NASA Contractor Report 145326

Flight Service Evaluation of Kevlar-49 Epoxy Composite Panels in Wide-Bodied Commercial Transport Aircraft - Fourth Annual Flight Service Report

R. H. Stone

LOCKHEED-CALIFORNIA COMPANY
BURBANK, CA 91520

CONTRACT NAS 1-11621

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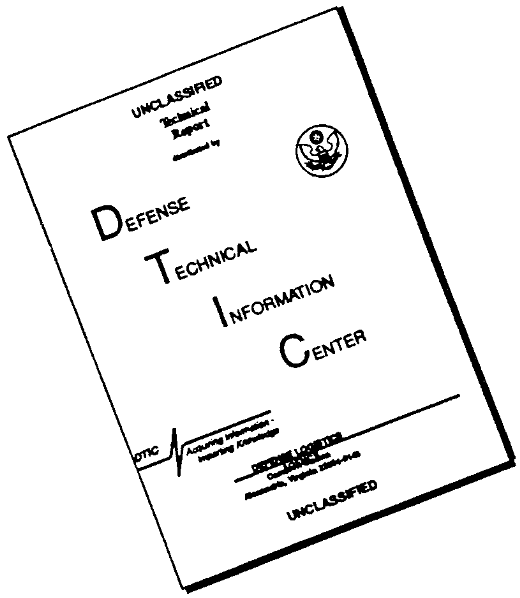

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FOREWORD

This is the fourth annual flight service evaluation report on the condition of Kevlar-49 fairing panels installed on three L-1011's under NASA Contract NAS 1-11621, "Flight Service Evaluation of Kevlar-49 Composite Panels in Wide-Bodied Commercial Transport Aircraft." The manufacture and installation of these panels was completed in February 1973 and reported in NASA CR-112250 dated March 1973 (Ref. 1). The results of inspections after the first three years of flight service were reported in Refs. 2, 3, and 4. Since the last annual report was issued, a five year program extension has been received from NASA. Annual reports will be issued describing service performance after each year of service through the ten year duration of the program.

This program is being administered by the Langley Research Center, National Aeronautics and Space Administration with Mr. Benson Dexter of the Materials Division as the Project Engineer.

This program is being performed by the Lockheed-California Company with Robert H. Stone the Program Leader, with assistance provided by T. L. Crawford, D. H. Horadam, R. S. Beck, and J. Luney of the Product Support Branch.

The ground-based environmental exposure data included herein was prepared by H. B. Dexter and R. A. Pride, NASA Langley Research Center.

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ABSTRACT

Kevlar-49 fairing panels, installed as flight service components on three L-1011s, were inspected after four years' service, and found to be performing satisfactorily. There are six Kevlar-49 panels on each aircraft, including sandwich and solid laminate wing-body panels, and 150°C (300°F) service aft engine fairings. The three L-1011s are one each of Eastern, Air Canada, and TWA aircraft. The fairings have accumulated a total of 32,472 hours, with one ship set having 13,347 hours service as of January 31, 1978. The inspections were conducted at the airlines' major maintenance bases with the participation of Lockheed Engineering.

The Kevlar-49 components were all found to be performing satisfactorily in service with no major problems, or any condition requiring corrective action. The only defects noted were minor impact damage, and a minor degree of fastener hole fraying and elongation. These are for the most part comparable to damage noted on fiberglass fairings.

A concurrent investigation has been conducted by NASA-Langley on Kevlar-49/epoxy coupons exposed to outdoor environment over a three year period at various locations providing a variety of climatic conditions. Weight changes and retention of mechanical properties were determined after one and three years exposure. A net weight loss has occurred due to ultraviolet effects on the unpainted specimens. Mechanical property retentions have been satisfactory with most specimens retaining well over 80% of their original value.

The service history to date indicates that Kevlar-49 epoxy composite materials have satisfactory service characteristics for use in aircraft secondary structure.

SECTION 1

INTRODUCTION AND BACKGROUND

The subject program on flight service evaluation of Kevlar-49 fairings consists of fabrication, installation and flight service of eighteen secondary structural panels; six on each of three L-1011's. The three participating airlines are Eastern, TWA, and Air Canada. Fabrication and installation of the panels was completed in February 1973, with initiation of flight service occurring in early 1973 on all three aircraft.

