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NASA MEMO 3-15-59L

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NASA MEMORANDUM

for the

U. S. Air Force

RESULTS OF A CYCLIC LOAD TEST OF AN RB-47E AIRPLANE

COORD. NO. AF-AM-171

By Wilber B. Huston

Langley Research Center
Langley Field, Va.

CLASSIFICATION CHANGED FROM SECRET TO UNCLASSIFIED

AUTHORITY: NASA TECHNICAL PUBLICATIONS ANNOUNCEMENTS NO. 55

EFFECTIVE DATE: SEPTEMBER 1, 1961 WHL

CLASSIFIED DOCUMENT

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON

JAN 30 1959

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MEMORANDUM 3-15-59L

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RESULTS OF A CYCLIC LOAD TEST OF AN RB-47E AIRPLANE*

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SUMMARY

Results of a cyclic load test made by NASA on an RB-47E airplane are given. The test reported on is for one of three B-47 airplanes in a test program set up by the U. S. Air Force to evaluate the effect of wing structural reinforcements on fatigue life. As a result of crack development in the upper fuselage longerons of the other two airplanes in the program, a longeron and fuselage skin modification was incorporated early in the test.

Fuselage strain-gage measurements made before and after the longeron modification and wing strain-gage measurements made only after wing reinforcement are summarized. The history of crack development and repair is given in detail. Testing was terminated one sequence short of the planned end of the program with the occurrence of a major crack in the lower right wing skin.

INTRODUCTION

At the request of the U. S. Air Force, the NASA has participated in a test program designed to evaluate some wing structural reinforcements of the B-47 airplane. The loading program was established by agreement between the Air Force and the Boeing Airplane Company. Operational data were used to estimate rate of fatigue damage and test loadings were designed to reproduce this rate of damage by applying cyclic loads of four types. The program, as outlined in reference 1, called for the cyclic loading to be applied to three airplanes. Testing was performed

*Title, Unclassified.

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by three agencies: the Boeing Airplane Company at Wichita, Kansas, the Douglas Aircraft Company, at Tulsa, Oklahoma, and the NASA at the Langley Research Center, Langley Field, Virginia. For convenience these airplanes will be referred to as the Boeing, Douglas, and NASA test airplanes.

As originally set up, the cyclic loading program was designed to evaluate structural reinforcements of two major splice areas in the airplane wing. One such splice, located at wing station 354 (W.S. 354), joins two sections of the wing lower skin in the region outboard of the inboard nacelle structure. The other splice, located at buttock line 45 (B.L. 45) is at the wing-fuselage junction where the wing principal structural elements change direction. Reinforcement of these splices had been deemed necessary on the basis of evidence of fatigue. During the course of the tests a new location, the upper fuselage longerons, was found to be critical in fatigue. It is the purpose of the present paper to describe the course of the investigation on the NASA test airplane, to report the stress measurements made, and to summarize the findings of the periodic crack inspections.

In order to minimize differences in applied loads due to equipment, the loading rigs for the three test airplanes were fabricated from the same drawings. Principal structural components of the loading equipment utilized in the NASA phase were fabricated by Boeing at Wichita and shipped to Langley for erection.

APPARATUS

Airplane

The airplane for the NASA phase of the program was an RB-47E airplane, with 918 hours of flying time recorded when delivered to Langley. The maintenance log showed no major structural modifications except for a programed replacement (at 519 flight hours) of the lower drag angles, which tie the wing center section to the fuselage sides. Extracts from the log and maintenance reports are given in the appendix together with available information on the prior utilization of the airplane.

The airplane was stripped of nonstructural items, such as fuel tanks and navigational, communication, and reconnaissance equipment, and was thoroughly inspected for fatigue cracks in accordance with the procedures specified in reference 2. Two cracks were found in the W.S. 354 splice area on the left side. The wing-splice reinforcements at W.S. 354 and B.L. 45 were then installed; these reinforcements are described in reference 3 and photographs are given in figure 1.

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For the test, the outboard engines and nacelles were removed, fairings were removed from the inboard nacelles, and salvageable J47-23 engines substituted for the operating J47-25 engines. Wing weight changes due to these modifications were compensated for by adding dead weights and by the weight of the wing formers (installed for load application) at wing stations 715, 515, and 258. Wing dead weight in the test was equivalent to the dead weight of the wing in flight.

Test Setup

General views of the airplane in the test position and of the loading equipment are shown in figure 2. The airplane was in a level flight attitude. Front and rear landing gears were secured to tie-downs capable of transmitting vertical loads to the floor. (See fig. 3.) Loads were applied to the airplane by hydraulic rams at seven stations. Rams at wing stations 715 and 515 were supported from the towers shown in figure 2(a); rams at wing station 258 pushed up from the floor. A fuselage loading point was incorporated at body station 1166. Points of load application for each level in the loads spectrum were as specified in reference 1.

Strain-Gage Instrumentation

Strain-gage bridges were installed in the wing and fuselage to measure stresses in various members. In general, wing strain-gage bridges incorporated two active arms installed on inner and outer surfaces of the lower wing skin. Both rosette and axial bridges were installed. All bridges were installed at the locations and with the orientations specified in reference 1.

Some of the strain-gage bridges called for in reference 1 were, by agreement, omitted from the NASA installation. Some of those specified in reference 1 were installed after the start of the tests as the need for further stress measurements was indicated. Some of the bridges were monitored continually throughout the program on recording oscillographs, whereas others were read at particular points in the program on a spotlight galvanometer. Stress measurements were made at a total of 69 locations. The bridge numbers, general locations, and point in the program at which the bridges were installed are shown in the following table:

Strain-gage bridge number	Approximate location	Strain-gage bridges installed -
23, 24, 29	W.S. 635.5	At start of test
10, 17, 19	W.S. 375	
34, 36, 37	W.S. 125	
49, 50, 55, 56	W.S. 112	
44, 45, 46	B.L. 56	
59, 60, 67, 68, 69, 72	Inboard of B.L. 45	
73 to 88, 190	Right longeron	
193 to 208, 191	Left longeron	
94, 95, 96, 97	Right longeron	After longeron modification
214, 215, 216, 217	Left longeron	
100, 101, 102, 103	Right crown-skin reinforcing plate	

For convenience a pictorial representation of the strain-gage locations on the wing and longeron is given in figure 4.

TEST PROCEDURE

Cyclic Loading

The cyclic loading of the airplane was accomplished in sequence as specified in reference 1. The loads were applied in spectrums, each spectrum consisting of three types of loading, with a specified number of cycles for each type as follows:

	Number of cycles
Ground-air-ground (GAG)	28
High-gust (Hi).	202
Low-gust (Lo)	2,926
Total	3,156

An additional type of loading, a "90 percent limit load," was applied at the start of the program and, in general, after every fourth spectrum. The actual loading history is summarized in table 1, which shows how the order of loading was varied in successive spectrums and the points in the program at which the 90 percent limit loads were applied. The applied forces associated with the maximums and minimums of these loadings are given in table 2 together with the points of application of the forces. The moments and torques associated with these forces are given in table 3. Also given in tables 2 and 3 is a loading denoted "Inspection." These inspection loads were applied and maintained during inspections when it was necessary to have the wing in a zero stress or jig position for strain reference, or in order to facilitate the removal of bolts during inspections.

Hydraulic pressure for the loading rams was supplied by a pump through a solenoid-operated valve. At a control panel the pump output was supplied to four individually controlled reducing valves which in turn supplied the two rams at each of the three wing stations and the single ram at the fuselage station. The loads applied by each ram were monitored by means of strain-gage-type load cells whose outputs were recorded continuously. The recorders were located at the control panel and were constantly under surveillance by the operating engineer.

Maximum conditions for any loading were fixed by setting the hydraulic pressures while monitoring the load recorders. Minimum conditions were fixed in two ways. At wing stations 715, 515, and 258, the minimum conditions were fixed by jacks with load cells attached under the wing formers. The minimum load at W.S. 258 was also limited by collars on the rams. At body station 1166 (B.S. 1166), auxiliary supports were provided which limited the ram stroke when the hydraulic pressure to the rams was reduced to zero.

Cyclic loading was controlled by timers which alternately placed a solenoid-operated valve in the load or dump position. Cycling rates were 4 cycles per minute in the low-gust loading, 2 cycles per minute in the high-gust loading, and 1 cycle per minute in the ground-air-ground loading.

The 90 percent limit load was applied in increments by manually controlling the hydraulic pressures to the rams. Beginning with no ram pressure, and with the wing in the droop position, the pressures were applied incrementally in such a way that each load step was 10 percent of the 90-percent-limit-load value. The load was returned to zero in similar steps.

Inspection

Prior to the start of cyclic loading, a uniform inspection procedure was established for all three test airplanes. Inspections were scheduled following each two spectrums. A careful visual inspection was given to all external skin and to the interior portion of the wing in the root region. In addition, 71 bolts were removed at each W.S. 354 splice and 50 bolts were removed at each B.L. 45 splice in order to permit inspection of the bolt holes with a borescope. Following the installation of the upper longeron modification, two bolt holes were also inspected in each of the upper longeron splices.

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Operation

Cyclic loads were applied to the airplane 24 hours a day, 7 days a week, except as interrupted for load level changes, routine maintenance, inspections, or repairs. During the Boeing phase of the program, fatigue cracks in the upper longerons resulted in a fuselage failure during spectrum 13. In order to determine the stresses in the region of failure, additional strain gages were installed on the NASA test airplane and stress data were recorded during the first two 90 percent limit loadings which were applied prior to cycling. (See table 1.) Continuous cycling of the NASA test airplane followed. Further on in the program, fatigue cracks were found in the upper longerons of the repaired Boeing test airplane and in the Douglas test airplane at spectrums 17 and 12, respectively. Cyclic loading of the NASA test airplane was halted at this point (low gust, spectrum 3) to await completion of design of a longeron modification kit. Once this modification was installed, testing was resumed and completed without any major shutdowns.

RESULTS AND DISCUSSION

Stress Measurements

Stress measurements prior to longeron modification.- Wing and longeron stress measurements made in the initial phases of cyclic loading, prior to longeron modification are given in table 4. The reference condition for all wing stresses is the droop position in the low-gust loading, as indicated by the zeroes in the first line of data. For convenience in comparing stress measurements, the bending moment at W.S. 120 is also given in table 4 as a percentage of the limit bending moment at this station. All data of table 4(a) were obtained from recording oscillographs. The highest values of wing stress measured were shown by the primary arm of strain-gage bridge 36 (i.e., 36-3). The maximum value was 41.6 ksi from the reference condition or 35.4 ksi from the inspection position, corresponding to 39.3 ksi at limit load and 59.0 ksi at ultimate load.

The variation of stress with load is shown in figure 5 for representative gages from each of the wing stations surveyed during the present study. Gages 23 and 10 are at wing stations 635.5 and 375, respectively; they represent the bending stresses just inboard of the splice at W.S. 642 and outboard of the splice at W.S. 354. Gages 49 and 60 are outboard and inboard, respectively, of the splice area at B.L. 45. For comparison of the results at the four different loading configurations, the measured stresses are plotted against applied bending moment at W.S. 120, expressed as a percent of limit bending moment at that station. There is little evidence of nonlinearity with load in these plots. Torque differences

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associated with the various load configurations appear to have a noticeable effect on the stress only at W.S. 375.

Longeron stress data in tables 4(b), 4(c), 5(a), and 5(b) were recorded on a spotlight galvanometer in order to secure more rapid data workup and to reduce the number of recorder channels required. The reference condition for each load configuration is the wing in the droop position for that configuration. Although a correction due to the small differences in droop bending moment for the different configurations is indicated, any correction would be smaller than the reading error of the spotlight galvanometer.

In table 4(b), the strain-gage bridges were located on the right longeron; the installation was duplicated on the left longeron, but stress measurements on the left, tabulated in table 4(c), showed no essential difference from those on the right. The largest longeron stresses were measured at B.S. 508 (see fig. 4(c)), which also showed the largest differences between adjoining gages at the same cross section. It was at this station that the longeron cracks occurred on the Boeing and Douglas test airplanes. Gage 79 showed the largest measured stress, 44.2 ksi, or from the inspection position, 38.3 ksi, corresponding to 42.6 ksi at limit load and 63.8 ksi at ultimate load. Gage 74 located at this same fuselage station showed only a maximum of 7.2 ksi from the inspection position for 90 percent limit load.

The variation of stress with load is shown in figure 6 for the four gages at B.S. 508. The abscissa is again arbitrarily chosen as the percent of limit load for W.S. 120. There is no evidence of nonlinearity with load, as indicated by the solid lines faired through the data.

Stress measurements following longeron modification.- After installation of the longeron splice modification, strain gages disturbed by the modification were replaced, in some instances with a slight change of position. The installation is shown pictorially in figures 4(c) and 4(d). The stresses measured during the high-gust loading and the 90 percent limit loading are given in table 5. The installation on the left side was less extensive than that on the right; however, right and left sides are in essential agreement. The variation of stress with load for the strain gages at B.S. 508 is shown in figure 6 by the dashed line faired through the test data. As may be seen by the results for gage 79, the effect of the modification has been to reduce the maximum stress by 34 percent. On the other hand the stress at gage 74 has been increased slightly, resulting in a very much more uniform stress distribution through the cross section.

Gages 100 to 103 are located on the upper-skin reinforcement plate. These gages were single axial gages located on the outer surface of the skin only. Gages 100 and 101, at B.S. 442.5, appear to be responding primarily to stress induced by wrinkling of the underlying original skin. The stresses for gages 102 and 103, at B.S. 518.5, vary linearly with load and indicate a reasonably uniform stress distribution across the plate.

Throughout the program, stresses were checked periodically. Results of this check for a representative strain gage, number 37-3, are shown in figure 7, where the incremental (difference between maximum and minimum) stresses measured during cycling at each spectrum are compared with the incremental stresses measured during the static loadings made prior to the start of cycling. (See tables 4 and 5.)

Fatigue Crack Observations and Repairs

The cracks which were observed during the course of the test program are reported in table 6. This table contains 22 entries, each indicates an area in which cracks developed during the test program on the NASA airplane. Each crack area is given an identifying number from 1 to 22 in the order of observation, and these numbers are repeated in figure 8 to show the general area of crack location. Of these crack locations, 12 are on the wing and 10 are on the fuselage. Also indicated in table 6 is the spectrum during or after which the crack was first observed, and, in general, there is a reference to a figure number showing a photograph of the crack. In cases where special disposition was made, such as stop-drilling or application of a doubler, the disposition is described. Such repairs were generally made in conformance with applicable structural repair instructions. (See ref. 4.) Cracks similar to those numbered 6, 10, and 12 occurred in similar locations on one or both of the other test airplanes. Repairs to the NASA airplane were coordinated with repairs to the other airplanes. Figures 9 to 21 are photographs of cracks or repairs, the more important ones being discussed in detail in the following paragraphs.

Cracks 1 and 2 existed in the airplane at the time it was received, and crack 3 which was discovered early in the program is considered to have been in the airplane as received. Cracks in the outboard shear ties could be detected only by drilling a special 1-inch-diameter inspection hole in the upper fuselage crown skin. Photographs of the crack area (item 3) showing its differing appearance under minimum and maximum load were made through the inspection hole and are shown in figures 9(a) and 9(b), respectively. Since these holes were covered by the upper-skin reinforcing plates installed in the longeron-modification program, crack-propagation information is not available on the outboard shear ties after spectrum 3.

Cracks in the fuselage forward of the front landing gear, such as cracks 4 and 7, are believed to result from the airplane support system in which the flight-type loads such as the high-gust and low-gust loading conditions are reacted to the floor through the landing gear.

Crack 6 which occurred in the fuselage crown skin 10 inches aft of the upper-skin reinforcement plate was the first crack to be observed which may be considered to be associated with the reinforcements. It occurred during the 13th spectrum, 10 spectrums after the longeron modification. The crack which is illustrated in figure 13(a) occurred on the right side. The reinforcement plate shown in figure 13(b) was also installed on the left side at the same time to determine whether it might forestall development of a corresponding crack on the left side. When this doubler was removed at the end of the program, a crack was found in the skin beneath the doubler similar to the one shown in figure 13(a).

Cracks 8 and 9 which involved the upper inboard shear ties on both sides were not repaired. Crack 8 is illustrated in figure 14.

Crack 10 at B.S. 648 involved both the right fuselage crown skin and the adjacent stiffener that supports the walkway door hinge. Crack occurrence was marked by a loud report, which was observed by one of the test personnel who happened to be standing in the vicinity on the floor beside the fuselage. The cracks occurred during the high-gust portion of the 20th spectrum, 17 spectrums after the longeron modification. The cracks and repairs are illustrated in figure 15.

During inspection 12, following spectrum 24, small cracks (item 13) were found in the upper-crown-skin reinforcing plate. Barely visible to the unaided eye in the inspection position, they were discovered during the night by the chance observation of disturbances in the reflection pattern from an overhead light bulb.

During spectrum 23 a special check of the aft lower drag angles was made because some cracks had been observed in the other test airplanes. No cracks were found at that time, but small cracks were found following spectrum 28, listed in table 6 as items 17 and 18.

Item 11 in table 6 records 20 distinct cracks in the leading-edge skin, all located between wing stations 114 and 389. Typical examples are shown in figure 16. These cracks were first observed during cycling in spectrum 20; they varied in length between 1/2 inch and 6 inches at that time. Easily visible under load, they were nearly all virtually invisible when the wing was in the inspection position. At the end of the program, cracks, frequently multiple, could be found at 22 distinct stations. At eight stations the leading-edge attach angle was involved, and at three of these stations it was failed completely. Inspection of the leading-edge skin during spectrum 17 did not reveal leading-edge cracks; thus, it is virtually sure that these cracks appeared subsequent to spectrum 17. Their occurrence during the present program indicates that the life of the magnesium leading-edge sections may be less than the augmented life of the wing splices and that close inspection of the wing leading edge would be warranted as airplane flight time increases.

The first crack in a wing splice developed at the left B.L. 45 area in a rear outboard finger of the reinforcement. This crack area, item 15, is pictured in figure 19(a) and was found at spectrum 26. An additional crack, item 16, in this area was found at spectrum 28. No cracks were found in the wing splices at W.S. 354 during the program.

During the application of the 90 percent limit loading of spectrum 29, the right wing closeout panel failed. The failure was marked by a clearly audible report just as the applied load reached the load maximum. The wing was returned to the droop position by dropping the ram pressure to zero. Inspection of this crack area (item 19) showed that the whole of the lower closeout panel had failed at W.S. 179 between stiffeners 7 and 8 (see fig. 20), a distance of 21 inches, and that the inboard rivets at the ends of the fingers on the doubler of the hole for the water-injection-pump access door were involved. The crack is clearly evident in the droop position because of the marked necking-down of the material at the lower surface. Preliminary inspection of spars and stringers in the crack area has revealed no secondary failures associated with this crack.

The B.L. 45 bolt hole inspection was repeated and two cracks, items 20 and 21, were found in the outboard aft fingers of the splice plate at B.L. 45 on the right side. These cracks were not present on the preceding inspection following spectrum 28. A crack (item 22) was also found in the closeout panel on the left side at the hole for the water-injection-pump-access door at W.S. 212. The crack, which is approximately 3.5 inches long, apparently originated in a tool mark in the radius of the cutout and propagated rearward to a bolt hole in stringer 7. (See fig. 21.)

Because of the magnitude of the crack at right W.S. 179, and because of the number of cracks which had developed in the fingers of the B.L. 45 splice, as well as in the fuselage crown-skin reinforcing plate, it was considered that further cyclic loading would not contribute to the objectives of the program. Cyclic loading was, therefore, terminated.

CONCLUDING REMARKS

A cyclic load test has been completed on a B-47 airplane as part of a program to evaluate reinforced wing splices designed to increase the fatigue life of the airplane.

During the course of the tests no cracks were found in the area of the wing splice at wing station 354. Small cracks developed in the fingers of the splice area outboard of buttock line 45 after spectrum 26. Testing was terminated by the sudden development of a 21-inch crack in the right closeout panel at wing station 179 during the application of the 90 percent limit loading of spectrum 29.

Cracks were found in the magnesium leading-edge skin after spectrum 20. These cracks indicate that the fatigue life of the leading-edge skin may be less than the augmented life of the wing splices, and that close inspection of the wing leading edge would be warranted as airplane flight time increases.

Because of crack development in the upper fuselage longeron splice area of other airplanes in the test program, a longeron modification was made during the third spectrum. During the application of spectrums 13 and 20, secondary cracks developed in the upper fuselage skin which appear to be associated with this longeron modification. These cracks were repaired using standard aircraft structural repair procedures. When the tests were terminated, 25 complete spectrums had been applied to the modified longeron splice and no cracks had been revealed during inspections.

Because of the necessity for installation of the wing-splice reinforcements prior to the start of cyclic testing, stress measurements could not be obtained on the original configuration. Wing stress measurements after reinforcement are given for comparison with the stresses measured on the other airplanes in the program. Longeron stress measurements obtained before and after installation of the longeron modification showed that the modification produced more uniform stress distribution across the longeron and reduced the maximum values of stress measured.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., January 26, 1959.

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APPENDIX

EXTRACTS FROM LOG AND MAINTENANCE REPORTS OF AIRPLANE

The following data pertinent to airplane assignments and structural history have been abstracted from the historical records carried on the airplane:

RB-47E-26-RW
SERIAL NUMBER 52-736A

Prime Contractor - Boeing Airplane Company

Acceptance Date: 16 July 1954

DATE	FLIGHT HOURS ON AIRCRAFT	REMARKS
16 July 1954	10:50	Boeing delivered aircraft to 90th S.R. Wg. at Forbes AFB, Kansas
25 January 1956	511:50	Aircraft transferred to Douglas-Tulsa at Tulsa, Oklahoma, for IRAN
15 May 1956	519:05	Aircraft returned to 90th S.R.Wg. at Forbes AFB, Kansas, and assigned to 320th S.R. Sq.
1 November 1957	907:35	Aircraft transferred to Arizona Aircraft Storage Branch, Davis Monthan AFB, Arizona, for storage
2 June 1958	915:40	Aircraft transferred to AMC for cyclic loading tests
4 June 1958	917:55	Aircraft delivered to NASA at Langley Research Center

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Technical Order Compliance (TOC) Record

DATE OF TOC	DATE COMPLIED WITH	TOC NUMBER	TITLE
29 June 54	9 Aug 54	1B-47-534	Reinforcements of bulkhead at station 329.28 B-47
27 Sept 54	18 Sept 54	1B-47-605	Inspection of wing-body drag angles
15 July 55	15 May 56	1B-47-606	Replacement of wing-body angle (drag angle) B-47 series aircraft
28 Apr 58	15 May 56	1B-47-1022	Inspection of lower skin of wing center section at buttock line 35 for cracks
15 May 58	Unknown	1B-47-1024	Inspection of lower wing skin interspar area, wing station 354

Correspondence with the Air Force relative to any details on the service history of the airplane indicated that, in general, this aircraft was used predominately for high-altitude navigation and bombing missions. Also, no low-level bombing was accomplished. The only intentional unusual maneuvers which the aircraft possibly experienced were demonstrated stalls, control crossover demonstrations, and "hi-jinks" breakaway involving an approximate 1 g turn with moderate buffeting.

[redacted]

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3. Knight, J. Homer, and Woodin, Charles J.: Inspection and Structural Rework of Wing Sta. 354, Buttock Line 45, and Body Sta. 515. Doc. No. D3-1663, Boeing Airplane Co., May 25, 1958.

4. Anon.: Handbook, Structural Repair Instructions - USAF Series B-47A, B-47B, TB-47B, B-47E, RB-47E, RB-47H, RB-47K, DB-47B, YDB-47E, Aircraft. T.O. IB-47A-3, U. S. Air Force, Jan. 30, 1957 (rev. Jan. 15, 1958).

[redacted]

TABLE 1.- LOADING SCHEDULE

[Order of loading is given for each spectrum]

1	2	3	4	5	6
90% 90% GAG Hi Lo	Hi Lo GAG	Lo 90% GAG Hi	GAG Lo Hi	(*) Lo Hi GAG	Hi GAG Lo

7	8	9	10	11	12
GAG Hi Lo	Hi Lo GAG	90% Lo GAG Hi	GAG Lo Hi	Lo Hi GAG	Hi GAG Lo

13	14	15	16	17	18
90% GAG Hi Lo	Hi Lo GAG	Hi GAG Hi	GAG Lo Hi	90% Lo Hi GAG	Hi GAG Lo

19	20	21	22	23	24
GAG Hi Lo	Hi Lo GAG	90% Lo GAG Hi	GAG Lo Hi	Lo Hi GAG	Hi GAG Lo

25	26	27	28	29
90% GAG Hi Lo	Hi Lo GAG	Lo GAG Hi	GAG Lo Hi	90%

*90% omitted from spectrum 5 because of extra calibrate loading required after longeron modification in spectrum 3.

