, the major portion of the fatigue marks emanate from the plugged rivet holes. It is clearly evident that there was crack initiation at the forward edge of the forward plugged rivet hole and at the aft edge of the aft plugged rivet hole. There was also some crack initiation at the jack pad hole. It is impossible to scientifically determine where the first crack initiated, whether at the jack pad hole or the plugged rivet hole. However, it is believed that the crack originates at the plugged rivet hole. It then progresses a certain amount at which time a crack initiates and grows in the area between the jack and plugged holes. This latter crack progresses to the point where the material remaining between the holes yields. During this time, the cracks which originated at the plugged holes have grown through the area which eventually became washed out. When the material between the holes yielded, the crack from the plugged hole jumped and and then continued growing until final failure occurred. This theoretical crack propagation history is sketched in figure 9.

The length of time, in flight hours, for which the spar cap was cracked can be estimated by counting the fatigue crack striations. Because of their distinctive shape these are sometimes called "beach marks" or "clam shell marks". By knowing the loading spectrum and assuming that the crack jumps with the application of a 105% load level it can be shown that, prior to wash out, the crack in the right-hand wing existed for 1,500 hours and that in the left-hand wing for 4,300 hours. The term "wash out" is used to connote the area of early fatigue cracking wherein continued working of the crack erases or washes out the fatigue striations. Since there are no striations in the wash out area, and it is known that crack propagation is a non-linear phenomenon, it is impossible to arrive at the actual time of crack initiation. Examination of the left-hand and right-hand wing fracture surfaces show that the wash out area on the right-hand wing is approximately .02 inches while that on the left-hand wing is .12 inches. Measurements taken on these fracture surfaces indicate three striations in the .02 inches adjacent to the wash out area. Three striations equals 300 flight hours, therefore, by erroneously assuming crack propagation as being linear, a minimum amount of time that the crack is in the wash out area can be estimated as follows:

	Left-hand	Right hand
Time to Failure	12,500 test hrs	8,200 test hrs
Striation Count	-4,300 test hrs	-1,500 test hrs
Time to Wash-out Boundary	8,200 test hrs	6,700 test hrs
Minimum Time in Wash-out Area	-1,800 test hrs	-300 test hrs
Maximum Time to Crack Initiation	6,400 test hrs	6,400 test hrs

On this basis, it can be assumed that the crack on the left-hand wing and the one on the right-hand wing started at approximately the same time, 6,400 test hours. However, it must be understood that this is an assumed time and that actual crack initiation should have theoretically occurred prior to the 6,400 test hours. It is possible that the crack existed prior to start of test.

Considering the severity of exfoliation in the spar caps (the left-hand wing cap being extremely more exfoliated than the right-hand wing cap), the fact that the heavier exfoliated left-hand wing did not fail first, and the premise that the fatigue crack in each spar cap started at approximately the same time, it is indicated that the presence, but not necessarily the severity of exfoliation was a determining factor in the wing fatigue failure. That is, once the material experiences any amount of exfoliation type corrosion a fatigue crack can and most likely will originate. As soon as the crack has started, the stress level at the tip of the crack is sufficiently high so as to negate any further stress raising effects due to exfoliation. This hypothesis, which would require an experimental program to verify, explains why the less exfoliated right-hand wing failed first.

The use of the GAC designed strap on the left-hand wing definitely provided a life extension of 4,000 test hours for the test wing. This was accomplished by the fact that the strap reduced the critical prime structure stress level in the jack fitting hole areas by about one-third, as can be seen by examining the curves of figures 6 and 7 in the vicinity of W. S. 155. As discussed previously, it is believed that a crack existed in the left-hand spar cap at the time the strap was installed (8,200 test hours). It would therefore be logical to assume that if the strap were installed prior to the spar cap crack initiation it would provide a greater life extension. However, without test results this assumption cannot be verified.

As discussed under Results, a crack was observed in the aft-cover skin at 12,300 test hours. It is apparent that this crack was an indication that complete failure was imminent. If the GAC strap is installed on a service aircraft and an aft-cover skin crack occurs, the aircraft should be grounded. It should be borne in mind, however, that this crack occurred only 200 test hours prior to final failure. Therefore, its use as a failure indicator is extremely limited.

Various NDI (Nondestructive Inspection) techniques were tried on the test vehicle. As indicated in the Results section of this report no definite cracks were found prior to catastrophic failure. If the theory indicated in figure 9 is correct, an eddy current inspection of the jack pad holes might prove to be an acceptable inspection technique. However, depending on the crack propogation rate in each individual aircraft, crack indications in the jack pad hole might not occur until just prior to catastrophic failure. A more acceptable method of inspection would be to eddy current inspect the plugged rivet holes. This, of course, would necessitate some structural disassembly.

An eddy current inspection of the jack pad holes of in-service aircraft has recently been performed by the U. S. Air Force with the results that crack indications were obtained in the following aircraft at the times listed:

Air National Guard Long Wing HU-16 ------ 5,300 hours
U. S. Air Force Long Wing HU-16 --------- 7,500 hours
Two Nationalist Chinese Short-Wing HU-16's -- unknown

The extent of cracking and the absolute certainty of the existence of a crack due to fatigue is currently unknown on all but the National Guard airplane. On this airplane, it has been determined by an Air Force materials analysis that the indication was due to a gap between grain boundary planes caused by intergranular corrosion.

Three different HU-16 spectra are shown in figure 10 along with the positive spectrum C of MIL-A-8866 and the positive S-2D/E and P-3A/B spectra. As indicated on figure 10, the Coast Guard spectrum is based on 950 hours of VGH (velocity, acceleration, altitude) data from six aircraft at three different locations (Brooklyn, N. Y., Salem, Mass., and Miami, Fla.) and included 67 smooth water landings. The Canadian spectrum is based on 27 hours of counting accelerometer data with no water landings and both the S-2D/E and P-3A/B spectra are based on counting accelerometer data for the hours indicated. Additional information on the average number of water landings per 1,000 flight hours normally flown by the various services is also shown on figure 10. It must be understood that the Canadian spectrum curve does not include this water landing data. As can be seen, and inferred by the water landing data, the test spectrum represents an envelope loading spectrum for the diversified use to which the HU-16 is put by the various operators concerned with its service life.

The HU-16 airplane wing service life, based only on the test results, can be considered in three different ways as follows:

1. Previous service hours + maximum test hours to crack initiation = total life.

   7,216 + 6,400 = 13,610

2. (Previous service hours + maximum test hours to crack initiation) / 2 = total life.

   (7,216 + 6,400) / 2 = 6,805

3. Previous service hours + (maximum test hours to crack initiation) / 2 = total life.

   7,216 + (6,400) / 2 = 10,416

The factor of 2, in two of the above methods, is a scatter factor. In fatigue tests performed for and by the U. S. Navy it is standard practice to assume an average airframe and then to use a severe loading spectrum and a scatter factor of 2. An average airframe is one that is of average quality in construction and materials used. Standard practice in fatigue tests for the U. S. Air Force is also to assume an average airframe but to then use an average loading spectrum and a scatter factor of 4. It is argued that both practices lead to the same life.

Since the aircraft tested was considered to be a representative Coast Guard vehicle it is assumed for the determination of service life, that all the Coast Guard HU-16's would have experienced a similar service environment. That is, in determining a model service life based on the results of a full-scale fatigue test, the actual service flight hours already imposed on the test aircraft need not be reduced or modified. For this reason it is felt that the third method for determining service life, as shown above, could be used for Coast Guard Aircraft. It is also felt that the second method, as shown above, should be used by the other services operating HU-16 aircraft. This is based on the consideration that the Coast Guard has the least severe total operational loading spectrum of all the user agencies, as indicated in figure 10, and that there is only a very limited amount of flight data available for these other agencies.

Estimates of service life for a new unmodified wing were made using a modification of the life estimation method described in reference (g). The results of the estimates are as follows:

Spectrum	Life
Coast Guard, positive loads only (fig. 10)	428,380 hours
Canadian, positive loads only (fig. 10)	88,760 hours
Test, positive loads only (fig. 10)	75,240 hours

To these lives would then be applied a scatter factor of 2, a corrosion factor of 2 (see reference (a)), and a factor of 2 to account for normal negative loading (see reference (h)). The estimated lives for new aircraft wings using the positive loading spectrum shown in figure 10 would then be as follows:

Coast Guard	53,500 hours
Canadian	11,100 hours
Test	9,405 hours

These life estimates apply for the model average. Individual airplanes, flying various loading spectra, would have lives in direct relationship to their spectra. That is, an aircraft experiencing a loading spectrum more severe than the average would have a shorter life than the model average and vice versa. These individual lives could vary widely from the model average. It must also be kept in mind that the life estimation method does not give highly accurate results, particularly in the regions of extreme long or short lives, and is best used as a measure of relativity between the various spectra. For example, barring all parameters but the positive loading spectra, it could be said that a Coast Guard aircraft would have a life almost five times that of a Canadian aircraft. These estimated lives must also be tempered by the knowledge that, as mentioned previously, crack indications (of undetermined origins) have been observed in operational aircraft having relatively low flight times.

CONCLUSIONS

The test vehicle sustained 8,200 test hours and 7,216 service hours prior to catastrophic failure of the right-hand wing. Examination of the fracture surface indicates that the failure originated in the main beam lower spar cap at the jack pad hole area which is approximately at wing station 155 and it is concluded that the crack initiated after approximately 6,400 test hours had been applied.

Installation, at the jack pad hole area on the left-hand wing, of the GAC designed steel strap allowed this semispan to sustain a total of 12,500 test hours prior to catastrophic failure.

Comparison of the amounts of exfoliation at the left-hand wing and right-hand wing fracture surfaces leads to the conclusion that the existence but not necessarily the amount of exfoliation is a determining factor in fatigue life. That is, the existence of even a minute amount of exfoliation can initiate a fatigue crack.

Eddy current examination of the jack pad hole is concluded to be the only non-destructive investigation method available for detection of a fatigue crack in this area without structural disassembly.

RECOMMENDATIONS

Based on a consideration of the flight loads and water landing data available, the test results, and the indications of cracks in four operational aircraft, it is recommended that an immediate eddy current inspection of the jack pad holes of all HU-16 airplanes be performed. If feasible (e.g. airplane is at a major rework facility) the plugged rivet holes should be unplugged and inspected, also using eddy current. Indication of cracks should be sufficient justification to "ground" the airplane pending a more thorough investigation. It is further recommended that the GAC steel strap be installed on all HU-16 airplanes at the earliest opportunity, such being no later than 9,500 flight hours for U. S. Coast Guard airplanes and no later than 6,500 flight hours for other user agencies airplanes. After installation of the steel strap an eddy current inspection should be performed every 200 flight hours or less. It is also recommended that after the steel strap installation the U. S. Coast Guard consider the feasibility of retiring their aircraft at 12,500 flight hours and that the U. S. Air Force, U. S. Navy and Canadian Armed Forces consider the feasibility of retiring their aircraft at 9,500 flight hours. If it is determined that there is some exfoliation in the spar cap at the time the steel strap is installed, it is recommended that the eddy current inspection be performed no later than every 100 flight hours.

It is also recommended that if it is required to continue use of the HU-16 aircraft beyond the above flight hours, either the spar caps be replaced with new ones or a major structural modification be performed for the critical jack pad hole area.

ACKNOWLEDGEMENTS

The author wishes to acknowledge the valuable assistance, during the test program, of the project team members. This team consisted of Messrs. H. Lystad, E. Kautz, P. Kozel, J. Bauer and J. Sebastian of the Naval Air Development Center.

REFERENCES

a. NAEC-AML-Report 2538, "Corrosion and Fatigue Evaluation of Spar Cap Specimens from HU-16 Wings", 1 Sep 1967

b. Military Specification, MIL-A-8866, "Airplane Strength and Rigidity Reliability Requirements, Repeated Loads, and Fatigue", 18 May 1960

c. Grumman Aircraft Engineering Corp. drawing DR246B, "Strap Installation, Wing C. S. Main Beam - Lower Cap - Reinforcement", 13 Jan 1969

d. AMD Report No. MA069002, "Analysis of Failure of Spar Cap from HU-16E Aircraft", 3 Mar 1969

e. GAEC Report No. 554-1, "Flight Test Wing Stress Survey of Model HU-16E Aircraft", 28 Aug 1967

f. GAEC Report No. 2931-1B, "Static Test of SA-16B Wing", 6 June 1956

g. NAEC-ASL Report 1096, "A Method for Estimating the Fatigue Life of 7075-T6 Aluminum Alloy Aircraft Structures", December 1965

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TABLE 1 -- MANEUVER SPECTRUM C OF MIL-A-8866

Positive maneuver (L.L. = 2.77g)		Negative maneuver (L.L. = -.96g)		
% L.L.	f/1000 hr	% L.L.	% of positive L.L.	f/1000 hr
50	5,500	0	0	0.7
55	3,000	10	-3.45	0.5
65	1,000	20	-6.90	0.25
75	300			
85	100			
95	30			
105	10			
115	3			
125	2			

TABLE 2 -- DETERMINATION OF GUST SPECTRUM

U_{d_e}	Δn	Σf/mile	f/1000 hr	% L.L. (1g +Δn)	% L.L. (1g -Δn)
11.7	.385	2.4×10^{-1}	37,940	50	-
15.9	.524	3.5×10^{-2}	6,220	55	-
24.3	.801	1.2×10^{-3}	191	65	-
32.7	1.078	1.6×10^{-4}	25	75	-2.7
41.1	1.355	2.5×10^{-5}	4	85	-12.7
49.5	1.632	5.5×10^{-6}	.8	95	-22.7
57.8	1.909	1.2×10^{-6}	.15	105	-32.7
66.2	2.186	3.8×10^{-7}	.05	115	-42.7
74.6	2.463	1.2×10^{-7}	.02	125	-52.8

Where: $\Delta n = .033\ U_{d_e}$ at 184 mph cruise
L.L. = 2.77g
Σf/mile taken from S2F data
U_{d_e} = gust velocity, feet per second, equivalent airspeed

TABLE 3 -- DETERMINATION OF LANDING SPECTRA

Sinking speed, fps	Ground landing (from MIL-A-8866 with max. load = 3.08g) % max. load	f/1000 landings	f/1000 hrs.	Water landing (assumed same as GL with max. load = 5.99g) % max. load	f/1000 landings	f/1000 hrs.
1	10	180	176.58	10	180	47.52
2	20	290	284.49	20	290	76.56
3	30	260	255.06	30	260	68.64
4	40	155	152.06	40	155	40.92
5	50	78	76.52	50	78	20.59
6	60	26	25.51	60	26	6.86
7	70	8	7.85	70	8	2.11
8	80	1.5	1.47	80	1.5	.40
9	90	1.0	.98	90	1.0	.26
10	100	0.5	.49	100	0.5	.13

Note: The following supplementary information was used:

from MIL-A-8866--1 sea landing/1 hr. at sea
10 land landings/7.5 hours on land

from records of HU-16 airplanes--264 hrs. at sea/1000 flight hrs.
736 hrs. on land/1000 flight hrs.

Therefore the landing spectra were corrected using the following information:
264 water landings/1000 hrs.
981 ground landings/1000 hrs.

TABLE 4 -- LOADING SPECTRA

Positive loading spectrum		Negative loading spectrum	
% limit load	f/1,000 hr.	% maximum load	f/1,000 hr.
50	43,400	10	450
55	9,220	20	420
65	1,191	30	250
75	325	40	90
85	104	50	27
95	30.80	60	9
105	10.15	70	2.75
115*	3.05	80	0.80
125*	2.02	90**	0.25
		100**	0.20

*Eliminated after block 30

**Eliminated after block 128

TABLE 5 -- POSITIVE LOADING TEST SPECTRUM

% limit load	f/20 hour block	Plus
50	868	--
55	184	1 in blocks 2, 4, 7, 9--repeat every 10 blocks
65	23	1 in blocks 1, 2, 3, 4, 6, 7, 8, 9--repeat every 10 blocks, plus 1 every 50th block
75	6	1 every other block (2, 4, 6-50)
85	2	1 in blocks 12, 25, 36, 50
95	-	1 in blocks 2, 3, 4, 7, 8, 9--repeat every 10 blocks, plus 1 every 50th block
105	-	1 every 5th block (5, 10, 15-50)
115*	-	1 in blocks 10, 25, 40
125*	-	1 in blocks 25, 50

Note: Spectrum repeats every 50 blocks.

*Eliminated after block 30

TABLE 6--NEGATIVE LOADING TEST SPECTRUM

% maximum load	F/20 hour block	Plus
10	9	-
20	8	1 in blocks 2, 4, 7, 9--repeat every 10 blocks
30	5	-
40	1	1 in blocks 1, 2, 3, 4, 6, 7, 8, 9--repeat every 10 blocks
50	-	1 every other block (2, 4, 6-50), plus 1 in blocks 1 and 25
60	-	1 every 5th block (5, 10, 15-50)
70	-	1 in blocks 10, 25, 40
80	-	1 in block 50
90**	-	1 in block 200--repeat every 200 blocks
100**	-	1 in block 250--repeat every 250 blocks

Note: Spectrum for 10% through 80% max-load repeats every 50 blocks.

**Eliminated after block 128

TABLE 7--WING LIMIT LOAD STRAINS

	GAEC-Flight Study (Ref.e)				GAEC-Static Test (Ref.f)				NADC--Ave. of Blocks 2,15,25, 50,200			
W.S.	MU	ML	FU	FL	MU	ML	FU	FL	MU	ML	FU	FL
5.5	-1,620	1,620	-1,440	1,520	--	--	--	--	-1,040	1,840	-1,750	1,930
37.5	-1,900	1,430	--	--	--	--	--	--	-2,430	1,840	--	--
43.5	--	--	-1,640	1,360	--	--	--	--	--	--	-1,720	1,790
56	--	--	--	--	-2,080	1,800	-1,870	1,730	--	--	--	--
71.5	--	--	--	--	--	--	--	--	--	2,040	--	--
79	--	--	--	--	--	--	--	--	--	--	-1,570	1,800
89	-2,200	1,700	-1,790	1,590	--	--	--	--	-2,330	2,020	-1,720	2,150
93	--	--	--	--	--	--	--	--	--	--	-1,530	2,150
100	--	--	--	--	--	--	--	--	--	1,960	--	--
115.5	--	--	--	--	--	--	--	--	--	--	--	2,050
135	--	--	--	--	--	1,930	--	--	--	--	--	--
149	--	--	--	--	--	--	--	--	--	--	--	--
151	--	--	--	--	--	--	--	--	--	--	--	--
152	--	--	--	--	--	--	--	--	--	--	--	2,090
154 11/32	--	--	--	--	--	--	--	--	--	--	--	--
156	--	--	--	--	--	--	--	--	--	--	--	--
158	--	--	--	--	--	--	--	--	--	--	--	--
160	--	--	--	--	--	--	--	--	--	--	--	--
164	--	--	--	--	--	--	--	--	--	--	--	--
168 3/4	--	--	--	--	--	--	--	--	--	--	--	--
171	--	--	--	--	--	--	--	--	--	--	--	--
175	--	--	--	--	--	--	--	--	--	--	--	--
180	--	--	--	--	--	--	--	--	-3,000	2,660	-2,400	1,990
185	--	--	--	--	--	--	--	--	--	--	--	--
190	--	--	--	--	-2,250	2,450	-2,210	2,150	--	--	--	--
193	-1,880	2,030	-1,650	1,700	--	--	--	--	-2,300	2,560	-1,710	2,120
230	--	--	--	--	--	--	--	--	-1,680	1,960	--	--

Notes:

1. MU -- Main beam, upper spar cap
   ML -- Main beam, lower spar cap
   FML - 2 5/16" forward of main beam, lower spar cap
   FU -- Front Beam, upper spar cap
   FL -- Front Beam, lower spar cap
2. Strains are given in micro-inches per inch
3. Gages are mounted as follows:

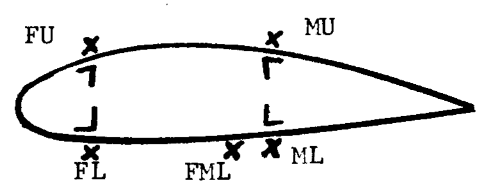


TABLE 7--WING LIMIT LOAD STRAINS (cont.)