The six fairings are all similar to baseline fiberglass designs in which Kevlar-49 fabric, (comparable in fabric weave and thickness per ply to the fiberglass), was substituted for the fiberglass on a ply for ply basis. This required no other design changes or development of new tooling for layup and cure, but still provided a potential weight savings of 25-30 percent. These six parts are as follows:

- A left-hand and right-hand set of a large 152 cm x 170 cm (60 inch x 67 inch) sandwich wing-body fairing panel. The exterior skin is 0.05 cm (0.020 inch) thick with 1 ply 181 style Kevlar-49 fabric and 2 plies 120 style Kevlar-49 fabric. The interior skin is 0.04 cm (0.015 inch) thick with three plies of 120 style Kevlar-49 fabric. The honeycomb core is Nomex with 0.3 cm (1/8 inch) cells, and 0.048 gm/cm^3 (3.0 lb/cu ft) density. Overall panel thickness is 2.24 cm (0.88 inch), with a solid laminate edge 0.25 cm (.100 inch) thick built up of 181 style Kevlar-49 plies.

- A left-hand and right-hand set of a small 23 cm x 84 cm (9 inch x 33 inch) approximately-solid laminate wing-body fillet panel. The laminate incorporates 9 plies of 181 style Kevlar-49 fabric and is approximately 0.2 cm (0.09 inch) thick.

- A left-hand and right-hand set of an aft engine sandwich fairing-76 cm x 183 cm (30 inch x 72 inch approximately). The skins are 0.05 cm (0.020 inch) thick with 1 ply 181 style Kevlar-49 fabric and 2 plies 120 style Kevlar-49 fabric. The Nomex core is identical to that used in the wing-body fairing, except for thickness, and the overall panel thickness is 0.64 cm (0.25 inch). The aft engine fairing also has a solid laminate edge member 0.25 cm (.100 inch) thick.

The Kevlar-49 panels all utilized the same resin system as the production fiberglass parts: A 120°C (250°F) curing, 82°C (180°F) service epoxy (Hexcel's F-155) for the wing-body fairing and fillet panels, and a 177°C (350°F) curing, 150°C (300°F) service epoxy (Hexcel's F-161) for the aft engine fairings. Two fabric weave styles of Kevlar-49 were used. Style 181 is a satin weave similar to the 181 fiberglass weave, 0.23mm. (9 mils) per cured ply and 0.17kg/m^2 (5.0 oz/yd^2) dry weight. Style 120 is a plain weave, 0.13mm. (5 mils) per cured ply and 0.06kg/m^2 (1.8 oz/yd^2) dry weight. Both fabric styles incorporate light denier Kevlar-49 yarns, 380 denier for Style 181, and 195 denier for Style 120.

All of the parts have an outer layer of flame sprayed aluminum and topcoat applied according to standard production procedures used on the baseline fiberglass parts. The actual weight savings achieved by this direct substitution of Kevlar-49 for fiberglass averaged 26 percent for the six parts. Further details on Kevlar-49 part design and fabrication are given in NASA CR-112250 (Ref. 1), which is the final report of the fabrication and installation phases of the program.

Under the original program plan, inspections of the Kevlar-49 parts were to take place annually in conjunction with regularly scheduled inspections at the airline maintenance bases. However, the first annual inspections of the TWA and Air Canada panels took place at Lockheed-California Company due to special circumstances, while the Eastern panels were inspected by Eastern personnel at Miami. Results of those inspections indicated no significant damage or deterioration of the parts other than minor impact damage, fastener hole elongation, and minor delaminations. Comparable damage was also noted on similar fiberglass parts. Further details are given in NASA CR-132647, the First Annual Flight Service Report (Ref. 2).

In order to obtain thorough information and documentation of part conditions, the inspection activity was expanded as follows for subsequent annual inspections:

1) A Lockheed Engineering representative is to be present for each annual inspection at the airlines' maintenance bases.

2) Three of the six panels (one of each left-hand and right-hand set) are to be removed for thorough inspection, weighing, inspection of fastener holes and interior surface conditions.