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TABLE 2.- FIXED-WEIGHT LOADS, RAM LOADS, JACK LOADS, AND POSITIONS OF LOAD APPLICATION FOR LOADING CONFIGURATIONS USED

[Loads in kips; positions in inches ahead of elastic axis (38 percent chord)]

Type of loading	Load level	W.S. 258						W.S. 515				W.S. 715				B.S. 1166
		Fixed weight		Ram		Jack		Ram		Jack		Ram		Jack		Ram load
		Load	Position	Load	Position	Load	Position	Load	Position	Load	Position	Load	Position	Load	Position	
90 percent limit	Max.	0	------	60.2	10.13	------	-----	35.9	19.46	----	-----	25.6	19.77	-----	-----	16.0
	Min.	0	------	0	-----	0	-----	0	-----	0	-----	0	-----	0	-----	0
Ground-air-ground	Max.	1.3	131.36	64.0	15.04	------	-----	23.0	16.28	----	-----	16.4	19.77	-----	-----	10.8
	Min.	1.3	131.36	6.5	15.04	.6	49.05	0	-----	0	-----	0	-----	0	-----	0
High gust	Max.	.85	131.36	57.4	15.04	------	-----	26.0	16.28	----	-----	18.5	19.77	-----	-----	11.6
	Min.	.85	131.36	26.0	15.04	11.6	49.05	0	-----	1.9	12.13	0	-----	5.3	19.90	2.5
Low gust	Max.	0	------	49.9	15.04	------	-----	20.7	16.28	----	-----	14.6	19.77	-----	-----	8.9
	Min.	0	------	38.6	15.04	4.7	49.05	0	-----	13.2	4.31	0	-----	9.2	19.90	5.2
Inspection	----	.95	131.36	----	-----	14.425	49.05	----	-----	5.8	12.13	----	-----	3.675	19.90	0

[redacted]

[redacted]

TABLE 3.- ABSOLUTE VALUES OF WING BENDING MOMENTS AND TORQUES AND FUSELAGE BENDING MOMENT FOR LOADING CONFIGURATIONS USED

[Moments and torques in millions of inch-pounds]

Type of loading	Load level	W.S. 635		W.S. 354		W.S. 120		B.S. 508
		Moment	Torque	Moment	Torque	Moment	Torque	Moment
90 percent limit	Max.	1.755	0.4320	12.69	1.1385	31.05	0.9675	37.46
Ground-air-ground	Max.	1.0179	.2500	7.3602	.6412	21.0105	.6547	27.39
	Min.	-.294	-.0742	-2.260	-.0572	-5.690	-.8810	-4.32
High gust	Max.	1.1876	.2919	8.5869	.7304	22.5630	.7031	28.46
	Min.	.1275	.0314	.9221	.0827	4.8645	.1516	8.39
Low gust	Max.	.8775	.2153	6.345	.5693	17.2845	.5386	22.35
	Min.	.4417	.1087	3.1936	.2865	10.1948	.3177	14.47
Inspection	----	0	------	0	------	0	------	1.27

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TABLE 4.- WING AND LONGERON STRESS MEASUREMENTS AFTER WING-SPLICE REINFORCEMENT,

AND PRIOR TO LONGERON MODIFICATION

[All stresses measured from wing-drooped low-gust configuration, rams unattached, in ksi]

(a) Wing stress measurements

Load level	Percent limit load at W.S. 120	Stress measured at strain-gage bridge -																								
		10	17	19	23	24	29	34	36-1	36-2	36-3	37-1	37-2	37-3	44-1	44-2	44-3	45-1	45-2	45-3	46-1	46-2	46-3	49-1	49-2	49-3
Low-gust loading																										
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	29.5	14.2	12.1	13.8	4.0	2.7	3.7	15.8	-5.6	6.2	18.0	-4.1	3.6	16.4	-3.0	2.5	13.2	-2.6	1.8	10.6	-2.5	2.1	9.1	-3.8	3.6	16.8
50%	39.9	19.1	16.2	18.6	5.2	4.5	4.9	19.8	-7.0	7.6	22.6	-5.2	4.6	20.5	-4.0	3.1	16.5	-3.4	2.4	13.3	-3.3	2.8	11.2	-4.9	4.6	20.9
Max.	50.0	22.8	19.4	22.5	6.1	5.6	6.1	23.3	-8.1	8.6	26.1	-6.1	5.4	23.8	-4.4	3.8	19.1	-3.5	3.1	15.3	-3.8	3.6	13.0	-5.6	5.3	24.3
Min.	29.5	14.3	12.2	13.9	3.8	3.4	3.7	15.8	-5.3	6.2	17.9	-4.3	3.8	16.4	-2.9	2.7	13.2	-2.6	1.8	10.8	-2.4	2.2	9.3	-3.6	3.8	16.5
Ground-air-ground loading																										
Droop	-19.3	0.6	0.1	0.1	0.1	0	0.2	-0.3	-0.1	0	-0.1	0.3	0	-0.1	0.2	0.4	0.1	0.1	0.2	-0.1	0.3	0.4	0.2	0.6	0.1	-0.1
Min.	-16.5	.2	-.3	.5	-.1	-.2	0	.4	-.4	.2	.6	0	.1	.4	-.2	.1	.5	-.3	-.2	.3	.2	.2	.4	.4	.3	.4
50%	22.2	13.0	10.9	12.7	3.5	3.2	3.4	13.8	-4.9	5.4	15.5	-3.6	3.1	14.0	-2.6	2.2	11.6	-2.5	1.6	9.1	-2.4	2.2	8.0	-3.3	3.0	14.5
Max.	60.9	25.9	22.4	25.9	7.1	6.4	6.8	27.9	-9.7	10.3	30.7	-7.2	6.4	28.0	-5.1	4.4	22.5	-4.5	3.6	18.4	-4.2	4.3	15.5	-6.9	6.3	28.1
High-gust loading																										
Droop	-19.1	0.6	0	-0.1	0	-0.2	0.2	-0.2	0.1	0	-0.1	0	0	-0.2	0.2	-0.2	-0.2	0.4	0.2	-0.3	0.2	0.2	-0.4	0.4	0	0
Inspect*	-.3	6.2	4.9	5.6	1.5	1.0	2.1	6.2	-2.1	2.5	6.8	-1.7	1.2	5.8	----	----	5.2	----	---	4.0	----	----	3.1	-1.3	1.3	6.4
Min.	14.1	9.1	7.4	9.2	3.4	1.8	2.2	10.9	-3.8	4.0	12.0	-3.1	2.2	10.9	-2.0	1.6	9.0	-1.3	1.2	7.7	-1.7	1.5	5.9	-2.6	2.2	11.1
Max.	65.5	28.6	24.8	28.5	8.1	6.8	7.6	29.4	-9.8	10.6	32.3	-7.8	6.4	29.4	-5.6	4.4	23.7	-4.4	4.1	19.4	-4.7	4.5	15.7	-7.3	6.6	29.7
Inspect	-----	6.9	5.3	6.3	1.2	.8	1.8	5.4	-1.8	2.3	6.2	-1.6	1.0	5.3	-1.2	1.0	4.7	-.6	.8	3.4	-1.0	.9	2.5	-1.1	1.1	5.7
90 percent limit loading																										
Droop	-18.8	1.8	0.6	0.1	0.1	-0.5	0.4	-0.7	1.0	-0.1	0.3	0	0	0.7	0.4	-0.3	-0.4	1.0	0.2	-0.4	0.3	-0.2	-1.5	1.2	0.4	1.3
20	5.4	10.2	7.6	8.5	2.3	1.6	2.7	7.1	-1.8	2.9	9.3	-2.1	1.6	8.5	-1.0	1.1	6.3	-.3	1.2	4.8	-1.1	1.0	3.2	-1.1	1.5	9.4
30	17.5	14.6	11.4	12.9	3.6	2.7	3.9	11.0	-3.3	4.4	13.6	-3.4	2.6	12.8	-2.0	1.8	9.5	-1.0	1.9	7.4	-1.7	1.8	5.6	-2.3	3.0	13.7
40	29.6	19.1	15.0	17.7	5.0	3.8	4.9	15.4	-4.9	5.4	18.5	-4.5	3.4	17.2	-2.9	2.4	13.1	-1.8	2.5	10.2	-2.5	2.5	7.8	-3.4	3.9	18.4
50	41.7	23.3	19.4	21.9	6.2	4.9	6.3	19.8	-6.3	7.3	23.3	-5.8	4.6	21.7	-3.6	3.1	16.4	-2.3	3.0	13.1	-3.1	3.3	10.2	-4.6	5.0	22.4
60	53.7	27.4	23.0	26.3	7.3	5.9	7.1	23.9	-7.7	8.8	27.5	-6.8	5.6	25.7	-4.3	3.7	19.6	-3.2	3.4	15.8	-3.7	3.7	12.4	-5.7	6.0	26.3
70	65.8	31.5	26.6	30.4	8.7	7.0	8.5	28.1	-8.9	10.3	31.8	-7.7	6.6	29.5	-5.0	4.3	22.9	-3.7	4.0	18.6	-4.4	4.6	14.6	-6.7	7.0	30.1
80	77.9	36.0	30.7	35.0	9.7	8.0	9.7	32.8	-10.2	12.0	36.8	-9.0	7.7	34.2	-5.7	5.1	26.4	-4.3	5.0	21.7	-4.8	5.5	17.2	-7.7	8.1	34.5
90	90.0	40.4	34.5	39.7	10.9	9.2	10.7	37.2	-11.8	13.4	41.6	-10.4	8.7	38.6	-6.7	5.7	29.7	-5.1	5.6	24.6	-5.8	6.1	19.4	-9.0	9.0	38.7
80	77.9	36.3	30.8	35.5	10.2	8.2	9.8	33.1	-10.5	11.8	37.1	-9.0	7.8	34.4	-5.7	5.1	26.7	-4.2	5.3	22.2	-4.8	5.5	17.4	-7.9	8.2	34.7
70	65.8	32.2	27.2	31.1	8.9	7.3	8.8	28.9	-9.0	10.5	32.7	-8.0	7.0	30.5	-5.3	4.4	23.2	-3.8	4.7	19.3	-4.6	4.6	14.8	-6.7	7.4	30.7
60	53.7	27.9	23.0	26.6	7.3	6.0	7.3	24.4	-7.7	8.8	28.0	-7.0	5.9	26.2	-4.3	4.0	20.0	-3.3	3.9	16.6	-3.7	4.0	12.8	-5.9	6.1	26.4
40	29.6	19.2	15.6	17.7	5.2	4.0	5.2	15.8	-4.8	5.9	18.8	-4.8	3.8	17.5	-2.7	2.6	13.2	-1.9	2.7	11.0	-2.3	2.7	8.0	-3.6	4.2	18.2
20	5.4	10.3	7.7	8.7	2.4	1.8	2.7	7.6	-2.1	3.0	9.7	-2.4	1.7	8.9	-1.3	1.1	6.5	-.4	1.3	5.3	-1.1	1.2	3.4	-1.4	2.1	9.6
Droop**	-18.8	1.1	0	-.6	-.1	-.6	.3	-1.2	1.0	-.3	-.1	0	-.2	0	.4	-.4	-.5	1.0	0	-.6	.3	-.2	-1.6	1.2	.3	.8

*Load held on rams.

**Rams attached.

[redacted]

TABLE 4.- WING AND LONGERON STRESS MEASUREMENTS AFTER WING-SPLICE REINFORCEMENT,

AND PRIOR TO LONGERON MODIFICATION - Continued

(a) Wing stress measurements - Concluded

Load level	Percent limit load at W.S. 120	Stress measured at strain-gage bridge -																								
		50-1	50-2	50-3	55-1	55-2	55-3	56-1	56-2	56-3	59-1	59-2	59-3	60-1	60-2	60-3	67-1	67-2	67-3	68-1	68-2	68-3	69-1	69-2	69-3	72
Low-gust loading																										
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	29.5	-3.5	1.0	13.7	-2.4	1.0	11.2	-2.7	.6	10.8	-3.4	8.8	15.6	-3.6	7.6	16.7	-3.0	5.8	13.7	-2.5	3.9	11.2	-1.8	2.6	7.4	17.1
50%	39.9	-4.4	1.6	17.2	-3.1	1.5	14.0	-3.4	1.1	13.7	-4.1	10.8	19.2	-4.4	9.5	20.6	-3.6	7.4	16.9	-3.0	4.9	13.7	-2.4	3.2	9.0	20.9
Max.	50.0	-5.2	2.0	19.8	-3.6	2.0	16.4	-4.0	1.6	15.9	-4.6	12.6	22.0	-5.0	10.8	23.4	-4.2	8.6	19.3	-3.6	5.6	15.8	-2.8	3.8	10.5	23.9
Min.	29.5	-3.5	1.0	13.7	-2.4	.9	11.4	-2.7	.6	10.9	-3.0	8.6	15.1	-3.2	7.6	16.4	-2.9	6.1	13.8	-2.4	4.1	11.5	-1.8	2.8	7.5	17.0
Ground-air-ground loading																										
Droop	-19.3	-0.2	0.1	-0.1	-0.1	0.3	0	-0.1	0.1	-0.2	0.3	0	-0.2	0.2	-0.1	0	0.2	-0.1	-0.1	0.3	0	0	0.4	0.3	-0.3	-0.3
Min.	-16.5	-.4	-.1	.1	-.4	-.1	.4	-.3	-.3	.3	0	.5	.6	0	.2	.7	0	.2	.4	.1	.3	.5	0	0	-.4	.6
50%	22.2	-3.2	1.0	11.8	-2.2	1.3	9.8	-2.4	.6	9.3	-2.8	7.7	13.6	-3.0	6.7	14.4	-2.5	5.0	11.5	-1.9	3.2	9.4	-1.5	2.3	6.2	14.9
Max.	60.9	-6.1	2.6	23.4	-4.3	2.4	19.6	-4.6	1.4	18.4	-5.2	15.2	26.0	-5.9	12.9	27.7	-4.9	9.8	22.8	-4.1	6.7	18.4	-3.2	4.5	12.3	27.8
High-gust loading																										
Droop	-19.1	0.2	0	-0.1	0	0	-0.3	-0.3	0	-0.4	0.2	-0.2	-0.2	0	-0.1	0	-0.3	-0.1	-0.1	0	-0.1	-0.5	0.2	0	-0.8	-0.2
Inspect*	-.3	----	----	5.1	----	---	4.1	----	---	3.7	-1.5	3.6	6.2	-1.3	2.8	6.5	-1.4	2.1	5.4	-.9	1.5	4.0	----	----	2.6	7.4
Min.	14.1	-2.3	.4	9.0	-1.8	.4	7.6	-2.0	0	7.0	-2.2	5.9	10.4	-2.4	5.0	11.4	-2.5	4.0	9.8	-2.1	2.6	8.0	-1.6	1.9	5.3	12.5
Max.	65.5	-6.2	2.8	24.7	-4.8	2.4	20.5	-5.2	1.9	19.3	-5.7	15.6	27.2	-6.4	13.4	28.9	-5.5	10.4	24.5	-4.6	7.2	19.4	-3.7	4.6	12.9	29.9
Inspect	-----	-1.0	.4	4.7	-.9	.6	3.7	-1.1	.6	3.4	-1.3	2.9	5.4	-1.1	2.4	5.8	-1.2	1.8	4.7	-.8	1.0	3.4	-.4	.8	2.0	6.6
90 percent limit loading																										
Droop	-18.8	-0.2	0.1	0.3	0.2	0	-0.3	-0.7	0	-0.3	-0.3	-0.4	0.4	0.3	-0.3	-0.3	-1.4	-0.1	0.6	-1.0	-0.8	0.2	-0.3	-0.2	-1.2	0
20	5.4	-1.4	1.2	7.0	-.9	.8	5.5	-2.3	.8	5.0	-1.9	3.8	8.1	-1.4	3.6	7.7	-2.6	2.8	7.0	-2.2	1.2	5.5	-1.1	1.2	2.2	8.7
30	17.5	-2.3	1.4	10.5	-1.7	1.1	8.2	-2.7	.8	7.7	-2.8	5.9	11.6	-2.4	5.4	11.7	-3.3	4.4	10.4	-2.8	2.3	8.2	-1.8	1.8	4.0	12.5
40	29.6	-3.5	1.7	14.4	-2.5	1.7	11.4	-3.5	1.3	10.7	-3.6	7.9	15.6	-3.5	7.3	16.0	-4.2	6.0	13.9	-3.5	3.2	11.0	-2.4	2.5	5.8	16.8
50	41.7	-4.4	2.1	17.8	-3.1	2.0	14.1	-4.1	1.7	13.4	-4.4	10.2	19.2	-4.2	9.4	20.0	-4.9	7.4	17.4	-4.1	4.3	13.8	-2.9	3.0	7.7	20.7
60	53.7	-5.1	2.8	21.3	-3.8	2.4	17.1	-4.8	2.0	16.0	-4.9	12.3	23.0	-5.1	11.3	23.8	-5.6	8.9	20.6	-4.8	5.2	16.5	-3.5	3.6	9.6	24.4
70	65.8	-6.0	3.5	24.7	-4.4	3.0	19.9	-5.4	2.4	18.7	-5.5	14.6	26.5	-5.9	13.1	27.6	-6.3	10.0	23.8	-5.4	6.1	19.0	-3.8	4.4	11.6	28.0
80	77.9	-7.0	4.0	28.6	-5.2	3.4	22.8	-6.2	2.9	22.0	-6.2	16.8	30.4	-6.6	15.0	31.6	-7.0	11.8	27.5	-5.9	7.3	22.0	-4.4	5.1	13.6	31.7
90	90.0	-7.8	4.4	32.4	-6.0	3.7	26.3	-7.0	3.4	24.8	-7.2	18.9	34.1	-7.8	16.8	35.6	-8.0	13.1	30.7	-6.8	7.[illegible]	24.7	-5.0	5.7	15.2	35.6
80	77.9	-6.8	4.3	28.8	-5.2	3.7	23.6	-6.2	3.0	22.1	-6.4	17.0	30.4	-6.8	15.0	32.0	-7.2	11.8	27.7	-6.2	7.1	22.2	-4.3	5.1	13.7	32.0
70	65.8	-6.2	3.6	25.6	-4.7	2.8	21.0	-5.6	2.6	19.6	-5.5	14.9	26.9	-6.1	12.9	27.8	-6.6	10.3	24.4	-5.7	6.1	19.6	-4.3	4.5	12.1	28.3
60	53.7	-5.4	3.1	21.8	-4.4	2.4	17.8	-4.9	2.3	16.5	-5.1	12.6	22.7	-5.1	11.0	24.0	-5.9	8.7	21.0	-5.0	5.2	16.7	-3.5	3.8	10.1	24.7
40	29.6	-3.5	2.1	14.7	-2.6	1.5	11.8	-3.4	1.4	11.0	-3.5	8.0	15.1	-3.3	7.3	15.9	-4.5	5.7	14.0	-3.8	3.3	11.4	-2.6	2.6	6.3	16.8
20	5.4	-1.8	.9	7.3	-1.3	.6	5.9	-2.1	.7	5.2	-2.0	3.7	7.6	-1.7	3.4	8.0	-2.9	3.0	7.3	-2.5	1.4	6.0	-1.4	1.3	2.7	8.7
Droop**	-18.8	.3	0	-.3	.2	-.1	-.4	-.6	0	-.7	-.2	-1.1	-.3	.3	-.7	-.6	-1.4	-.3	0	-1.0	-.8	0	-.4	-.2	-1.4	-.8

*Load held on rams.

**Rams attached.

[redacted]

[redacted]

TABLE 4.- WING AND LONGERON STRESS MEASUREMENTS AFTER WING-SPLICE REINFORCEMENT,

AND PRIOR TO LONGERON MODIFICATION - Continued

(b) Right-longeron stress measurements

Load level	Percent limit load at W.S. 120	Stress measured at strain-gage bridge -																
		73	74	75	76	77	78	79	80	81	82	83	84	85	86	87	88	190
Low-gust loading																		
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	29.5	16.7	3.5	12.7	13.6	14.0	8.7	19.7	15.0	15.1	14.8	15.0	5.9	6.7	9.4	11.1	13.0	5.8
50%	39.9	20.9	4.4	15.6	16.9	17.2	10.8	24.2	18.6	18.6	18.4	18.5	7.4	8.4	11.9	13.6	16.0	7.0
Max.	50.0	23.8	5.1	17.7	19.6	19.5	12.3	27.9	21.2	21.2	21.1	21.2	8.7	9.5	13.8	15.7	18.3	7.7
Ground-air-ground loading																		
Droop	-19.3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	-16.5	.9	.1	.6	.8	.7	.5	.3	.8	.8	.9	.8	.4	.4	.5	.5	.6	.5
50%	22.2	14.6	3.1	11.4	12.1	12.3	7.8	17.2	13.2	13.2	13.1	13.2	5.2	6.2	8.1	9.2	11.3	5.4
Max.	60.9	28.2	6.1	21.4	23.4	22.8	14.8	34.0	25.6	25.7	25.6	25.6	10.5	11.4	16.6	18.8	21.4	8.6
High-gust loading																		
Droop	-19.1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	14.1	12.2	2.7	9.2	9.8	10.2	6.3	14.0	10.9	10.9	10.6	10.6	4.3	5.2	6.6	9.8	9.4	4.6
Max.	65.5	29.1	6.2	21.2	24.2	23.3	14.9	33.8	25.8	25.6	25.5	25.8	10.1	11.7	17.0	19.3	21.9	8.8
Inspect	-----	5.2	1.0	4.0	4.3	4.2	2.6	5.9	4.6	4.5	4.9	4.8	2.0	2.4	2.8	3.0	4.0	2.2
90 percent limit loading																		
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
20	5.4	8.2	1.6	6.3	6.3	6.9	4.2	9.4	7.3	7.0	7.2	7.1	2.8	3.7	4.3	4.3	6.0	3.3
30	17.5	12.1	2.4	9.2	9.9	10.2	6.3	14.7	11.1	10.8	10.7	10.8	4.2	5.2	6.7	7.1	9.2	4.4
40	29.6	17.0	3.4	12.7	13.7	14.0	8.6	20.5	15.3	14.8	14.8	14.9	5.8	6.9	9.3	10.5	12.9	5.9
50	41.7	21.0	4.3	15.7	17.1	17.3	10.6	25.5	18.9	18.4	18.4	18.6	7.3	8.3	12.1	13.4	16.0	7.0
60	53.7	24.8	5.1	18.5	20.4	20.3	12.7	30.2	22.3	21.8	21.8	22.1	8.8	9.9	14.3	16.2	19.0	7.9
70	65.8	28.7	6.0	21.0	23.8	23.0	14.7	34.6	25.6	25.1	25.1	25.5	10.3	11.4	16.7	18.8	21.6	8.6
80	77.9	32.6	7.0	24.0	27.6	26.0	17.0	39.2	29.2	28.6	28.5	29.0	11.9	13.4	19.5	21.6	24.6	9.6
90	90.0	36.9	8.2	27.0	31.1	29.4	19.2	44.2	32.8	32.3	32.1	32.7	13.6	15.5	22.3	24.5	27.6	10.4
80	77.9	33.4	7.3	24.3	28.0	26.4	17.2	39.7	29.4	29.1	28.8	29.4	12.2	13.6	20.1	22.0	24.8	9.4
70	65.8	29.5	6.5	21.6	24.8	23.4	15.3	32.7	26.2	26.0	25.6	26.0	10.9	11.5	17.8	19.6	21.9	8.3
60	53.7	25.4	5.5	18.4	19.1	20.3	13.2	30.0	22.4	22.2	21.8	22.3	9.4	9.8	15.2	16.7	19.1	7.5
40	29.6	16.8	3.5	12.6	14.2	13.7	8.7	19.8	15.1	14.8	17.2	14.8	6.2	6.5	10.0	10.8	12.4	5.8
20	5.4	8.4	1.8	6.3	7.2	7.1	4.4	9.5	7.6	7.2	7.4	7.4	3.0	3.5	4.9	5.1	6.0	3.4
Droop*	-18.8	.2	-.1	0	.1	.1	0	-.3	-.1	-.3	-.4	-.2	-.2	.3	-.1	0	.2	.3

*Rams attached.

TABLE 4.- WING AND LONGERON STRESS MEASUREMENTS AFTER WING-SPLICE REINFORCEMENT, AND PRIOR TO LONGERON MODIFICATION - Concluded

(c) Left-longeron stress measurements

Load level	Percent limit load at W.S. 120	Stress measured at strain-gage bridge -																
		193	194	195	196	197	198	199	200	201	202	203	204	205	206	207	208	191
Low-gust loading																		
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	29.5	16.8	2.9	12.5	12.7	13.6	7.2	20.2	15.6	14.9	12.7	13.9	6.2	8.0	9.3	9.8	12.5	6.3
50%	39.9	20.9	3.8	15.5	16.1	16.8	9.2	25.0	19.4	18.5	16.0	17.5	8.0	11.1	11.9	12.2	15.4	7.8
Max.	50.0	24.0	4.5	17.7	18.7	19.3	10.6	28.8	22.3	21.2	18.4	20.0	9.4	12.5	13.7	14.3	17.6	8.6
Ground-air-ground loading																		
Droop	-19.3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	-16.5	1.5	.2	.9	.9	.9	.4	1.6	.9	.9	.8	.9	.5	.6	.6	.5	.7	.6
50%	22.2	15.7	3.0	10.2	12.3	12.7	7.0	18.1	14.4	14.1	11.9	13.0	6.1	8.8	8.8	8.6	11.3	6.7
Max.	60.9	29.2	5.9	21.6	23.7	23.4	13.5	35.1	27.0	25.9	22.7	24.4	11.9	15.3	17.2	18.0	21.3	10.5
High-gust loading																		
Droop	-19.1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	14.1	15.1	5.2	12.6	12.7	13.1	8.4	17.7	14.4	14.0	12.2	13.0	7.7	9.8	9.6	9.3	11.6	8.4
Max.	65.5	32.5	8.6	25.5	27.4	26.5	16.5	39.5	30.3	29.3	25.8	27.8	15.4	18.5	20.6	21.5	24.8	13.5
Inspect	-----	5.4	1.3	4.3	4.7	4.3	2.4	6.3	5.0	4.6	4.4	4.8	2.4	3.2	3.0	2.8	3.8	2.6
90 percent limit loading																		
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
20	5.4	8.6	1.6	6.6	6.6	6.8	4.0	9.7	0	7.4	6.6	7.0	3.5	4.9	4.5	4.2	5.9	4.1
30	17.5	12.7	2.6	9.8	10.2	10.6	6.0	14.6	12.0	11.5	9.8	10.6	5.0	7.3	7.0	6.6	9.1	5.9
40	29.6	17.4	3.4	13.1	14.2	14.4	8.0	20.3	16.0	15.4	13.3	14.5	6.6	9.6	9.6	10.0	12.7	7.2
50	41.7	21.6	4.2	16.2	17.5	17.6	9.9	25.6	19.9	19.0	16.5	18.0	8.4	11.4	12.2	12.9	16.0	8.4
60	53.7	25.7	5.1	19.2	20.9	20.8	11.8	30.4	23.5	22.5	19.6	21.3	10.3	13.4	14.7	15.6	18.9	9.7
70	65.8	29.7	5.9	22.0	24.0	23.7	13.5	34.9	27.0	25.7	22.3	24.4	12.2	15.4	17.2	18.3	21.3	10.6
80	77.9	34.0	6.9	25.0	26.0	27.0	15.8	40.0	30.9	29.5	25.5	28.1	14.1	17.5	20.0	21.2	29.4	11.8
90	90.0	38.4	8.0	27.8	31.3	30.3	18.0	45.0	34.9	33.8	28.7	32.5	16.1	20.0	22.9	24.1	28.1	13.2
80	77.9	34.3	7.0	25.1	28.1	27.2	16.1	40.8	31.4	30.0	25.7	28.4	14.6	17.6	20.4	21.5	24.7	11.8
70	65.8	30.6	6.1	22.2	24.8	24.2	14.1	35.0	27.9	26.7	22.8	25.1	13.2	15.3	17.8	19.0	21.8	10.3
60	53.7	26.2	5.1	19.3	21.4	20.8	11.9	31.3	24.1	23.0	19.4	21.4	11.6	13.2	15.3	16.1	18.9	9.0
40	29.6	17.7	3.5	13.1	14.1	14.2	8.0	20.5	16.6	15.4	12.8	14.2	7.7	9.0	9.9	10.4	12.7	6.4
20	5.4	9.1	1.8	6.9	7.0	7.4	3.9	10.1	8.2	7.7	6.3	7.1	3.8	4.6	4.8	4.9	6.2	3.6
Droop*	-18.8	.3	0	.2	.2	.2	0	.1	.1	-.1	-.3	-.3	-.2	-.2	-.1	-.2	.2	0

*Rams attached.