	NADC-- After RH Wing Failure Before Fix, **Block 411**					**NADC**-- After RH Wing Failure After Fix, **Block 411**					**Block 435**				
W.S.	MU	ML	FML	FU	FL	MU	ML	FML	FU	FL	**MU**	**ML**	**FML**	**FU**	**FL**
5.5	- 520	940	--	- 730	1,050	-1,080	2,430	--	-1,830	1,870	-1,200	1,843	--	-1,925	1,898
37.5	-1,460	1,100	--	--	--	-2,530	1,840	--	--	--	-2,530	1,815	--	--	--
43.5	--	--	--	-1,100	1,130	--	--	--	-1,780	1,840	--	--	--	-1,850	1,815
56	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
71.5	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
79	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
89	-1,750	1,020	--	-1,380	1,780	-2,180	1,350	--	-1,780	2,180	-2,250	1,953	--	-1,825	2,250
93	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
100	--	300	--	--	--	--	320	--	--	--	--	303	--	--	--
115.5	--	--	--	--	1,870	--	--	--	--	2,090	--	--	--	--	2,035
135	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
149	- 630	1,000	2,250	-2,300	2,350	- 680	--	2,080	-2,150	2,500	- 650	--	2,000	-2,575	2,525
151	--	--	--	--	1,980	--	--	--	--	2,120	--	--	--	--	2,090
152 11/32	-3,150	580	2,950	-1,200	2,130	-3,350	1,050*	2,050	-1,400	2,250	-3,400	1,150*	2,125	-1,300	2,350
154	-3,430	2,500	2,750	-1,800	2,200	-3,580	880*	1,800	-1,950	2,430	-3,850	950*	1,800	-1,950	2,350
156	-2,480	1,500	1,880	-1,280	1,950	-2,230	--	1,150	-1,300	1,950	-2,200	--	1,125	-1,350	1,950
158	- 380	30	430	- 150	350	- 480	--	--	--	330	- 450	--	--	--	400
160	- 380	0	530	- 50	- 280	- 400	--	480	--	- 430	- 325	--	500	--	- 300
164	-2,200	1,130	1,000	- 800	1,280	-2,030	--	1,580	- 830	750	-2,100	--	1,150	- 850	750
168 3/4	- 350	- 100	550	350	100	- 440	--	800	--	100	- 500	--	750	475	200
171	-4,550	3,050	2,480	-1,400	1,350	-4,890	3,480	3,050	-1,430	1,600	-4,900	2,375	2,900	-1,575	1,550
175	-1,600	2,380	2,400	-1,380	1,700	-1,680	2,730	2,880	-1,480	1,830	-1,800	2,600	2,750	-1,575	1,800
180	-3,050	2,700	2,430	-2,450	1,980	-3,250	2,980	2,880	-2,630	2,080	-3,425	2,925	2,775	-2,400	2,125
185	-2,050	2,630	2,530	- 800	2,050	-2,250	2,850	2,880	-1,100	2,000	-2,250	3,775	2,800	- 750	1,975
190	--	--	--	--	--	--	--	--	--	--	--	--	--	--	--
193	-2,230	2,550	2,450	-1,950	2,150	-2,380	2,800	2,700	-2,030	2,200	-2,450	3,650	2,700	-2,050	--
230	--	-1,460	--	--	--	--	-1,600	--	--	--	--	-1,500	--	--	2,250

4. For "ASD -- after RH wing failure, before fix," only the loading, and therefore the strain, from W.S. 120 outboard simulated design condition.

5. For "ASD -- After RH wing failure, after fix," the loading on the entire wing simulated design condition.

6. The values marked with an asterisk (*) are gages mounted on the steel strap.

TABLE 7--WING LIMIT LOAD STRAINS (Cont)

	NADC--After RH Wing Failure Block 610				
W.S.	MU	ML	FML	FU	FL
5.5	-1,100	1,760	--	-1,980	2,035
37.5	-2,558	1,760	--	--	--
43.5	--	--	--	-1,700	1,760
56	--	--	--	--	--
71.5	--	--	--	--	--
79	--	--	--	--	--
89	-2,250	1,100	--	-1,750	2,150
93	--	--	--	--	--
100	--	165	--	--	--
115.5	--	--	--	--	2,090
135	--	--	--	--	--
149	- 700	--	2,350	-2,250	2,650
151	--	--	--	--	2,145
152	-3,350	1,750*	2,500	-1,450	2,250
154 11/32	-3,700	1,300*	2,200	-1,950	2,200
156	-2,150	--	1,760	-1,250	2,100
158	- 300	--	--	--	--
160	- 300	--	-300	--	- 300
164	-2,150	--	1,450	- 800	700
168 3/4	- 250	--	--	--	100
171	-4,650	--	1,700	-1,760	1,500
175	-1,850	1,300	1,700	-1,760	1,700
180	-3,250	2,500	2,350	-2,650	2,100
185	-2,100	2,550	2,700	-1,150	2,100
190	--	--	--	--	--
193	-2,250	950	2,600	-1,950	2,150
230	--	-1,430	--	--	--

TABLE 8--WING DEFLECTIONS

CALCULATED

W. S.	1g	95	105	-80	-70	-60	-50
91	.2	.5	.5	- .2	- .18	- .15	- .13
208	1.4	3.7	4.0	- 1.6	- 1.4	- 1.2	- 1.0
334	4.1	10.8	11.9	- 4.8	- 4.2	- 3.6	- 3.0
462	7.6	19.9	22.0	- 8.9	- 7.8	- 6.6	- 5.5
571.5	11.0	28.9	31.9	-12.8	-11.2	- 9.6	- 8.0
TIP	11.3	29.6	32.8	-13.2	-11.6	- 9.9	- 8.3

PRE-RH FAILURE

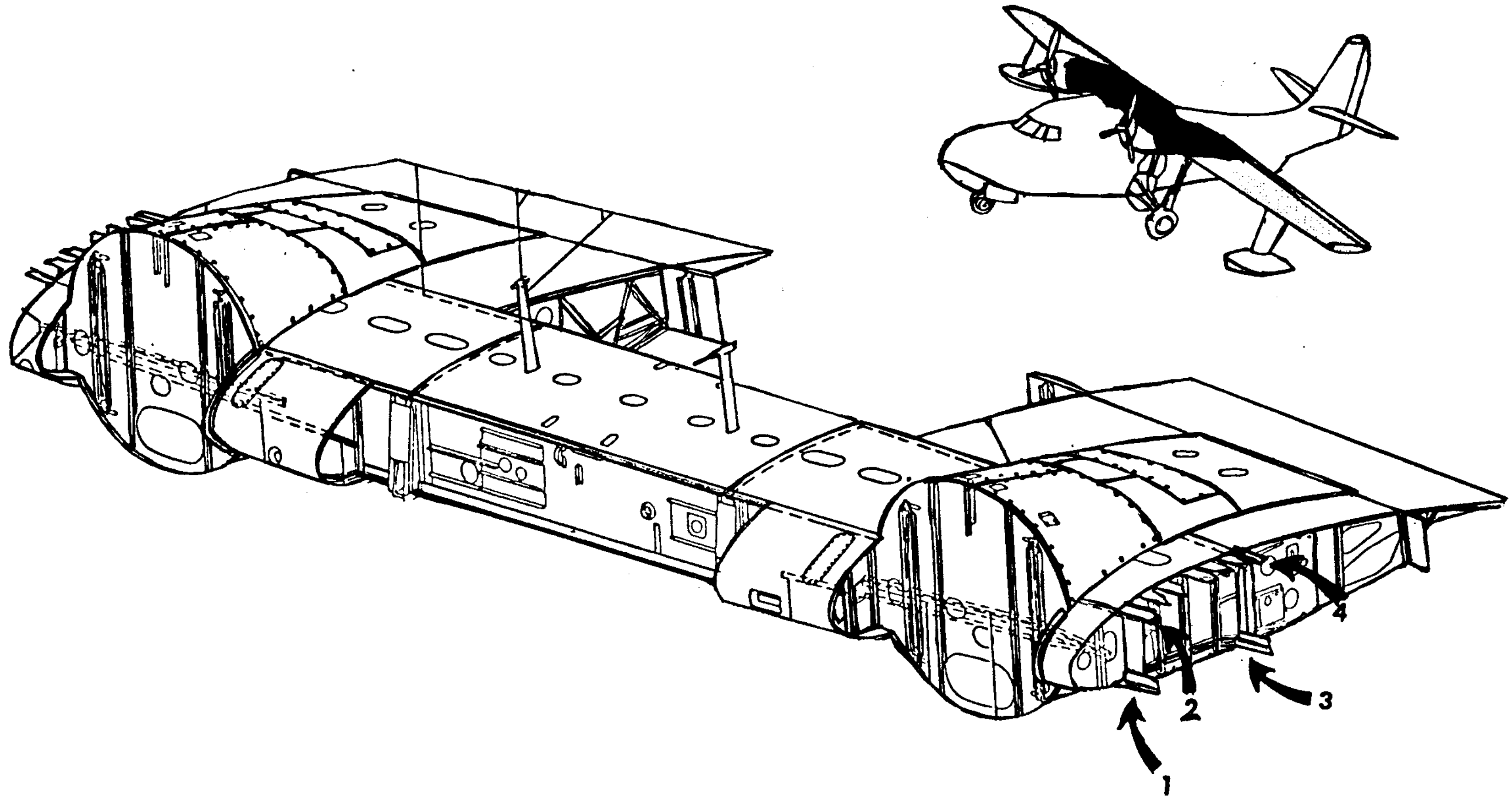
W. S.	1g	95	105			-60	-50
91	.3	.7	.8			- .3	- .3
208	1.5	4.2	4.6			- 1.4	- 1.2
334	4.0	11.0	12.1			- 3.8	- 3.1
462	7.5	20.4	22.6			- 6.8	- 5.5
571.5	10.8	29.2	32.5			- 9.8	- 8.0

POST - RH FAILURE

W. S.	1g	95	105			-60	-50
91	.2	.7				- .3	- .3
208	1.5	4.1				- 2.0	- 1.7
334	4.0	11.0				- 4.3	- 3.6
462	7.4	20.7				- 7.9	- 6.5
571.5	10.6	29.7				-11.1	- 9.2

Notes:

1. Left Hand Wing deflections listed
2. Wing deflections are in inches, measured from the chord plane of an unloaded wing