3) The airlines are to report incidences of damage and repair occurring in service using special formats provided by Lockheed.

The second annual inspections in 1975 and the third annual inspections in 1976 were conducted in accordance with this expanded scope, and are reported in NASA CR-132733 (Ref. 3) and CR-145141 (Ref. 4).

The TWA panels were removed after approximately one year (2400 hours) of service when the aircraft was taken out of service in April 1974, because of a cabin interior fire. The parts were not damaged and were returned to Lockheed for inspection. The parts were subsequently installed on a second TWA L-1011 for continuation of flight service testing. The reinstallation on TWA aircraft N31030 required some rework and repair of the panels, particularly in the case of the aft engine fairing panels, where relocation of all fastener holes was required. This rework activity is reported in detail in the Second Flight Service Report (Ref. 3). The aircraft on which these parts were reinstalled was delivered to TWA in August 1975, and have since been inspected annually in accordance with the expanded program scope.

During 1977, a five year extension to the program was received from NASA for a total of ten years' flight service of the Kevlar-49 fairings. This extension will carry the program from 1979 through 1983, and annual inspections of the three ship sets will take place in accordance with the expanded program scope outlined above.

Since this flight service program was initiated in 1973 a considerable number of Kevlar-49 components have been installed as production components on the L-1011 and other aircraft, and many other applications are being considered. The fairings in this program remain the longest service life Kevlar-49 components in commercial aircraft flight service, where they see over 2000 flight hours per year. Kevlar-49 has unique chemical and mechanical characteristics as the only organic filamentary reinforcement being used in aircraft structures. Two characteristics which have been of concern are the pick up of moisture in the fiber, and the low resin/fiber interface bond. The detailed monitoring of the fairings' performance in this program thus provides information on long-term mechanical behavior and environmental durability which has applicability to many other programs, and adds significantly to confidence in the use of this material.

Concurrent with the flight service evaluations, various composite materials coupons are being subjected to long-term environmental exposures at various aircraft terminals and at the Langley Research Center. The coupons are collected by the Boeing Commercial Airplane Company under NASA contract NAS1-11668. The coupons are tested at the Langley Research Center. Details of the environmental exposure program for both graphite/epoxy and Kevlar/epoxy materials systems are reported in Reference 5. The results after 1 and 3 years exposure for the L-1011 materials, Kevlar-49/F-155 and Kevlar-49/F-161, are presented herein.

SECTION 2

PANEL INSPECTIONS

The fourth annual inspection of the Kevlar-49/epoxy fairings on Eastern Ship N314EA (Serial #1022) took place at the Miami Maintenance Base on May 10, 1977. The panels at that time had been in flight service approximately four years with 11,400 flight hours and 5903 flights. In the intervening year since the previous inspection, the panels accumulated 2664 flight hours.

This inspection took place during an overnight maintenance check with the aircraft outside the hangar in rainy conditions. Eastern Maintenance could not provide access to the aft engine fairing panels under these conditions. Eastern Maintenance Engineering agreed to perform inspection of these panels later in the year at a "C" check or at a scheduled modification.

The left-hand wing-body fairing and underwing fillet panels were removed for inspection, as the right-hand panels had been removed in 1976. The right-hand wing-body fairing and underwing fillet panels were inspected in place on the aircraft. Inspection was by visual examination and coin tapping for delaminations and skin-core disbonds. The panels taken off the aircraft were cleaned to remove excessive dirt and residue, and then dried and weighed. These panels were also inspected for condition of the fastener holes and the inner surface.

The left-hand aft engine fairing was subsequently inspected by Eastern Maintenance on September 15, 1977. The panel was removed from the aircraft and inspected visually. On that date, the panel had approximately 12,300 flight hours service.

The fairings installed on Air Canada Ship CF-TNB-502 (Serial 1021) were inspected at the Montreal Maintenance Base on September 1, 1977. The fairings at that time had been in flight service for 9999 flight hours and 4796 flights. In the intervening year since the last inspection, the panels accumulated 2547 flight hours. The three panels scheduled for removal were the right-hand wing-body fairing and underwing fillet panels, and the left-hand aft engine fairing. Air Canada was also able to remove the left-hand wing body fairing and underwing fillet panels. Thus, five of the six panels were inspected off the aircraft. Inspection was by visual examination, coin tapping for delaminations and disbonds, and weighing as described for the Eastern panels.