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TABLE 5.- LONGERON AND SKIN-PLATE STRESS MEASUREMENTS

AFTER LONGERON MODIFICATION

[All stresses measured in ksi]

(a) Right side

Load level	Percent limit load at W.S. 120	Stress measured at longeron strain-gage bridge -																				Stress measured at skin plate strain-gage bridge -			
		73	74	75	76	77	78	79	80	81	82	83	84	85	86	87	88	94	95	96	97	100	101	102	103
High-gust loading																									
Droop	-19.1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Min.	14.1	6.4	3.1	6.7	7.1	7.5	3.6	8.1	6.9	6.9	8.3	8.7	7.6	9.2	4.9	4.5	5.5	4.5	4.3	6.6	7.3	.2	.8	6.7	5.9
25%	27.0	9.4	5.0	9.3	10.2	10.3	5.8	11.4	9.8	9.6	11.8	12.4	10.2	12.5	7.2	6.1	7.9	6.6	6.6	9.4	10.4	0	.6	9.0	8.2
50%	39.8	12.1	7.0	11.7	13.3	12.9	7.8	14.6	12.5	12.3	15.0	15.8	13.2	16.0	9.6	7.4	10.0	8.8	8.8	12.2	13.3	0	.8	11.6	10.6
75%	52.7	14.6	8.9	14.0	16.3	15.3	9.8	17.8	15.2	14.9	18.2	19.1	15.8	18.9	12.1	8.5	12.0	10.8	11.0	14.6	15.9	.2	1.1	13.9	12.9
Max.	65.5	17.4	10.7	16.3	19.3	18.0	11.8	21.1	18.1	17.6	21.6	22.5	18.6	22.3	14.7	9.8	14.2	13.0	13.3	17.1	18.6	.5	1.5	16.3	15.4
Min.	14.1	5.8	3.8	6.0	7.2	6.4	3.7	6.9	6.6	6.8	7.4	8.1	7.2	7.6	5.8	3.2	4.8	4.5	4.6	6.0	6.3	1.0	1.7	6.3	5.4
Inspect	-----	2.7	1.6	3.0	3.3	3.3	1.6	3.3	3.3	3.1	4.1	4.0	2.9	3.8	2.5	1.8	2.7	2.2	2.0	2.7	2.7	-.2	-.1	2.8	2.7
90 percent limit loading																									
Droop	-18.8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10	-6.7	2.4	1.4	2.0	2.3	2.3	1.8	2.7	2.2	2.2	3.2	3.0	2.0	2.3	1.7	1.4	1.9	2.0	1.9	2.3	2.3	-.3	-.5	2.0	2.1
20	5.4	5.0	3.1	4.6	5.0	5.3	3.5	5.7	4.8	4.8	6.1	6.2	4.6	5.9	3.7	2.8	4.0	3.7	3.8	4.8	5.3	-.4	-.3	4.2	4.1
30	17.5	7.7	4.4	6.8	7.5	8.0	5.1	8.8	7.2	7.3	9.2	9.5	7.3	9.4	5.6	4.5	5.9	5.3	5.4	7.4	8.2	-.3	0	6.4	6.1
40	29.6	10.4	5.9	9.3	10.1	10.6	6.8	12.3	9.9	9.9	12.3	12.9	10.1	13.1	7.5	5.9	8.0	7.2	7.2	10.2	11.4	-.3	.1	8.7	8.2
50	41.7	13.1	7.4	11.6	13.0	13.0	8.6	16.0	12.5	12.3	15.5	16.0	12.9	16.5	9.5	7.5	10.2	9.1	9.1	12.9	14.2	-.3	0	11.1	10.6
60	53.7	15.7	9.2	13.9	15.8	15.5	10.6	19.2	15.2	14.8	18.6	19.2	15.6	20.0	11.7	8.8	12.4	11.2	11.2	15.4	17.0	-.3	.3	13.5	12.8
70	65.8	18.3	10.8	16.0	18.5	17.7	12.3	22.2	17.6	17.1	21.6	22.1	18.1	23.2	13.9	10.2	14.3	13.1	13.2	17.8	19.4	-.3	.5	15.8	15.1
80	77.9	20.9	12.5	18.1	21.2	20.1	14.0	25.2	20.2	19.6	24.6	25.3	20.6	26.6	16.1	11.7	16.5	15.1	15.2	20.1	22.1	0	.9	17.9	17.3
90	90.0	23.7	14.1	20.3	24.2	22.6	16.1	28.5	22.9	22.1	27.8	28.5	23.3	30.5	18.4	13.3	18.6	17.1	17.4	22.7	24.7	.5	1.5	20.1	19.4
80	77.9	21.1	12.6	18.3	21.6	20.2	14.3	25.6	20.5	19.9	24.7	25.5	21.2	27.2	16.5	11.9	16.7	15.3	15.6	20.5	22.3	.7	1.6	18.1	17.3
70	65.8	18.9	11.2	16.4	19.4	18.0	12.6	23.0	18.3	17.9	21.9	22.9	18.9	24.0	14.8	10.7	14.9	13.6	13.9	18.3	19.9	.8	1.5	16.1	15.2
50	41.7	13.5	8.2	11.6	13.8	12.8	9.0	16.3	12.9	12.8	15.5	16.4	13.5	16.8	10.5	7.7	10.5	9.7	10.0	13.0	14.2	1.0	1.3	11.3	10.4
30	17.5	7.8	4.7	6.8	7.9	7.4	5.2	9.4	7.4	7.5	8.8	9.6	7.9	9.9	6.1	4.5	5.9	5.6	5.7	7.7	8.4	.7	1.1	6.6	5.9
10	-6.7	2.3	1.3	1.9	2.5	2.1	1.4	3.0	2.3	2.3	2.8	2.8	2.6	3.2	1.9	1.5	1.7	1.9	1.9	2.6	2.7	.1	.3	2.4	1.9

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TABLE 5.- LONGERON AND SKIN-PLATE STRESS MEASUREMENTS

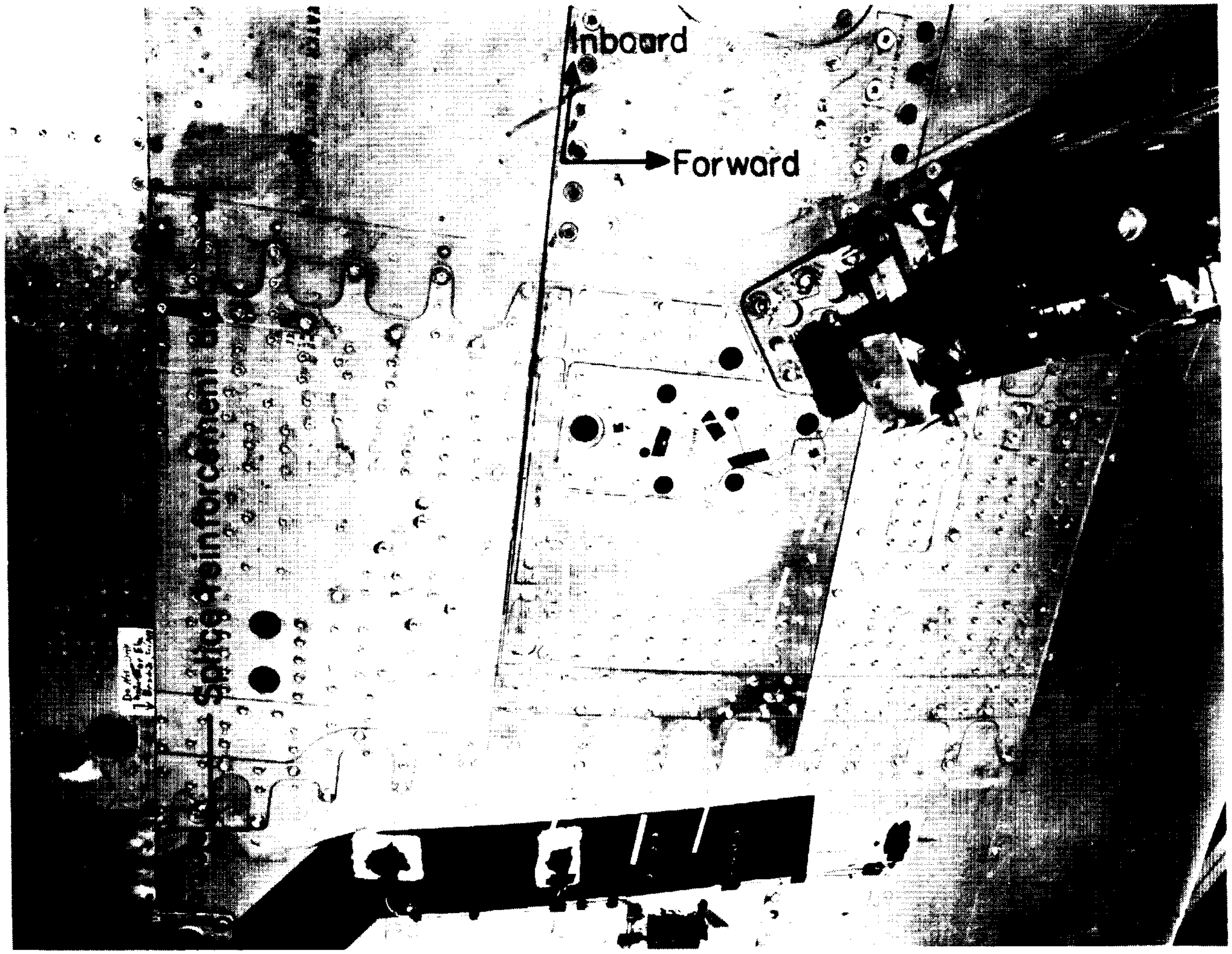
AFTER LONGERON MODIFICATION - Concluded

(b) Left side

Load level	Percent limit load at W.S. 120	Stress measured at longeron strain-gage bridge -							
		193	194	198	199	214	215	216	217
High-gust loading									
Droop	-19.1	0	0	0	0	0	0	0	0
Min.	14.1	6.7	3.7	3.3	9.4	4.8	4.4	6.8	7.6
25%	27.0	9.4	5.8	5.2	12.9	7.0	6.6	9.4	10.5
50%	39.8	12.2	7.3	7.0	16.5	9.2	8.8	12.1	13.6
75%	52.7	14.8	9.1	8.8	20.0	11.4	11.0	14.6	16.3
Max.	65.5	17.9	11.0	10.7	23.5	13.7	13.4	17.3	19.3
Min.	14.1	6.1	4.3	3.5	8.5	4.9	4.8	6.0	6.7
Inspect	-----	2.7	1.7	1.5	3.8	2.3	2.2	2.7	2.8
90 percent limit loading									
Droop	-18.8	0	0	0	0	0	0	0	0
10	-6.7	2.4	.9	1.5	3.2	2.0	1.9	2.2	2.3
20	5.4	5.0	2.8	3.1	6.5	3.9	3.8	4.8	5.3
30	17.5	7.6	4.2	4.6	10.0	5.7	5.6	7.5	8.3
40	29.6	10.5	5.7	6.1	14.0	7.6	7.3	10.3	11.4
50	41.7	13.3	7.4	7.8	17.6	9.6	9.3	12.9	14.6
60	53.7	15.9	8.8	9.5	21.0	11.8	11.3	15.6	17.4
70	65.8	18.5	10.3	11.2	24.2	13.7	13.3	17.8	20.2
80	77.9	21.1	12.0	13.1	27.4	15.8	15.5	20.3	23.0
90	90.0	24.0	13.5	14.9	31.0	18.0	17.8	22.9	25.8
80	77.9	21.5	12.0	13.2	27.9	16.1	15.9	20.6	23.3
70	65.8	19.1	10.9	11.7	25.0	14.3	14.0	18.2	20.6
50	41.7	13.6	7.8	8.4	17.7	10.3	10.1	12.9	14.7
30	17.5	8.0	4.3	4.9	10.4	6.0	5.8	7.7	8.8
10	-6.7	2.5	.9	1.5	3.4	2.0	1.9	2.3	2.8

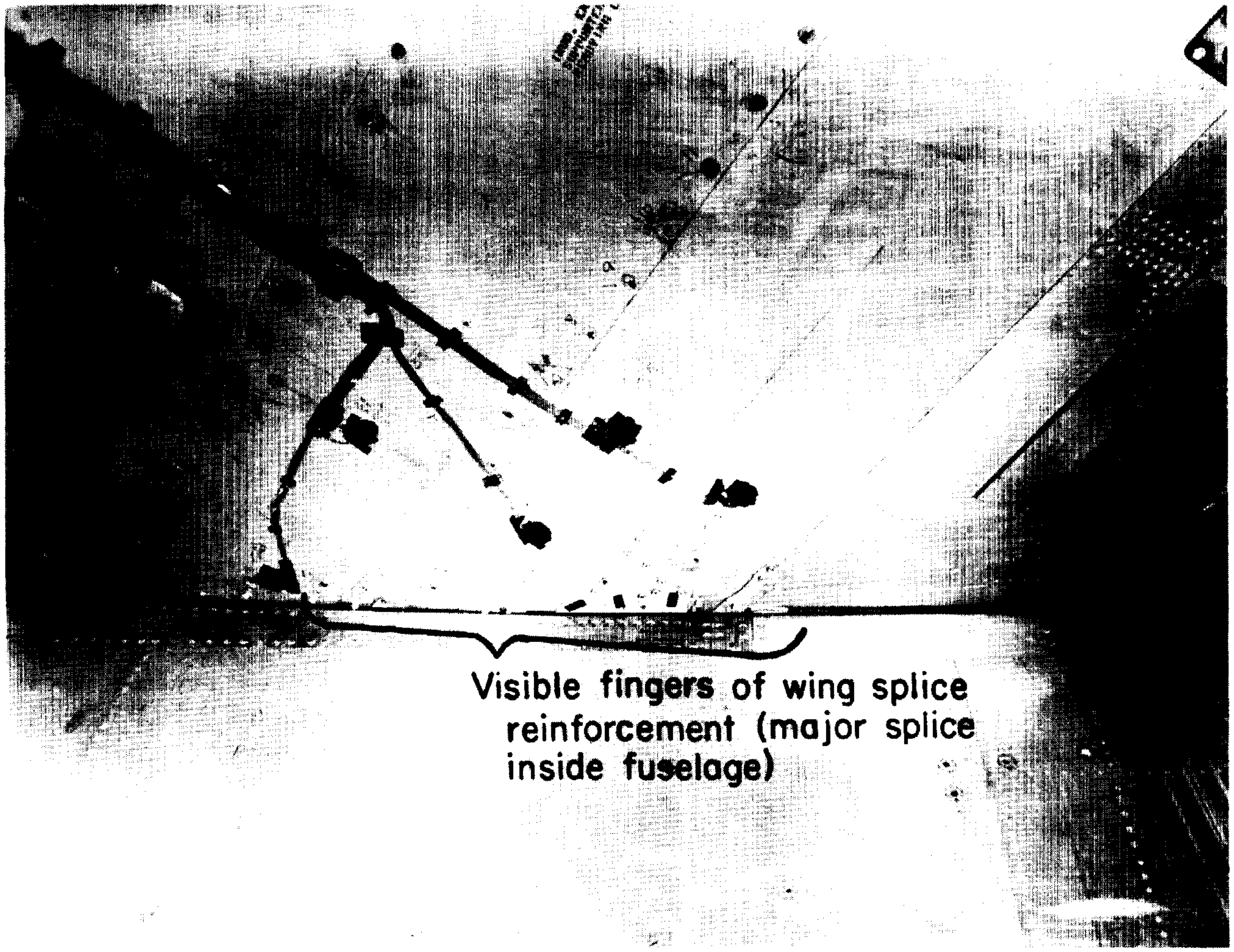
TABLE 6.- SUMMARY OF CRACK OCCURRENCES AND REPAIRS

Operation	Crack occurrence	Repair
Preliminary inspection	1. 1/16-inch vertical crack was found in lower skin in aft countersunk portion of bolt hole 99 at left W.S. 354.	Cracks were removed by reaming 1/32 inch. The wing reinforcement plates were installed as a result of this inspection. Reference 3.
	2. 1/4-inch vertical crack was found in lower splice plate in bolt hole 24 at left W.S. 354.	
Spectrum 1	3. 1/2-inch crack was found at the fillet of right outboard shear tie at B.S. 442.5. See figure 9.	The initiation of this crack occurred before the testing had commenced. No repair was made.
Spectrum 3		The longeron modification was installed on the right and left sides of fuselage during the low gust loading phase of this spectrum. See figure 10.
Spectrum 6	4. 4-inch crack was found at the forward corner of bomb bay section and the adjacent bulkhead at left B.S. 425. See figure 11.	This crack was stop drilled, but the growth continued with the test operation. It was necessary to stop-drill this crack on three additional occasions.
Spectrum 9	5. 1/2-inch crack occurred at the edge of screw hole in the trailing-edge skin at left W.S. 122. See figure 12.	No repair was made.
Spectrum 13	6. 1-inch crack occurred at a rivet hole in right fuselage crown skin at B.S. 601.	Crack was found during the cycling of the low-gust load condition and stop-drilled at each end. See figure 13(a). The reinforcement plate was installed on the area of crack 6 on both sides of fuselage during inspection 9. See figure 13(b).
Inspection 7	7. Four 1/2-inch to 1-inch cracks occurred in bottom skin of fuselage at B.S. 425.	No repair was made.
Inspection 9	8. 1/2-inch crack was found in fillet of the left upper inboard shear tie at B.S. 442.5. See figure 14.	No repair was made.
	9. 1/16-inch crack was found in fillet of the right upper inboard shear tie at B.S. 442.5.	No repair was made.
Spectrum 20	10. 12-inch crack occurred in the fuselage crown skin and the adjacent stiffener at right B.S. 648. See figures 15(a) and 15(b).	The crack occurred suddenly with an audible report during the cycling of the high-gust load condition of this spectrum. A reinforcing plate was added to the area. See figure 15(c).
	11. Twenty 1/2-inch to 6-inch cracks were found in the magnesium skin of the leading edge between W.S. 114 and W.S. 389. Twelve cracks occurred in the left wing and eight occurred in the right wing. See figure 16.	The cracks were stop-drilled after reaching 0.75 inch in length.
Spectrum 23	12. $1\frac{1}{4}$- inch crack occurred in the fuselage skin at the edge of the filler neck opening on right side at B.S. 550. See figure 17(a).	The crack was stop-drilled as a temporary repair. A reinforcement plate was installed to the area during inspection 12. See figure 17(b).
Inspection 12	13. 0.1-inch and 0.2-inch cracks were found at the Jo-bolt holes in the fuselage skin reinforcing plate at left B.S. 560 and right B.S. 567, respectively. See figure 18.	No repair necessary. Additional cracks occurred in this same region during the remainder of the test.
Inspection 13	14. 1/4-inch crack found at rivet hole in left fuselage crown skin at B.S. 600.	No repair was made.
	15. 1/4-inch crack found in the outboard finger of the reinforcing plate at bolt hole 406 at left B.L. 45. See figure 19(a).	The hole in the finger was enlarged to include the crack and hole left open. See figure 19(b).
Inspection 14	16. 1/4-inch crack was found in the outboard finger of the reinforcing plate at bolt hole 410 at left B.L. 45.	No repair was made.
	17. 1-inch crack was found in the drag angle at bolt hole 77 at left B.L. 45.	No repair was made.
	18. 1/2-inch crack was found in the drag angle at bolt hole 77 at right B.L. 45.	No repair was made.
Spectrum 29	19. 21-inch crack occurred in the right wing closeout panel between stringers 7 and 8 at W.S. 179. See figure 20.	This crack occurred suddenly with a loud report at the 90-percent-limit-load level. Testing terminated at this point.
Inspection 15	20. 1/8-inch crack was found in the outboard finger of the reinforcing plate at bolt hole 406 at right B.L. 45.	
	21. 0.015-inch crack was found in the outboard finger of the reinforcing plate at bolt hole 407 at right B.L. 45.	
	22. 3.5-inch crack was found in the left wing closeout panel at W.S. 212. See figure 21.	This crack apparently originated in a tool mark in the radius of the cutout.



(a) Wing station 354. L-58-1309a.1

Figure 1.- Wing-splice reinforcements on lower surface.



(b) Outboard of buttock line 45. L-58-1313a.1

Figure 1.- Continued.

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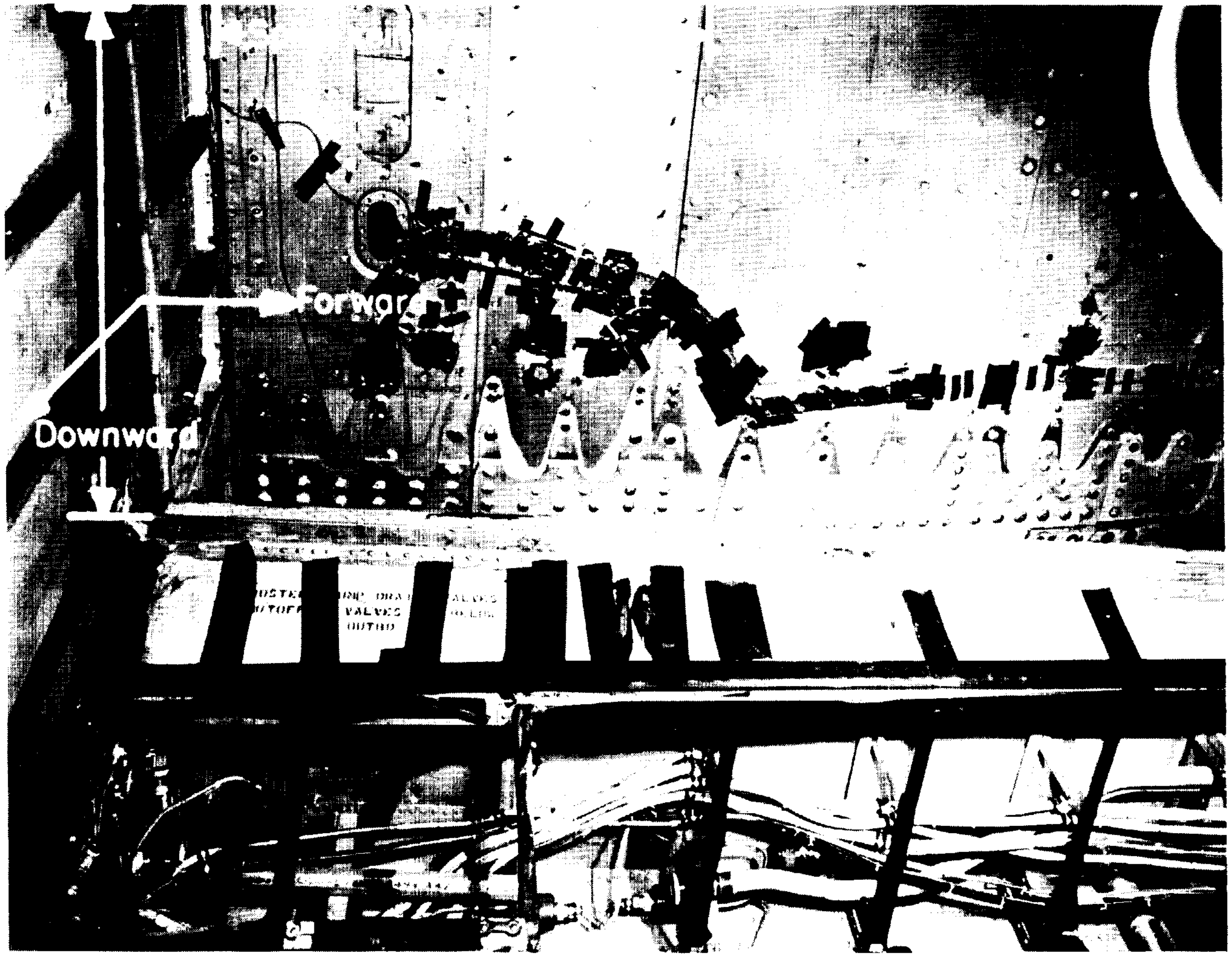

(c) Inboard of buttock line 45. L-58-1311a.1

Figure 1.- Concluded.