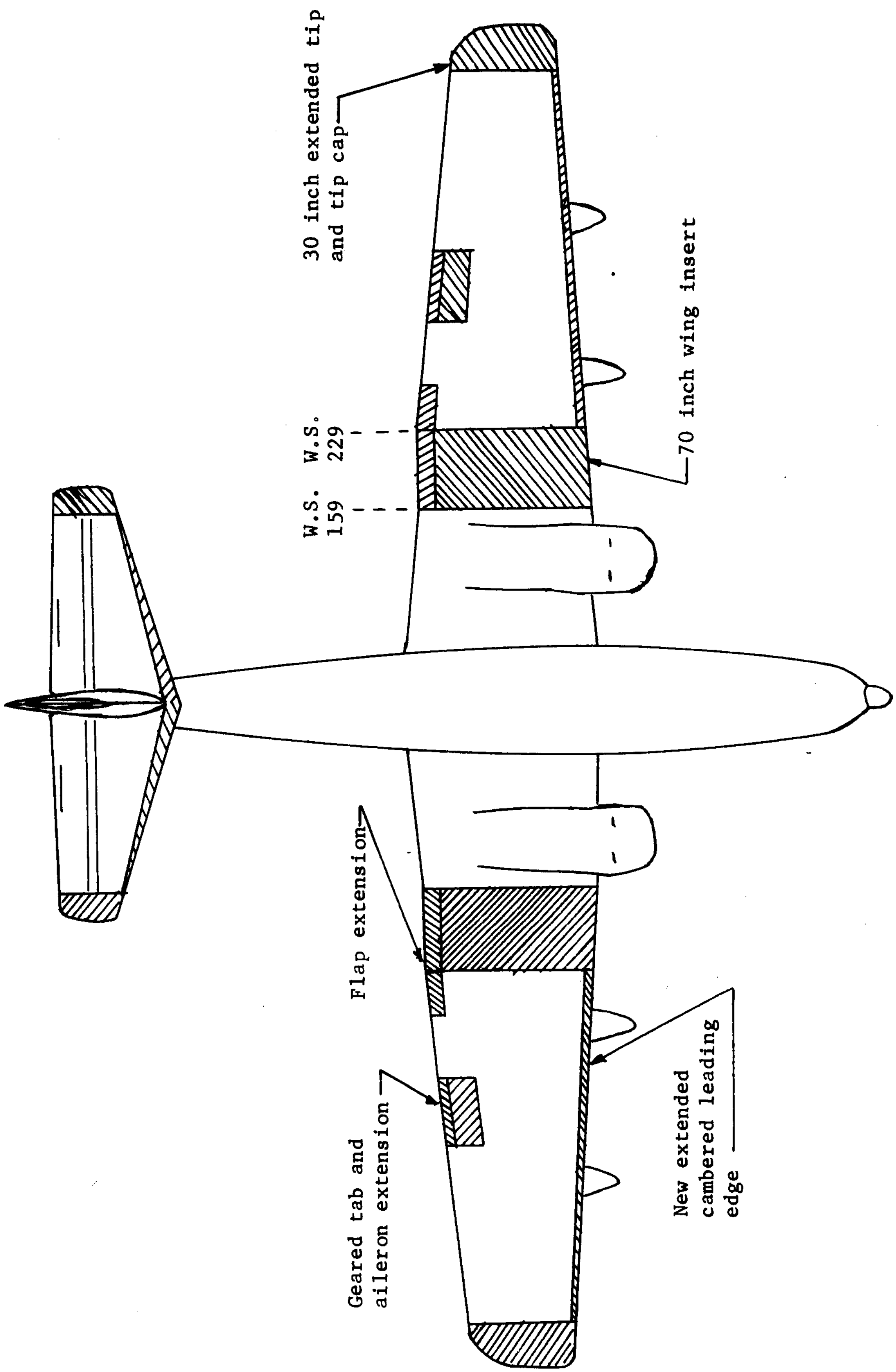


Spar Caps

1 - Lower forward
2 - Upper forward
3 - Lower Main
4 - Upper Main

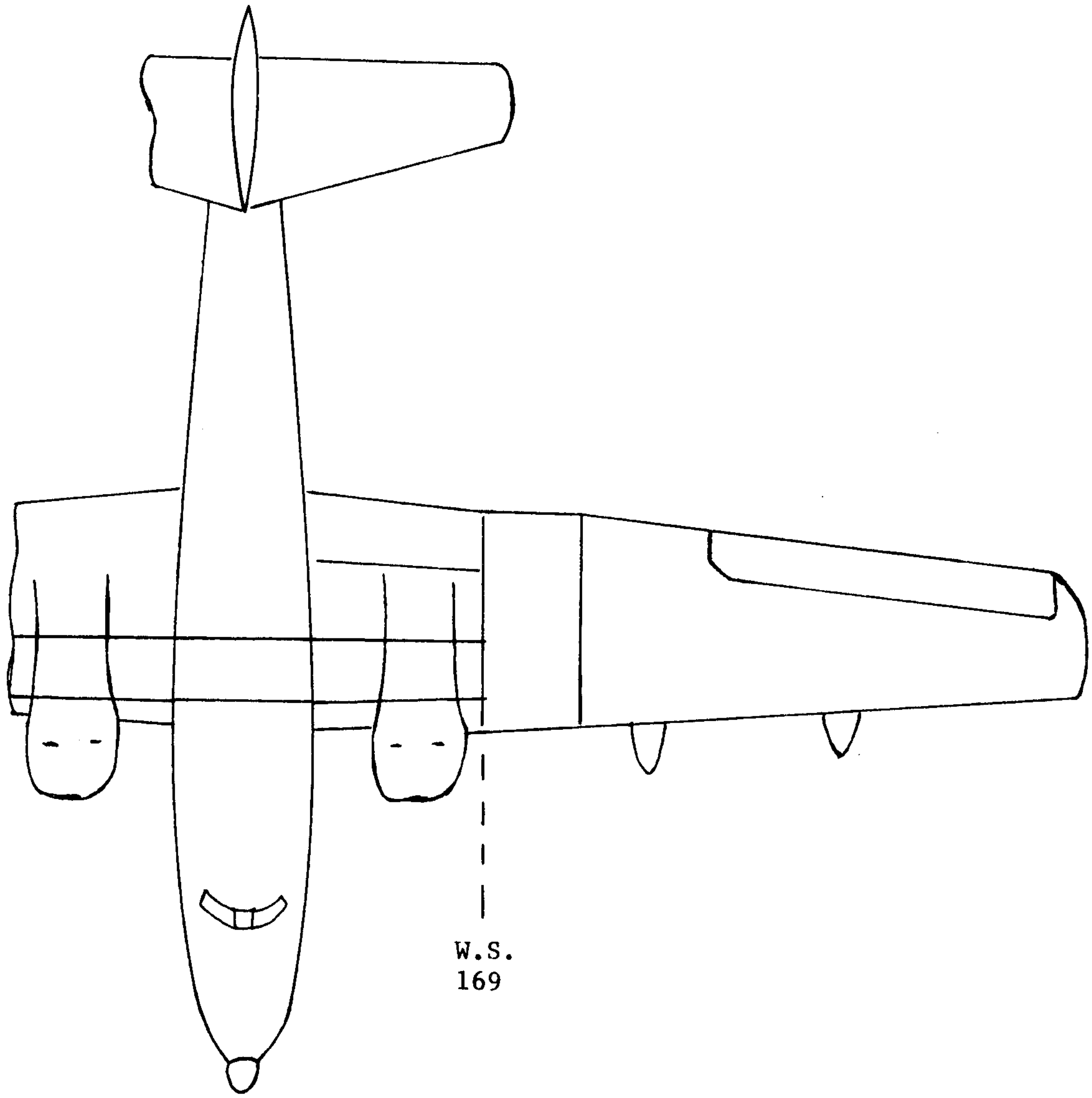
WING CENTER SECTION

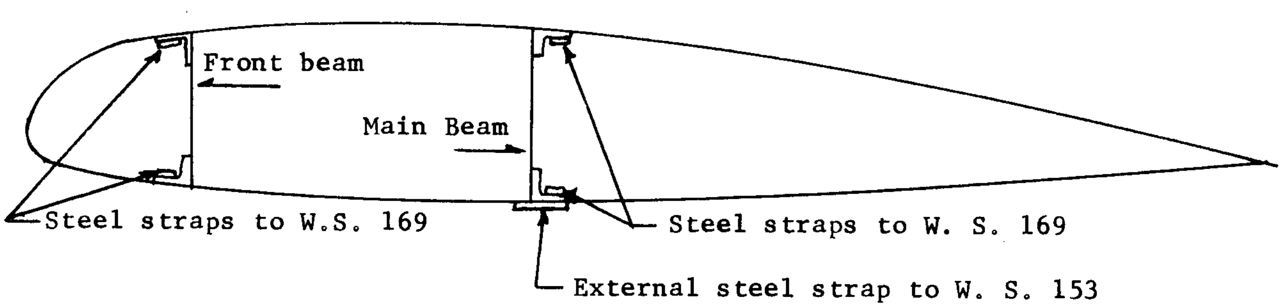
Figure 1



WING EXTENSION

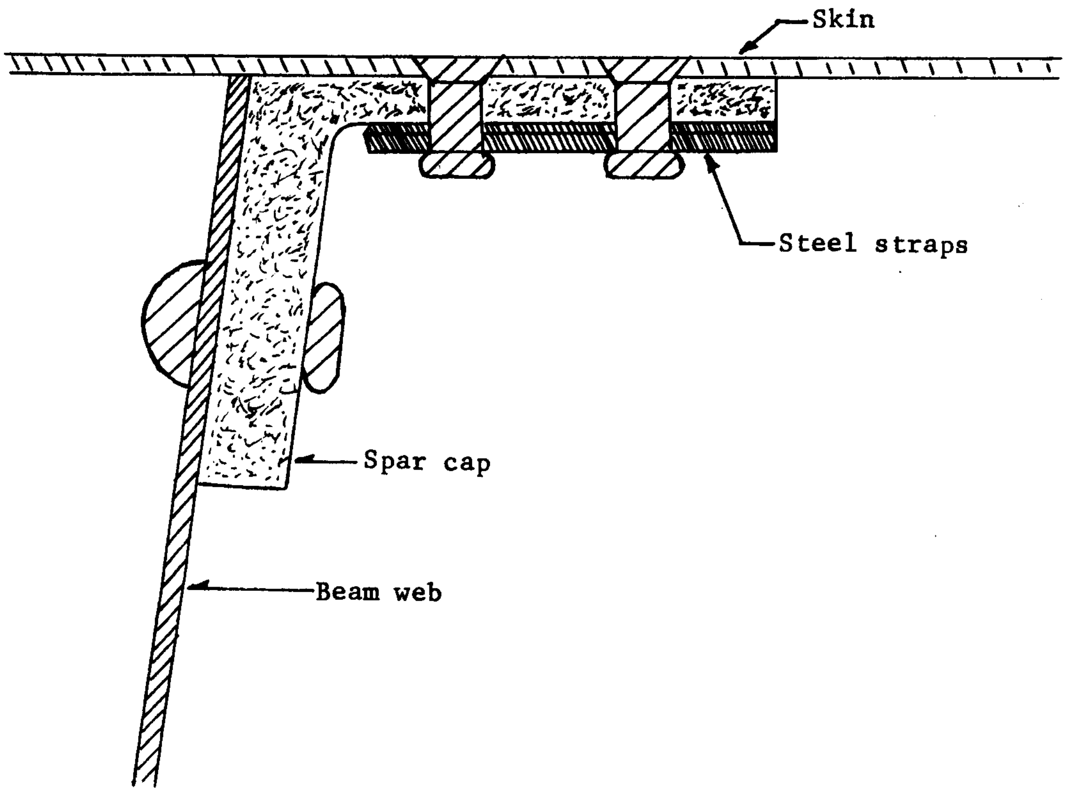
Figure 2





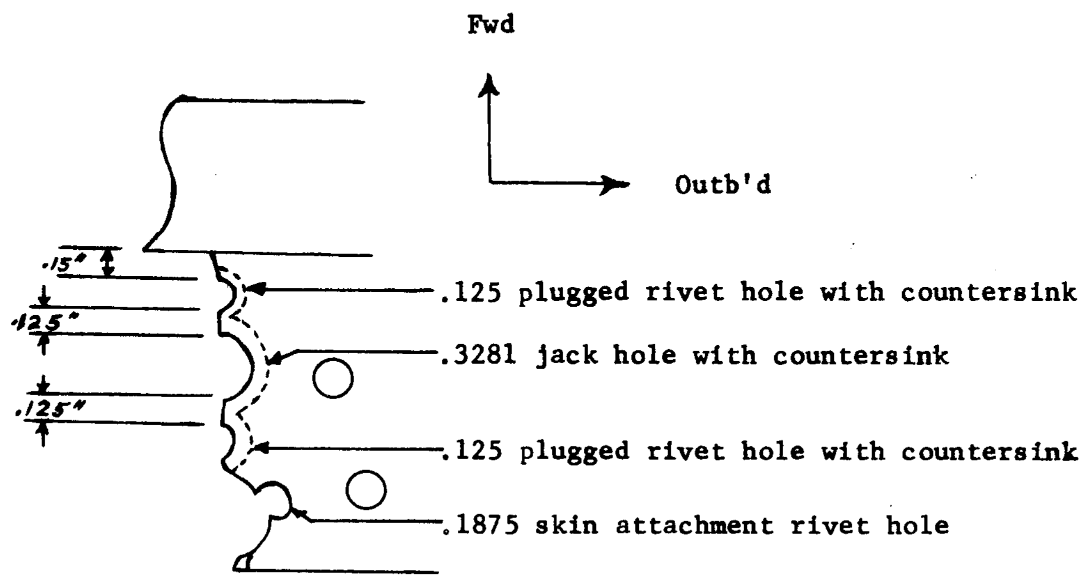
WING CENTER SECTION STEEL STRAPS

Figure 3



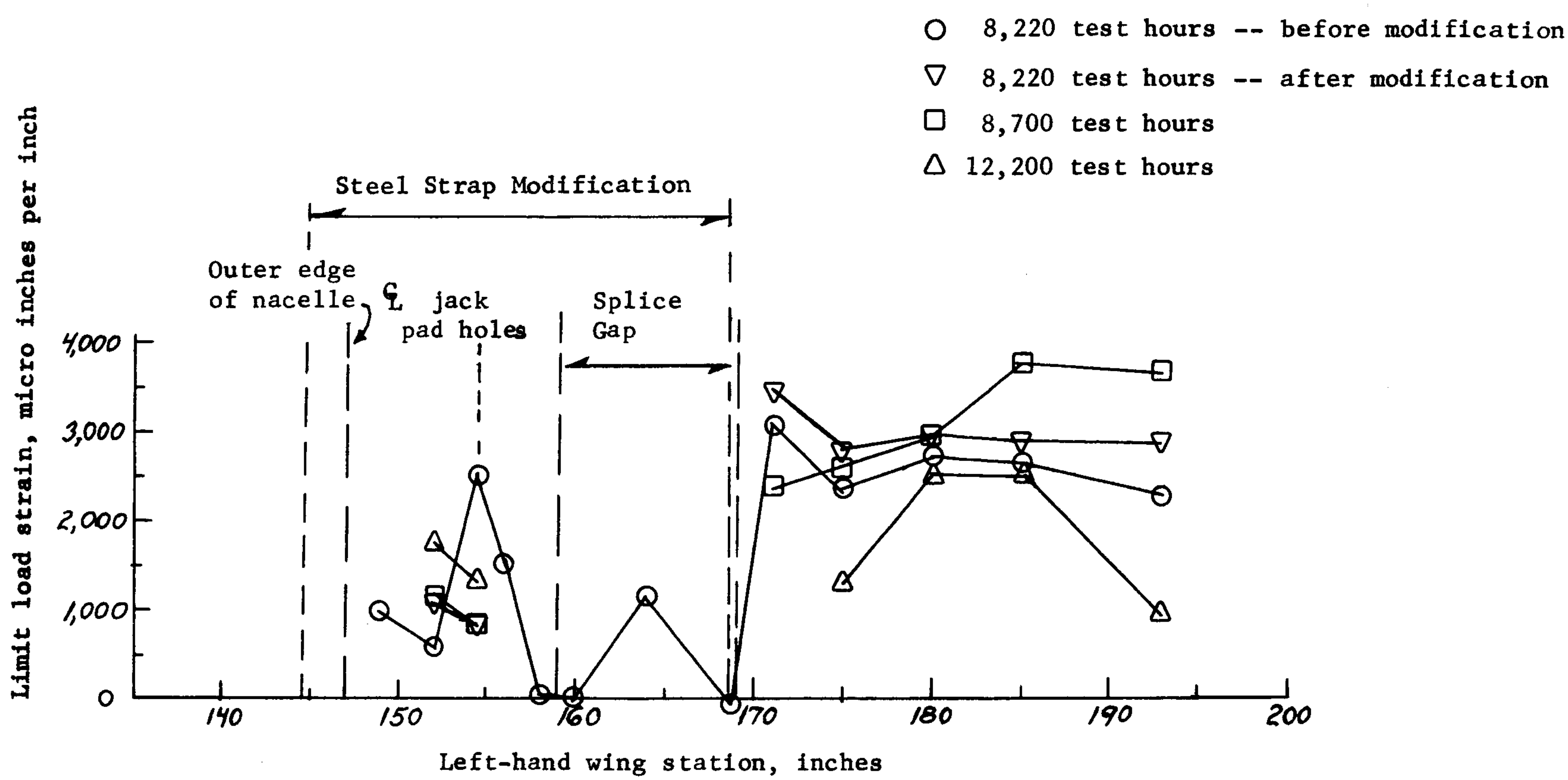
SECTION THRU SPAR CAP

Figure 4



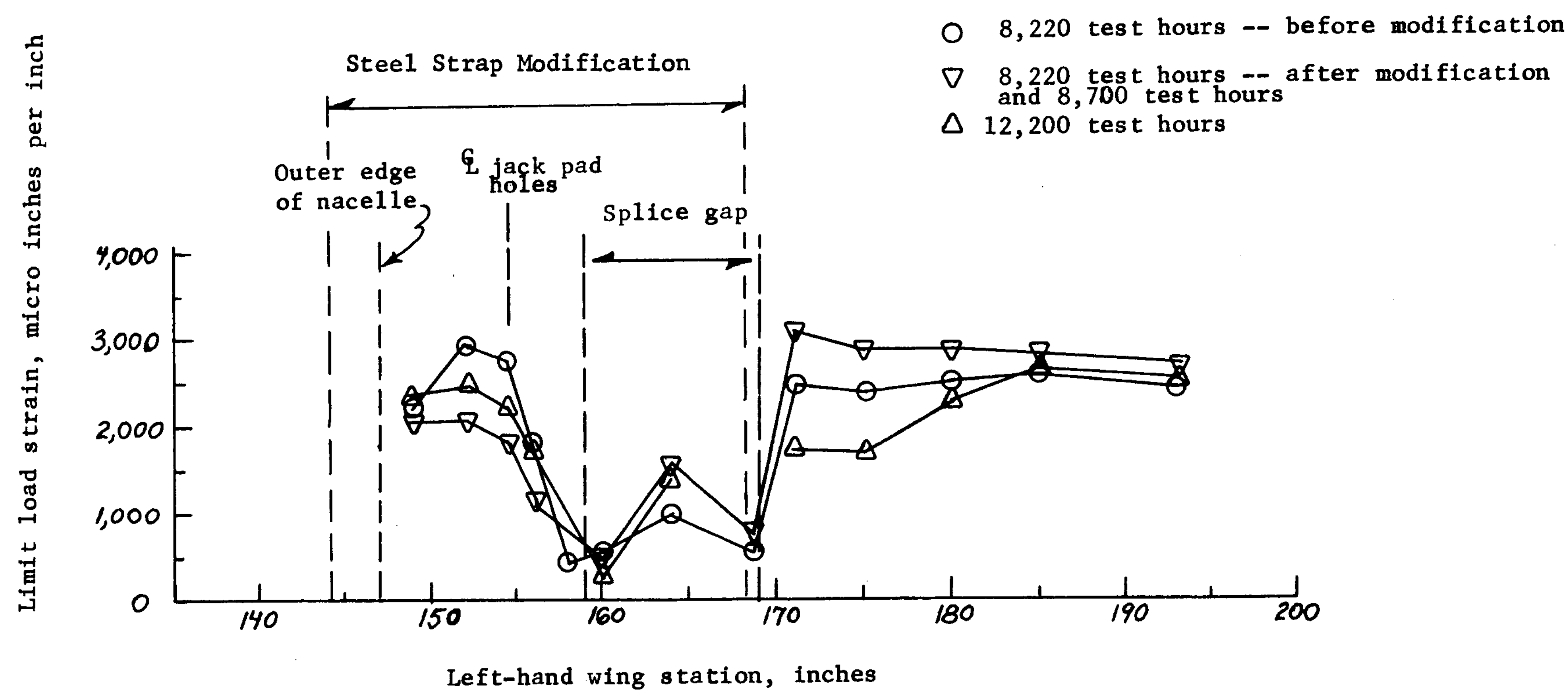
RIGHT-HAND WING, MAIN BEAM LOWER SPAR CAP FAILURE

Figure 5



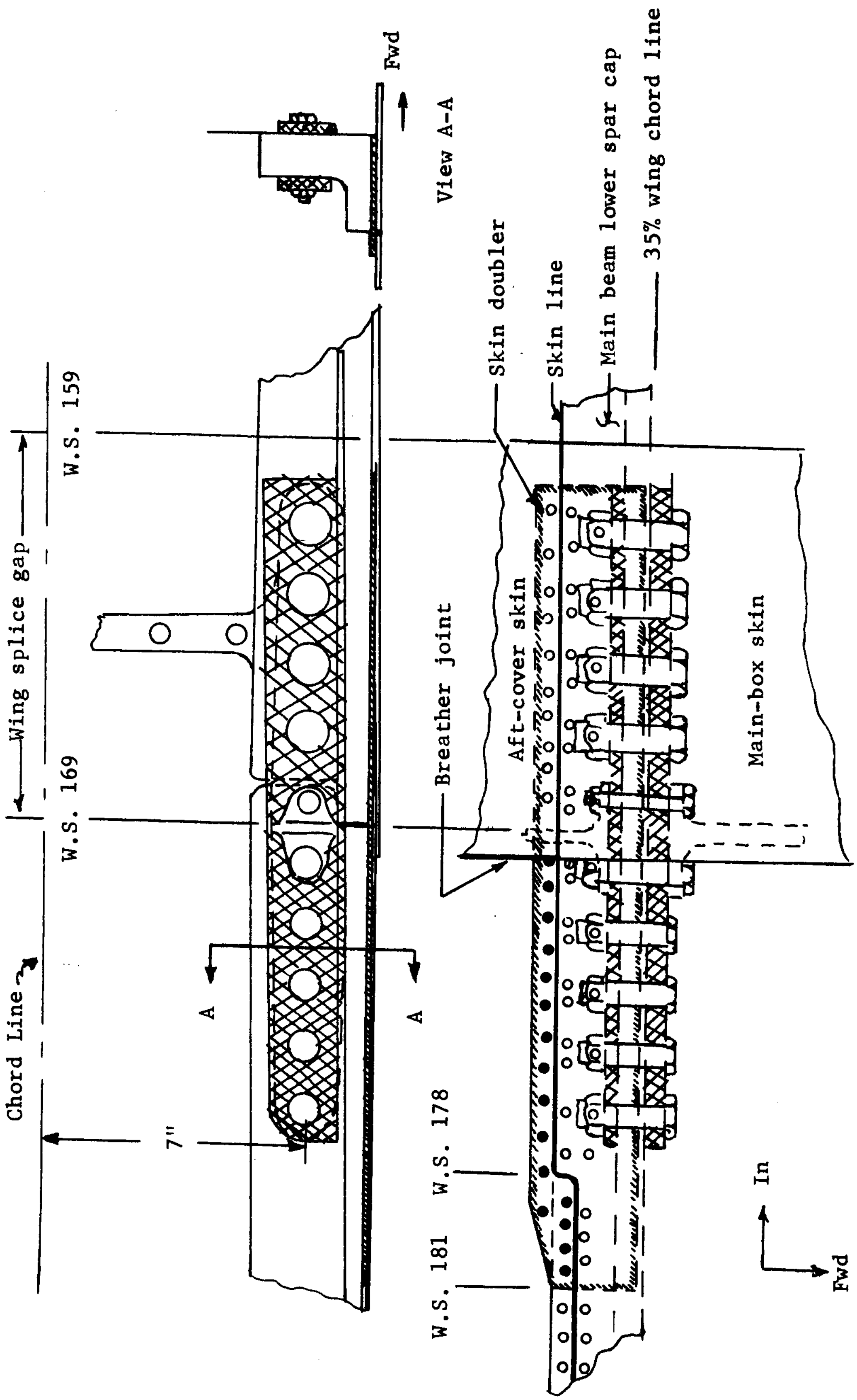
STRAIN DISTRIBUTION, MAIN BEAM LOWER SPAR CAP

Figure 6



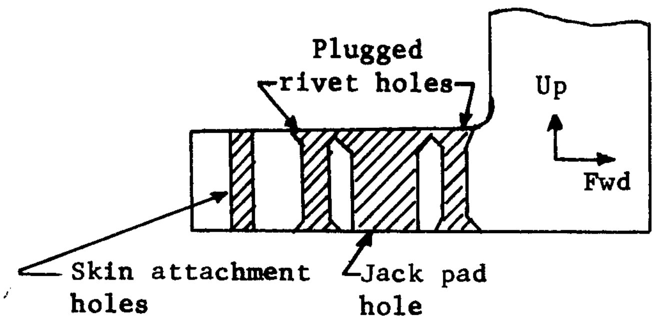
STRAIN DISTRIBUTION, FORWARD OF MAIN BEAM LOWER SPAR CAP

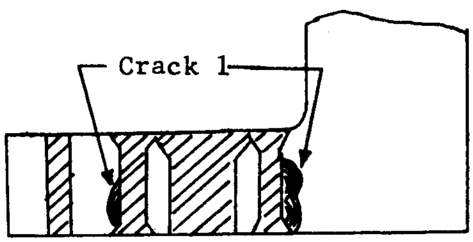
Figure 7



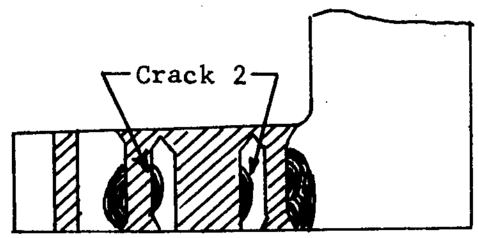
● Failed rivets

BREATHER JOINT RIVET FAILURES
Figure 8

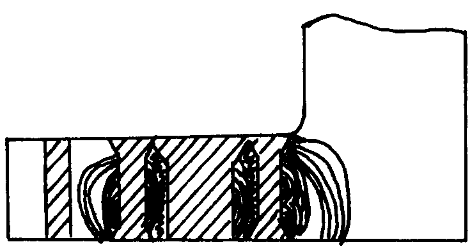




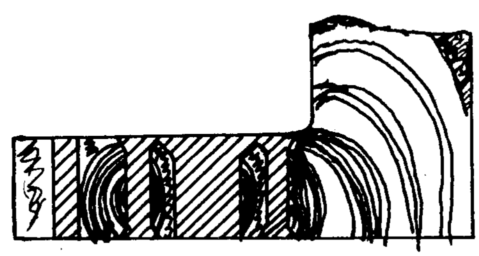
(a) Crack 1 initiates at plugged holes



(b) Crack 1 propagates, crack 2 initiates in material between jack and rivet holes

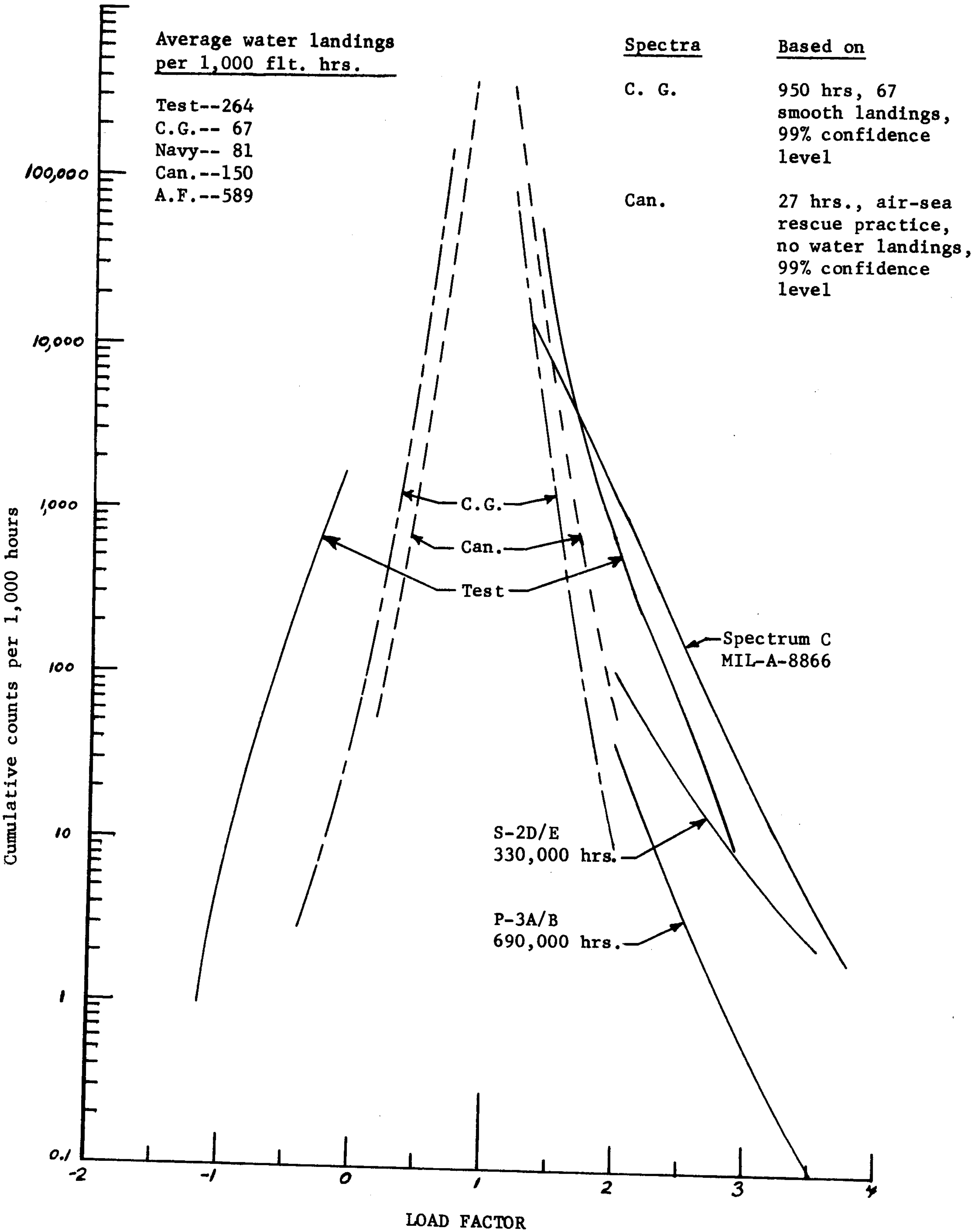


(c) Material between holes fails, crack 1 jumps and continues propagation



(d) Crack 1 propagation to final failure

THEORETICAL CRACK PROPAGATION
Figure 9



HU-16 SPECTRA
Figure 10

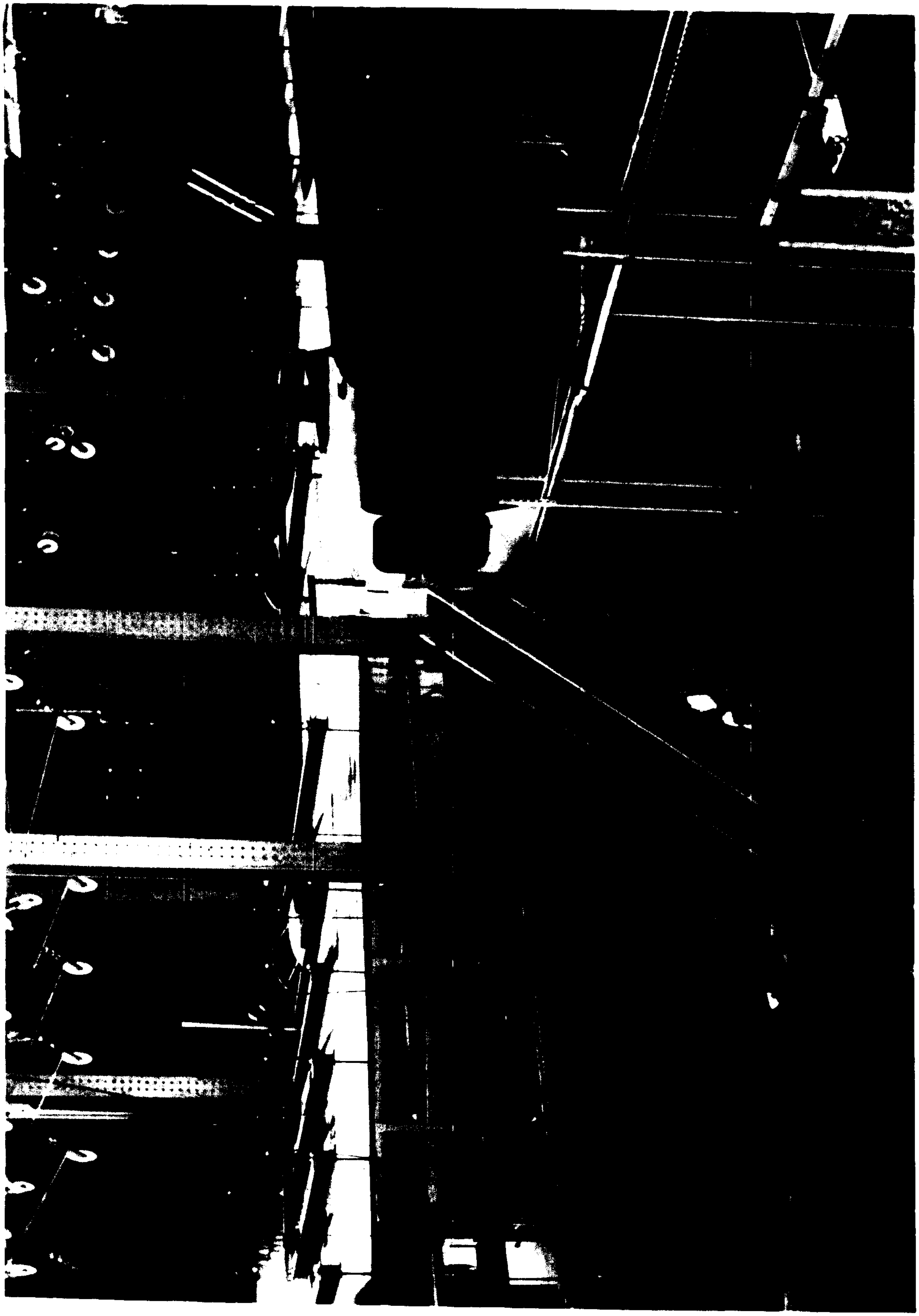


PHOTO 1. TEST SETUP, VIEW LOOKING FORWARD

PHOTO NO: CAN-387439(L)-5-68

PHOTO 2. TEST SETUP, VIEW LOOKING AFT

PHOTO NO: CAN-388392(L)-7-68

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PHOTO 3. LOAD CONTROL EQUIPMENT