The fairings installed on TWA Ship N31030 (Serial 1111) were inspected at the Los Angeles Maintenance Base on November 1, 1977. The fairings at that time had 5082 flight hours and 1831 flights on Ship 1111. These panels had seen 2404 hours on Ship 1026 prior to their removal and reinstallation for a total of 7486 flight hours. In the intervening year since the previous inspection the panels accumulated 2804 flight hours. The three panels removed for inspection were the left-hand wing-body fairing and underwing fillet panels, and the right-hand aft engine fairing. This was the opposite set of those removed in 1976. Inspection procedures were the same as described above for the Eastern and Air Canada panels.

All three inspections were conducted with the participation of Lockheed Engineering, and with the assistance of airline maintenance personnel in removal and reinstallation of the panels. Photographs were taken of all panels and areas containing defects, damage, or other conditions of special interest. Photographs were provided by Air Canada in Montreal, by the Lockheed Photography Department at TWA in Los Angeles, and by a commercial photographer at Eastern.

SECTION 3

DISCUSSION OF INSPECTION RESULTS

The Kevlar-49 panels are all performing satisfactorily in service, with no major damage or defects requiring corrective maintenance. Minor impact damage has been noted throughout the program, primarily on the two wing-body fairing sandwich panels which are in an area subject to damage from objects thrown up from the runway and also from damage during loading operations. Two additional small cracks were noted in this years' inspection, but more significantly all cracks observed in previous inspections had not grown or propagated. This type of impact damage is comparable to similar damage on adjacent fiberglass panels.

A more extensive damage condition was observed on the TWA right-hand wing-body fairing. This is a fairly deep concave depression and skin-core disbond, but with no associated crack or surface damage, which does not appear to be the result of impact damage. This condition is probably related to a repair made in this part during its reinstallation on Ship 1111 (Ref. 3). This repair was not documented, but apparently consisted of replacement of a damaged core area, extending partially through the core thickness, with a microballoon filled potting compound. Crushing of this potting compound under in-service conditions is a possible explanation of this condition. In any case, this does not appear to be a Kevlar-49 related problem, but as it is highly visible it will be carefully monitored in future inspections.

The other types of minor damage observed were some instances of fraying and elongation of fastener holes. Elongation of fastener holes has been observed on a small percentage of holes in a random distribution, and is comparable to conditions observed on similar fiberglass panels. This condition continues to be observed primarily on the underwing fillet panels, and appears to be related to installation problems which result in concentrated or non-uniform bearing loads. In some cases, the degree of elongation has increased from one inspection to the next, but this has not occurred in all cases. There has also been some observed increase in the incidence of hole elongation in the fillet panels. The locations of these holes and measurements of elongation will continue to be noted for comparison in future inspections. It should again be noted that the elongation observed to date is a minor condition which does not affect part performance or require corrective maintenance.

In one instance, hole deformation has been observed on parts other than the underwing fillets, and these are the TWA aft engine fairings. These fairings, as described in the Second Annual Report (Ref. 3), required relocation of all fastener holes during reinstallation. The holes were filled with a chopped glass filled epoxy, and a layer of epoxy impregnated 120 glass cloth was applied to both surfaces. Many holes were redrilled through the filled area, and these holes show a significantly greater degree of elongation than any holes drilled through a solid Kevlar-49 laminate. This condition, while not affecting part performance or requiring corrective maintenance, does indicate an inadequacy of the repair; and does not reflect on the performance of Kevlar-49.

Fraying of fastener holes appears to be a general occurrence on the Kevlar-49 parts, and is the only condition not observed on similar fiberglass parts. As discussed in the previous Annual Reports, this appears to be the effect of a fiber which exhibits non-linear stress-strain behavior combined with a relatively brittle resin, and is the same condition observed after machining. The degree of fraying does not appear to be increasing, and probably represents an initial condition more than a service condition. It is significant that more fraying is observed on the aft engine fairings which incorporate a more brittle 177°C(350°F) curing epoxy. The TWA aft engine fairings, which had the surface overlay of glass showed no fastener hole fraying. The elongated holes in the underwing fillet had more fraying than the other holes, indicating that the non-uniform loading which caused the elongation also aggravated the initial degree of fraying.