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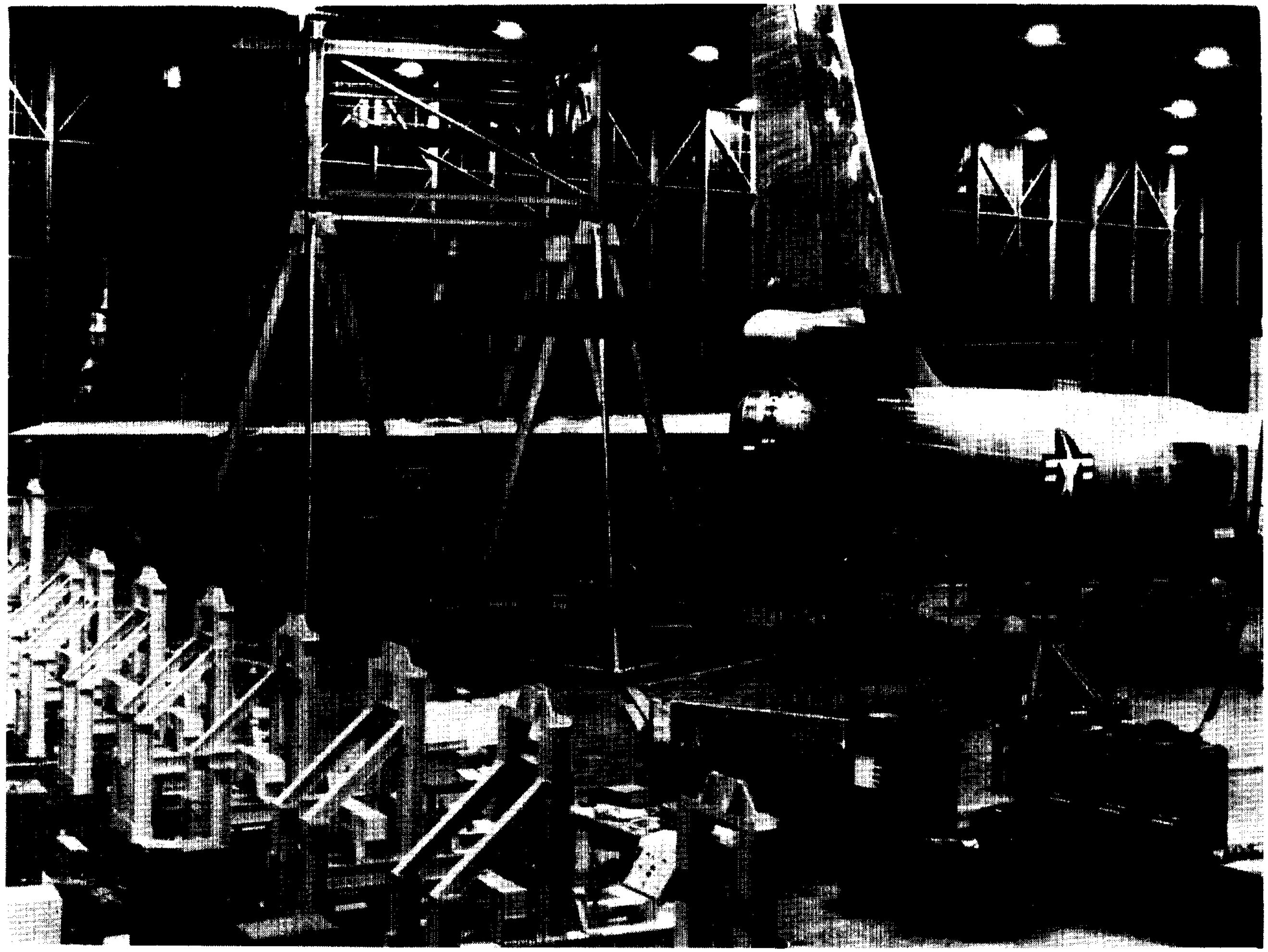
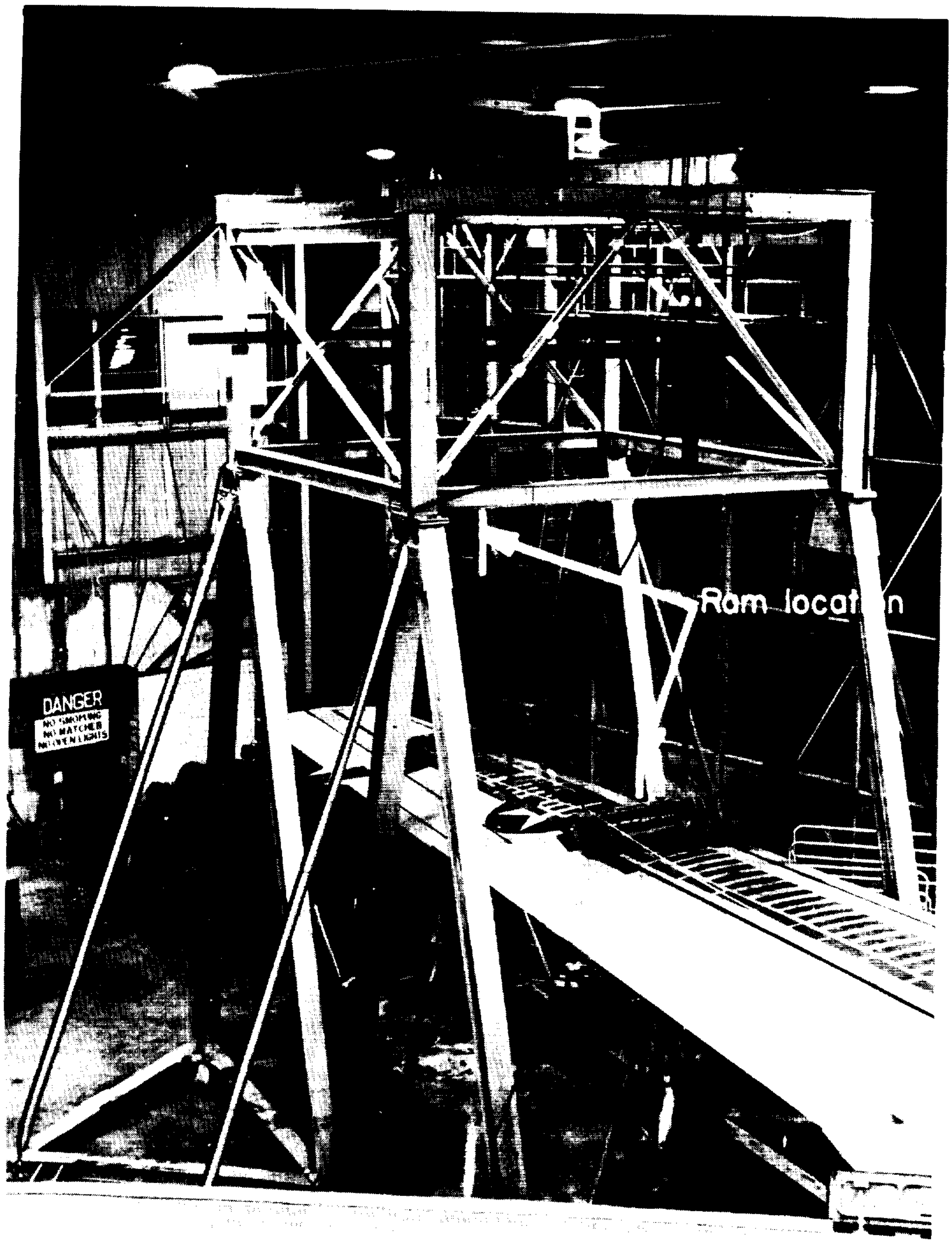


(a) General view of airplane mounted for testing. L-59-173

Figure 2.- Views of test setup.

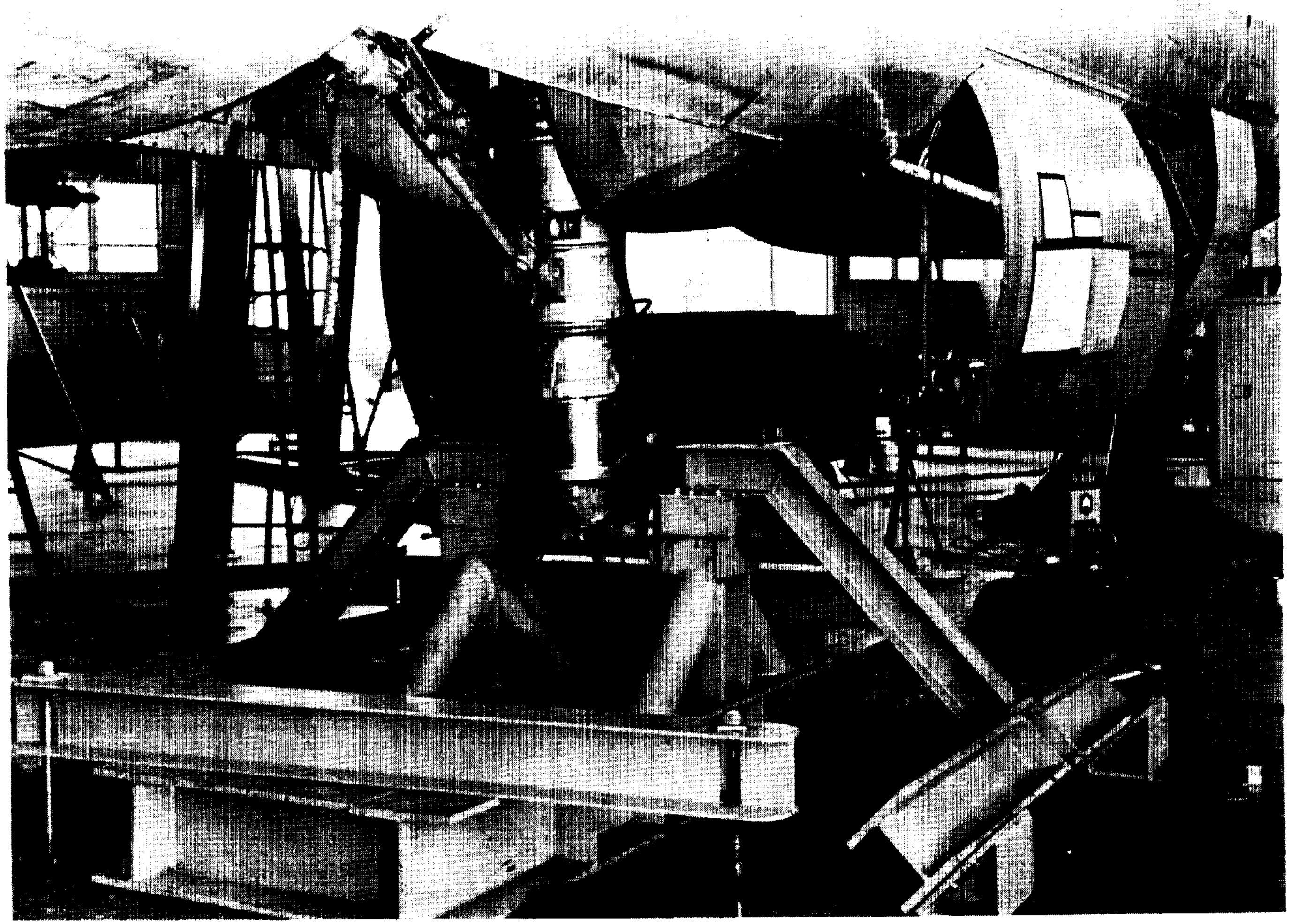
[redacted]



(b) View of outboard loading towers. L-59-174

Figure 2.- Concluded.

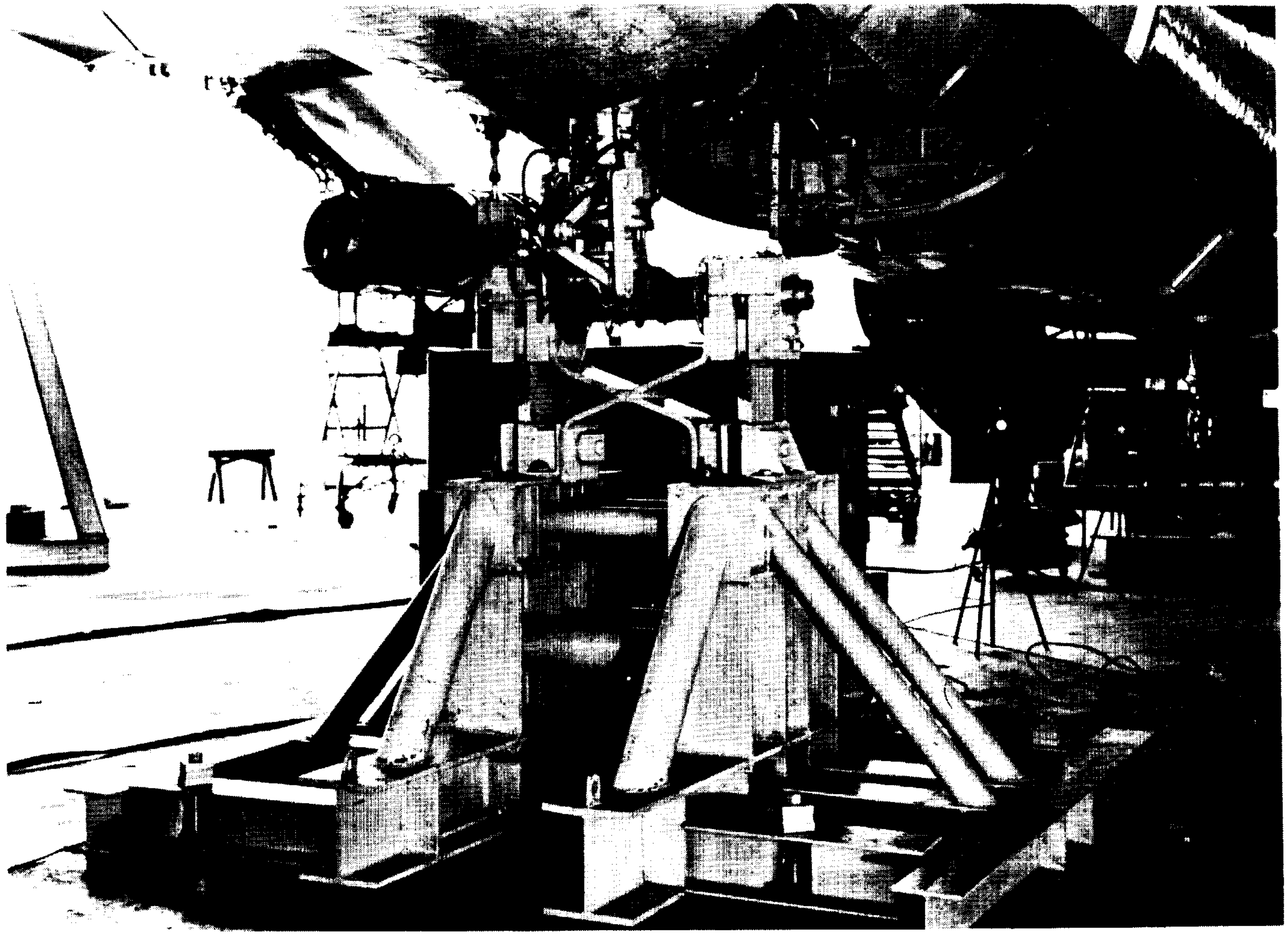
[redacted]



(a) Front tie-down. L-58-3664

Figure 3.- Landing-gear tie-downs.

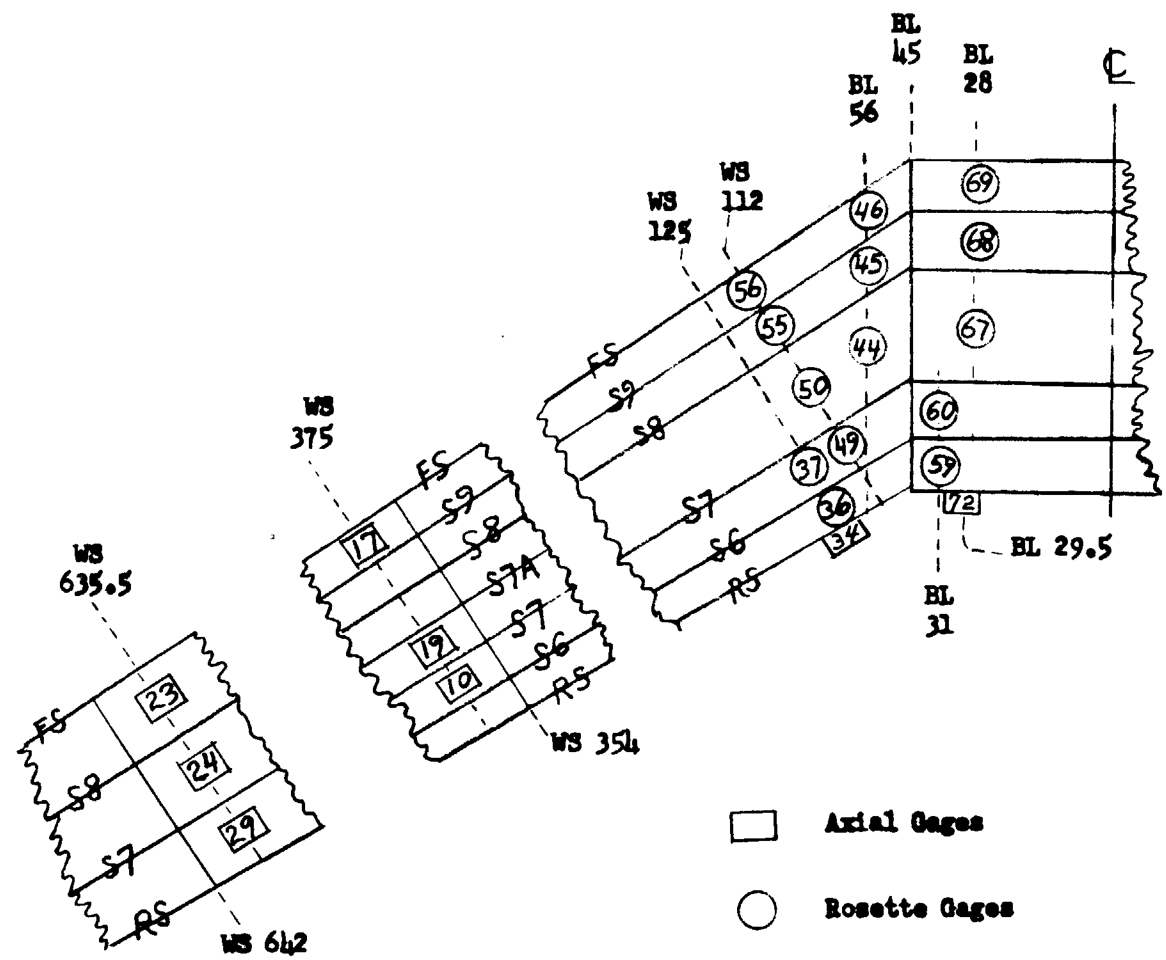
[redacted]



(b) Rear tie-down. L-58-3663

Figure 3.- Concluded.

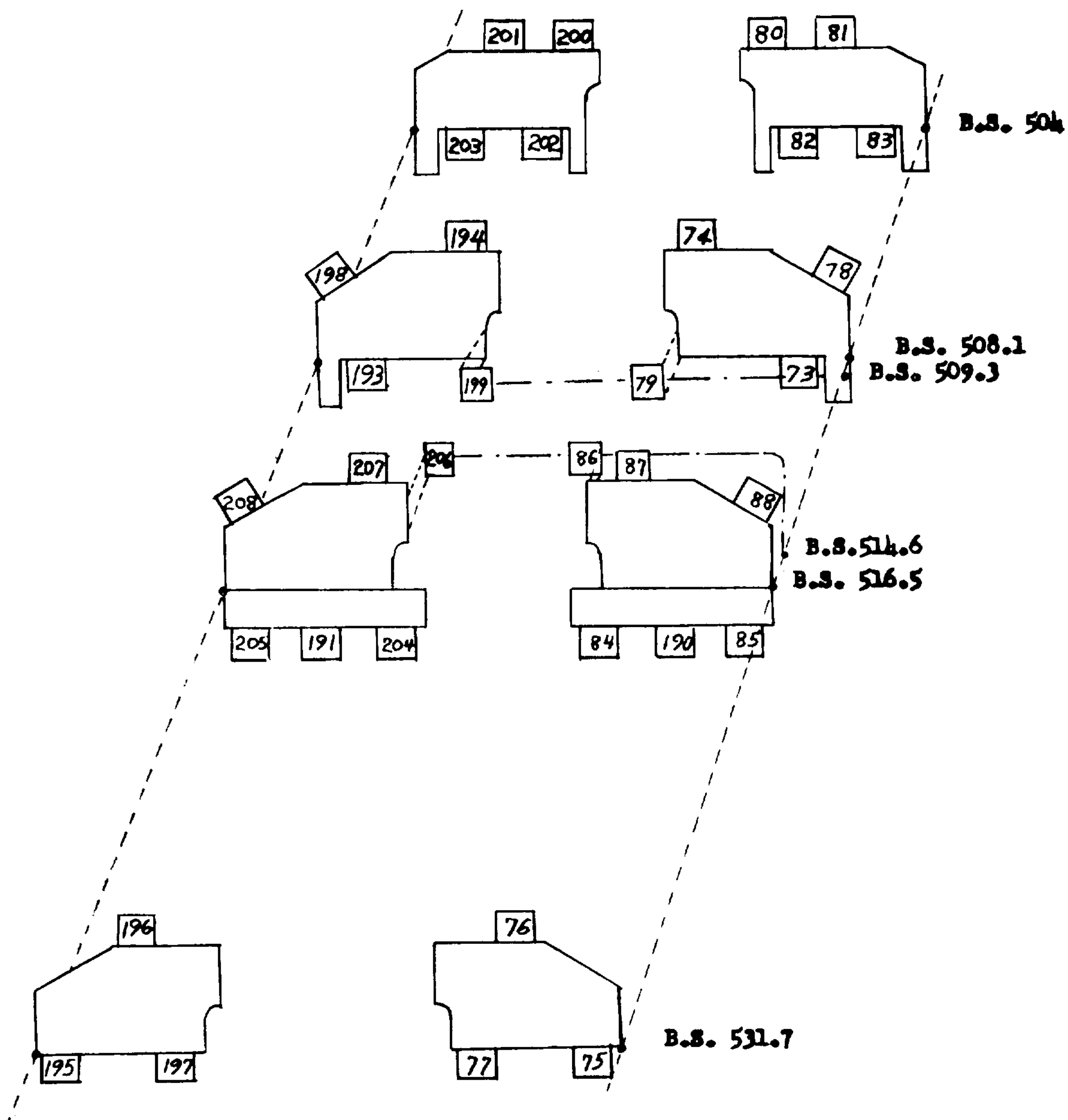
[redacted]



(a) Left wing.

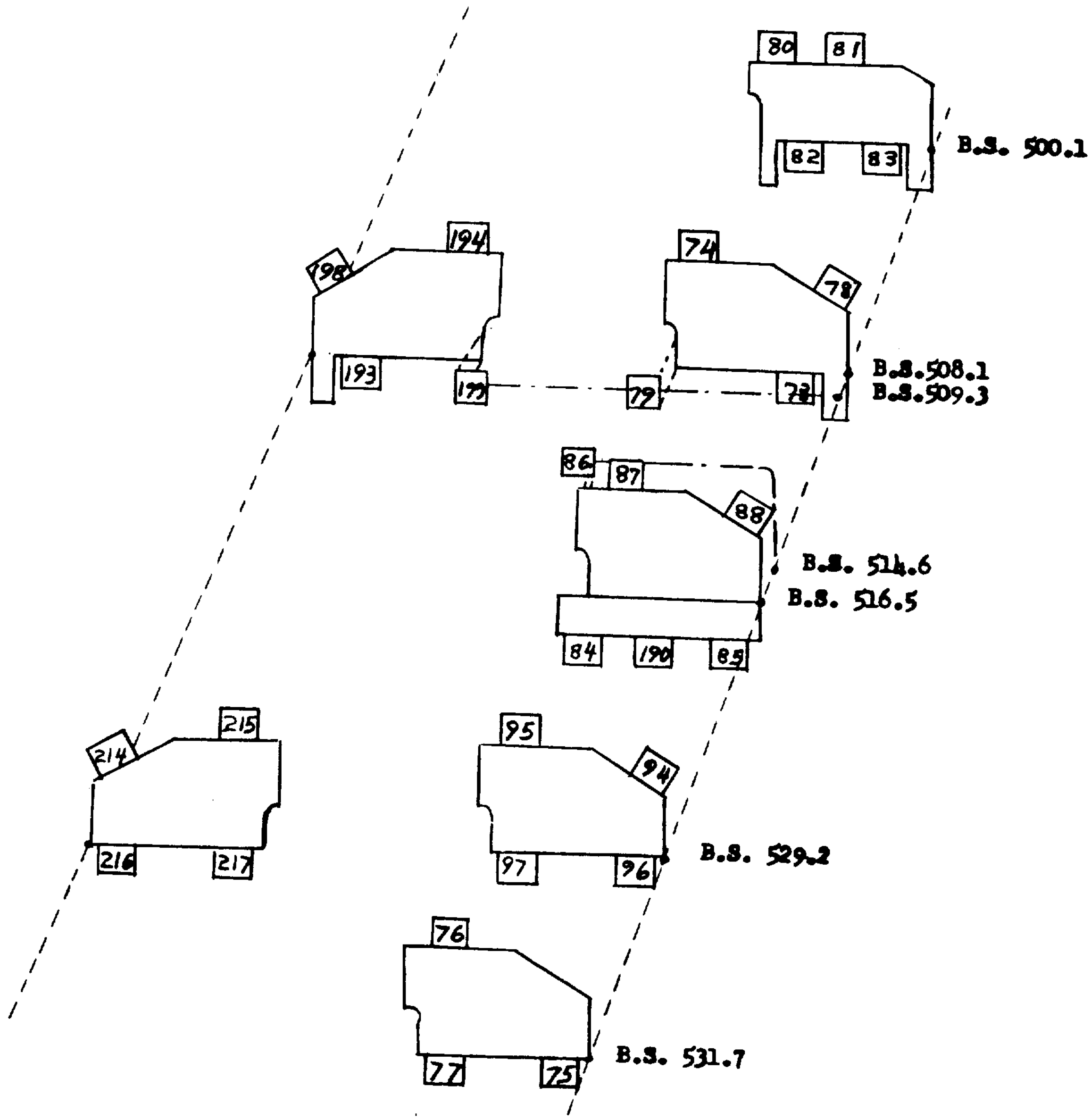
Figure 4.- Pictorial representation of strain-gage locations identifying stringers between front and rear spars.

[redacted]



(b) Longeron prior to modification.

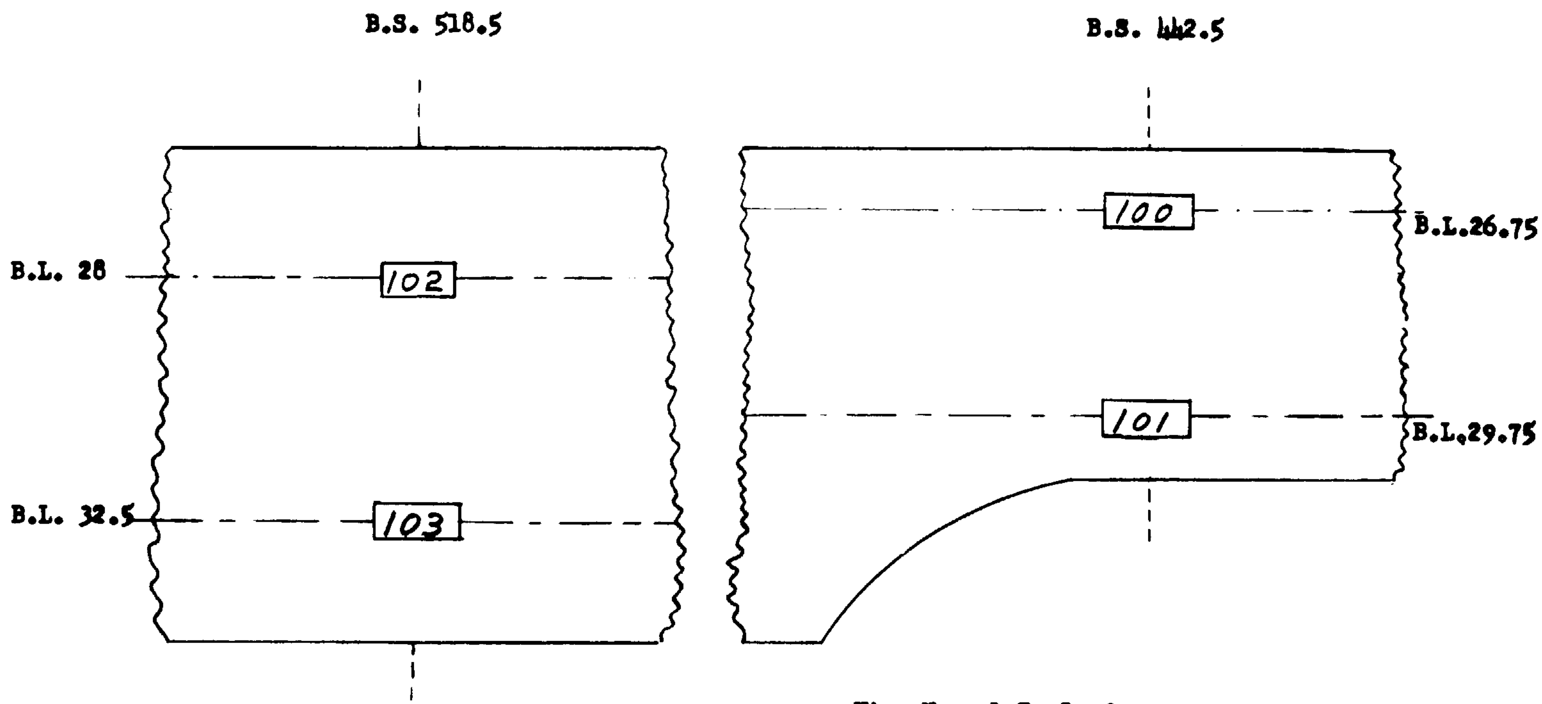
Figure 4.- Continued.