PHOTO NO: CAN-388395(L)-7-68

PHOTO 4. GAC STEEL STRAP ON LEFT-HAND WING

PHOTO NO: CAN-393999(L)-6-69

PHOTO 5. AREA OF FAILED RIVETS, RIGHT-HAND WING

PHOTO NO: CAN-387556(L)-5-68

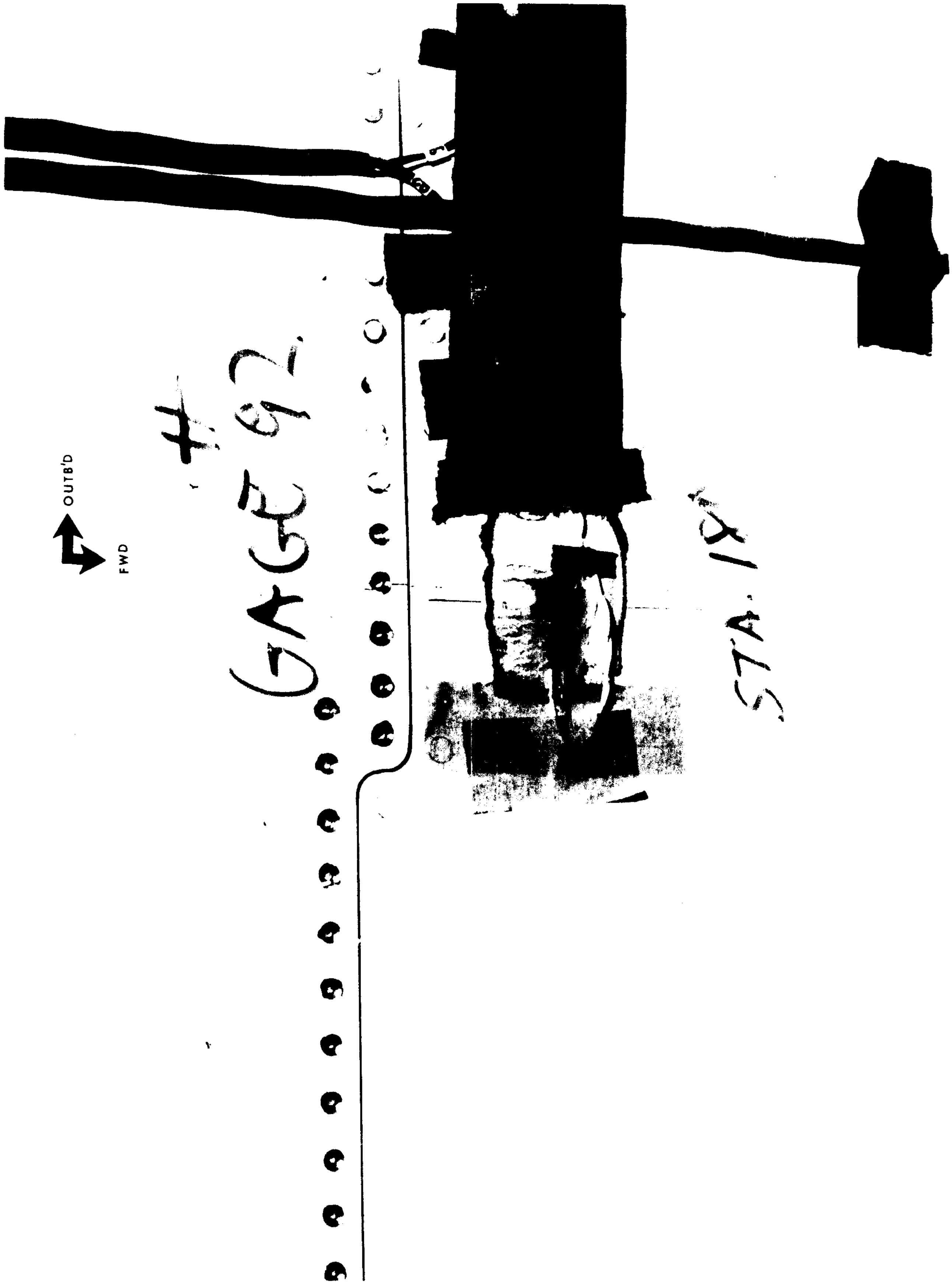


PHOTO 6. CLOSE-UP VIEW OF FAILED RIVETS, RIGHT-HAND WING

PHOTO NO: CAN-387419(L)-5-68

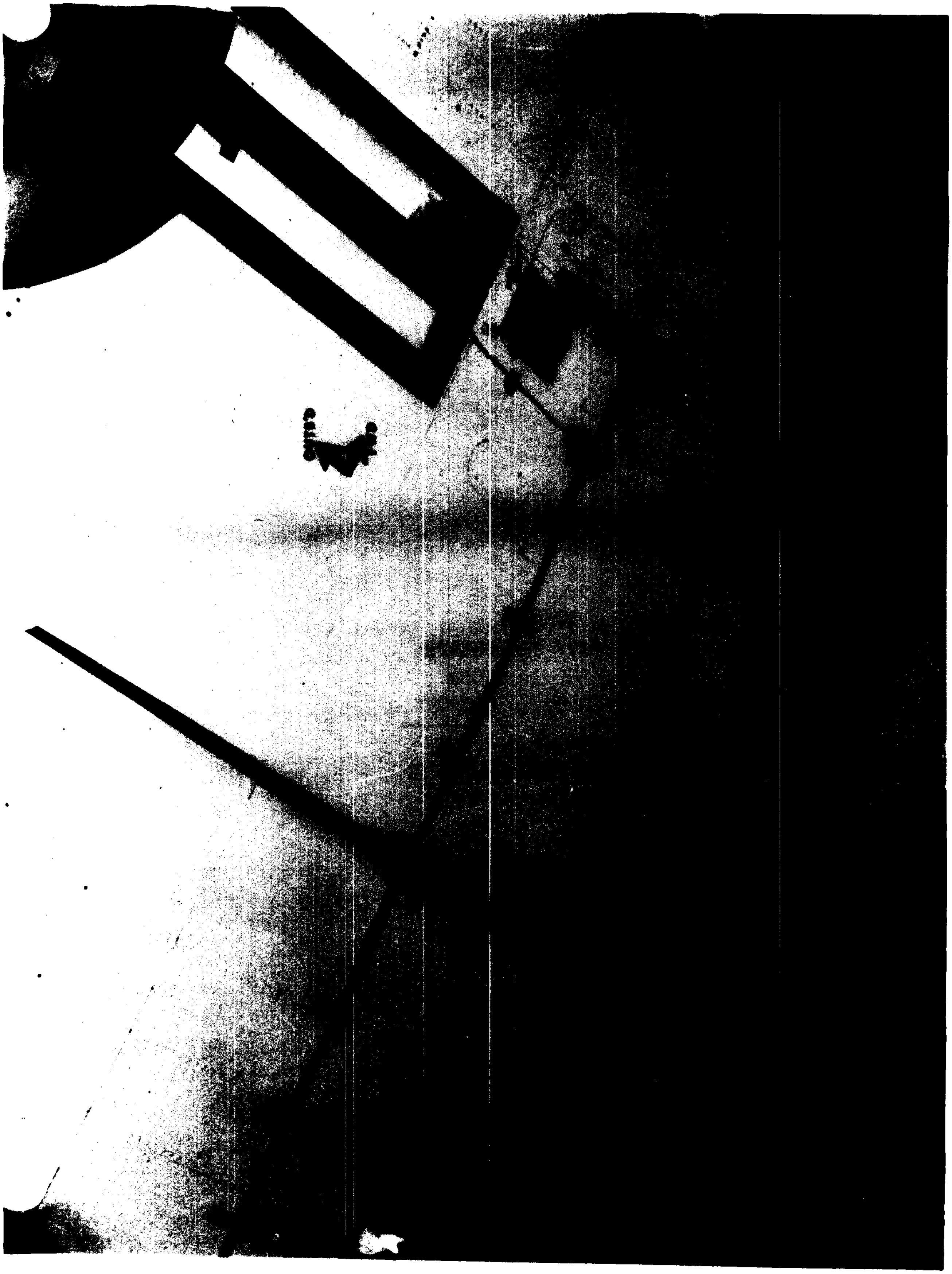


PHOTO 7. RIGHT-HAND WING FAILURE

PHOTO NO: CAN-390256(L)-11-68

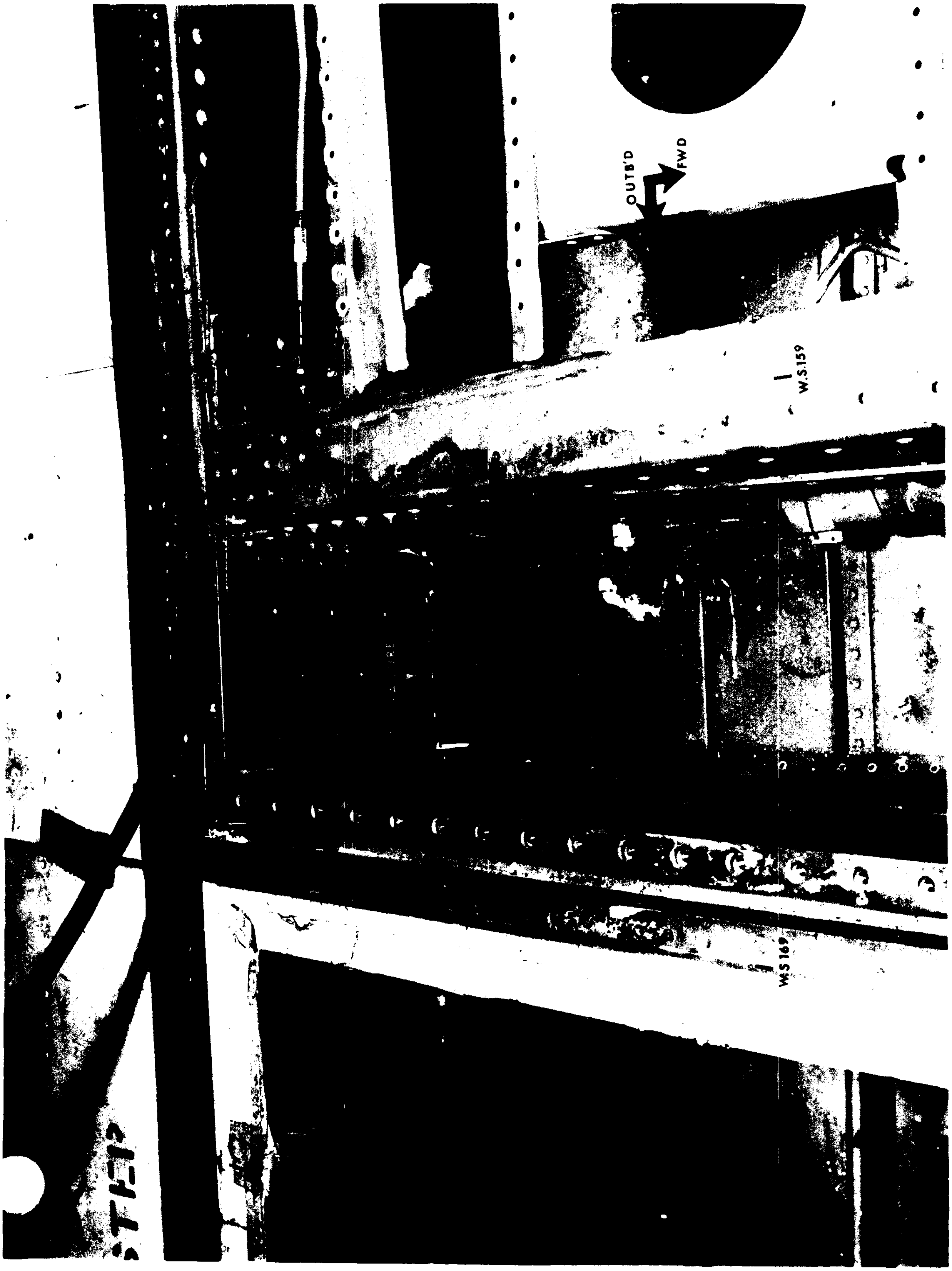


PHOTO 8. TAILING-EDGE BEAM FAILURE

PHOTO NO: CAN-390254(L)-11-68

PHOTO 9. FRONT BEAM FAILURE

PHOTO NO: CAN-389957(L)-10-68

PHOTO 10. FAILURE OF RIB 159 AND FRONT-BEAM

PHOTO NO: CAN-390258(L)-11-68

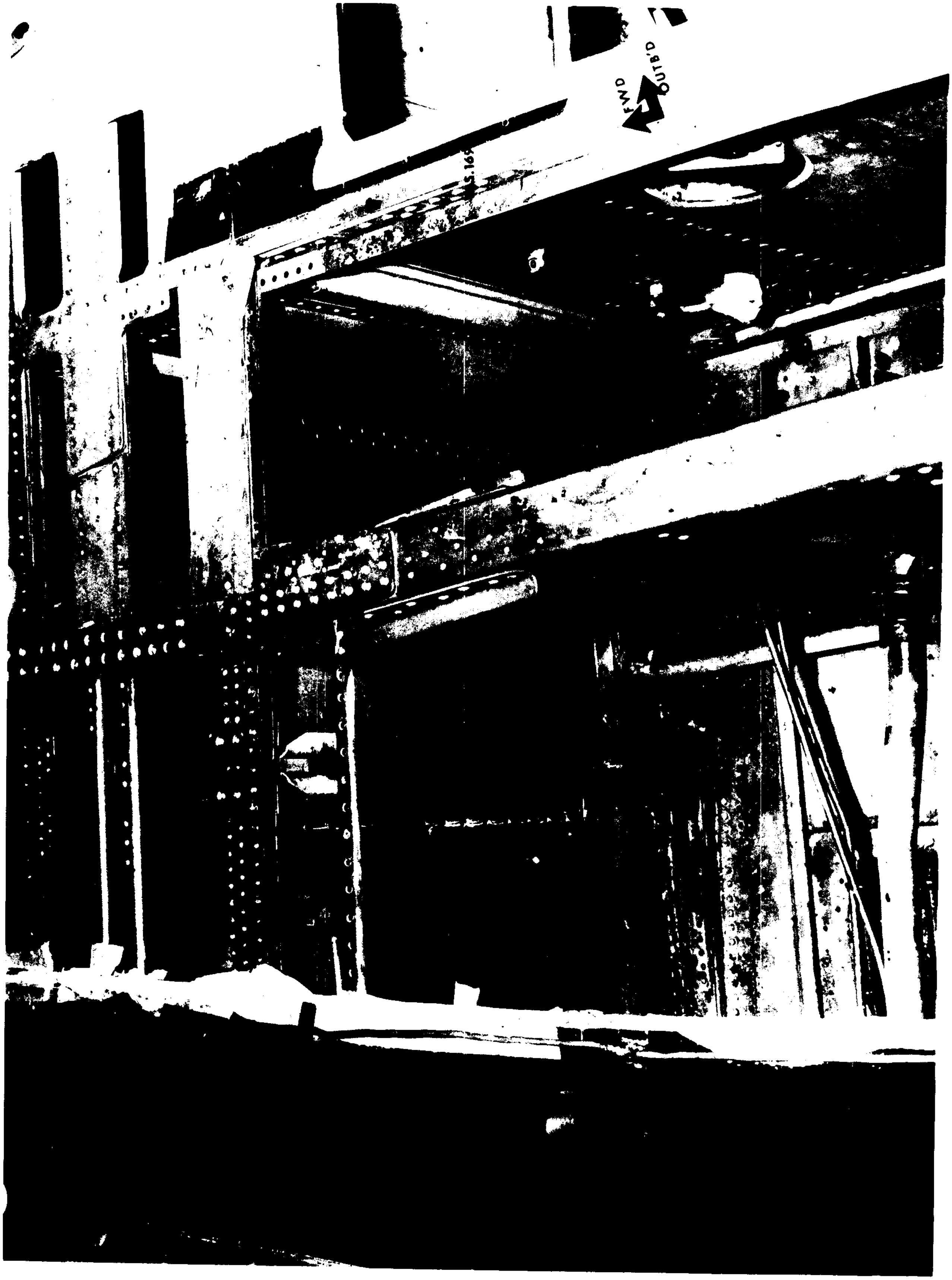


PHOTO 11. MAIN BEAM FAILURE

PHOTO NO: CAN-390262(L)-11-68

PHOTO 12. CLOSE-UP VIEW OF MAIN BEAM FAILURE

PHOTO NO: CAN-390260(L)-11-68

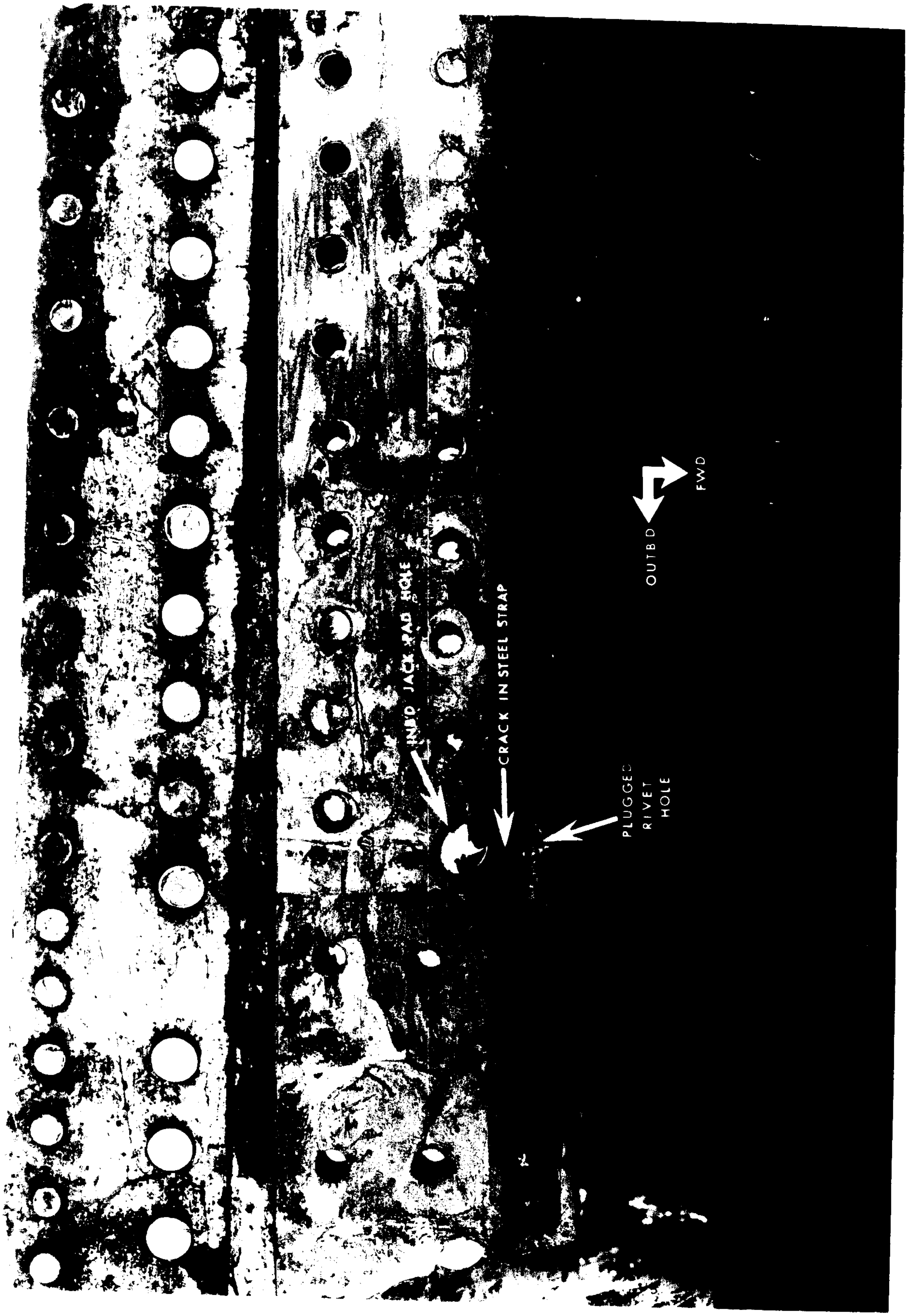


PHOTO 13. MAIN BEAM LOWER SPAR CAP WITH STEEL STRAP, VIEW LOOKING DOWN

PHOTO NO: CAN-390345(L)-11-68

PHOTO 14. MAIN BEAM LOWER SPAR CAP, VIEW LOOKING DOWN

PHOTO NO: CAN-390378(L)-11-68

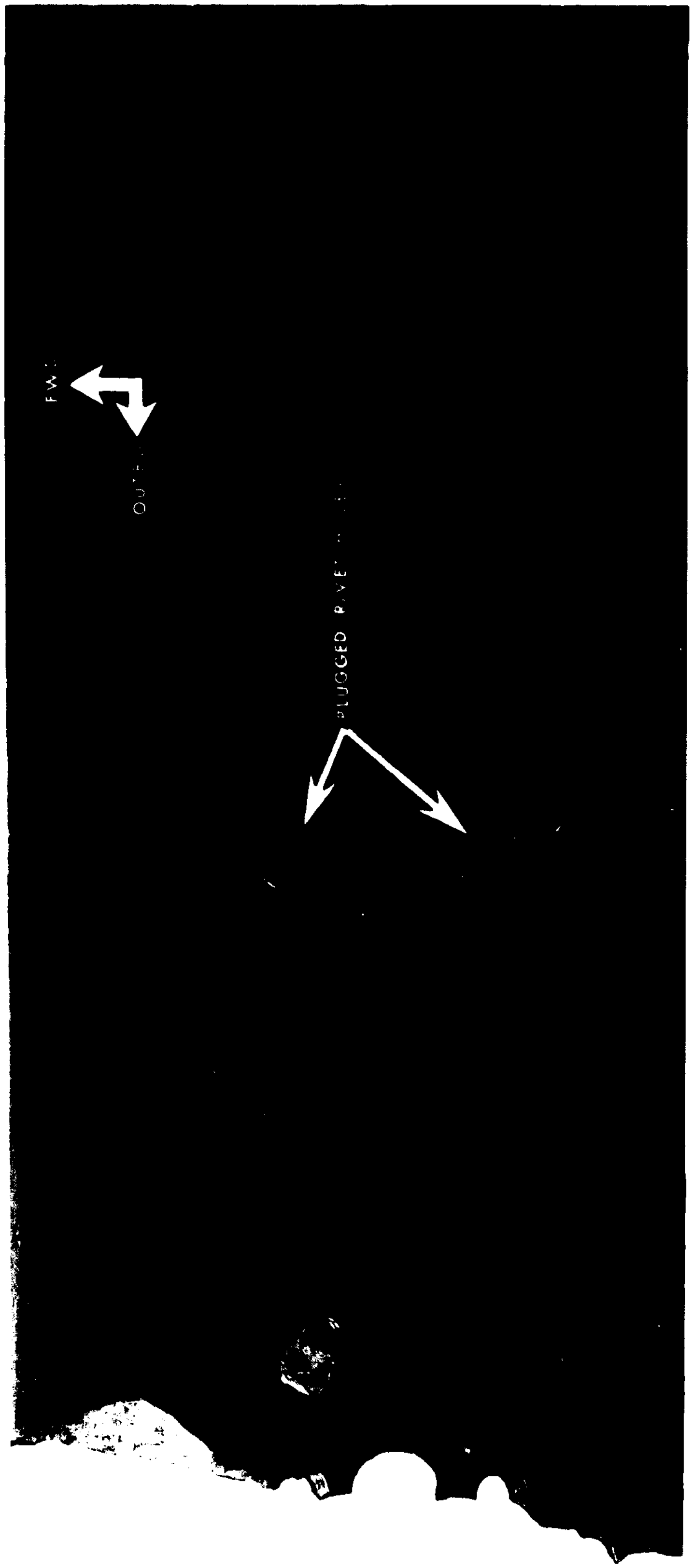


PHOTO 15. MAIN BEAM LOWER SPAR CAP, VIEW LOOKING UP

PHOTO NO: CAN-390379(L)-11-68