The inner surfaces of the Kevlar-49 fairings have been relatively free of any defects or damage. However, the Air Canada right-hand wing-body fairing, removed for the first time for inspection, had three skin-core disbond areas. This condition probably is the result of processing errors since all the other panels have been free of skin-core disbonds. This condition will be monitored in future inspections for any increase in disbond area.

There continues to be no evidence of contamination of Kevlar-49 surfaces with Skydrol, although hydraulic lines are located behind the wing-body fairings and underwing fillets. Occurrences of paint loss on the Kevlar-49 parts have been noted. It is unlikely that a substitution of Kevlar-49 for fiberglass in a given epoxy matrix would affect surface adhesion, and this appears to be the case. Paint loss is a possible indication of damage or contamination, but aside from obvious impact areas no Kevlar-49 defects have been associated with paint loss.

All of the Kevlar-49 parts removed for inspection have been weighed for determination of possible weight gains due to moisture pick-up. This does not appear to provide any true evidence of the relative moisture absorption of Kevlar-49 composites compared to fiberglass. The effects of paint loss, repainting, loss of sealant and resealing, repair patches and the accumulation of surface contaminants all mask any weight changes due to moisture. Airline maintenance bases lack suitable balances for accurate weighing, and in future TWA inspections in Los Angeles it is planned to bring in Lockheed equipment for this purpose which could be used with the relatively small fillet panels.

SECTION 4

GROUND-BASED ENVIRONMENTAL EXPOSURE (1)

Concurrent with the flight service evaluations, various composite materials coupons are being subjected to long-term environmental exposures at various aircraft terminals and at the Langley Research Center. Details of the environmental exposure program for both graphite/epoxy and Kevlar/epoxy materials systems are reported in Reference 5. The results after 1 and 3 years exposure for the L-1011 materials, Kevlar-49/F-155 and Kevlar-49/F-161, are presented herein.

Environmental exposure data are being obtained on interlaminar shear, flexure, and compression specimens. A rack designed to hold five replicate panels is shown in Figure 1. The Kevlar/epoxy specimens are held in the panels in a manner that provides a maximum exposure to sunlight on one surface but allows free circulation of air around the specimens. Panels for 1 and 3 years exposure data have been removed and the specimens have been tested. The remaining 3 panels are scheduled for removal after 5, 7, and 10 years, respectively. Data being generated include strength retention, moisture pickup, and ultraviolet weight loss.

All specimens were weighed and measured to obtain baseline data prior to environmental exposure. A separate group of specimens were dried in an oven to obtain an initial fully dry weight. The baseline weights of all exposed specimens were corrected to a fully dry weight. All specimens are weighed after removal from the exposure racks. Any weight changes are attributed to the combined effects of moisture pickup and ultraviolet weight loss. After the flexure specimens are tested, they are dried to determine the absorbed moisture content. The ultraviolet weight loss is taken to be the difference between the fully-dried weights before and after exposure. Similar data are

(1) Work performed by H. Benson Dexter and Richard A. Pride of the NASA-Langley Research Center.

not generated for the shear specimens because of their small size and for the compression specimens because of glass/epoxy tabs bonded to the specimens prior to testing.