(c) Longeron after modification.

Figure 4.- Continued.

[redacted]



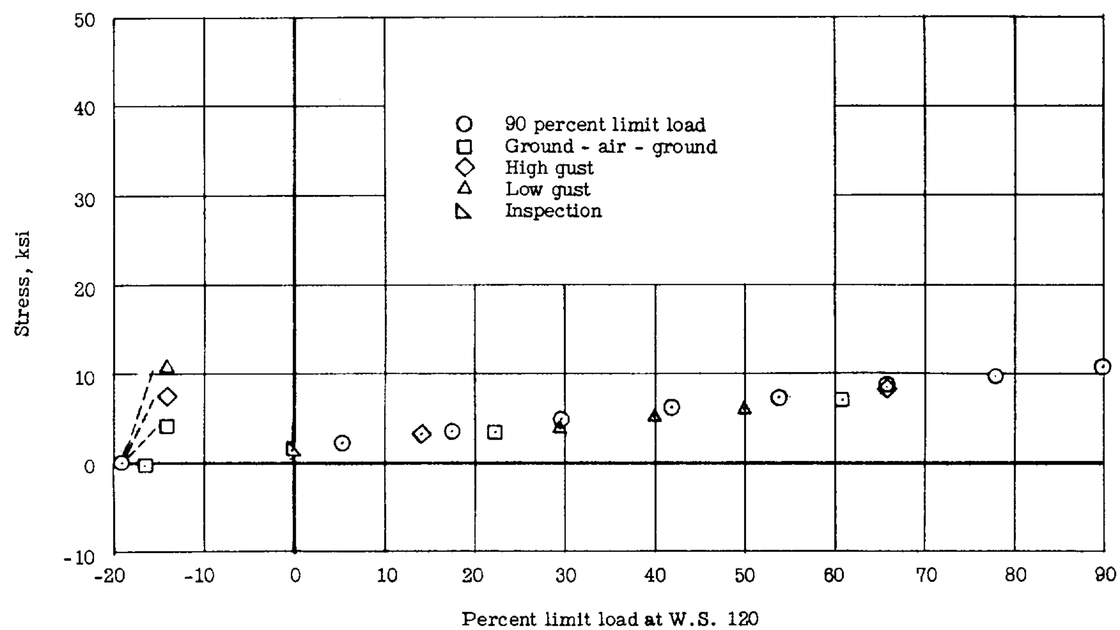
View Normal To Surface, Right Side

(d) Crown-skin reinforcing plate.

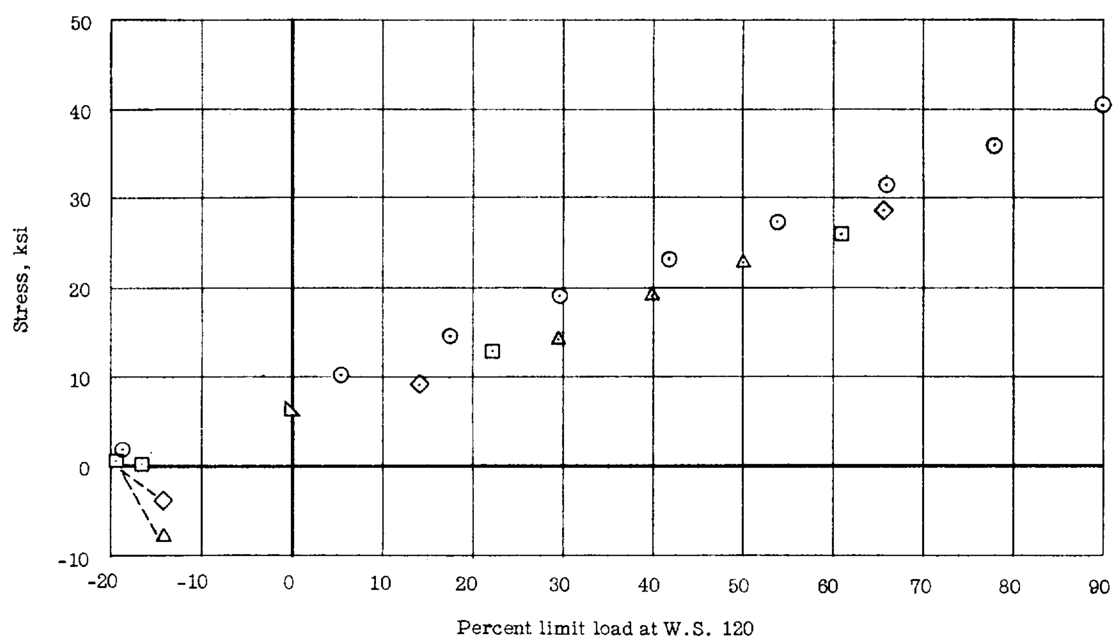
Figure 4.- Concluded.

[redacted]

[redacted]



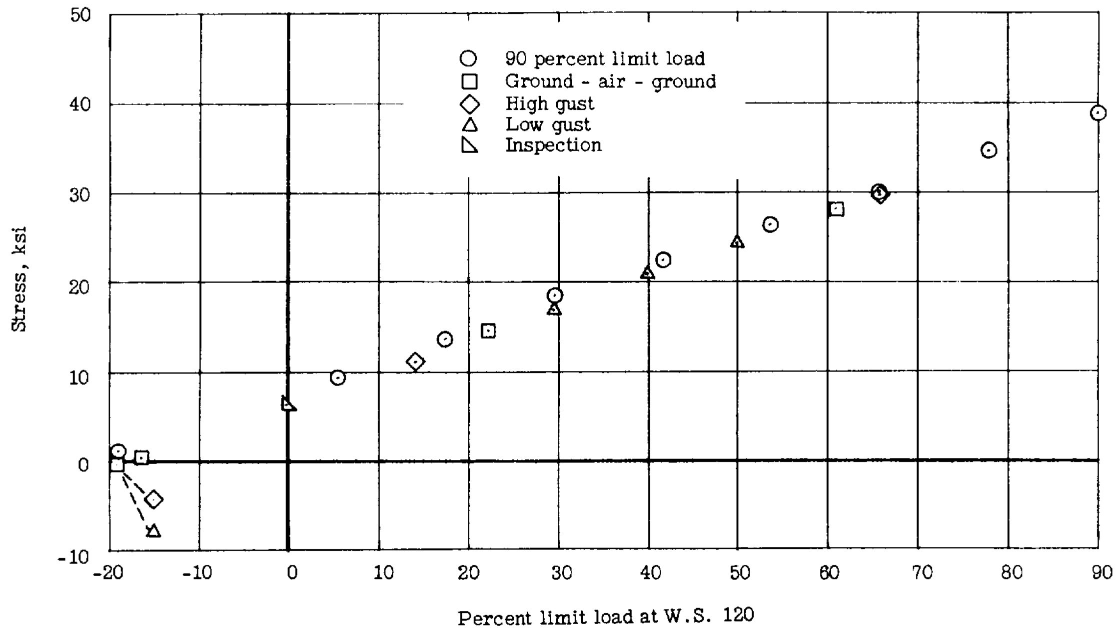
(a) Gage 23; wing station 635.5.



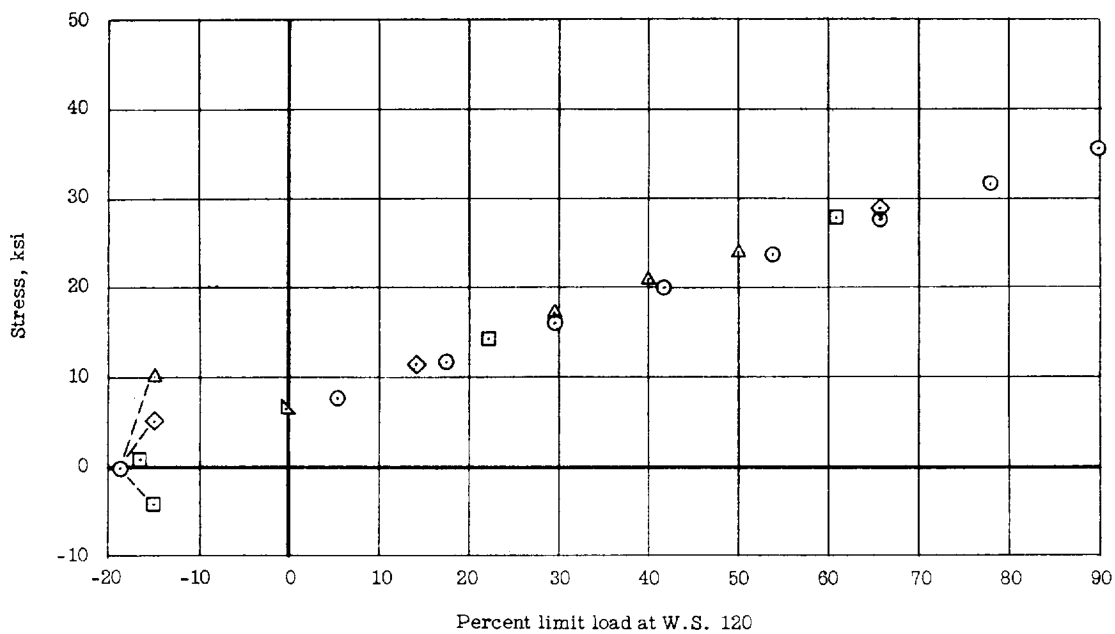
(b) Gage 10; wing station 375.

Figure 5.- Wing stress measurements following wing-splice reinforcement.

[redacted]



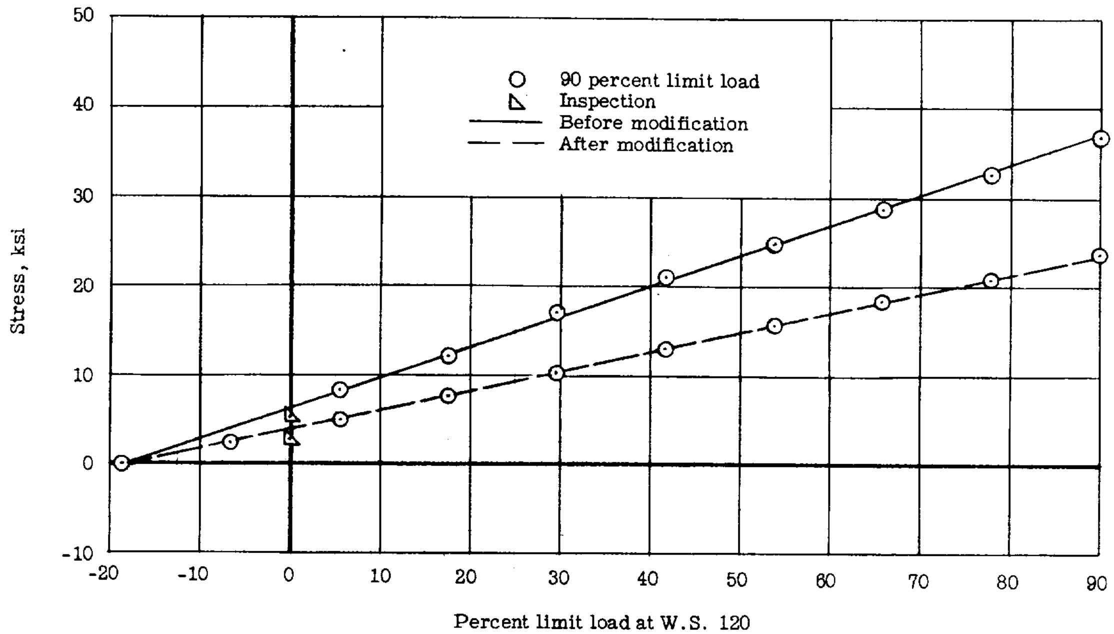
(c) Gage 49; wing station 112.



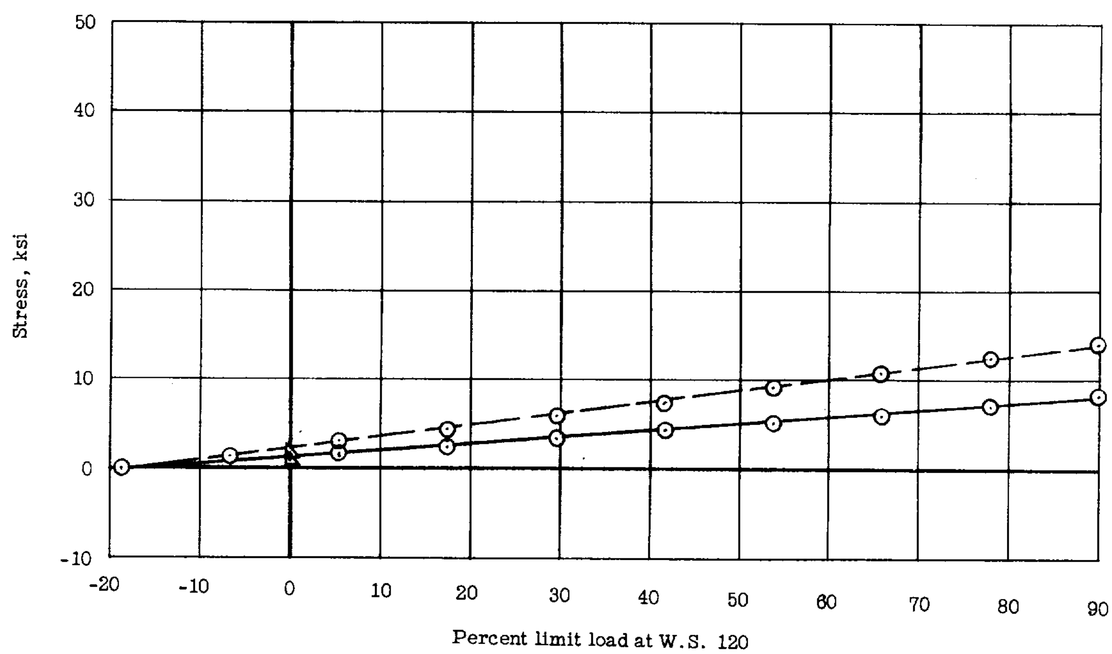
(d) Gage 60; buttock line 31.

Figure 5.- Concluded.

[redacted]



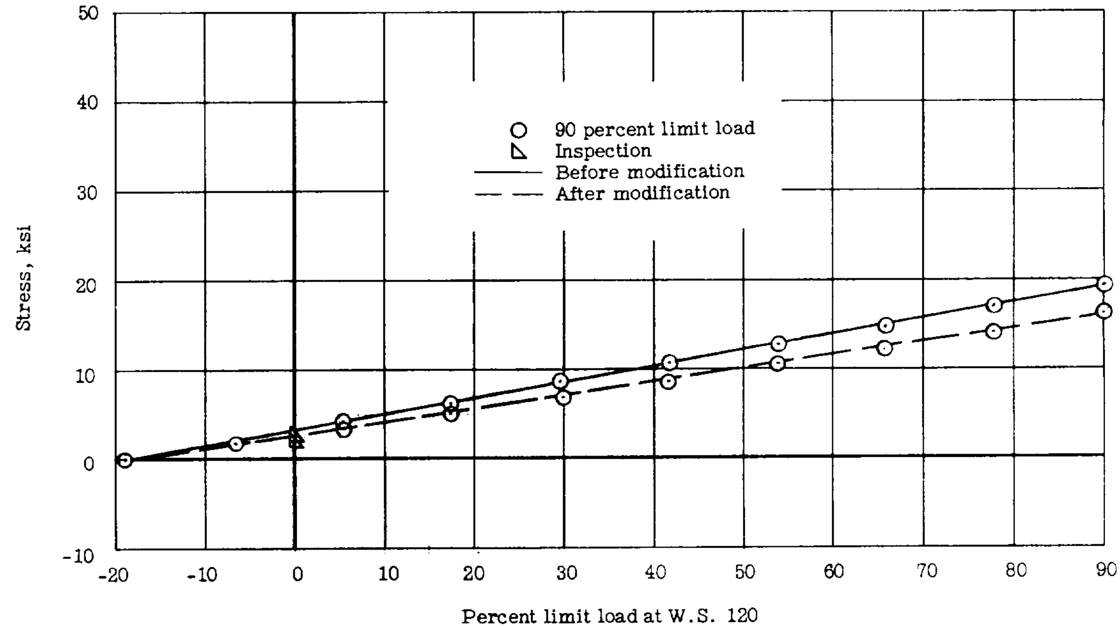
(a) Gage 73.



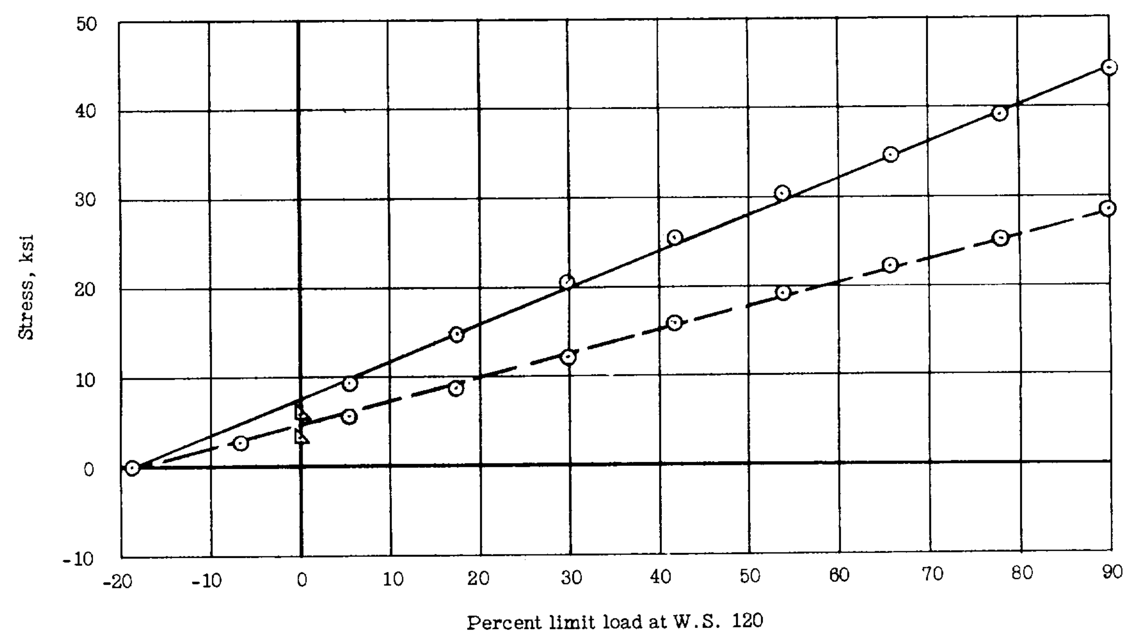
(b) Gage 74.

Figure 6.- Longeron stress measurements before and after repair at body station 508.

[redacted]



(c) Gage 78.



(d) Gage 79.

Figure 6.- Concluded.

[redacted]

[redacted]

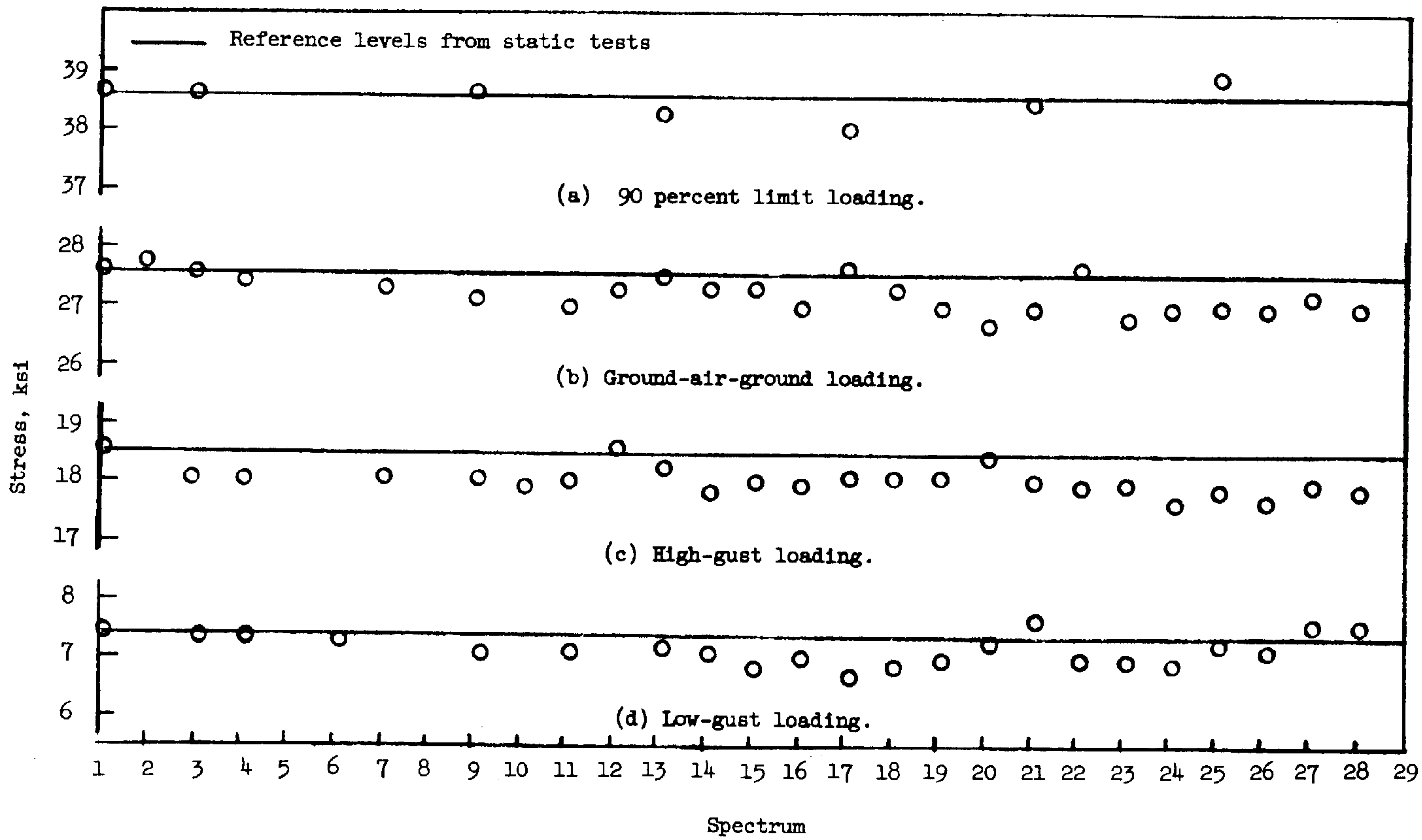
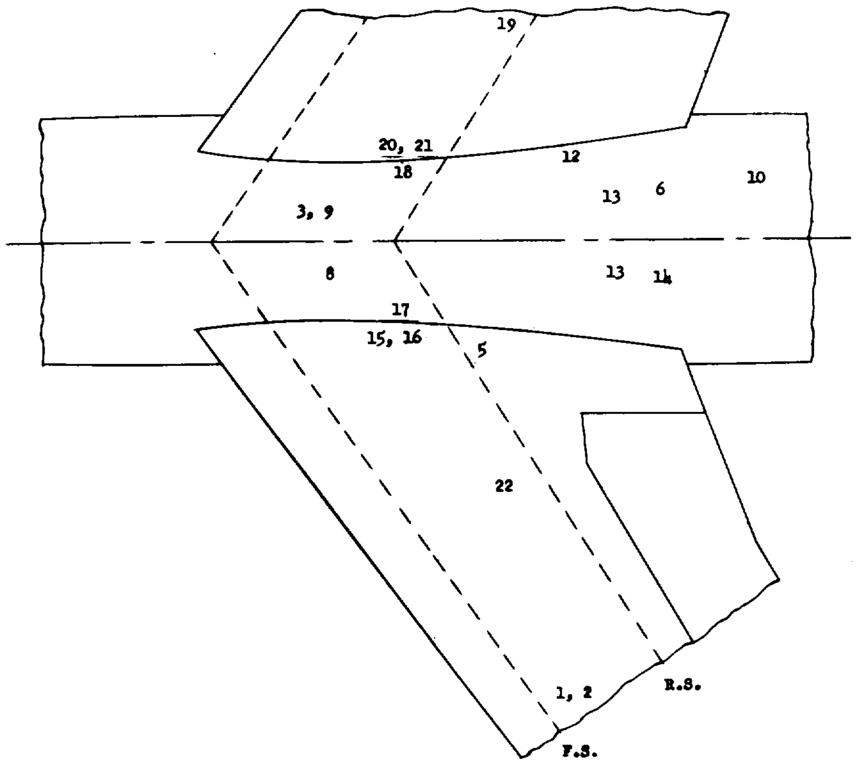
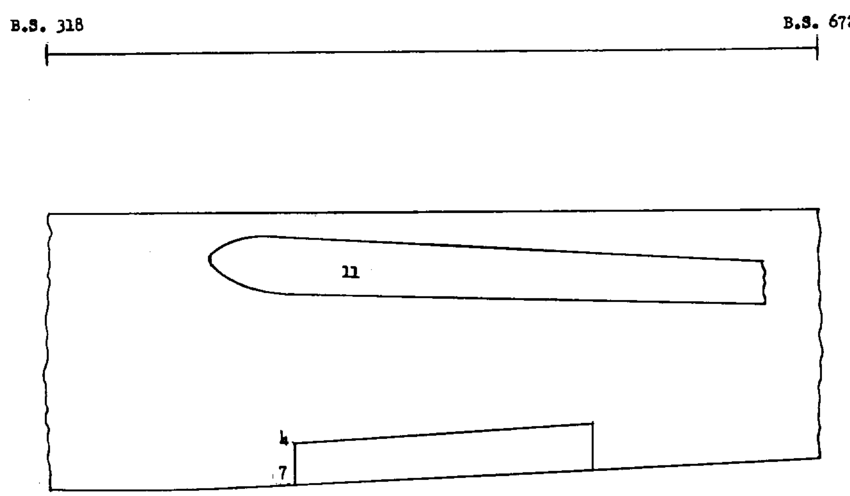


Figure 7.- Spot checks of incremental (difference between maximum and minimum) stress of a representative strain gage during cyclic loading. Gage 37-3 at wing station 125.