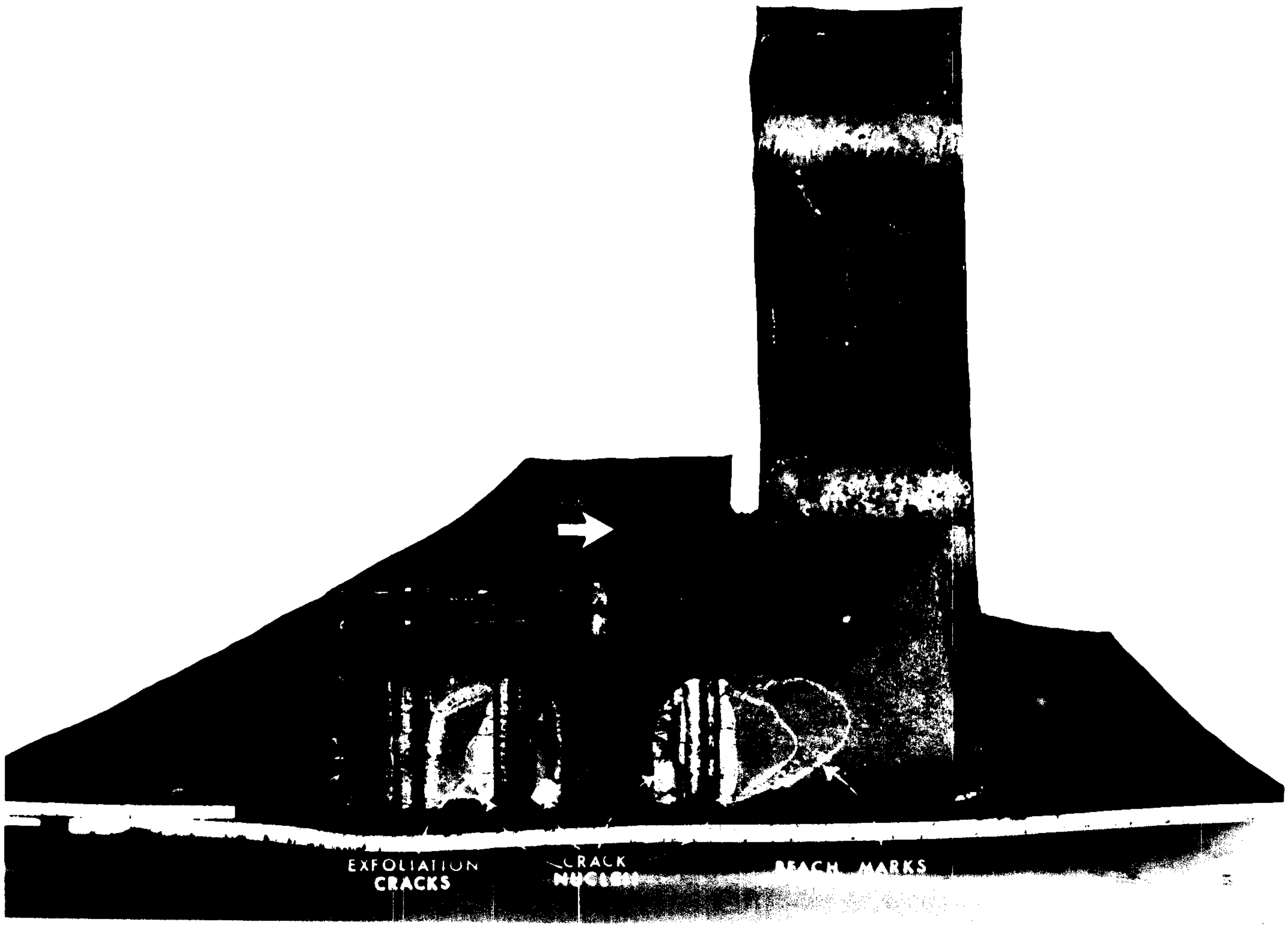


PHOTO 16. RIGHT-HAND WING FAILURE SURFACE, LOOKING INBOARD

PHOTO NO: CAN-390346(L)-11-68

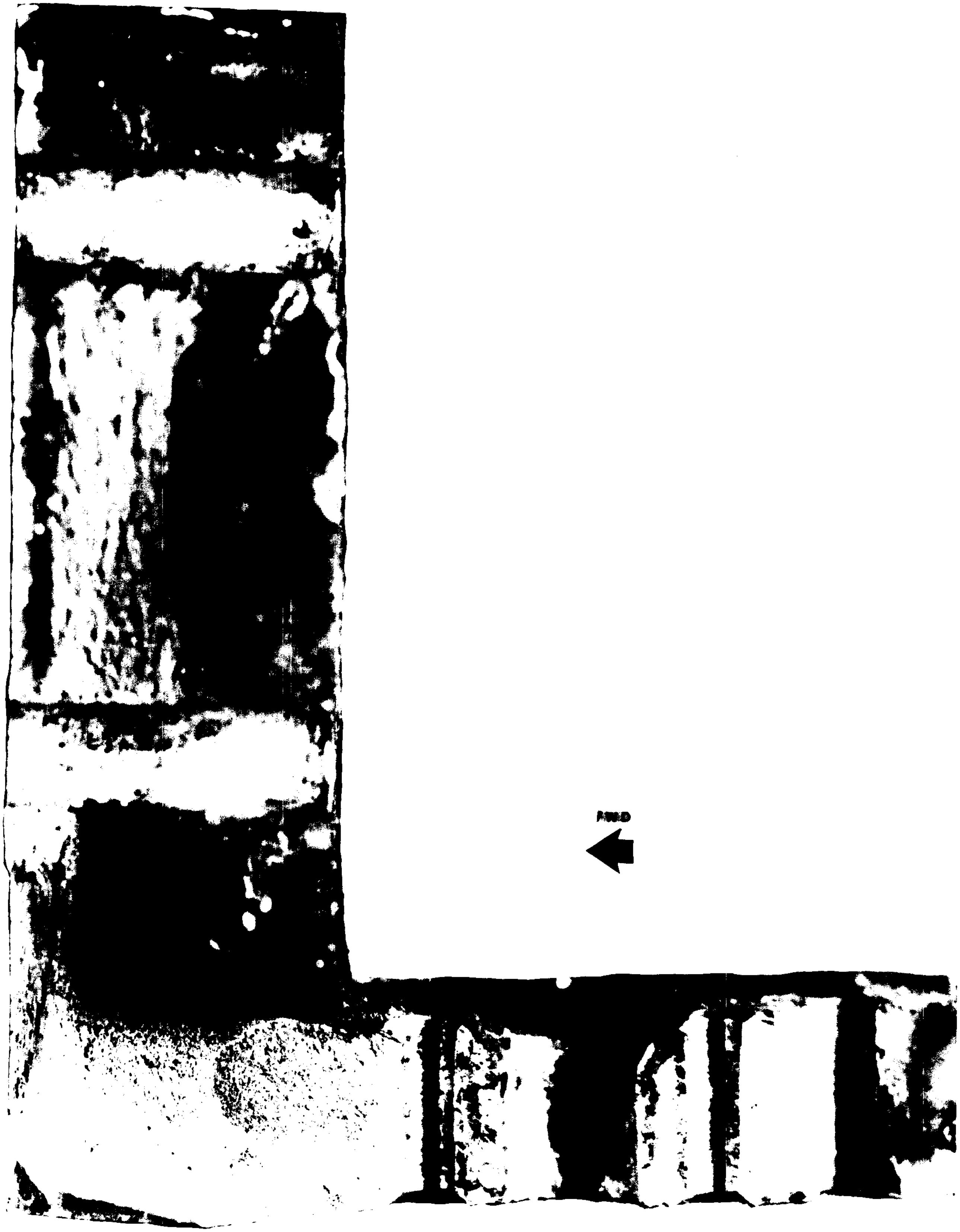
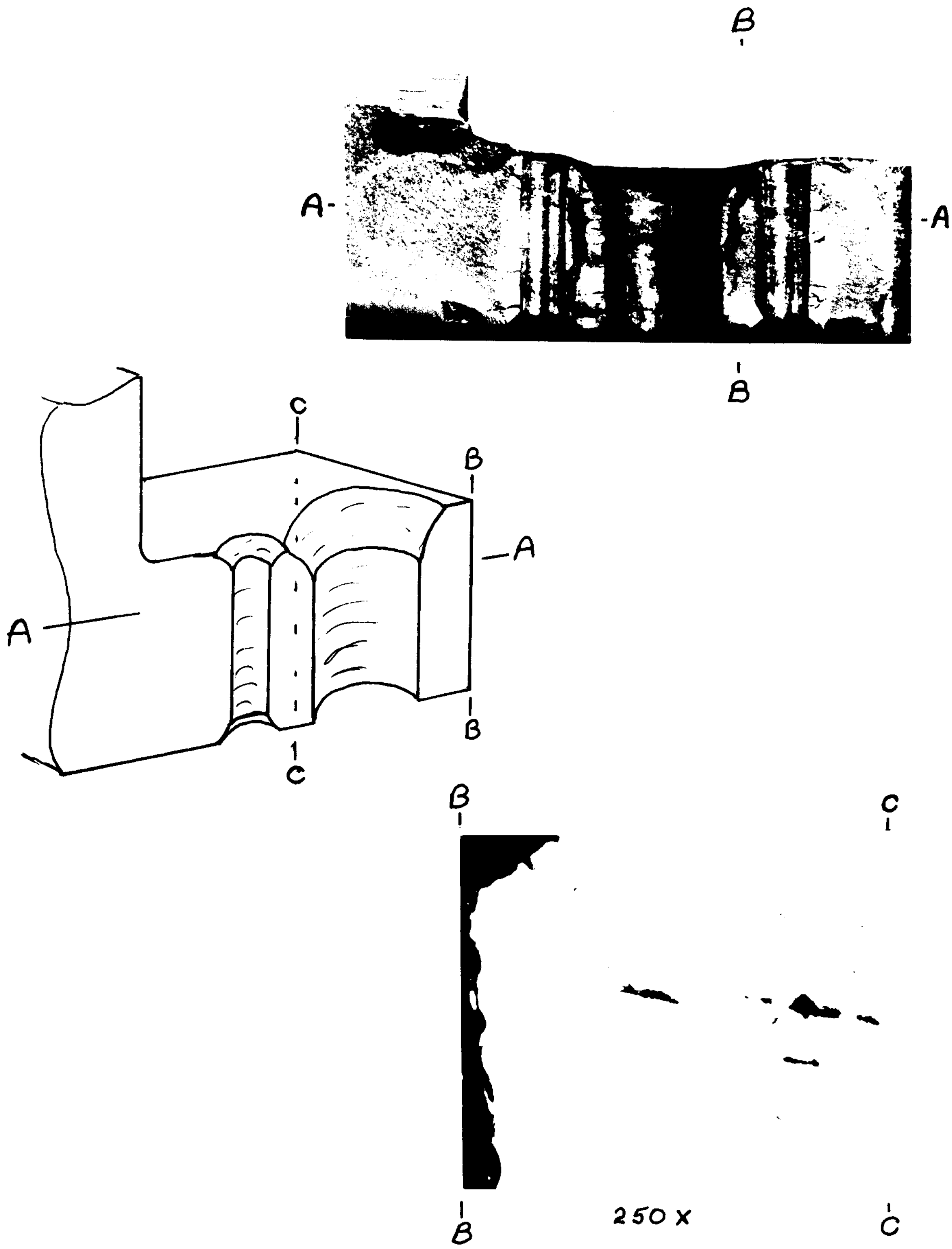


PHOTO 17. RIGHT-HAND WING FAILURE SURFACE, LOOKING OUTBOARD

PHOTO NO: CAN-390343(L)-11-68

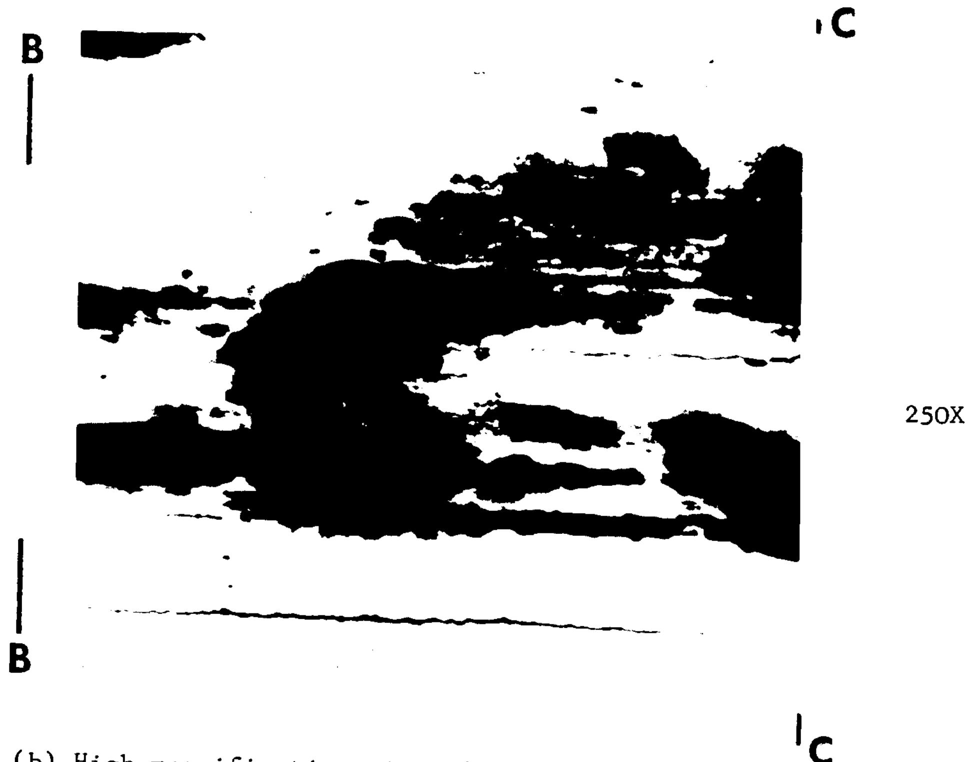


THE PHOTOMICROGRAPH REVEALS THE PRESENCE OF SECONDARY FATIGUE CRACKS THAT HAVE INITIATED AT SITES OF INTERGRANULAR CORROSION

PHOTO 18. COMPOSITE OF SPAR CAP SHOWING FRACTURE SURFACE AND METALOGRAPHIC SECTION

PHOTO NO: CAN-400705(L)-7-70

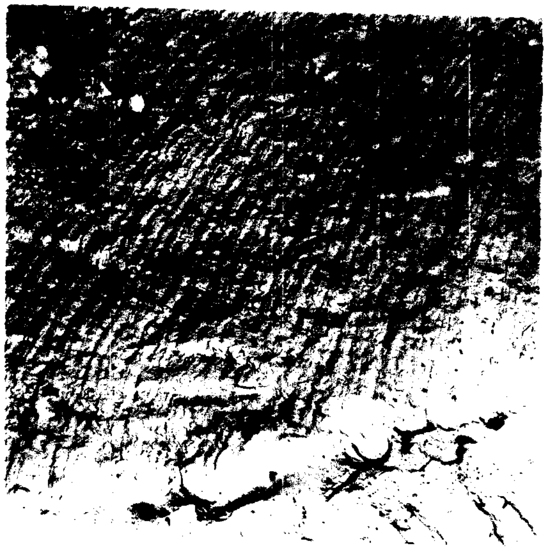
(a) Area showing widespread corrosion that initiated fatigue failure.



(b) High magnification view of small area from (a) showing extensive nature of corrosion and intergranular corrosion cracking.

PHOTO 19. PHOTOMICROGRAPHS SHOWING SEVERE CORROSION THROUGH THE FRACTURE SURFACE

PHOTO NO: CAN-400706(L)-7-70



3300X
(a)

2200X
(b)

2200X
(c)

Figs. (a) and (b) are fractographs showing the clearly defined fatigue striations found on the aluminum spar cap; (c) is a dimpled area representative of the ductile rupture mode of the steel straps.