Figure 2 shows the worldwide distribution of the environmental exposure racks. The average annual temperature and relative humidity for all exposure sites are 290K (62°F) and 75 percent, respectively. Figure 3 shows the moisture pickup data for Kevlar-49/F-155 and Kevlar-49/F-161 flexure specimens after 1 and 3 years exposure. The data shown is for all exposure sites except for the 3 year Brazil data which is not currently available. The average moisture pickup after three years exposure is 2.1 percent for the 450K (350°F) cure Kevlar-49/F-161 system and 1.9 percent for the 394K (250°F) cure Kevlar-49/F-155 system. Figure 4 shows the weight loss data resulting from 3 years outdoor ultraviolet exposure for all exposure sites except Brazil. The exposed surface dimensions for the flexure specimens was 2.54 cm x 6.48 cm (1.00 in. x 2.55 in.). The weight loss data are presented as a function of exposure site latitude. The ultraviolet weight loss varies from 1.5 mg/cm^2 for the Kevlar-49/F-161 material at the Germany rack location to 7.9 mg/cm^2 for the Kevlar-49/F-155 material at the Hawaii rack location. The limited data obtained to date indicates that weight loss due to ultraviolet exposure is approximately inversely proportional to the distance of the exposure site from the equator. The weight loss of 7.9 mg/cm^2 represents about 25 percent of the weight of one ply for the 3.17 mm (.125 in.) thick flexure specimens. Table 1 lists the moisture pickup and ultraviolet weight loss data presented in Figures 3 and 4. It should be pointed out that all the specimens had bare surfaces. Preliminary data from other tests indicate that standard commercial aircraft paint practically eliminates ultraviolet weight loss of composite laminates. However, the paint does not prevent moisture absorption.

Tables 2, 3, and 4 list the baseline, 1 year and 3 year average strength data for flexure, short beam interlaminar shear and compression secimens, respectively. In addition, average flexure modulus is presented in Table 2. All the strength data are presented in Figures 5-10 in bar-graph form for each of the six exposure sites after 1 and 3 years exposure. The largest flexure strength reduction occurred after 3 years exposure in Hawaii. An 84 percent strength retention for the Kevlar-49/F-155 Hawaii specimens is shown in Figure 6. Very little strength changes have occurred for the Kevlar-49/F-161 flexure specimens after 3 years exposure at all six locations as shown in Figure 6.

The largest short beam interlaminar shear strength reduction occurred after 3 years exposure in Brazil. A 76 percent strength retention for the Kevlar-49/F-155 Brazil specimens is shown in Figure 8. The largest strength reduction for the Kevlar-49/F-161 shear specimens occurred after 3 years exposure at the Langley Research Center. This stength reduction was only about 6 percent.

The 1 year compression strength increased compared to the baseline except for the specimens exposed in Brazil which showed a 10 percent strength reduction as shown in Figure 9. The 3 year compression strength decreased except for the specimens exposed in Brazil which showed a 2 to 4 percent strength increase compared to the baseline specimens as shown in Figure 10. Scatter in the data and a limited number of tests are probably the cause of variations in data trends.

TABLE I.- MOISTURE PICKUP AND ULTRAVIOLET LOSSES FOR KEVLAR/EPOXY FLEXURE COUPONS AFTER 3 YEARS WORLDWIDE EXPOSURES

Exposure site	Moisture pickup, percent		UV weight loss, mg/cm^2	
	F-155	F-161	F-155	F-161
LaRC	1.86	2.14	5.49	3.73
California	1.78	1.99	5.09	2.51
New Zealand	2.04	2.34	5.70	3.16
Hawaii	2.01	2.18	7.92	4.70
Germany	1.60	1.90	3.16	1.52
Brazil	Data not available			