[redacted]



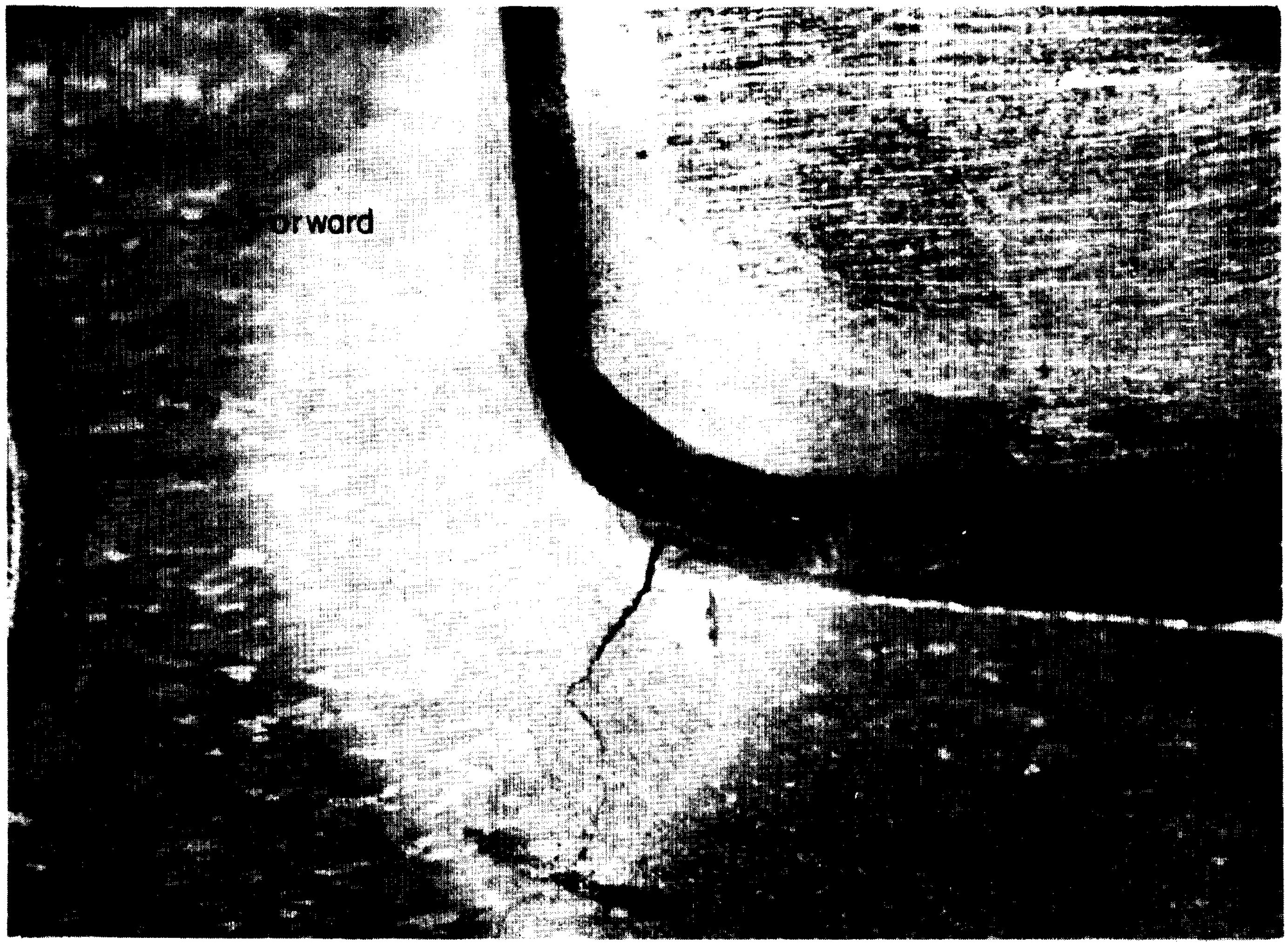
(a) Top view of wing center section.



(b) Left-side view of fuselage near wing root.

Figure 8.- Location of crack areas. Refer to table 6 for detailed descriptions of crack areas.

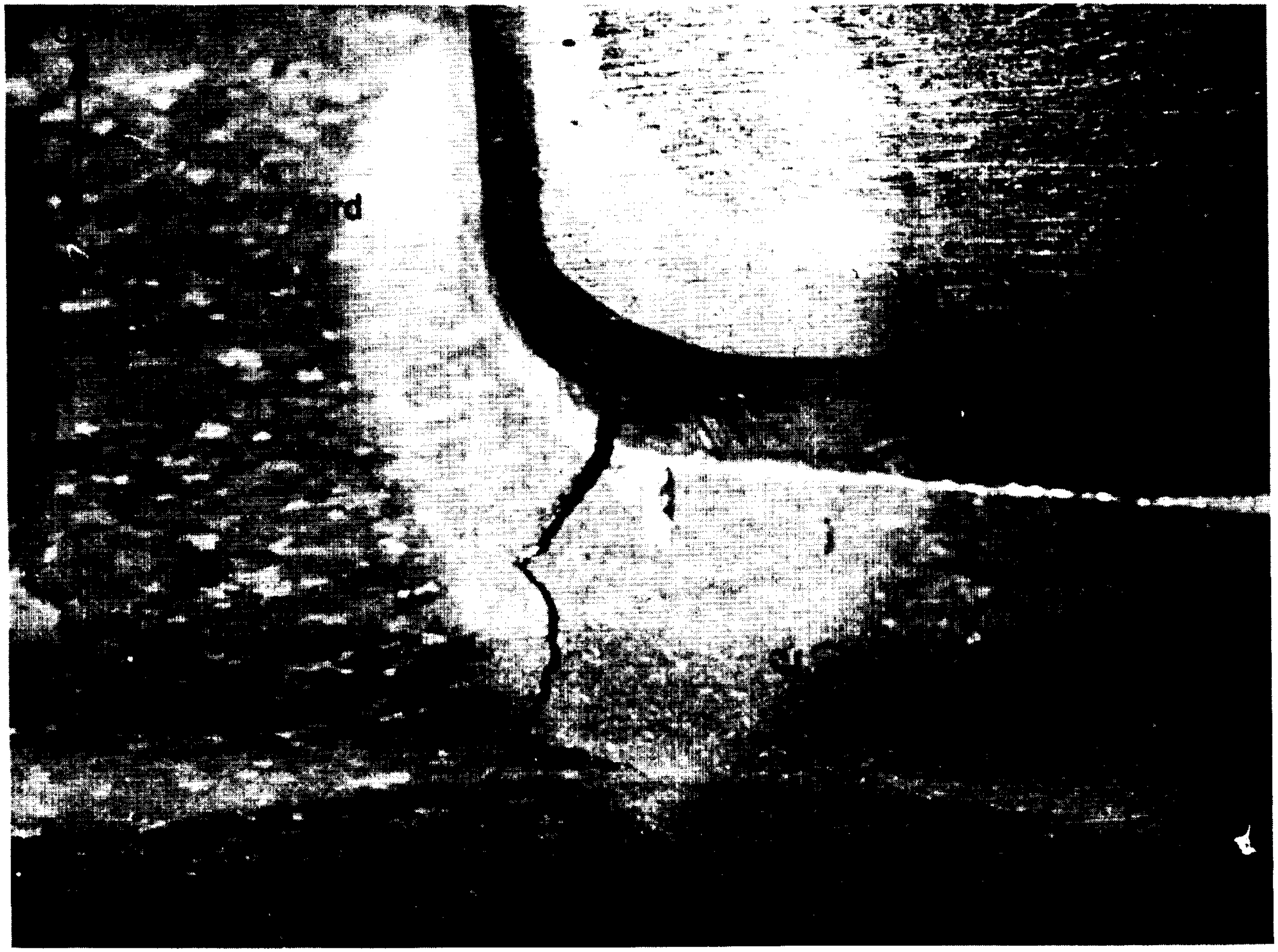
[redacted]



(a) Load at high-gust minimum. L-59-175

Figure 9.- Crack in fillet of right outboard upper shear tie at body station 442.5.

[redacted]



(b) Load at high-gust maximum. L-59-176

Figure 9.- Concluded.

[redacted]

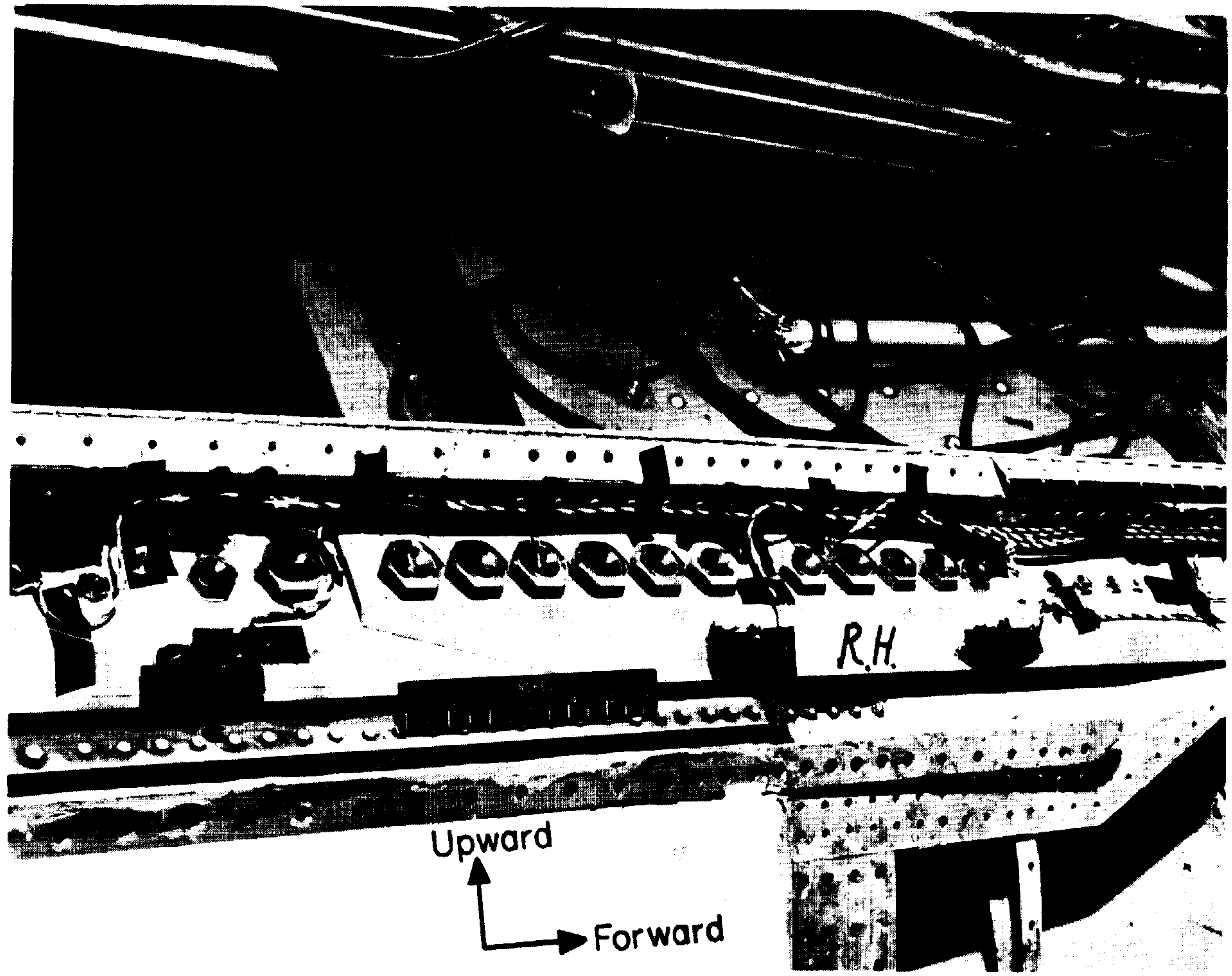
[redacted]

L-59-177
(a) Fuselage crown-skin reinforcing plate. Body station 425 to body station 591.

Figure 10.- Upper longeron modification.

[redacted]

[redacted]



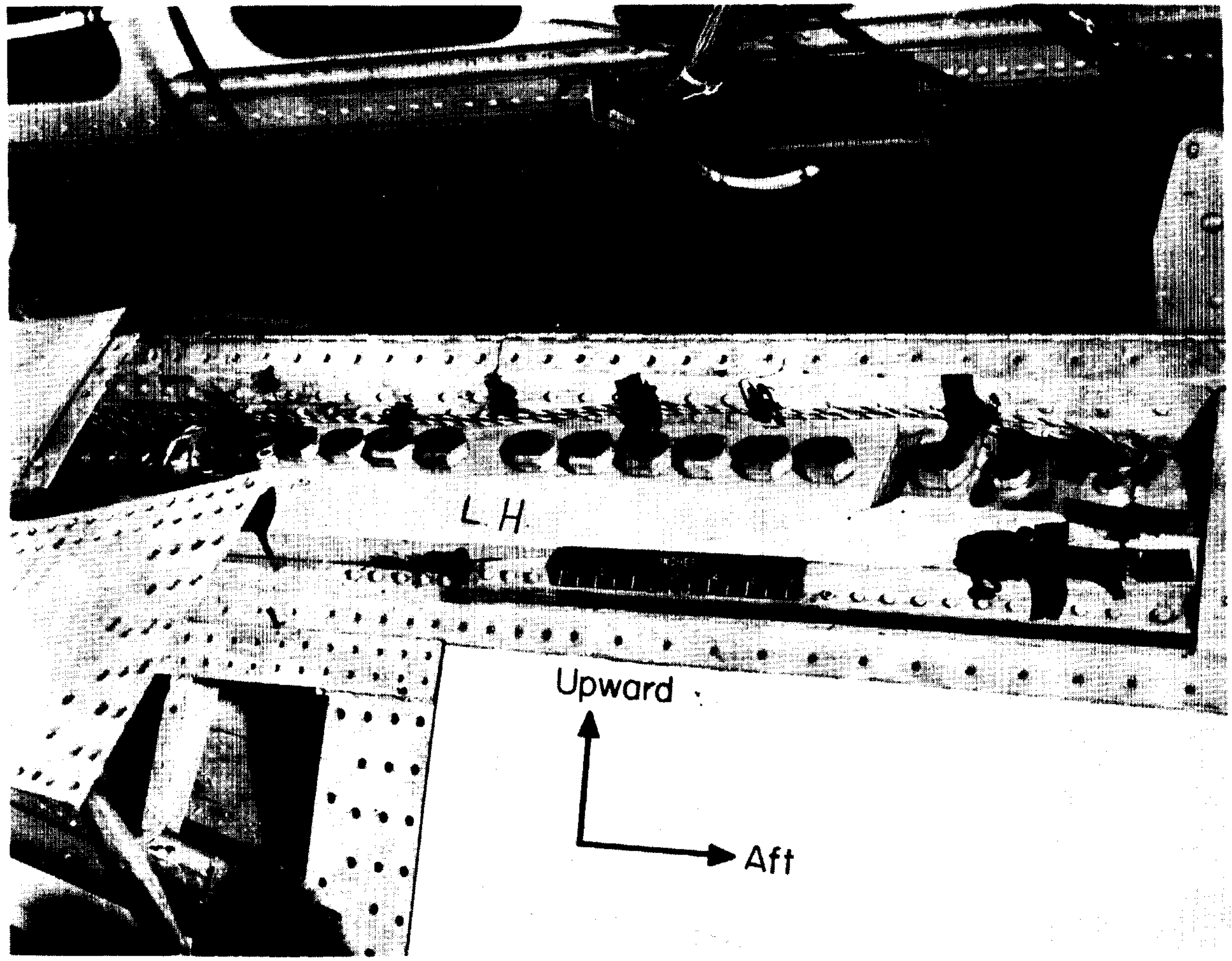
L-58-220a.1

(b) Right-longeron splice after modification showing strain gages.

Figure 10.- Continued.

[redacted]

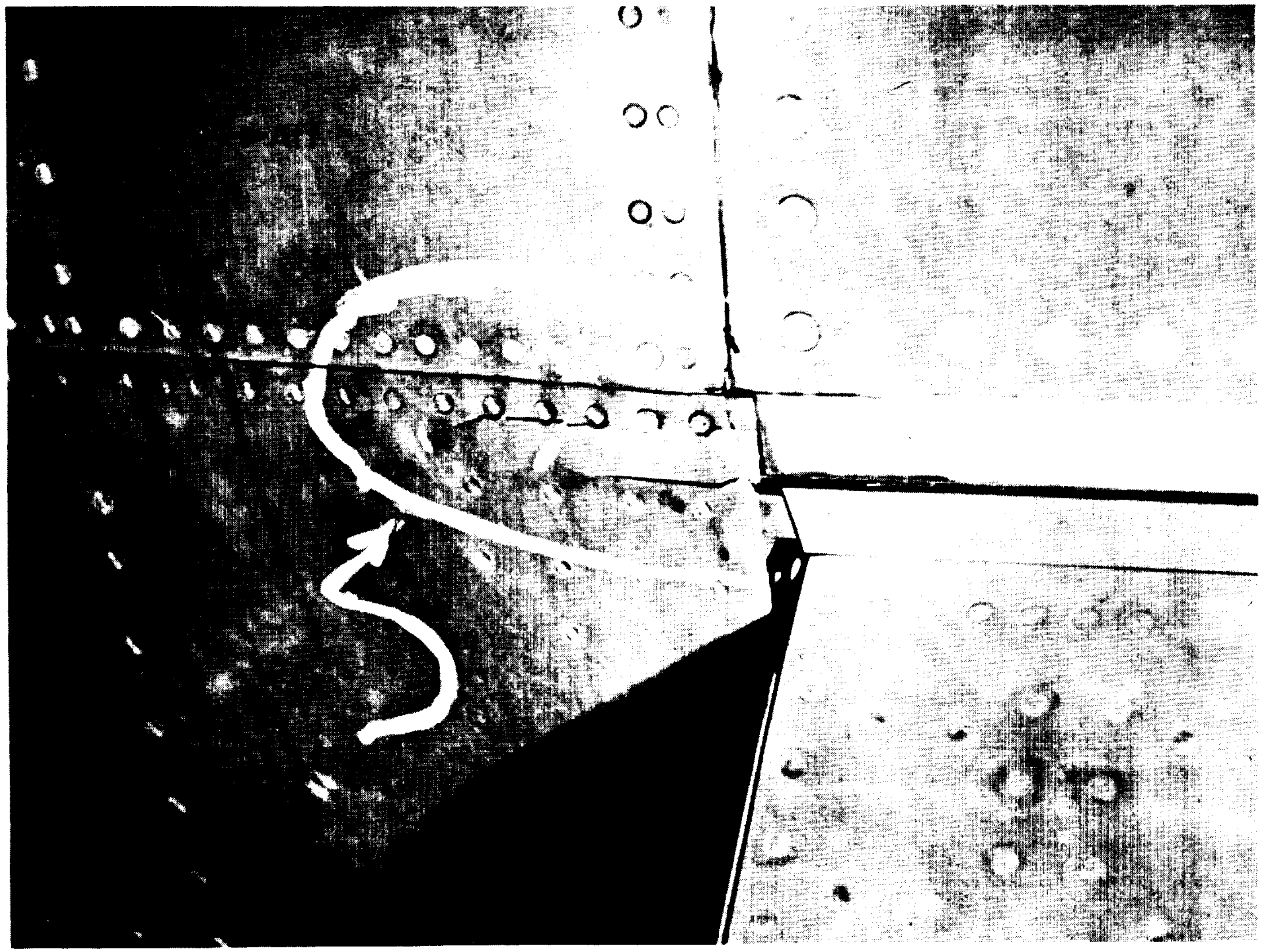
[redacted]



L-58-221a.1
(c) Left-longeron splice after modification showing strain gages.

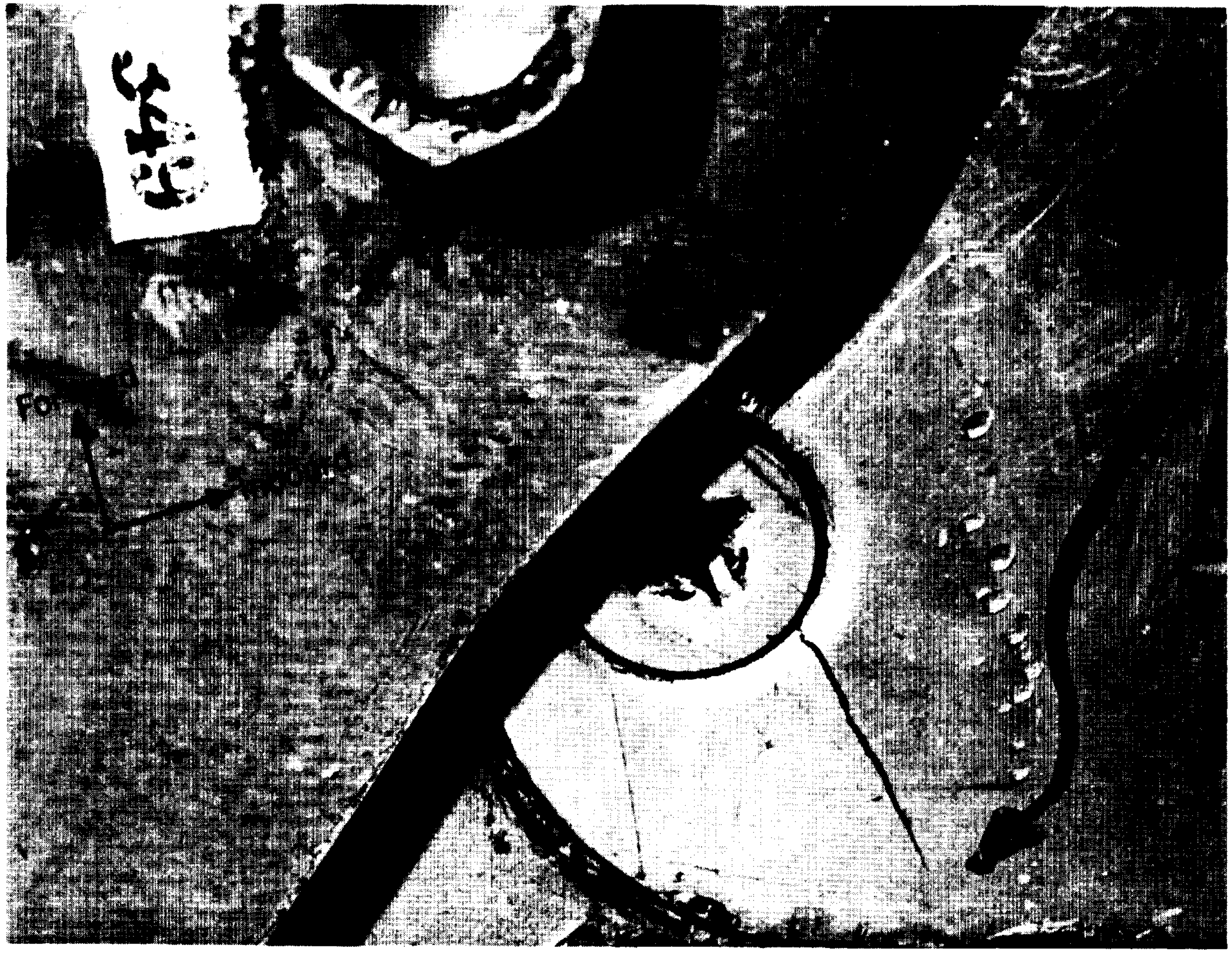
Figure 10.- Concluded.

[redacted]



L-59-178
Figure 11.- Cracks in lower left fuselage skin forward of bomb bay door. Body station 425.

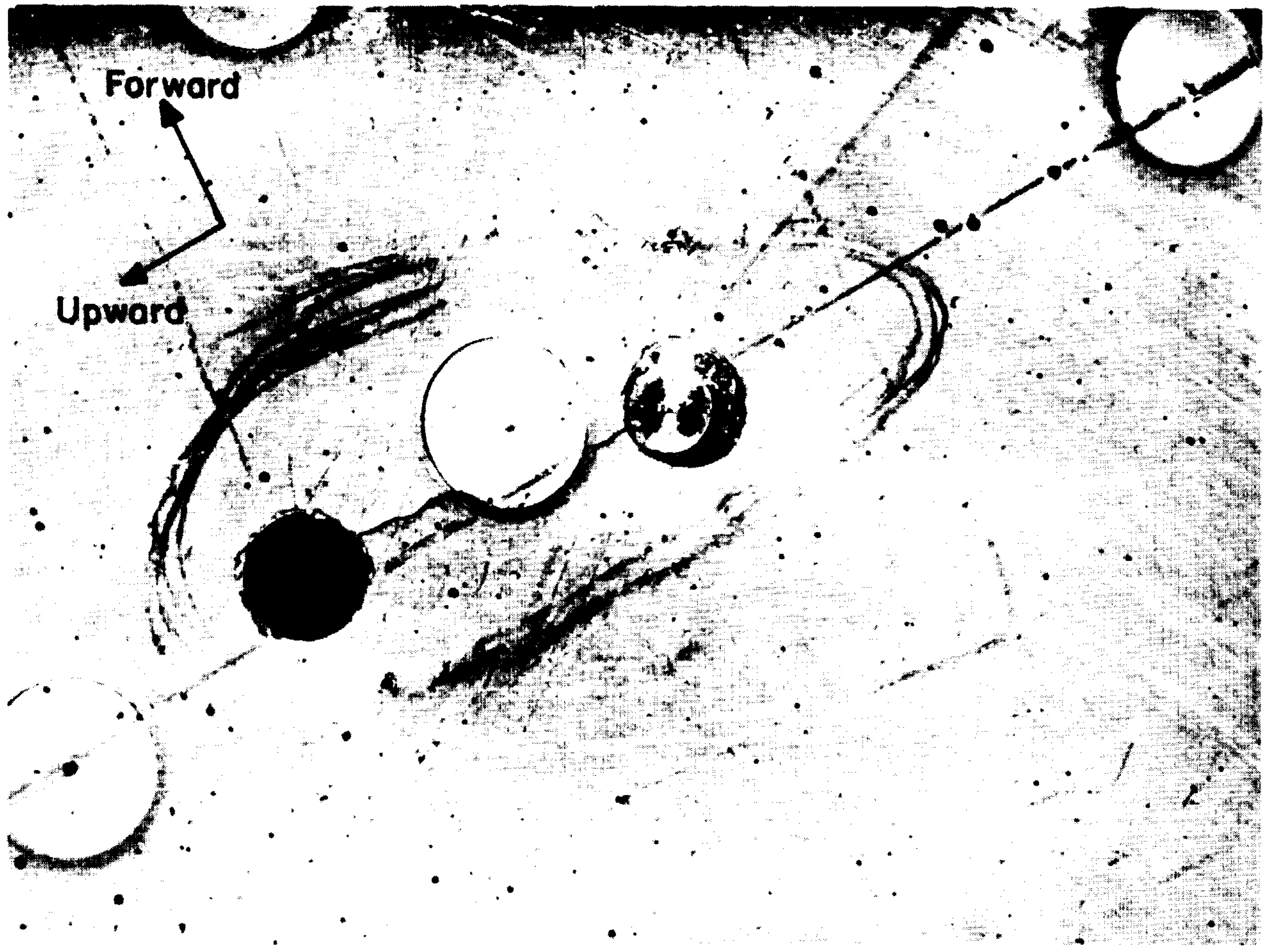
[redacted]



L-59-179

Figure 12.- Crack in left trailing-edge skin just aft of hole 401. Photograph made at low-gust maximum, spectrum 9.