PHOTO 20. FRACTOGRAPHIC OF FRACTURE SURFACE

PHOTO NO: CAN-400707(L)-7-70

PHOTO 21. CRACK IN LEFT-HAND WING AFT-COVER SKIN

PHOTO NO: CAN-397062(L)-12-69

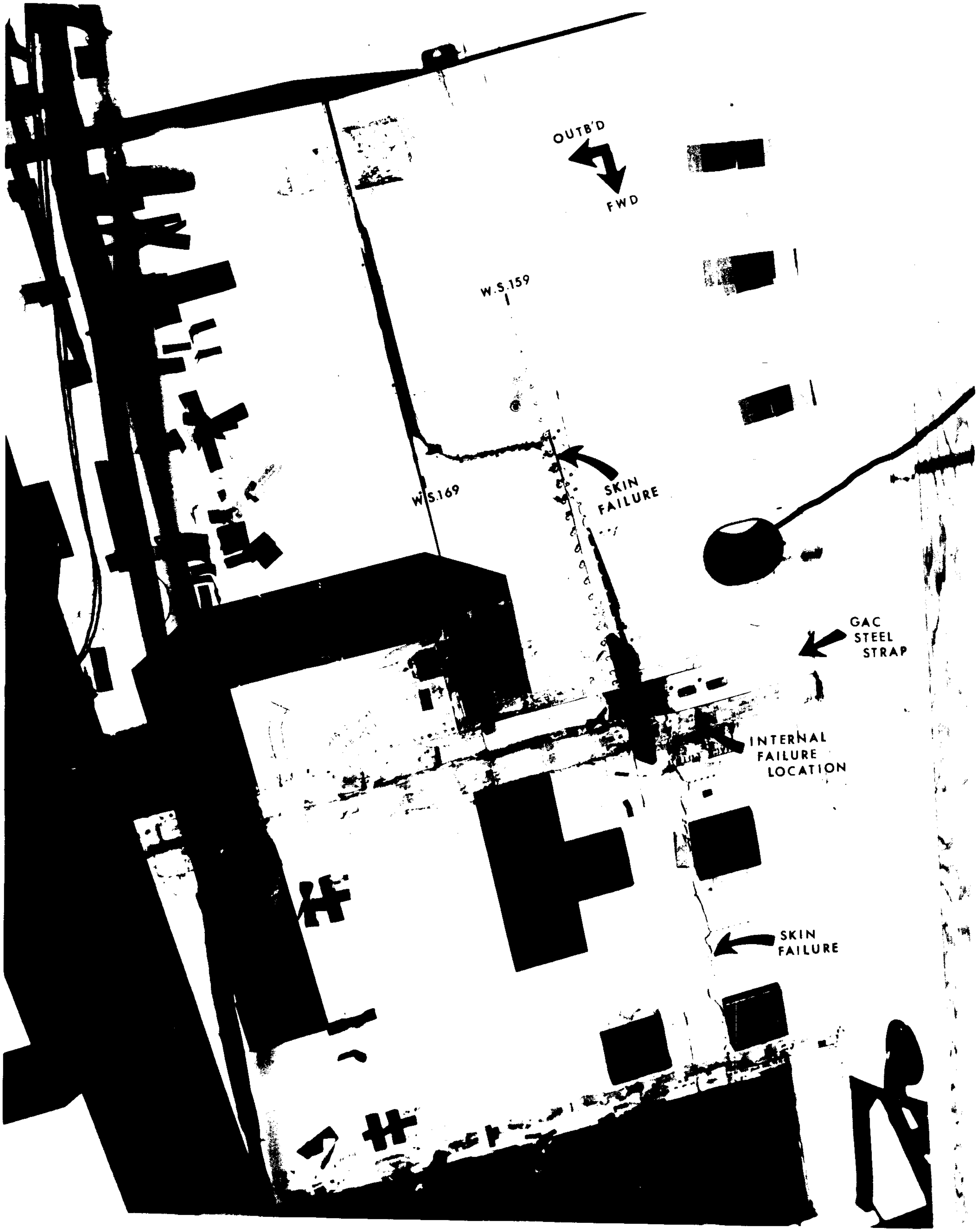


PHOTO 22. LEFT-HAND WING FAILURE

PHOTO NO: CAN-397090(L)-11-69

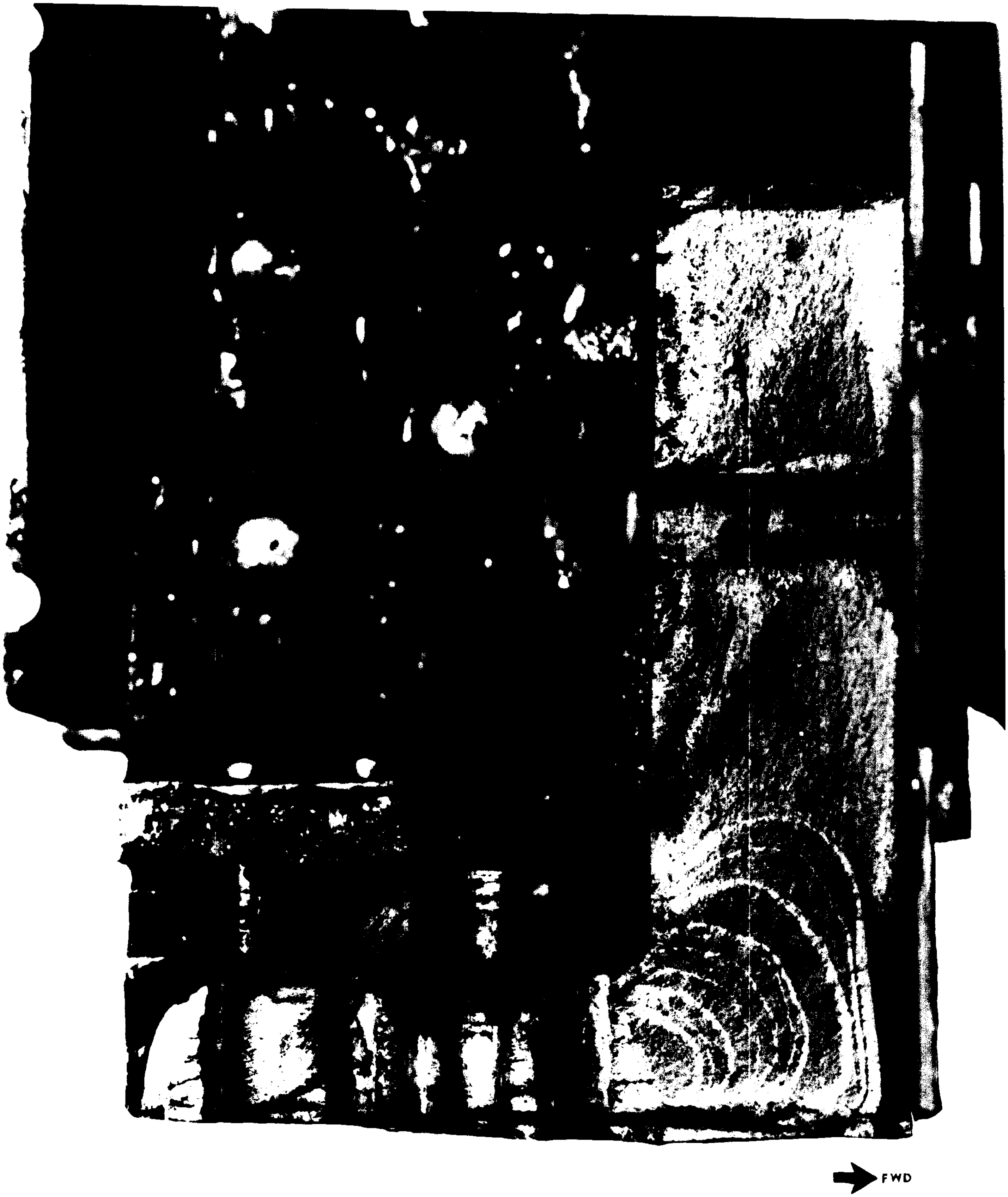


PHOTO 23. LEFT-HAND WING FAILURE SURFACE, LOOKING OUTBOARD

PHOTO NO: CAN-397228(L)-12-69

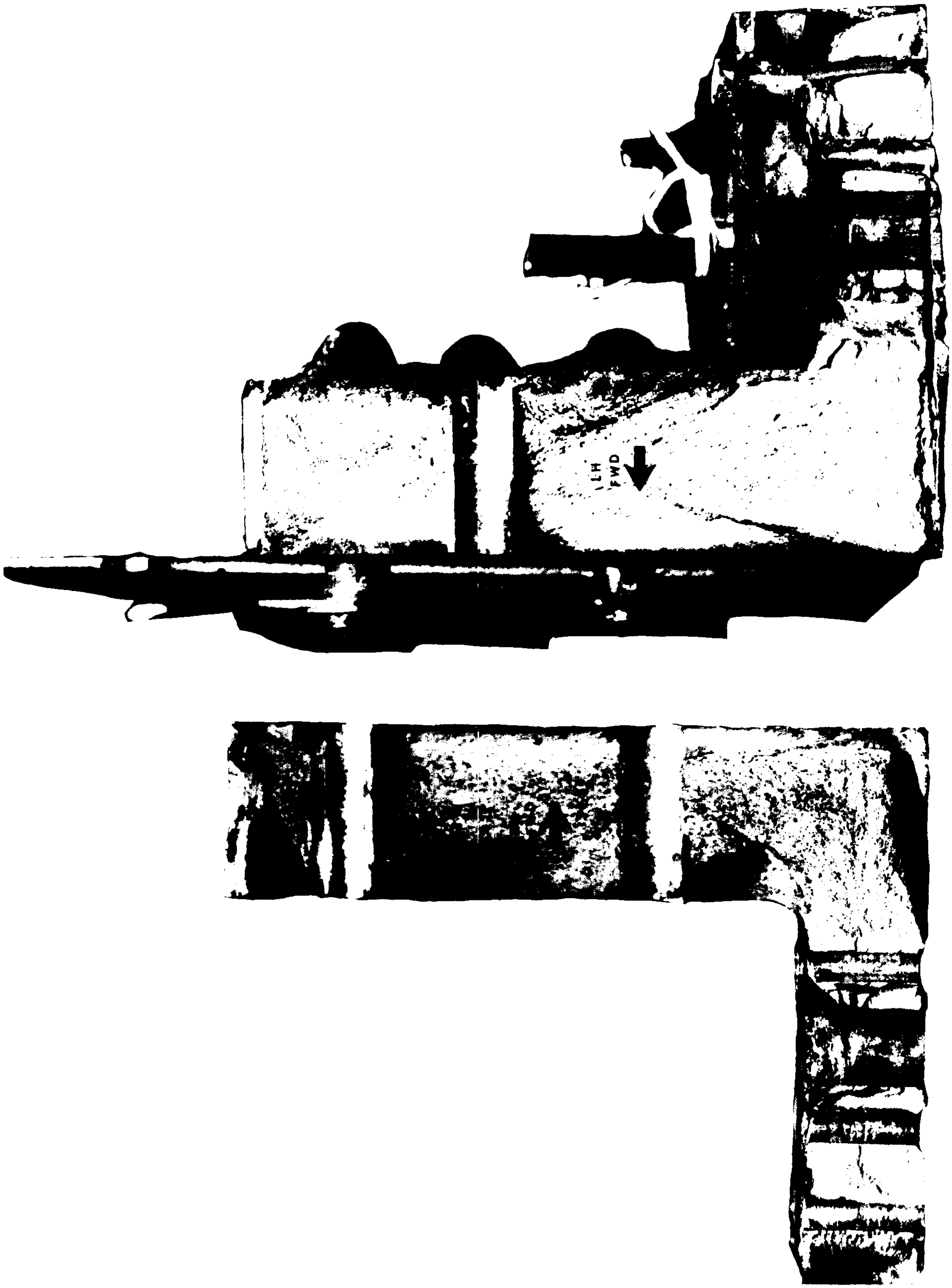


PHOTO 24. FAILURE SURFACES, LEFT-HAND WING AND RIGHT-HAND WING

PHOTO NO: CAN-398732(L)-3-70

APPENDIX A

TEST LOADS AND INSTRUMENTATION

TABLE OF CONTENTS

LIST OF TABLES AND FIGURES

TABLES

FIGURES

SYMBOLS

All symbols used herein are as defined below and in the text of this report:

W.S. . . . . wing station (measured along the Y'-axis)
H.S. . . . . hull station
W.L. . . . . water line
L.L. . . . . limit load

SIGN CONVENTION

The following sign convention was used:

Distance and forces: Positive when they are up, aft, and outboard with respect to the reference axes. (See figure A-1). Individual torsional moments about the Y-axis are positive as defined by the left-hand thumb rule.

Shear or torsional moments: Positive at a section when the summation of loads or individual torsional moments outboard of the section is positive.

Bending moments: Positive at a section when the moment produces compression on either the top or trailing-edge fibers.

REFERENCE AXES

Airplane (Fig. A-1)

X-axis: Lies in the plane of symmetry and is parallel to the hull reference line and lies 28 inches below the keel.

Y-axis: Perpendicular to the plane of symmetry at hull station 0, which is 21 inches ahead of the bow.

Z-axis: Perpendicular to the X-Y plane through the intersection of the X and Y axes.

Wing (Fig. A-1)

X'-axis: Lies in the X-Z plane and makes an incidence angle of 5° with the X-axis.

Y'-axis: Perpendicular to X'-axis (coincides with the 35% chord line which is at hull station 314.352) and makes a dihedral angle of 1° - 10.15' with the Y-axis at water line 155.774.

Wing (Fig. A-1) (Cont'd)

Z'-axis: Perpendicular to the X'-Y' plane at the origin of the X' and Y'-axes.

BASIC DATA

General Data (Ref. A-1)

Design flight gross weight	32,000 lbs.
Wing dihedral	1° - 10.15'
Wing incidence	5°
Wing span	1,160 in.
Wing gross area	1,043 sq. ft.
Wing mean aerodynamic chord	134.67 in.

Critical Conditions

Flight maneuver	PLAA Condition (ref. A-1)
Ground landing	Tail down landing, condition 2G (ref. A-2), plus tip float and drop tank fuel.
Water landing	Rough water landing (ref. A-3)

Discussions with the Grumman Aerospace Corp. (GAC) concerning the ground landing and water landing conditions led to the conclusion that a single negative loading condition could be used in place of the following conditions; negative flight maneuver, negative gust, ground landing, and water landing. This single negative loading condition used the maximum negative flight condition design curves and a spectrum that combines all the frequency of occurrence of loads from all the negative loading spectra.

This conclusion was reached based on the following:

1. The wing fatigue life is determined by the stresses resulting from the wing bending moment.

2. The expected critical fatigue location on the wing is outboard of the landing gear and is in the area of the wing splice (wing station 169 to wing station 239).

3. The negative bending moments shown in figure A-2 would be quite similar in shape if the water landing curve was smoothed out and the landing gear load was omitted from the ground landing condition.

4. The magnitude of the ground landing bending moment in comparison with the negative flight bending moment is such that the ground landing condition is estimated to have a negligible contribution to the fatigue damage.

5. Omission of the landing gear load affects only the local loading at the gear trunnion and the bending moment inboard of wing station 120, which is not an expected critical area. Omission of this load produces a conservative ground landing bending moment for fatigue life estimation and still has a negligible contribution to the overall wing damage.

6. Smoothing out the water landing bending moment curve so that it matches the maximum negative flight bending moment curve produces a conservative water landing bending moment. This smoothing out of the water landing curve has a negligible effect in changing the overall wing fatigue life.

7. Effects on the fatigue life were determined by using the Smith method of fatigue life prediction as found in reference A-4.

8. The negative loading spectra will be combined such that the water landing bending moment is equal to the maximum negative flight bending moment and the ground landing bending moment is equal to 61.3% of the negative flight bending moment.

9. This combination in a single negative loading condition simplifies the test setup with a negligible effect on the test results.

The critical conditions were therefore changed to the following:

Positive loading condition. . . . . PLAA condition (Ref A-1)

Negative loading condition. . . . . Combined negative flight, ground landing, and water landing.

TEST LOADS

Determination of Test Loads

The magnitudes of the test limit loads for the positive flight conditions were determined by simulation of the design curves for the HU-16E that were supplied by the Grumman Aerospace Corporation, and found in reference A-1. The vertical limit load factor is $n_{z_{LL}} =$ 2.77 g and the gross weight (G.W.) is 32,000 pounds.

The design loads for the negative flight condition were determined by ratioing the negative and positive flight bending moments for the UF-1 wing from reference A-2. The ratio is such that the negative flight limit load ($-F_{LL}$) equals 34.5% of the positive flight limit load ($+F_{LL}$).

Therefore:

$$-F_{LL} = -.345\ (+F_{LL})$$

$$n_{z_{LL}} = -.96g$$

$$G.W. = 32{,}000\#$$

The design loads for the ground landing condition were determined by using condition 2G of reference A-2, the GAC tables for inertia loads, the GAC curves for air loads, and the following fuel considerations:

1,200# fuel in tip float -- wing station 413

2,000# fuel in drop tank -- wing station 280

Therefore:

$$GL_{max} = \left[(\text{wing and fuel inertia load}).\ (n_z) + 2/3\ \text{wing lift} + \text{gear loads}\right].\ (1.15)$$

Where:

GL_{max} = ground landing load for a 10 fps sinking speed

n_z = 3.08g

G.W. = 29,500#

Vibratory factor = 15%

The design loads for the water landing condition were determined by using the GAC curves for rough water landing, the magnification factors of reference A-5, and the following:

$n_z = 5.99g$

G.W. = 25,000#

The design limit loads for the gust condition were the same as the positive flight condition (reference A-2) with $n_z = 2.77g$ and $\Delta n_z = .033$ Ude, where Ude = gust velocity.

In all cases, only the vertical shear loads were considered.

The design and test curves for the positive flight condition are shown in figures A-3, A-4, and A-5. A comparison of the bending moment curves for the negative loading conditions is shown in figure A-2. As explained previously, the maximum negative flight bending moment curve was used as the design curve for the combined negative loading condition. The test curves for this condition are shown in figures A-6, A-7, and A-8.

Magnitude of Test Loads

The magnitude of the test loads for each condition was determined by simulation of the design curves. The test loads for each condition are listed in table A-1. The balance diagrams are shown on figures A-9 and A-10.

TEST MEASUREMENTS

Axial strain gages were mounted on the front and main beams of both semispans at the locations shown in figure A-11. The number of gages and locations at each station are shown in tables A-2 and A-3. The gages on the skin were mounted as closely as possible to the beam indicated.

After the right-hand wing failure, additional gages were mounted on the left-hand wing between W.S. 149 and W.S. 193. These new gages were applied to the skin outer face at the upper and lower front and main beam skin intersections. A fifth row of gages was mounted 2-5/16" forward of the main lower beam line.

Deflection indicators were placed on the main beam at various locations in order to generate limit load deflection curves for each test condition.

REFERENCES

A-1 GAEC Report No. 2931.01B, Determination of Conditions for Wing Group Tests, of 20 Jan 1956.

A-2 GAEC Report No. 2907.1A, Stress Analysis of Wing Resultant Loads, Shears, Bending Moments and Torsions, of 1 Mar 1950.

A-3 GAEC Report No. 2902D, Strength Summary and Operating Restrictions, of 1 Jan 1956.

A-4 C. R. Smith, "A Method for Estimating the Fatigue Life of 7075-T6 Aluminum Alloy Aircraft Structures", U. S. Naval Air Engineering Center Report No. NAEC-ASL-1096, Dec 1965

A-5 GAEC Report RE-80, Investigation of the Dynamic Loads on the Wing of the SA-16B Airplane Due to Rough Water Landing, of 21 Dec 1966.

TABLE A-1--TEST LOADS

Wing former station, Y'	Chord station,	Positive loading condition limit loads, lb.	Negative loading condition maximum loads, lb.	1g base load, lb.
541	-6.00	2,000	-1,056.0	722.020
495	-9.20	2,500	-1,320.0	902.525
453	-10.00	2,500	-1,320.0	902.525
413	-14.55	2,200	-1,161.6	794.222
370	-12.67	3,000	-1,584.0	1,083.030
325	-11.84	3 800	-2,006.4	1,371.838
280	-13.95	3,800	-2,006.4	1,371.838
239	-14.69	3,200	-1,689.6	1,155.232
199	-35.00	3,000	-1,584.0	1,083.030
159	-50.00	3,000	-1,584.0	1,083.030
120	-25.75	2,000	-1,056.0	722.020
120-engine	-85.50	-6,000	+3,168.0	-2,166.060
82	+29.00	1,500	-792.0	541.515
		26,500	-13,992.0	9,566.765

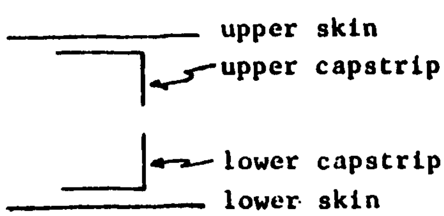
Where: 1g base load = positive limit load/2.77

Maximum negative load = (.528) positive limit load

TABLE A-2—STRAIN GAGES ON THE MAIN BEAM

Station	No. of gages	Location
5.5	2	U + L
37.5	2	U + L
71.5	2	L
89	3	U + L
100	2	L
180	2	U + L
193	2	U + L
230	2	U + L
	17	

Note:

1. Gages will be mounted symmetrically about the airplane ℄
2. U = upper capstrip or skin
   L = lower capstrip or skin
3. 

4. See figure A-11

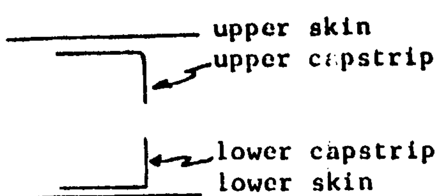
TABLE A-3--STRAIN GAGES ON THE FRONT BEAM

Station	No. of gages	Location
5.5	2	U + L
43.5	2	U + L
79	4	U + L
89	2	U + L
93	4	U + L
115.5	2	L
152	2	L
180	2	U + L
193	2	U + L
	22	

Note: 1. Gages will be mounted symmetrically about the airplane ℄

2. U = upper capstrip
L = lower capstrip

3.