TABLE 2.- RESULTS OF GROUND-BASED ENVIRONMENTAL EXPOSURE ON KEVLAR/EPOXY MECHANICAL PROPERTY TEST SPECIMENS

FLEXURE TESTS

Exposure time, yr	Exposure location	Kevlar/epoxy system	Number of specimens	Average failure stress		Average flexure modulus	
				MPa	ksi	GPa	psi (× 10^6)
0 (Baseline)	LaRC	F-155	6	396.2	57.46	25.0	3.63
1	LaRC	↓	3	369.1	53.53	23.9	3.46
1	California		3	357.0	51.77	23.4	3.40
1	New Zealand		3	366.9	53.21	23.3	3.38
1	Hawaii		3	335.9	48.72	23.2	3.36
1	Germany		3	382.5	55.48	24.1	3.50
1	Brazil	↓	(Unavail.)	-----	-----	----	----
3	LaRC	F-155	3	367.8	53.35	23.0	3.33
3	California	↓	3	372.3	54.00	23.2	3.36
3	New Zealand		3	349.8	50.73	22.4	3.25
3	Hawaii		3	333.6	48.38	22.0	3.19
3	Germany		3	391.6	56.80	22.8	3.30
3	Brazil	↓	3	353.2	51.23	23.0	3.33
0 (Baseline)	LaRC	F-161	5	375.4	54.45	24.4	3.54
1	LaRC	↓	3	363.4	52.70	25.3	3.67
1	California		3	368.4	53.43	26.3	3.81
1	New Zealand		3	376.4	54.59	25.2	3.65
1	Hawaii		3	358.5	52.00	25.7	3.73
1	Germany		3	389.3	56.46	25.4	3.69
1	Brazil	↓	(Unavail.)	-----	-----	----	----
3	LaRC	F-161	3	374.4	54.30	25.1	3.64
3	California	↓	3	374.1	54.26	25.5	3.70
3	New Zealand		3	365.9	53.07	24.2	3.51
3	Hawaii		3	358.5	52.00	23.6	3.42
3	Germany		3	378.7	54.92	25.2	3.65
3	Brazil	↓	3	349.7	50.72	25.3	3.67

TABLE 3.- RESULTS OF GROUND-BASED ENVIRONMENTAL EXPOSURE ON KEVLAR/EPOXY MECHANICAL PROPERTY TEST SPECIMENS

SHORT BEAM INTERLAMINAR SHEAR TESTS

Exposure time, yr	Exposure location	Kevlar/epoxy system	Number of specimens	Average failure stress	
				MPa	ksi
0 (Baseline)	LaRC	F-155	7	47.9	6.94
1	LaRC	↓	3	44.2	6.41
1	California		3	44.2	6.41
1	New Zealand		3	45.7	6.63
1	Hawaii		3	43.8	6.35
1	Germany		3	46.7	6.77
1	Brazil		3	42.5	6.16
3	LaRC	F-155	3	40.1	5.81
3	California	↓	3	42.7	6.19
3	New Zealand		3	38.3	5.55
3	Hawaii		3	41.6	6.03
3	Germany		3	44.1	6.40
3	Brazil		3	36.5	5.29
0 (Baseline)	LaRC	F-161	5	32.4	4.70
1	LaRC	↓	3	33.6	4.88
1	California		3	32.2	4.67
1	New Zealand		3	33.9	4.92
1	Hawaii		3	31.7	4.60
1	Germany		3	31.3	4.54
1	Brazil		3	33.8	4.90
3	LaRC	F-161	3	30.3	4.40
3	California	↓	3	31.8	4.61
3	New Zealand		3	33.2	4.82
3	Hawaii		3	32.4	4.70
3	Germany		3	32.4	4.70
3	Brazil		3	30.9	4.48

TABLE 4.- RESULTS OF GROUND-BASED ENVIRONMENTAL EXPOSURE ON KEVLAR/EPOXY MECHANICAL PROPERTY TEST SPECIMENS

COMPRESSION TESTS

Exposure time, yr	Exposure location	Kevlar/epoxy system	Number of specimens	Average failure stress	
				MPa	ksi
0 (Baseline)	LaRC	F-155	5	137.3	19.92
1	LaRC	↓	3	148.7	21.57
1	California		3	150.0	21.76
1	New Zealand		3	142.1	20.61
1	Hawaii		3	146.9	21.31
1	Germany		3	140.8	20.42
1	Brazil		3	124.3	18.03
3	LaRC	F-155	3	133.5	19.36
3	California	↓	3	134.3	19.48
3	New Zealand		3	125.3	18.17
3	Hawaii		3	126.9	18.40
3	Germany		3	139.6	20.24
3	Brazil		3	143.1	20.76
0 (Baseline)	LaRC	F-161	5	128.0	18.56
1	LaRC	↓	3	136.7	19.83
1	California		3	138.7	20.11
1	New Zealand		3	135.7	19.68
1	Hawaii		3	140.0	20.30
1	Germany		3	133.0	19.29
1	Brazil		3	114.3	16.57
3	LaRC	F-161	3	123.4	17.90
3	California	↓	3	125.1	18.15
3	New Zealand		3	120.7	17.51
3	Hawaii		3	121.1	17.56
3	Germany		3	118.5	17.18
3	Brazil		3	130.3	18.90