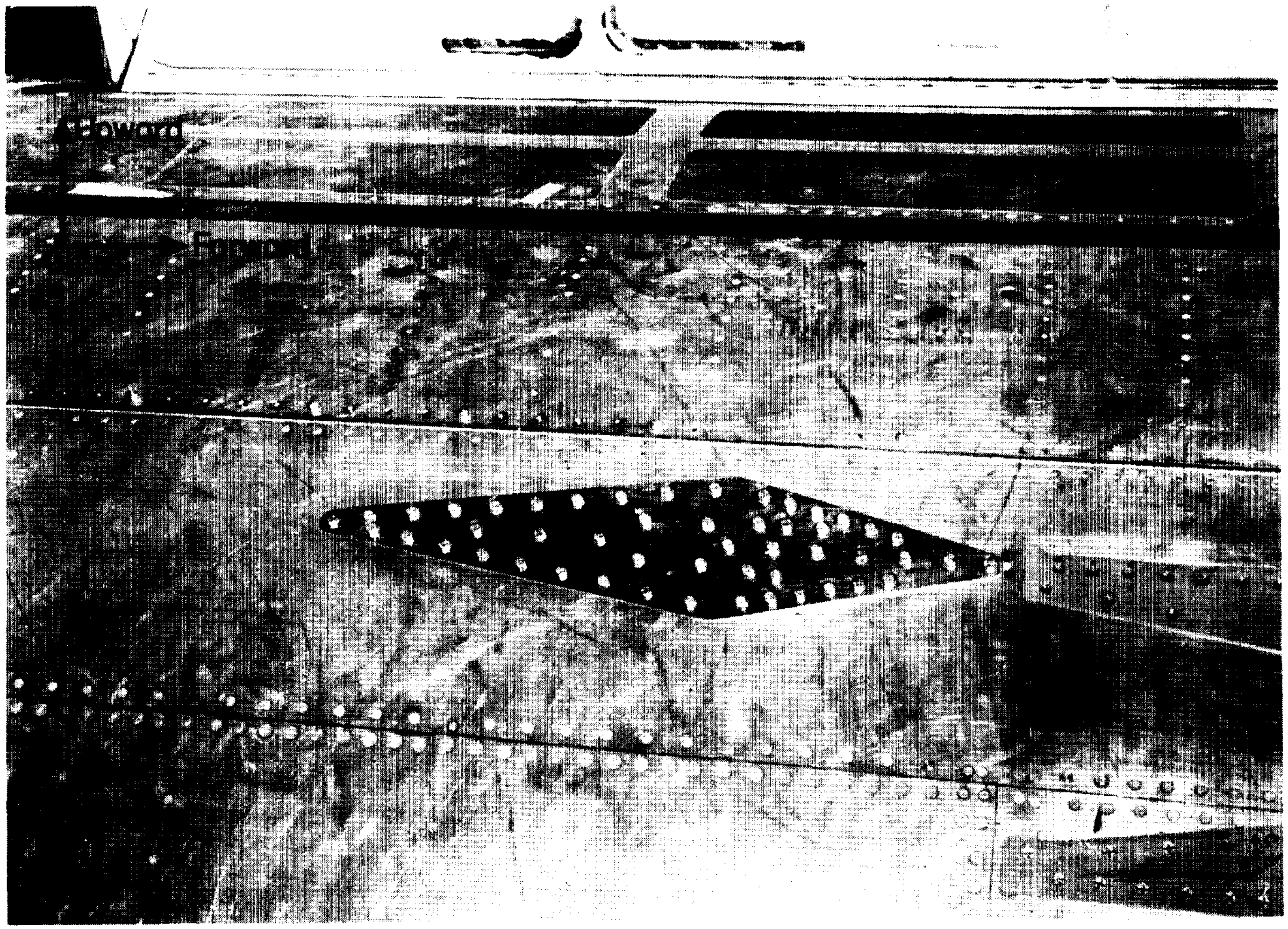
[redacted]



(a) Crack at spectrum 13, as stop-drilled. L-59-180

Figure 13.- Crack and repair in right fuselage crown skin at body station 601, 10 inches aft of reinforcing plate.

[redacted]

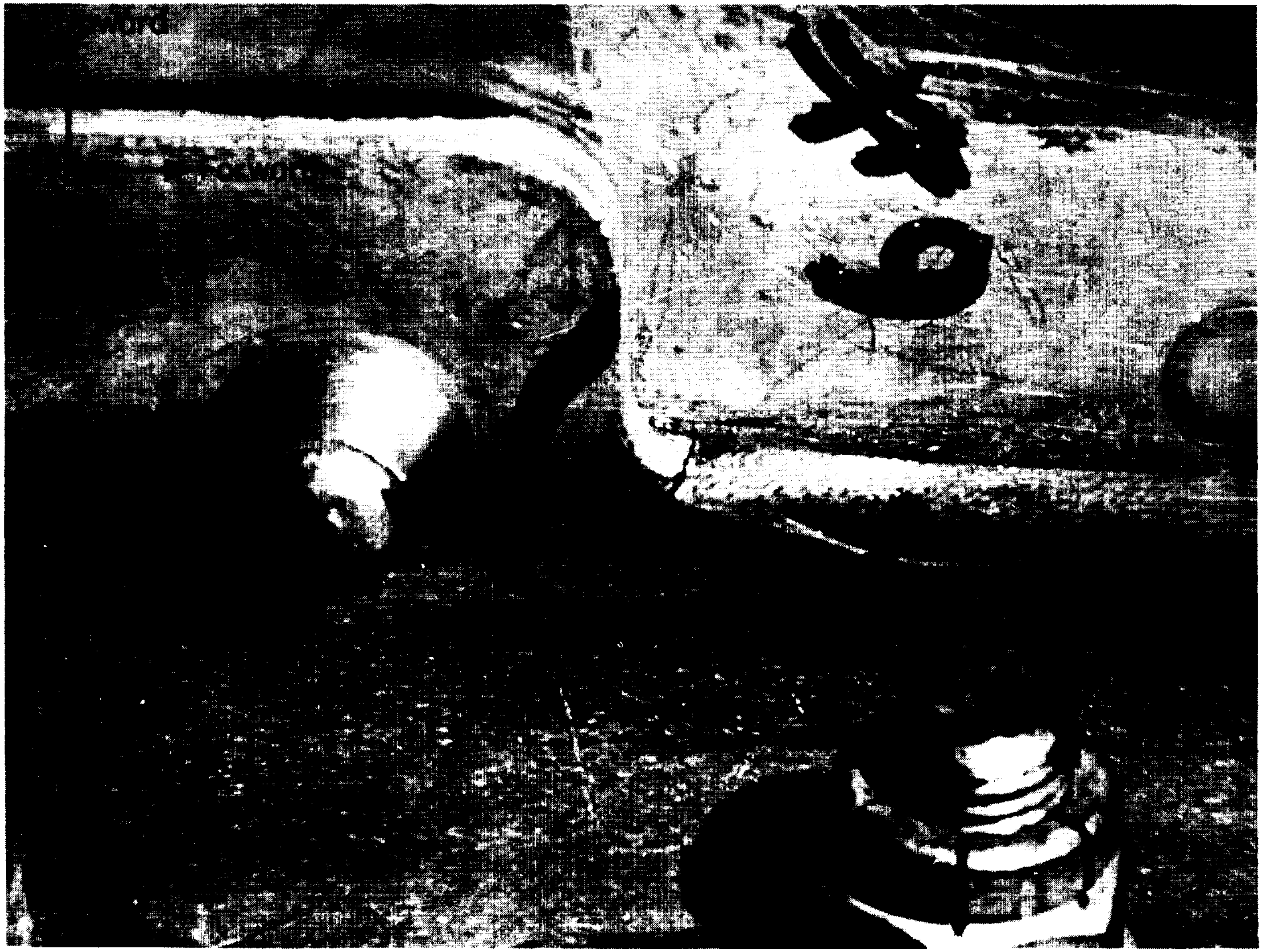


(b) Reinforcement plate installed during inspection 9. L-59-55.1

Figure 13.- Concluded.

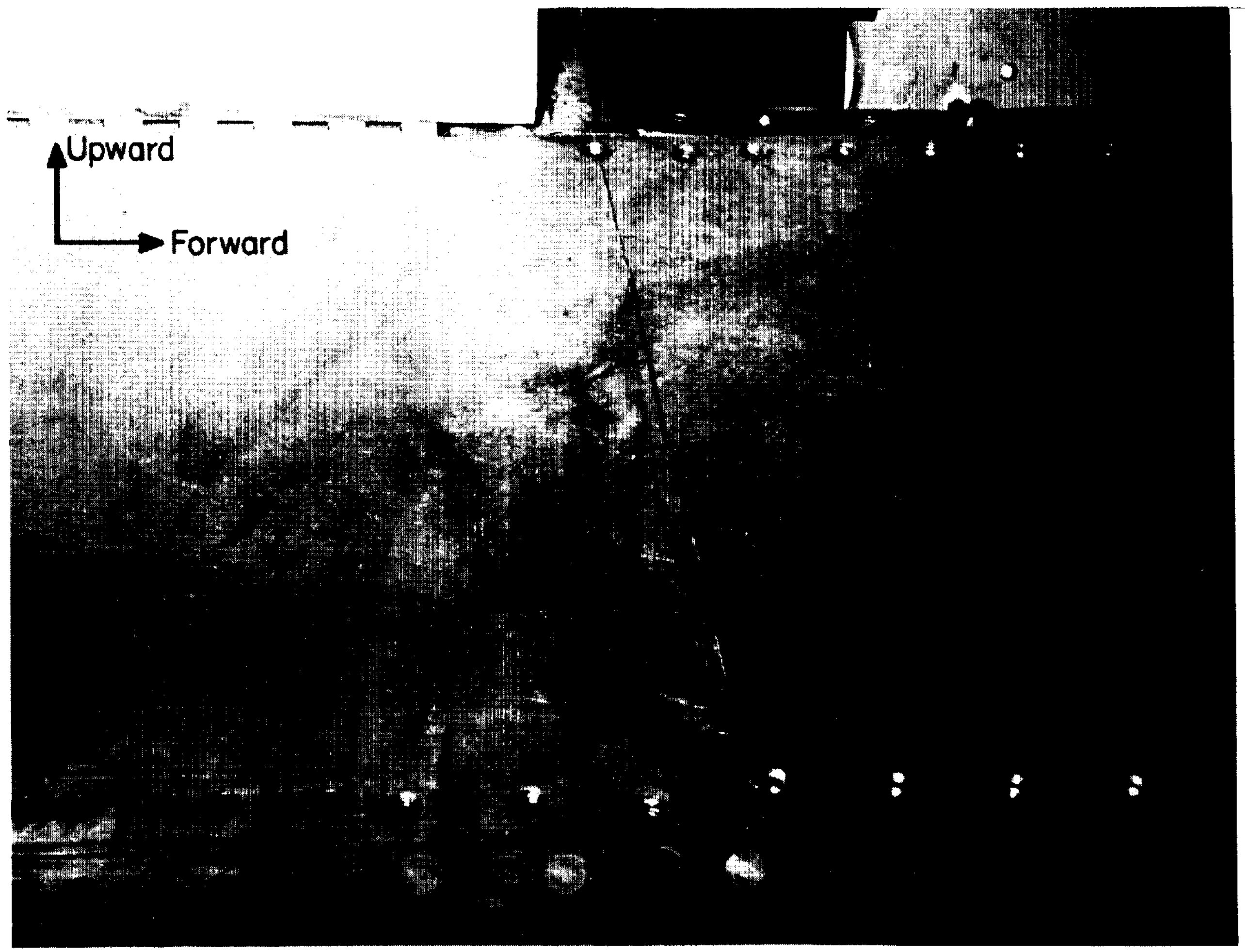
[redacted]

[redacted]



L-59-181
Figure 14.- Crack in fillet of left inboard upper shear tie at body station 442.5.

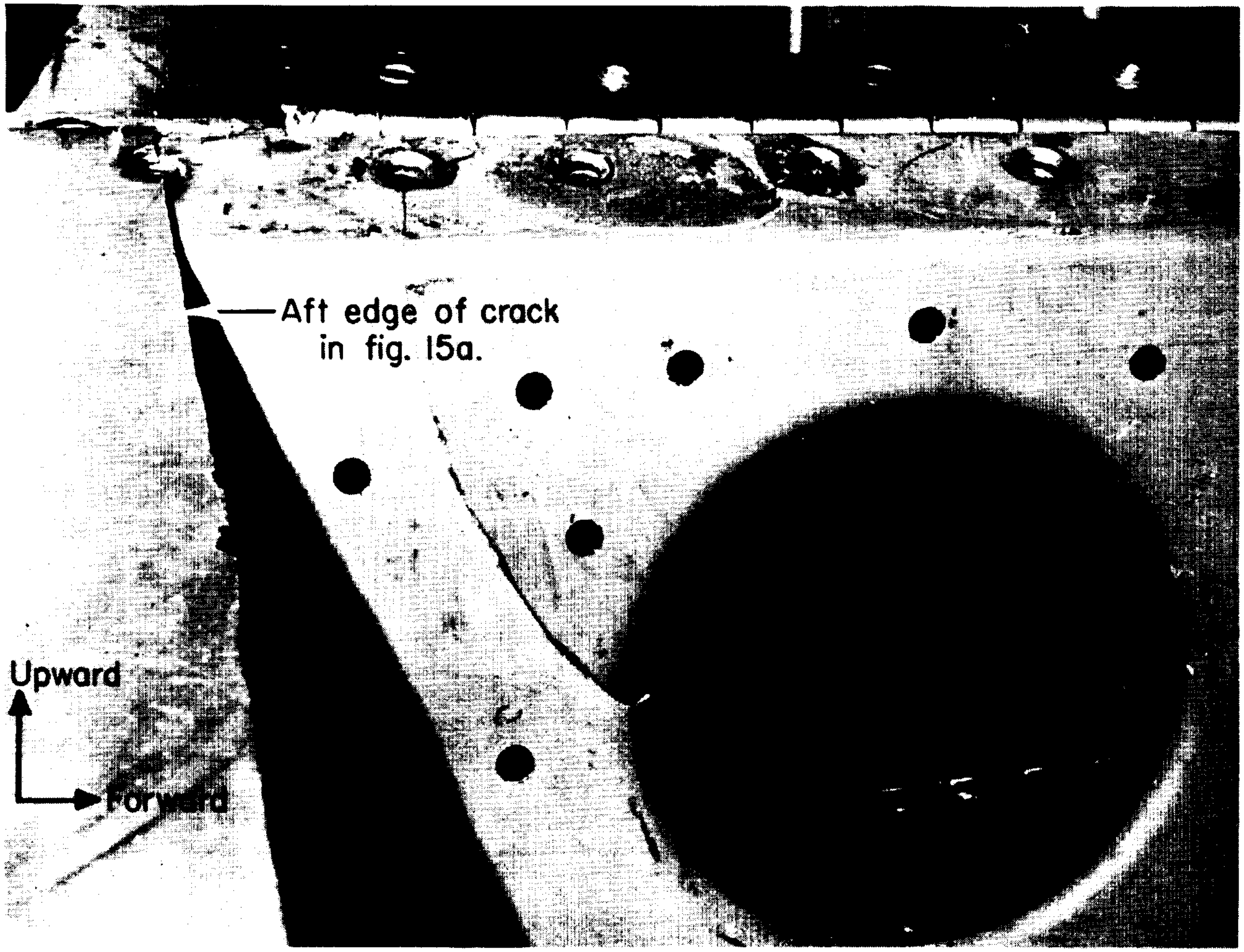
[redacted]



L-59-182

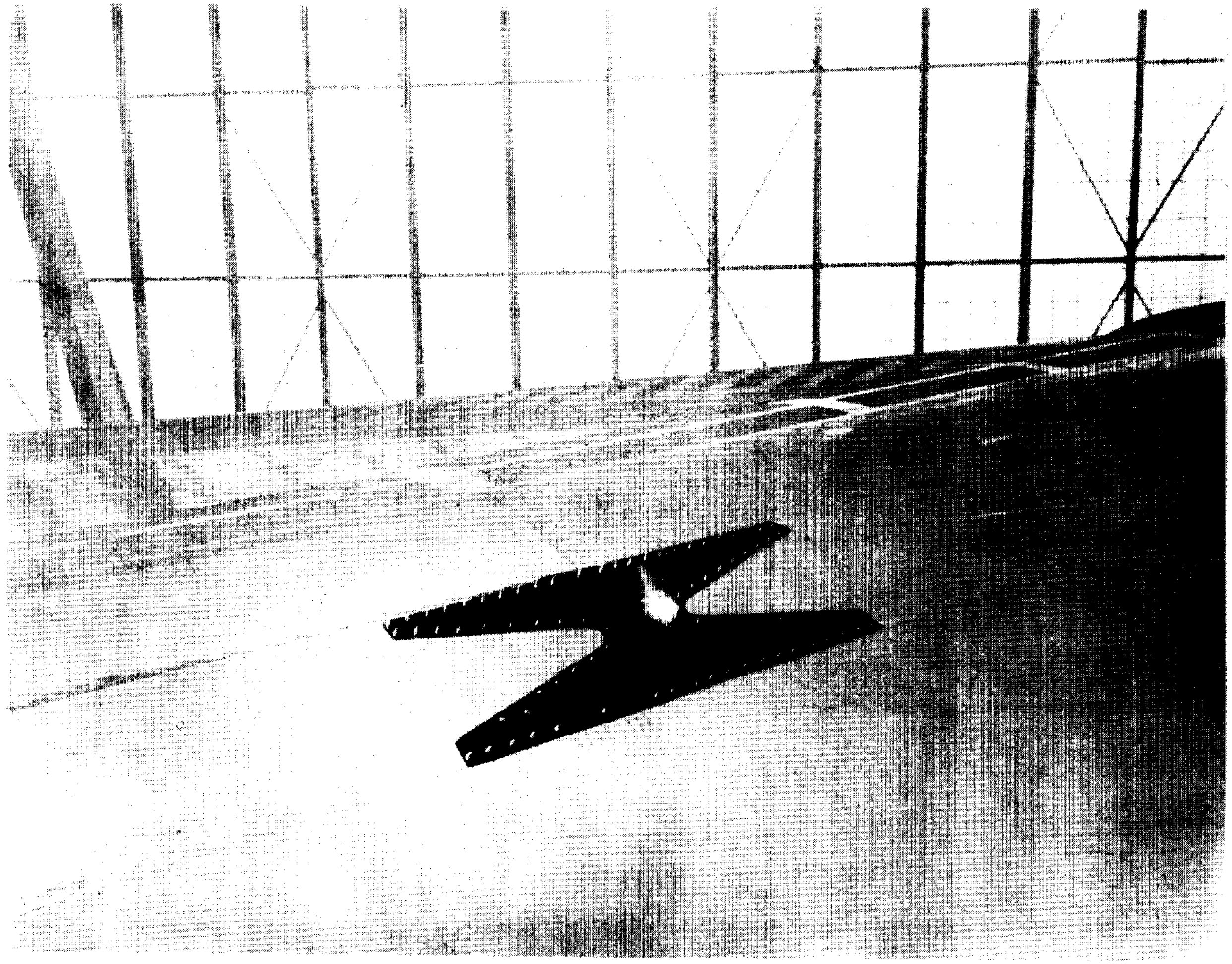
(a) Fuselage crown skin, 12 inches long. Crack did not extend beyond bottom rivet hole.

Figure 15.- Fuselage cracks at body station 648 on right side.



(b) Vertical stiffener in life-raft compartment. L-59-183

Figure 15.- Continued.



(c) Fishtail-doubler repair to backbone skin. L-59-184

Figure 15.- Concluded.

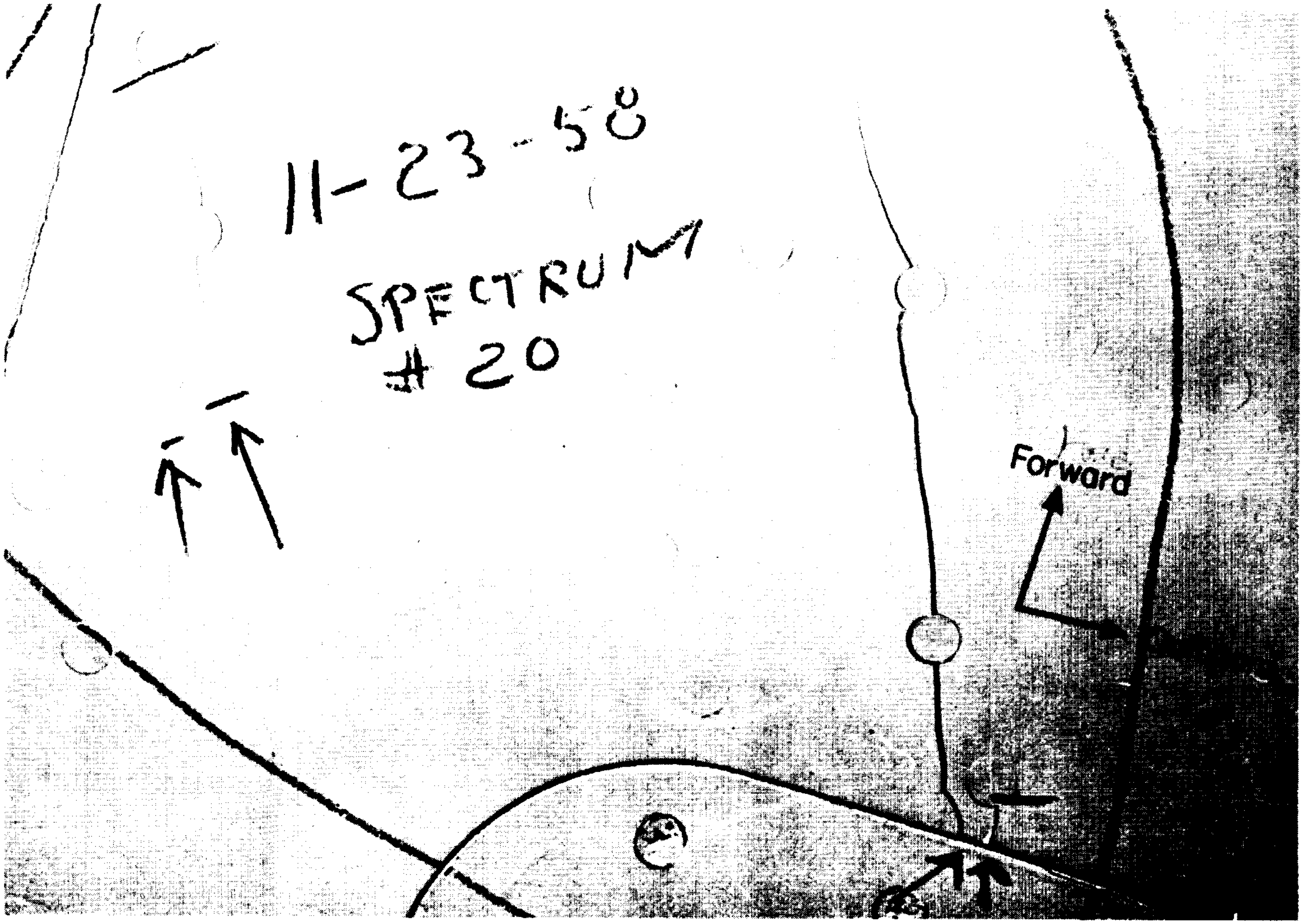
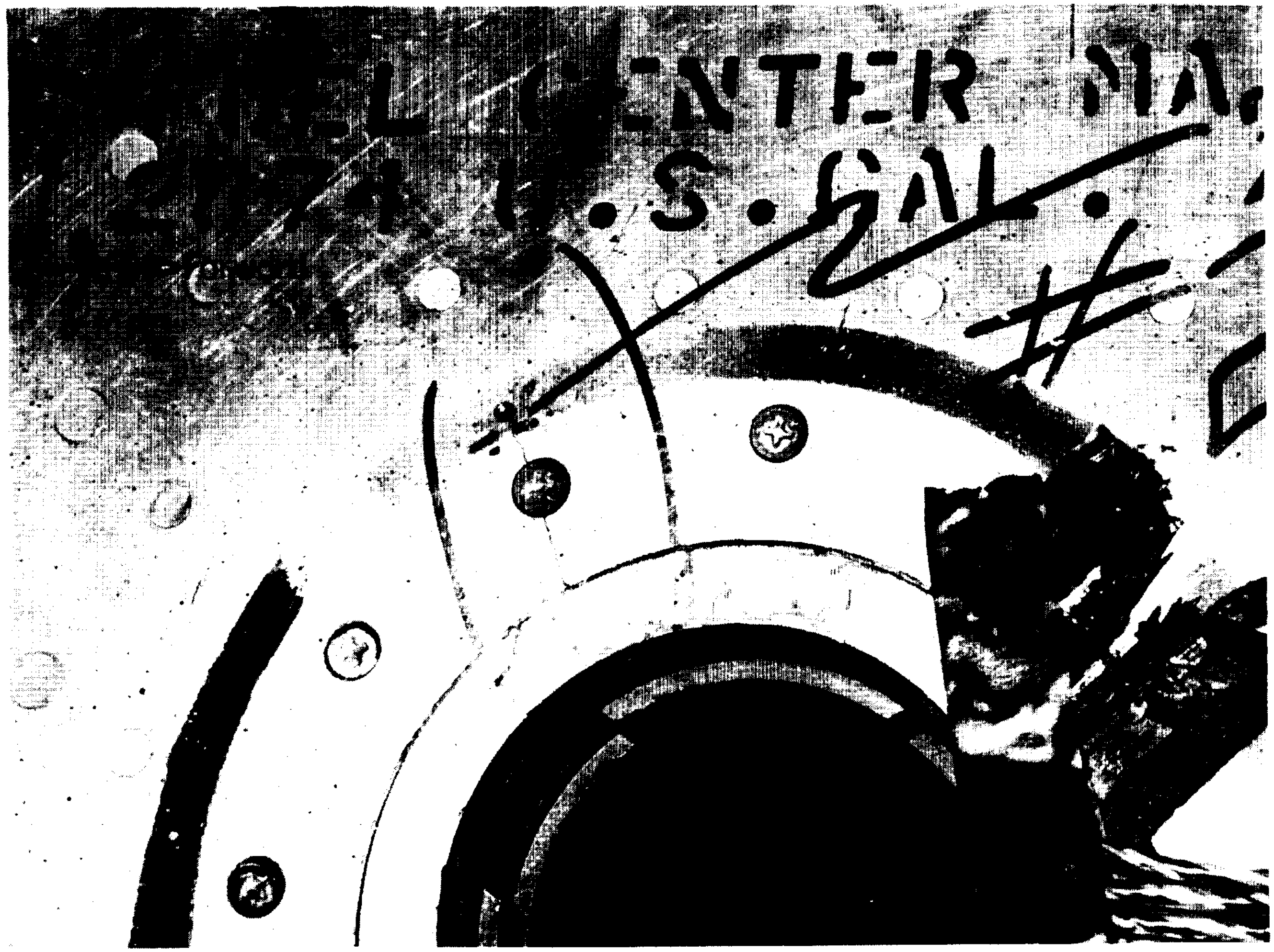