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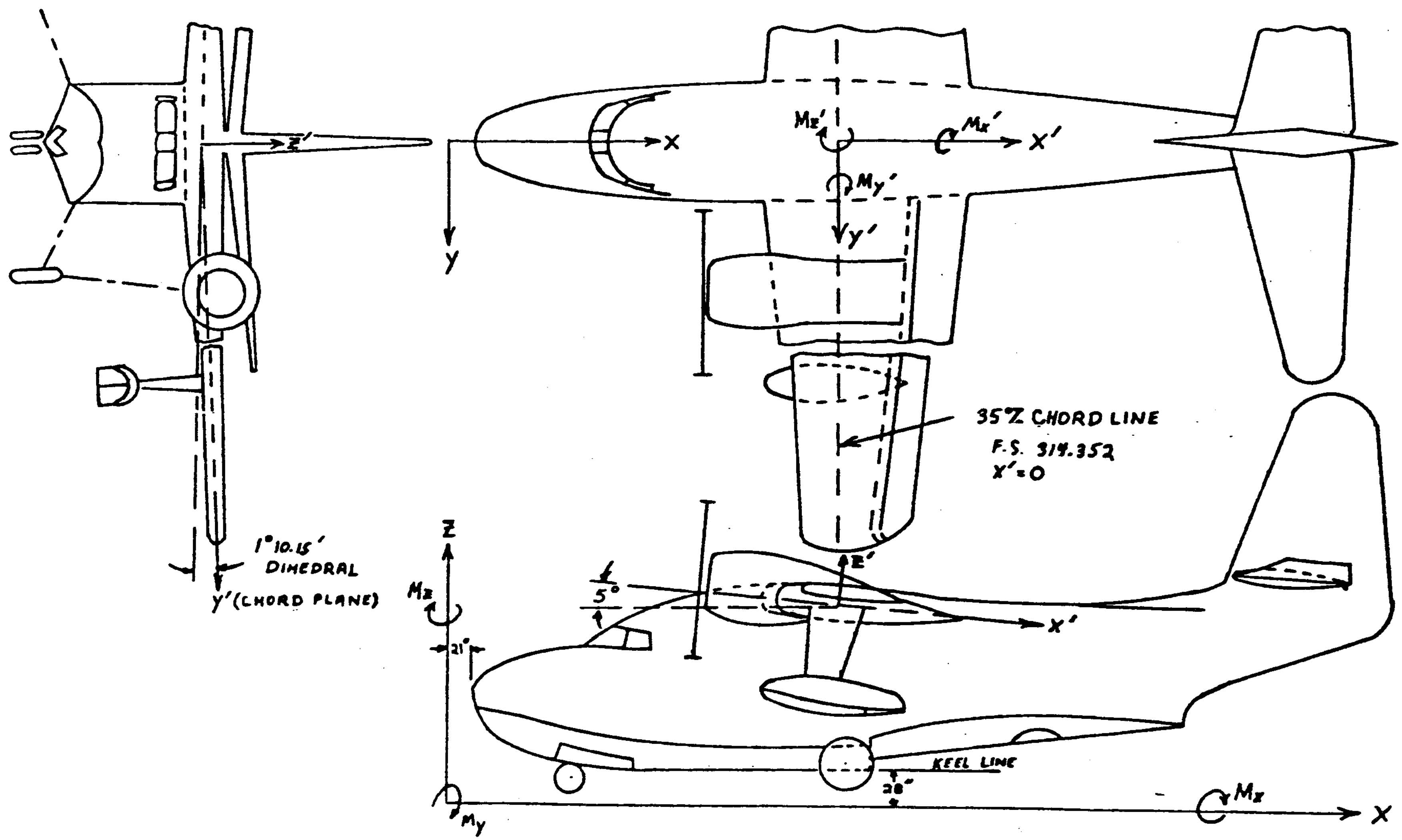


Fig. A-1--Reference axes and sign convention for individual forces and moments

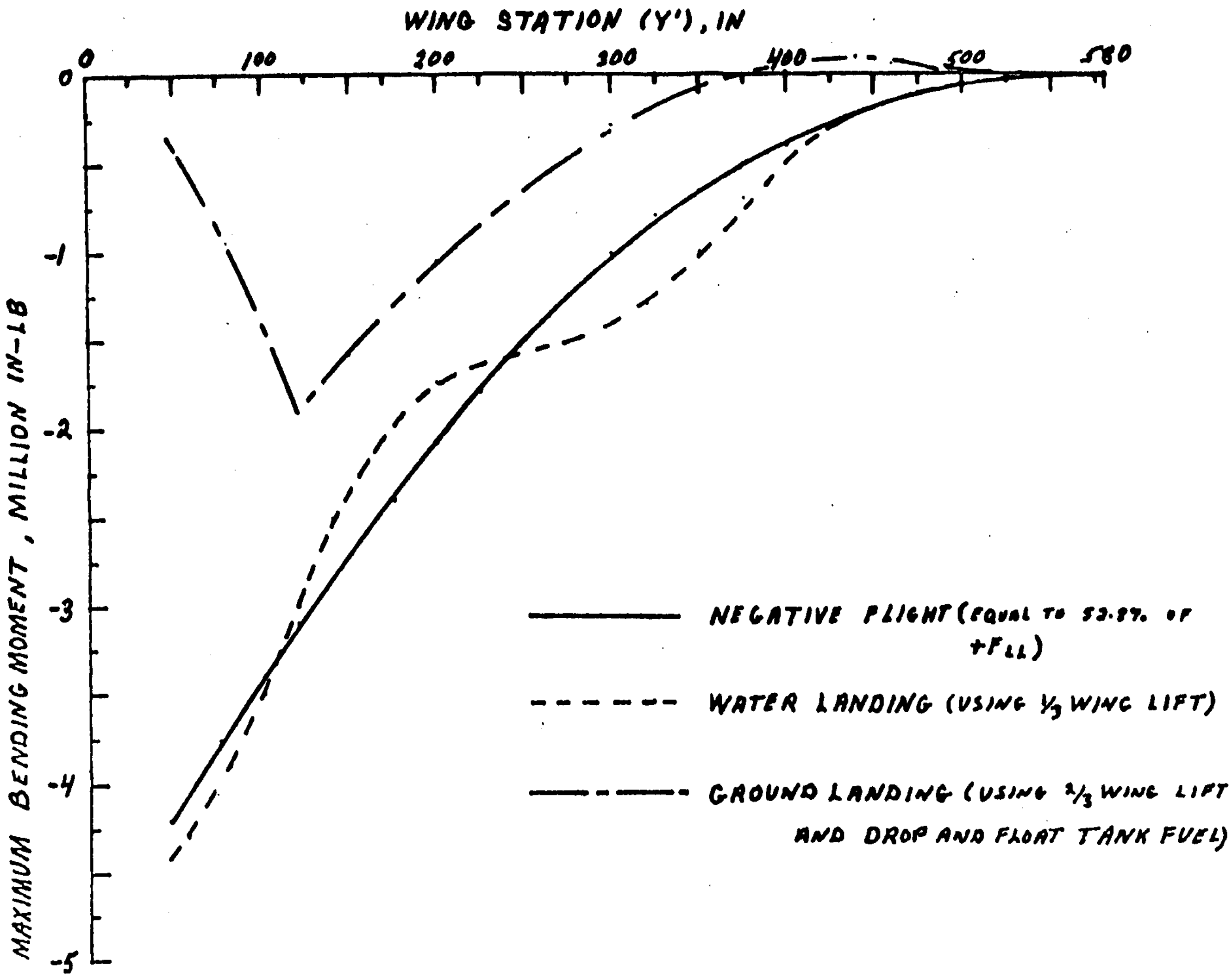


Fig. A-2--Comparison of negative bending moments.

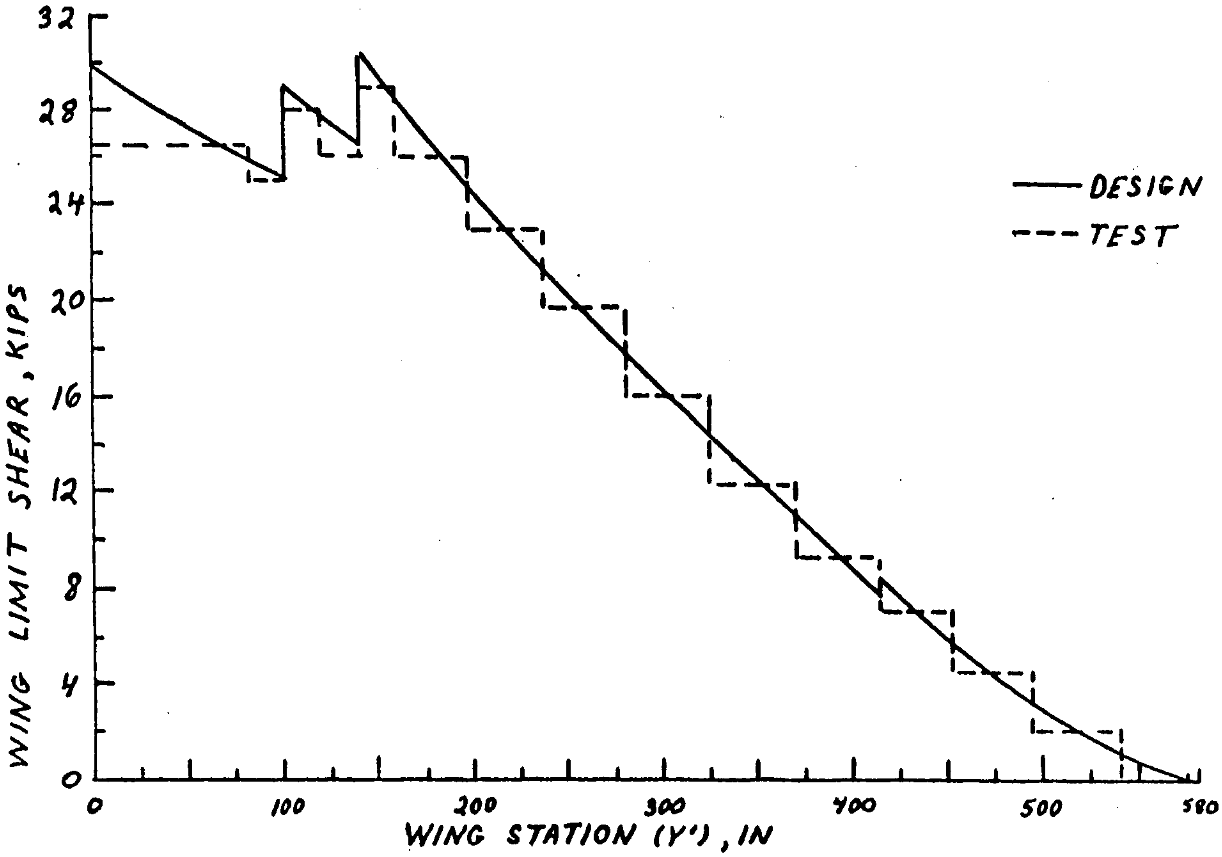


Fig. A-3-- Semispan vertical shear distribution--positive loading condition.

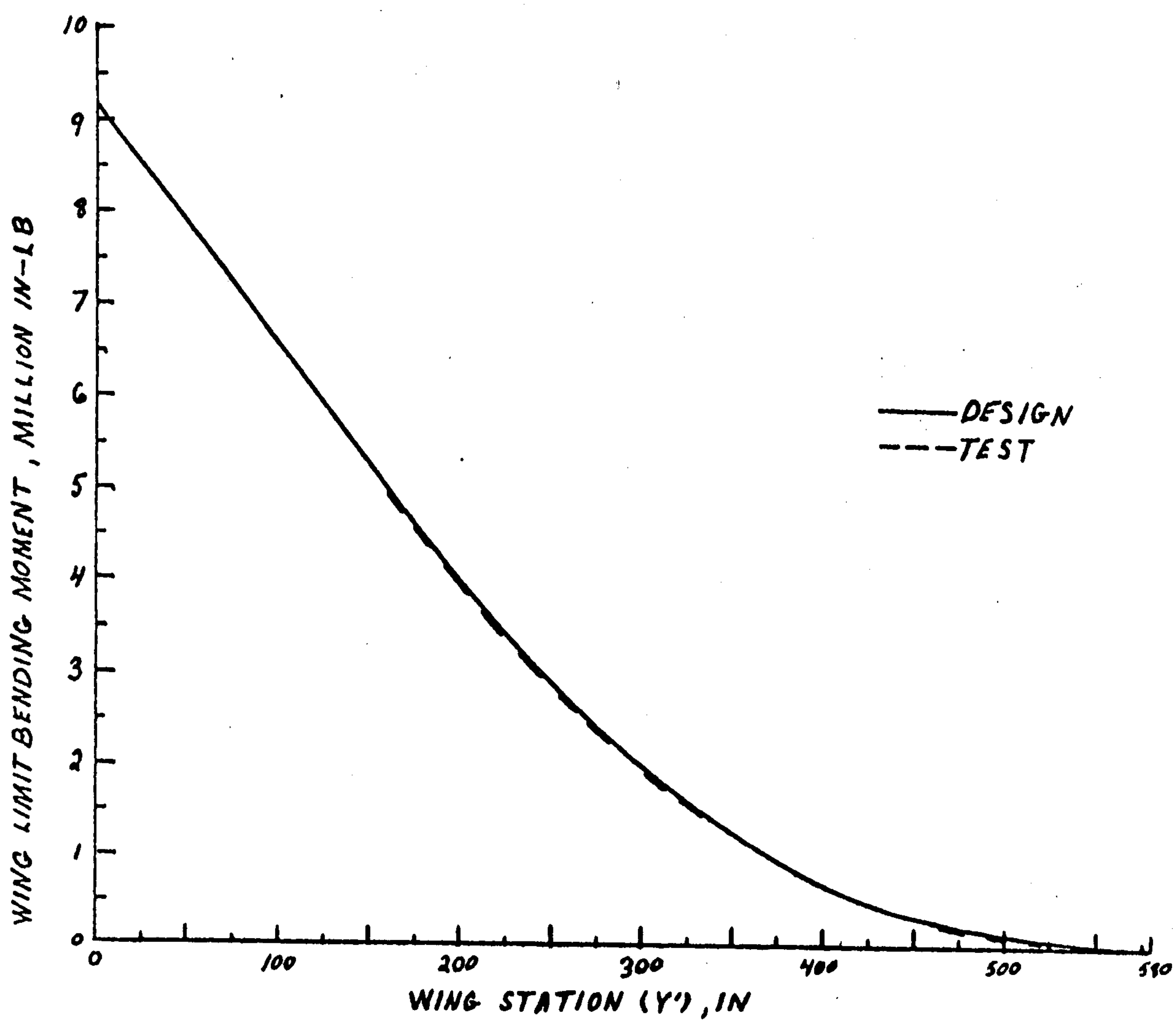


Fig. A-4--Semispan vertical bending moment distribution--positive loading condition.

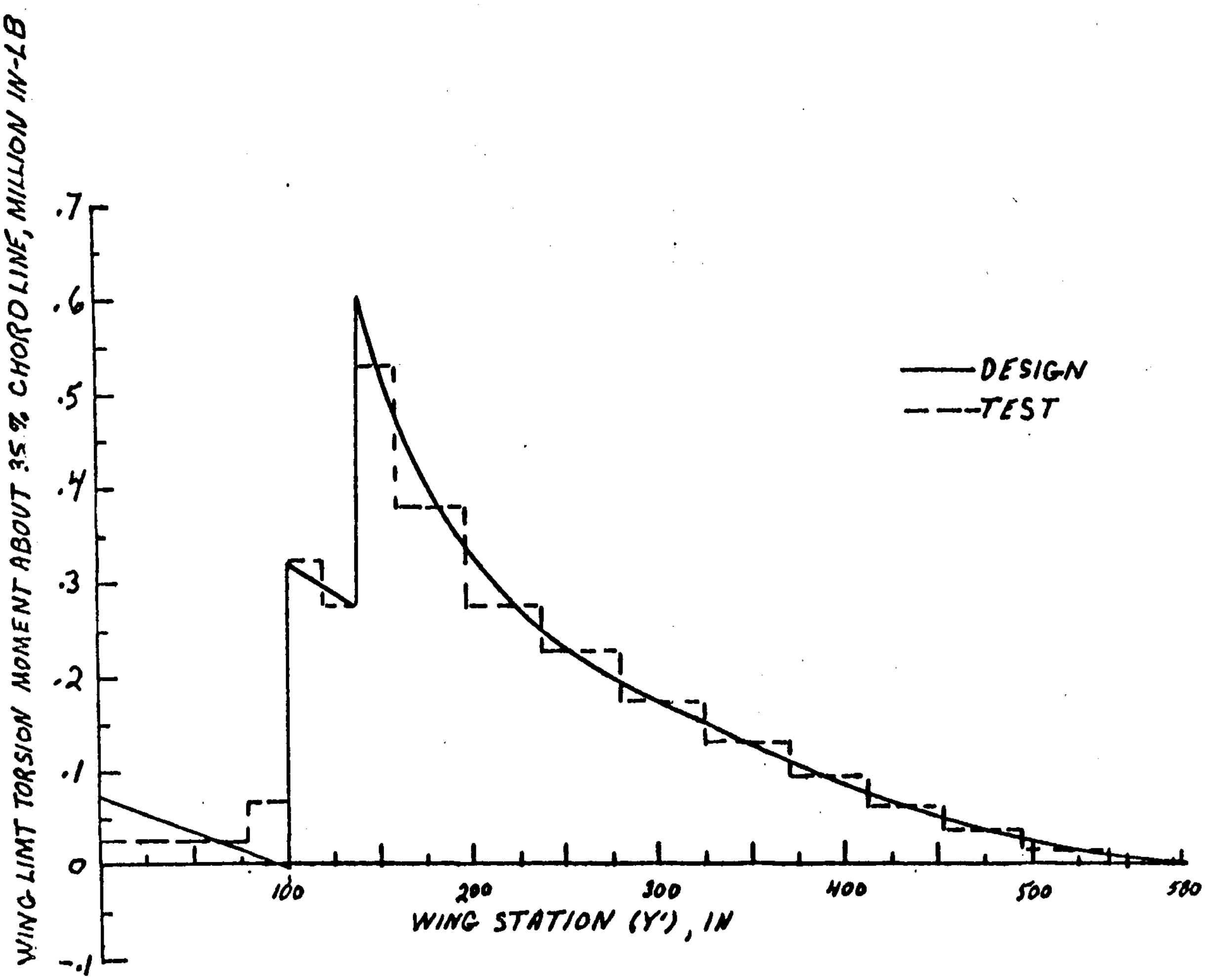


Fig. A-5--Semispan torsional moment distribution--positive loading condition.

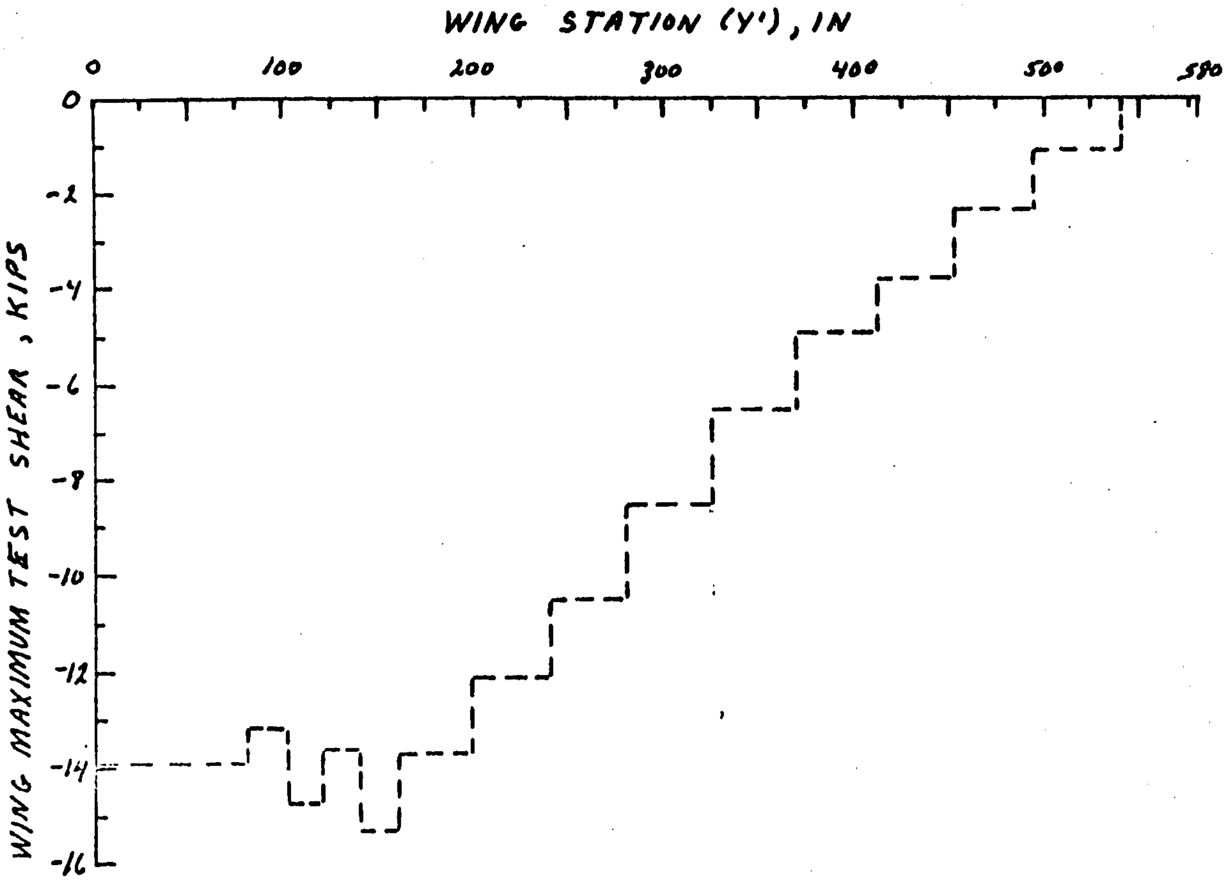


Fig. A-6--Semispan vertical shear distribution--negative loading condition.

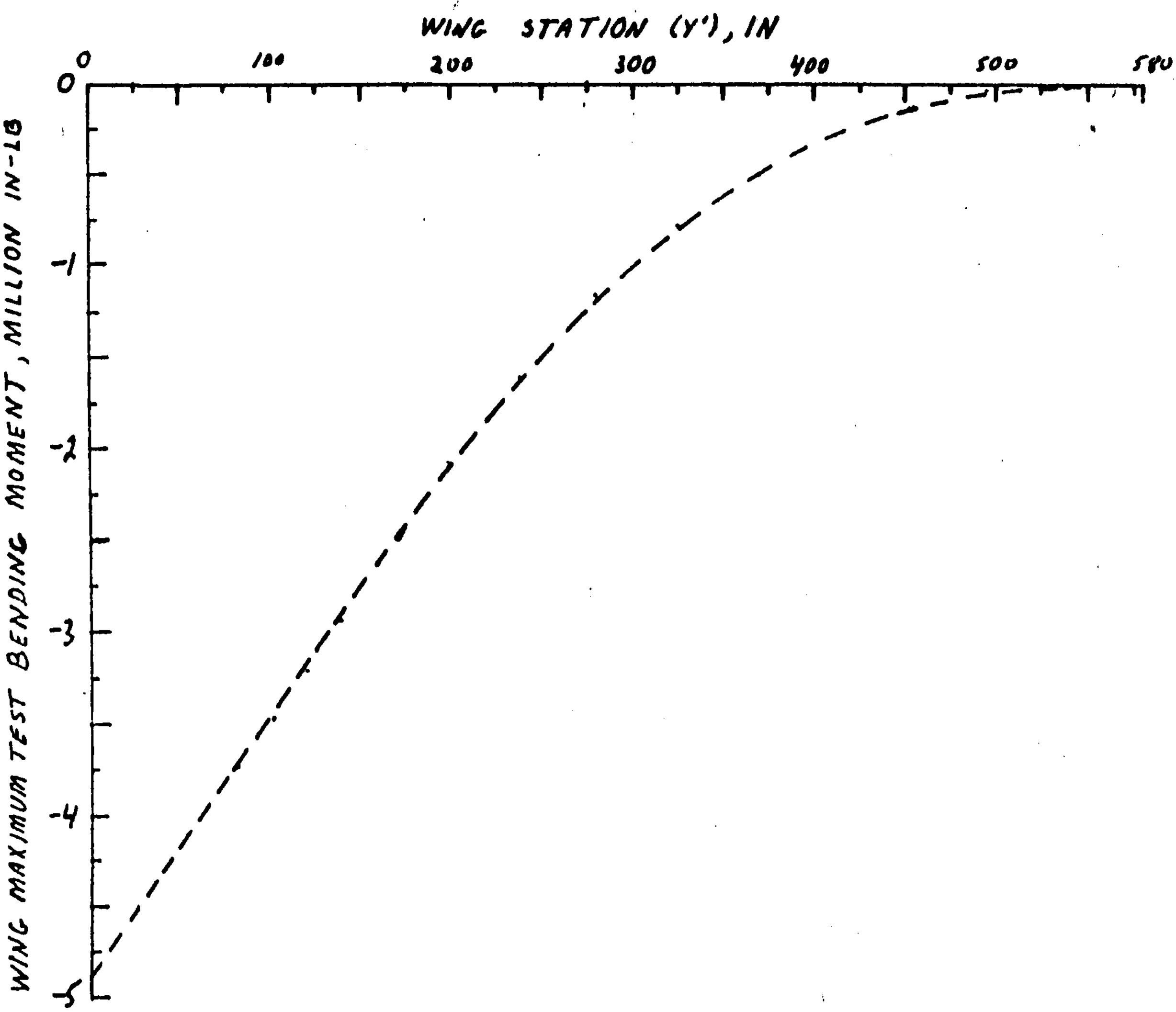


Fig. A-7-- Semispan vertical bending moment distribution--negative loading condition.

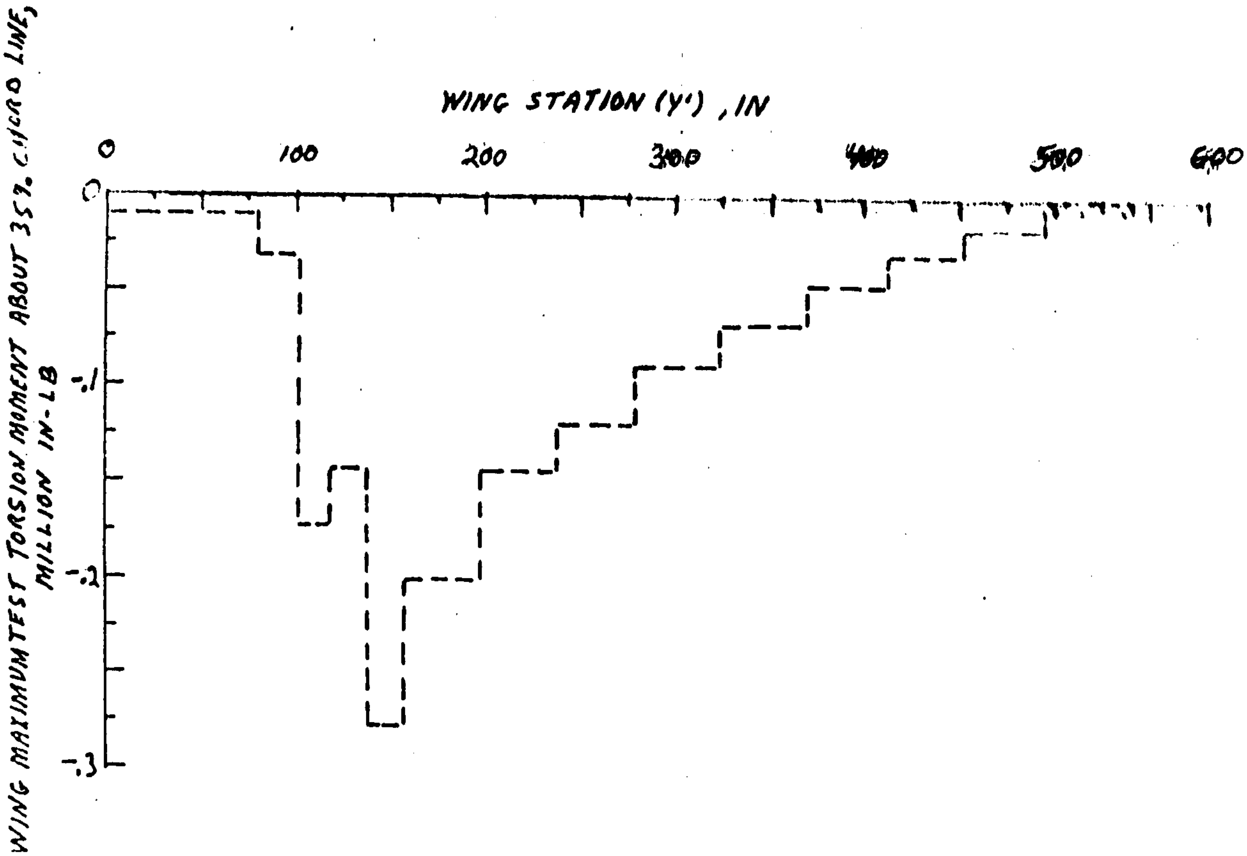
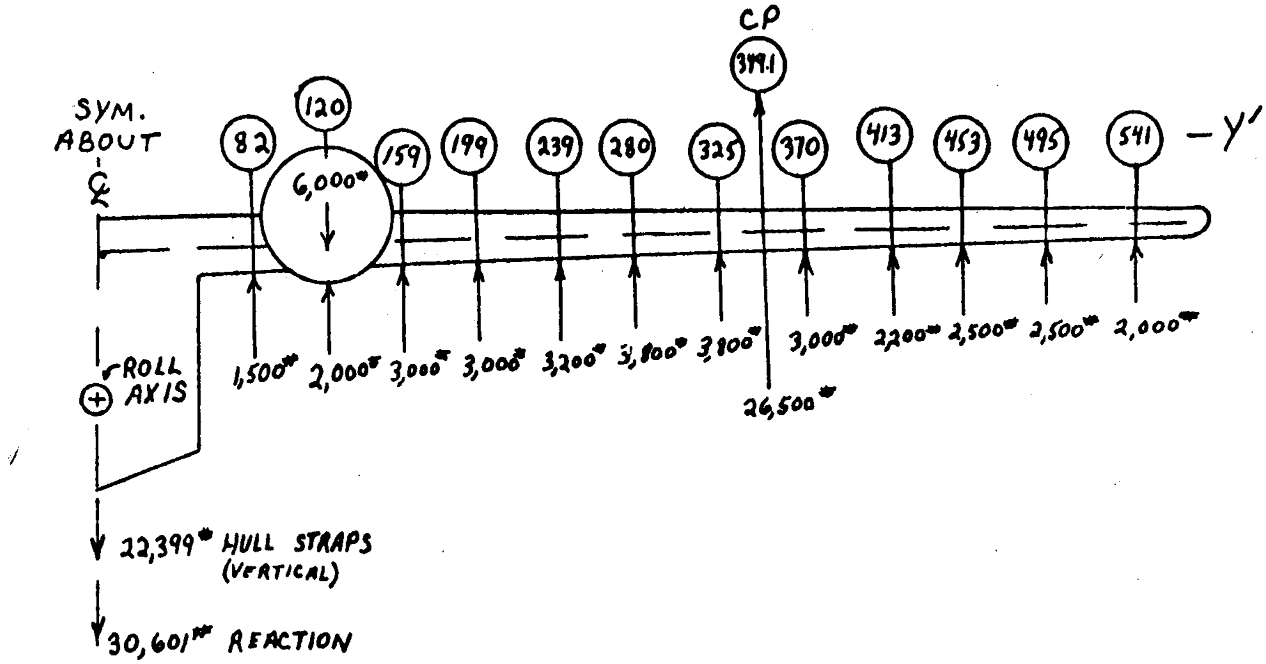


Fig. A-8--Semispan torsional moment distribution--negative loading condition.



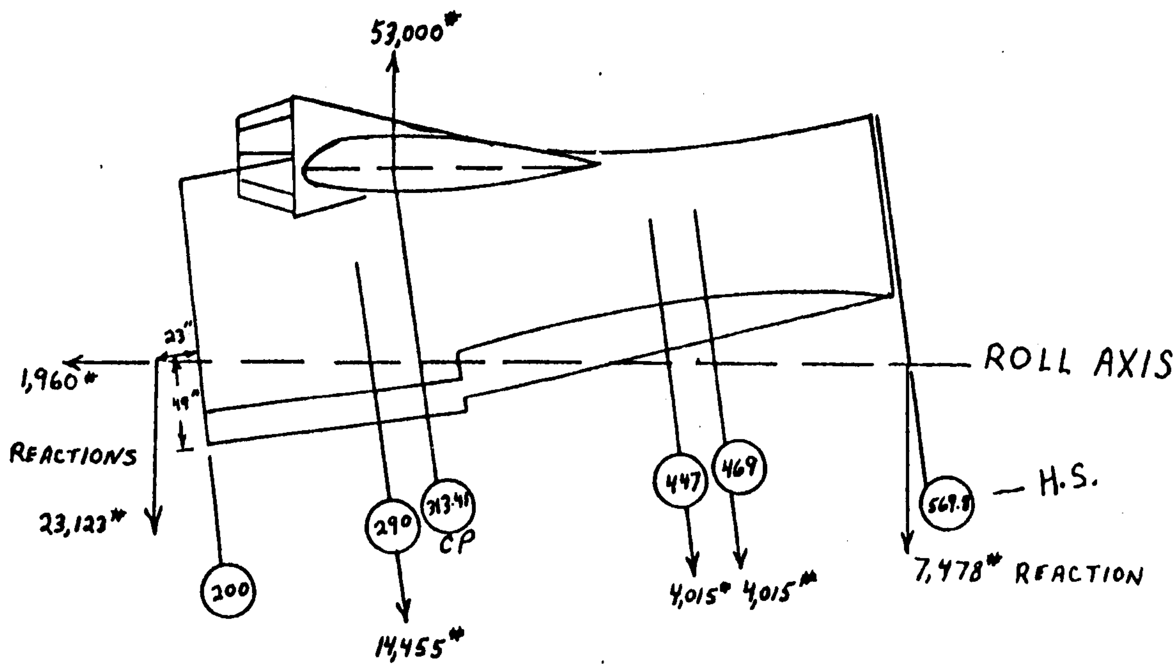
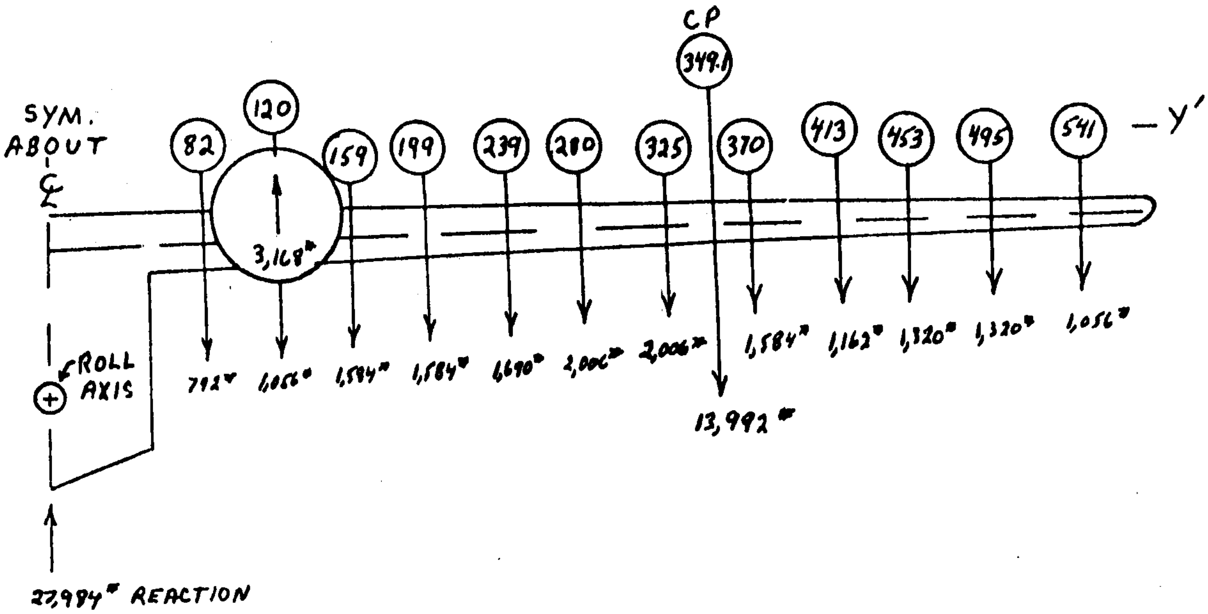


Fig. A-9--Test limit balance loads--positive loading condition



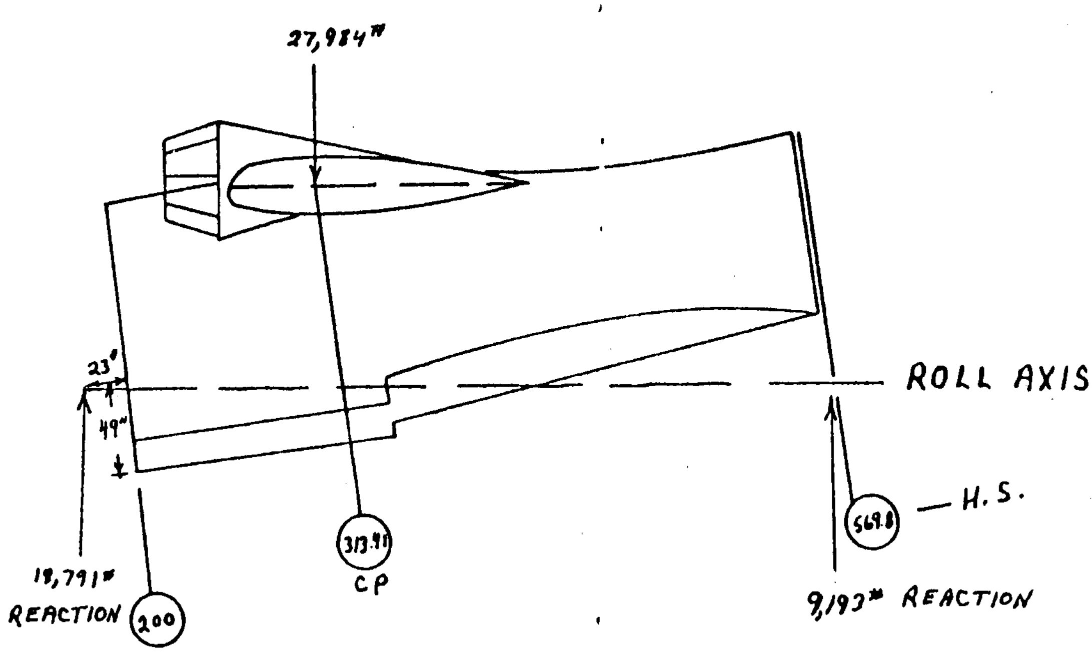


Fig. A-10--Test maximum balance loads--negative loading condition

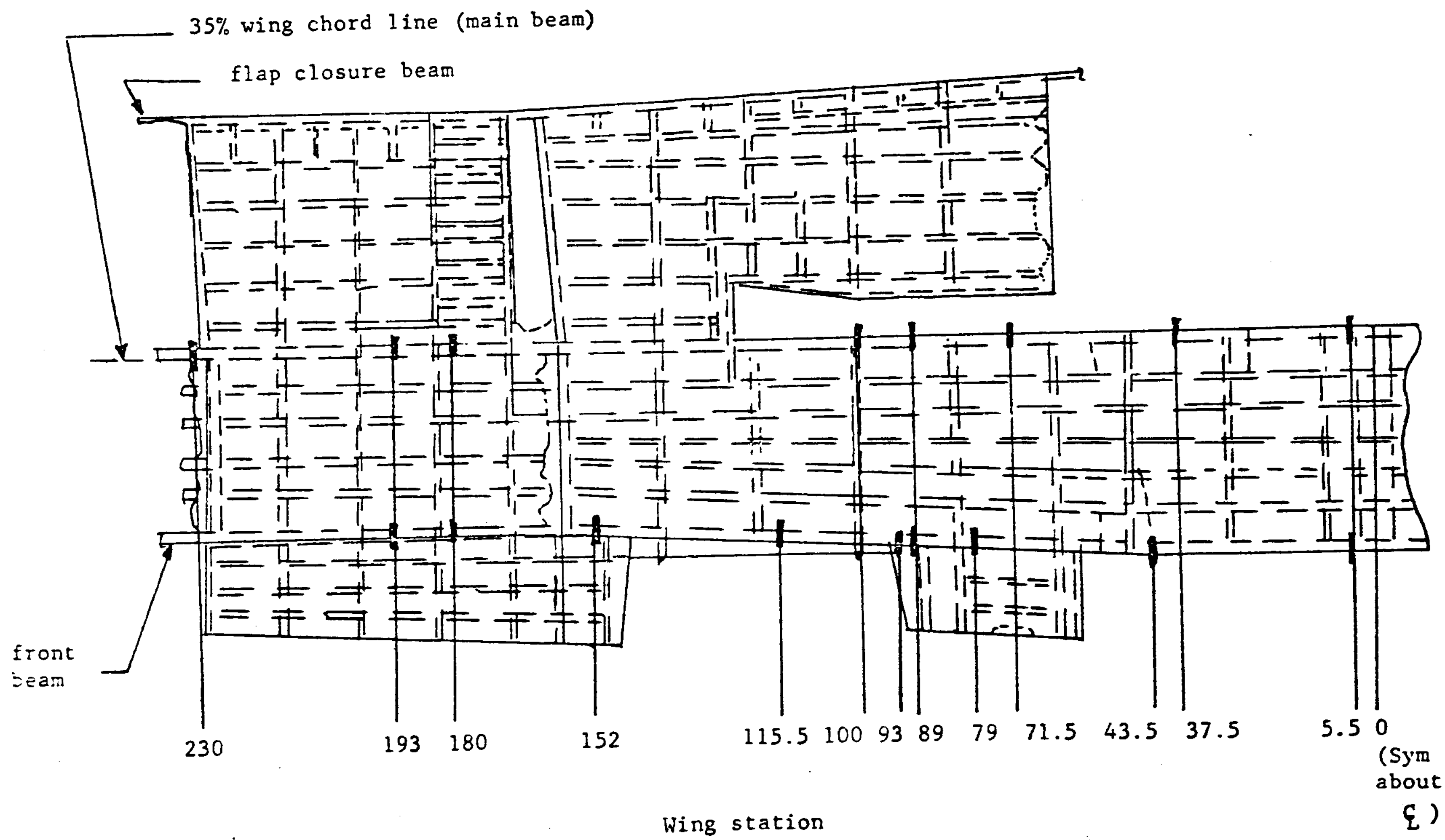


Fig. A-11--Strain gage locations

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Naval Air Systems Command Department of the Navy Washington, D.C. 20360

13. ABSTRACT

A laboratory fatigue test was performed to determine the remaining service life of the HU-16 airplane. A Coast Guard HU-16E airplane, considered to be representative of those in service and having a total 7,216 flight hours, was withdrawn from service and used as the test vehicle. During the fatigue test, a total of 8,200 test hours were accumulated on the test article prior to catastrophic wing failure. Post-failure examination revealed the presence of exfoliation (corrosion) in the wing main beam lower spar cap. The program results indicate that the presence, but not necessarily the amount, of exfoliation was a determining factor in the wing fatigue failure.

Unclassified

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Structural Fatigue Service Life HU-16 Airplane						

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