Figure 16.- Typical cracks in leading-edge skin. L-59-185

[redacted]

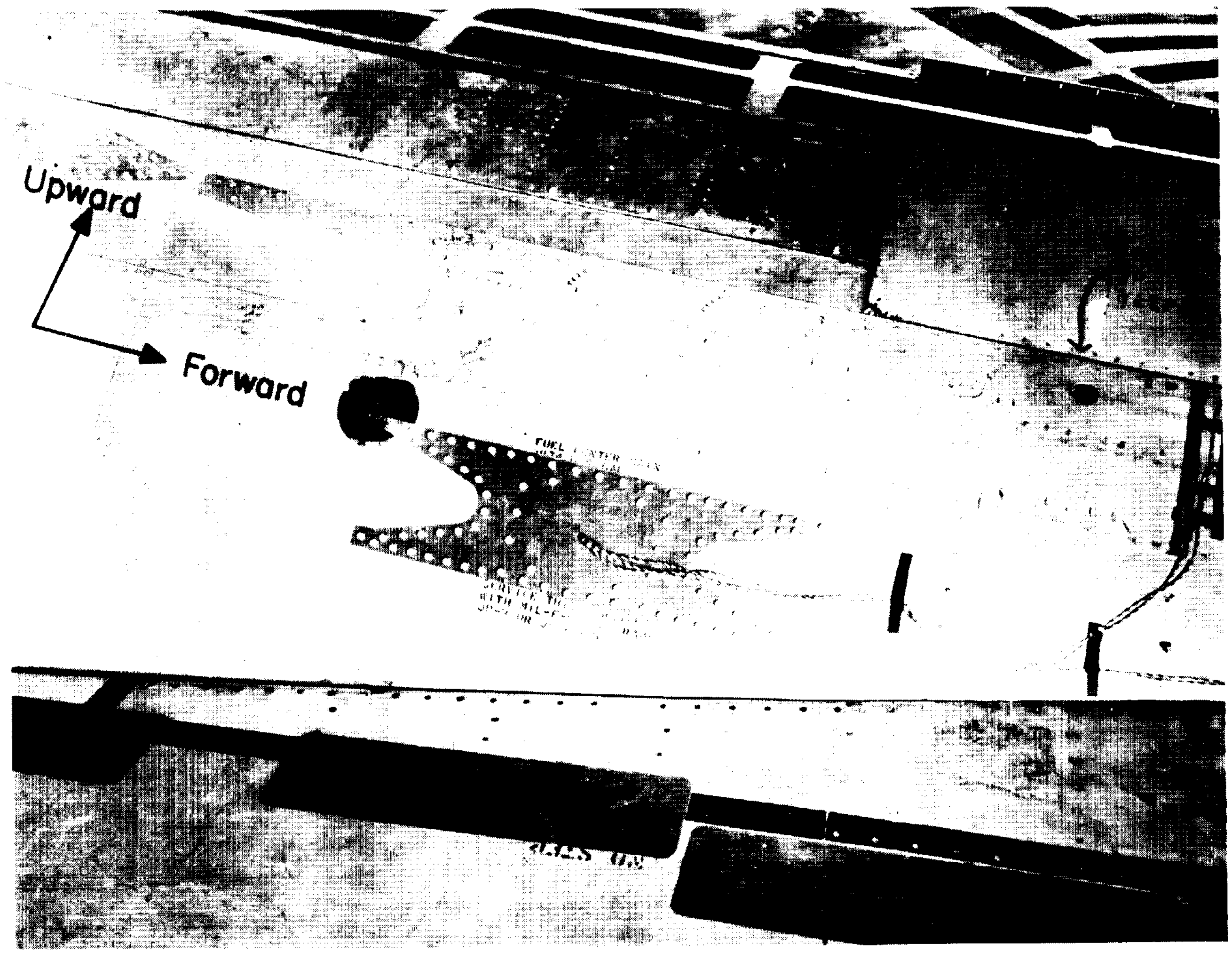


(a) Crack as stop-drilled, spectrum 23. L-59-186

Figure 17.- Crack at filler neck opening on right side at body station 550.

[redacted]

[redacted]

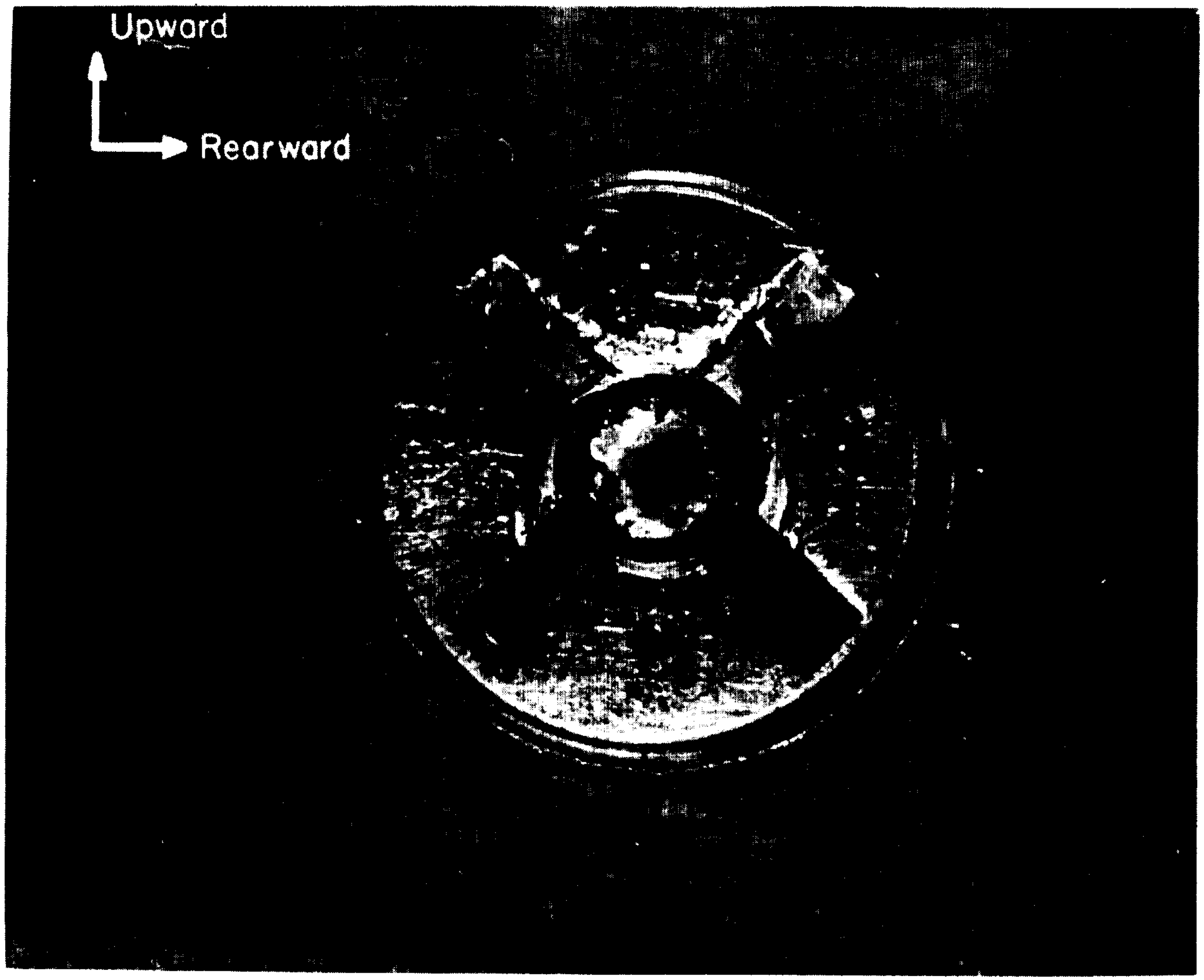


(b) Fishtail-doubler repair, inspection 12. L-58-1312a.1

Figure 17.- Concluded.

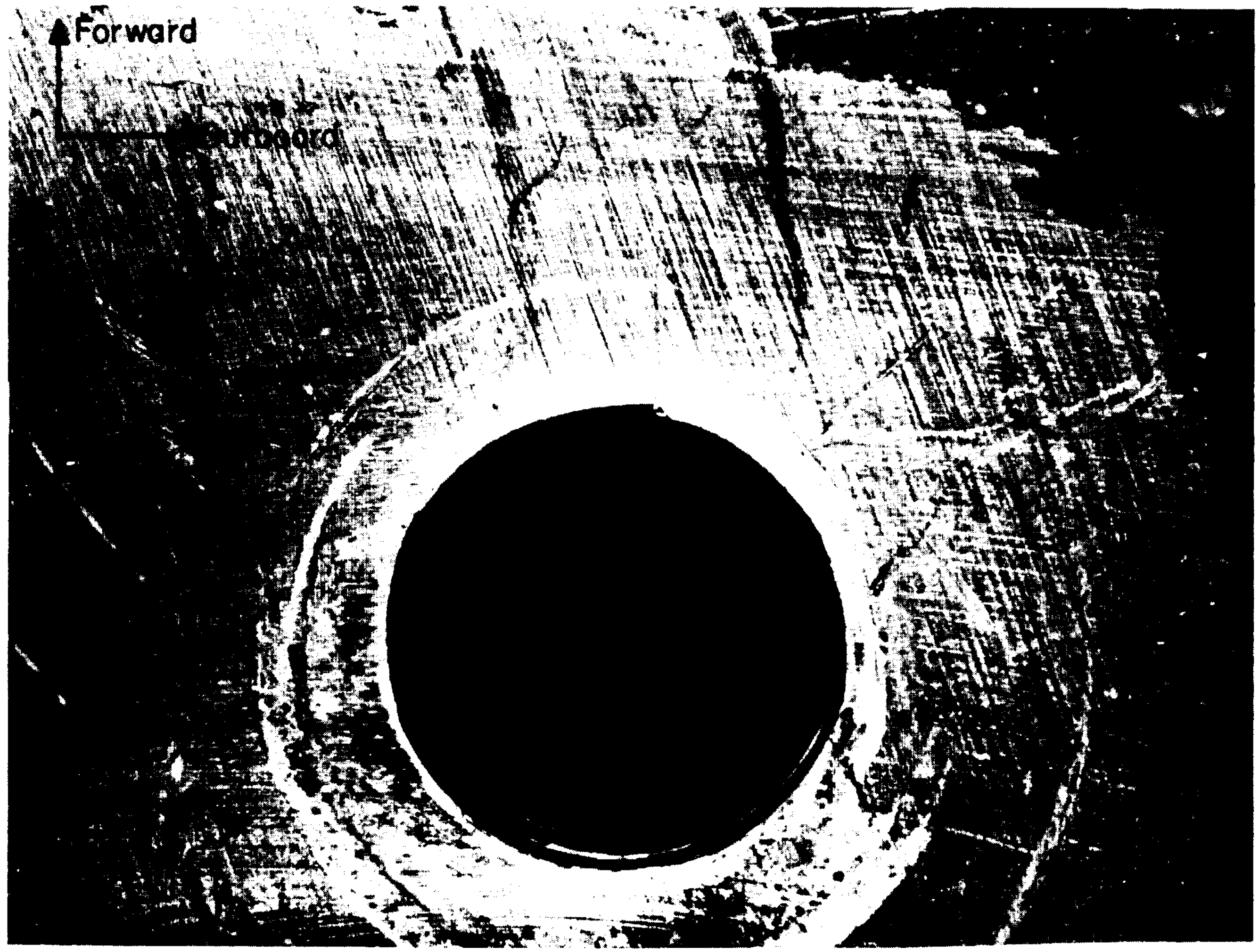
[redacted]

[redacted]



L-59-187
Figure 18.- Typical crack in lower leg of upper-fuselage crown-skin reinforcing plate, discovered during inspection 12 following spectrum 24.

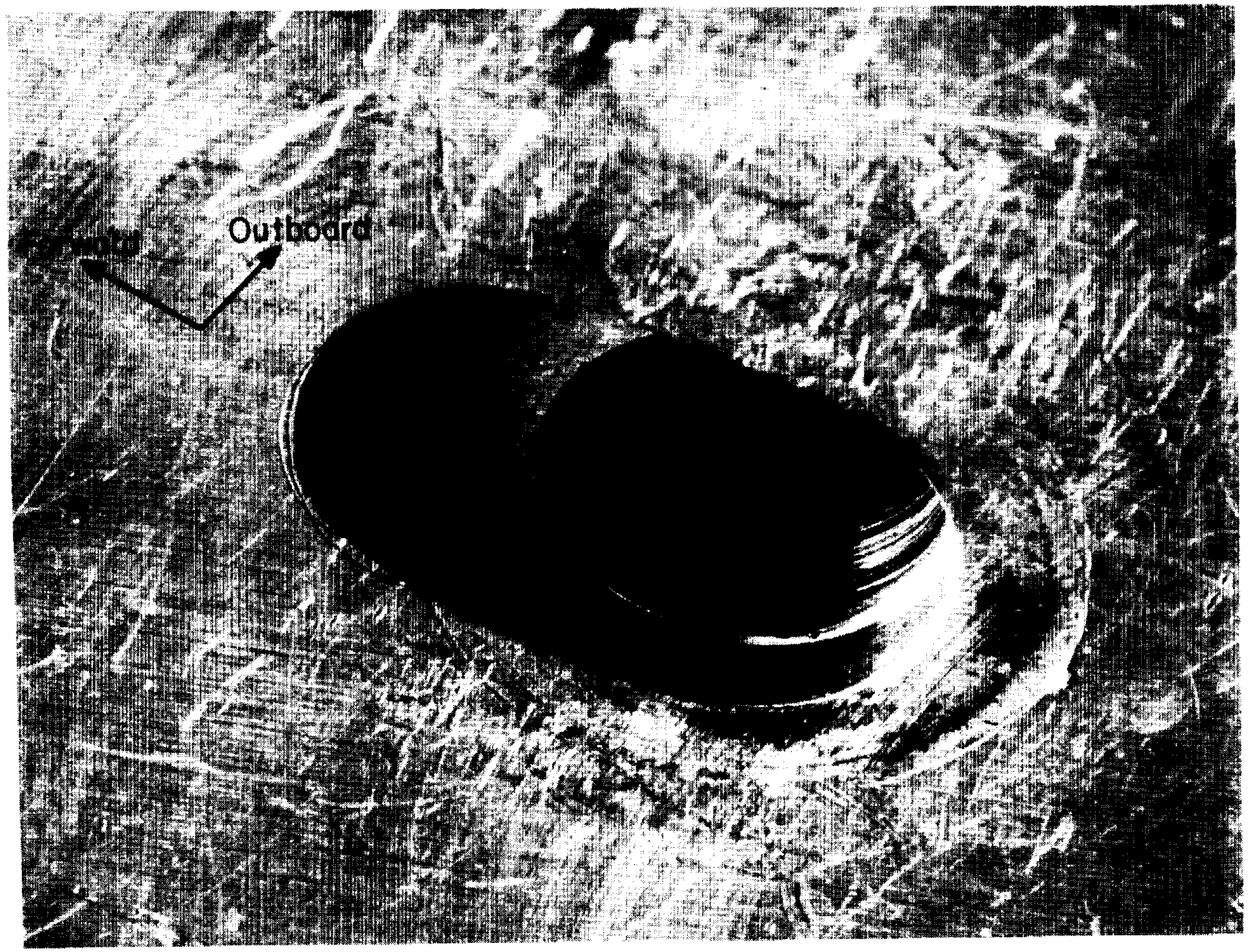
[redacted]



(a) Crack as found during inspection 13, following spectrum 26. L-59-188

Figure 19.- Crack in outboard finger of splice at hole 406 at left buttock line 45.

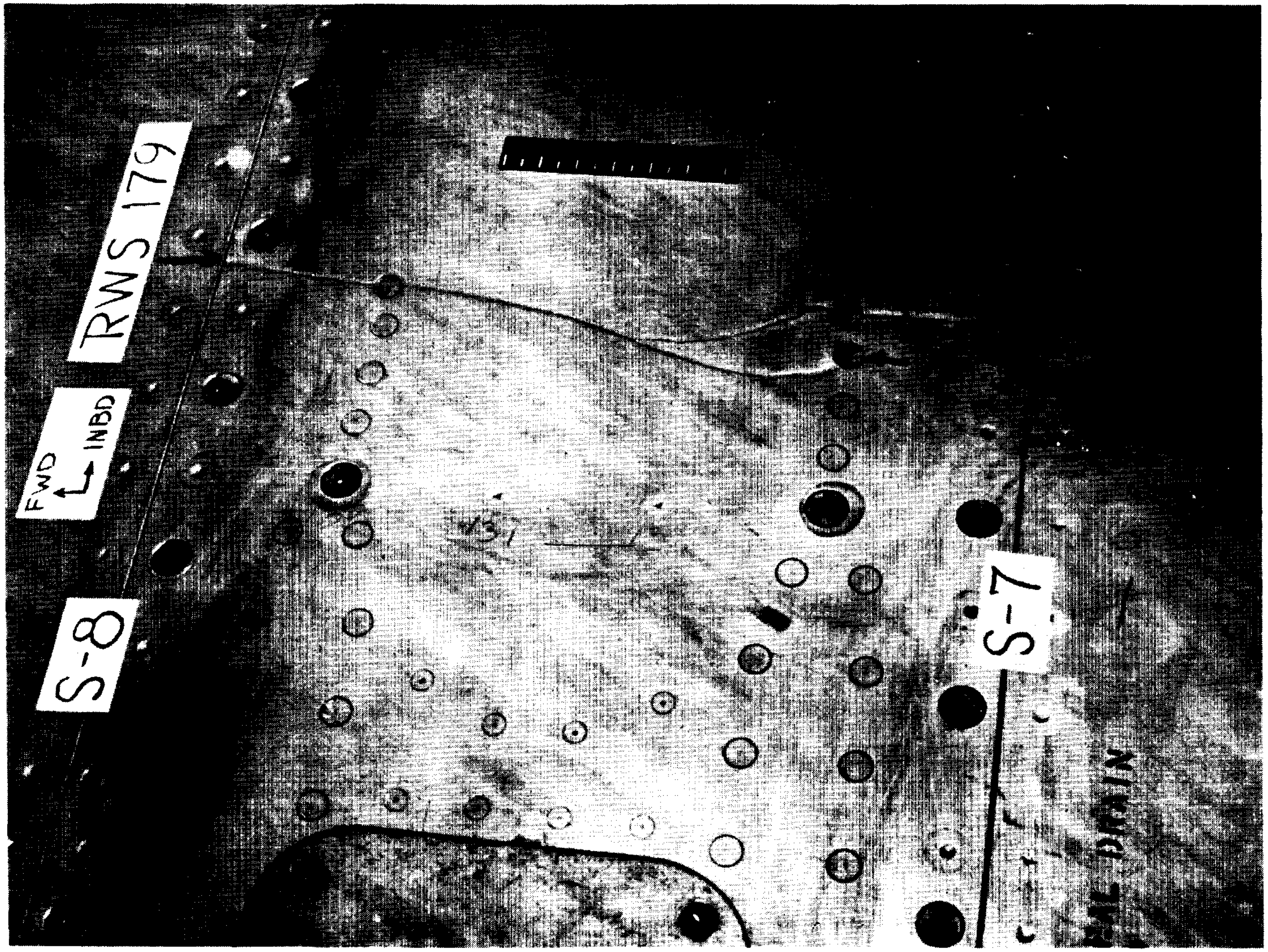
[redacted]



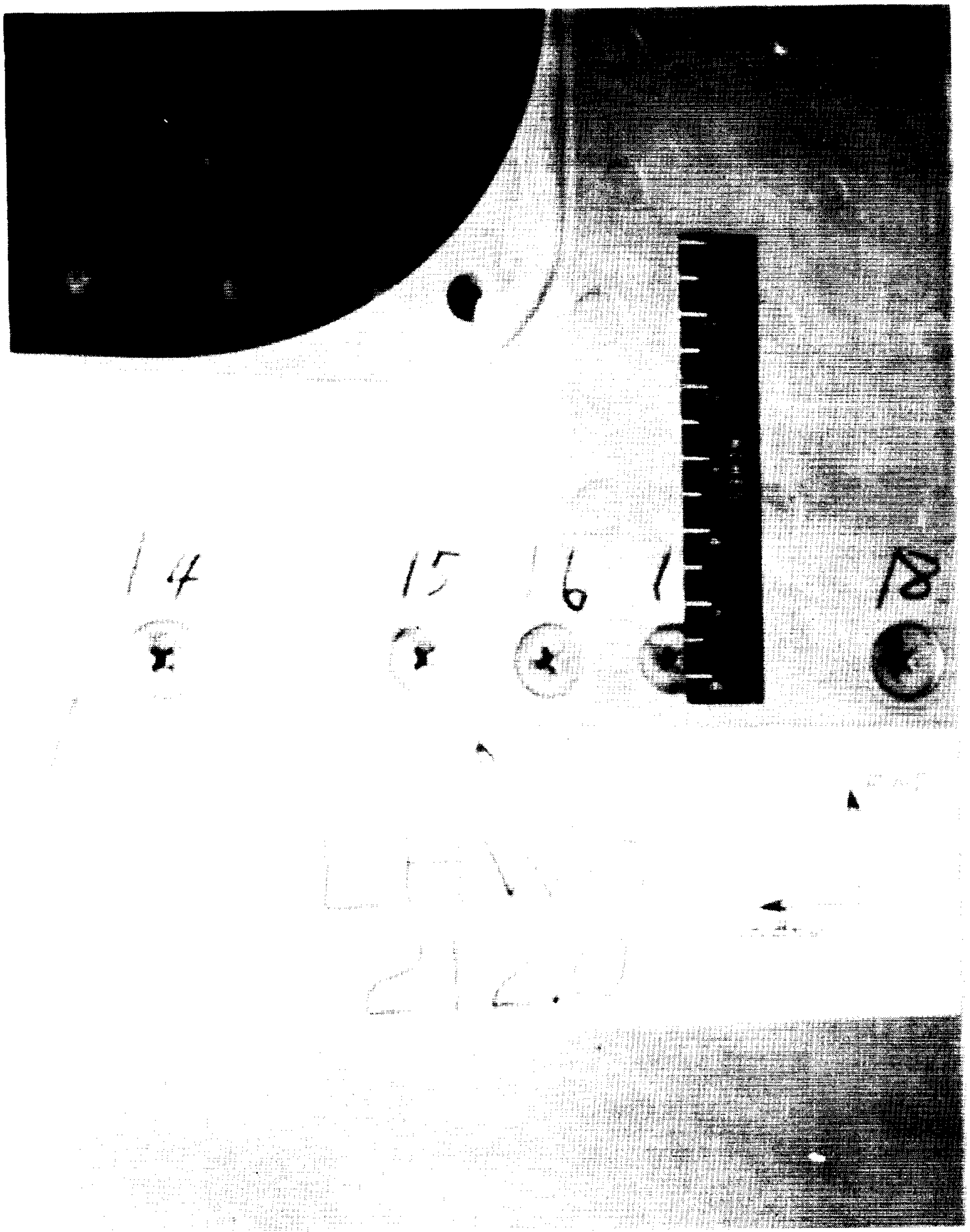
(b) Repair. Hole double drilled and contour smoothed. L-59-189

Figure 19.- Concluded.

[redacted]



L-58-1060a
Figure 20.- Crack in right wing closeout panel at wing station 179. Spectrum 29.



L-59-343
Figure 21.- Crack in left wing closeout panel at wing station 212, found during inspection 15.

[redacted]

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MEMORANDUM 3-15-59L

for the

U. S. Air Force

RESULTS OF A CYCLIC LOAD TEST OF AN RB-47E AIRPLANE*

COORD. NO. AF-AM-171

By Wilber B. Huston

ABSTRACT

An account is given of the NASA phase of a laboratory fatigue test on a B-47 airplane. The loadings used composed a spectrum of four levels. Stress measurements made on the wing and on the upper fuselage longerons are tabulated. A description is given of all cracks which developed during the tests.

INDEX HEADINGS

Airplanes - Specific Types	1.7.1.2
Connections, Bolted	4.3.6.1
Loads and Stresses, Structural - Bending	4.3.7.3
Materials, Properties - Fatigue	5.2.5
Materials, Airframe - Operating Stresses	5.3.1

[redacted]