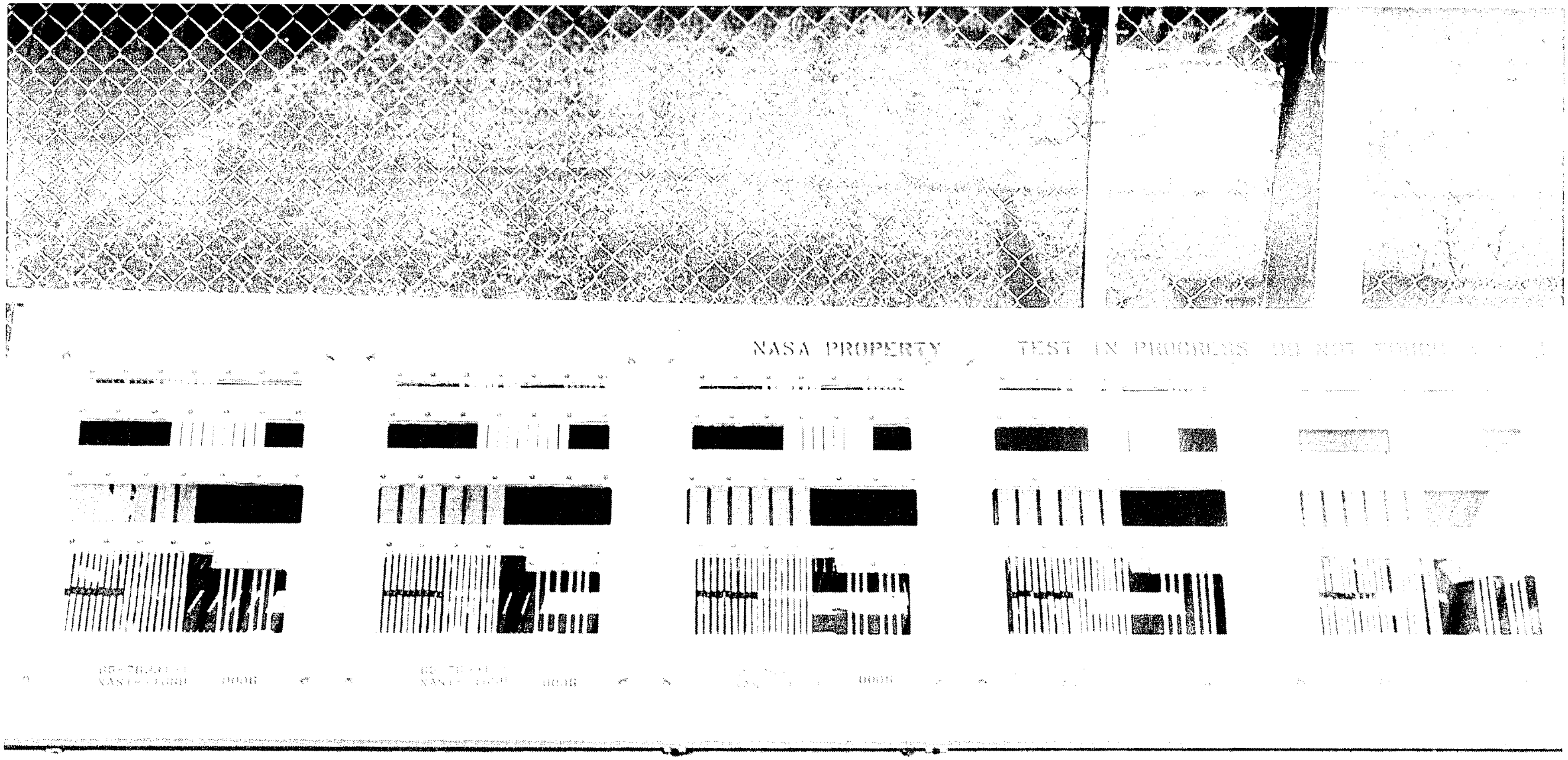

Figure 1. Rack for Holding Environmental Test Specimens

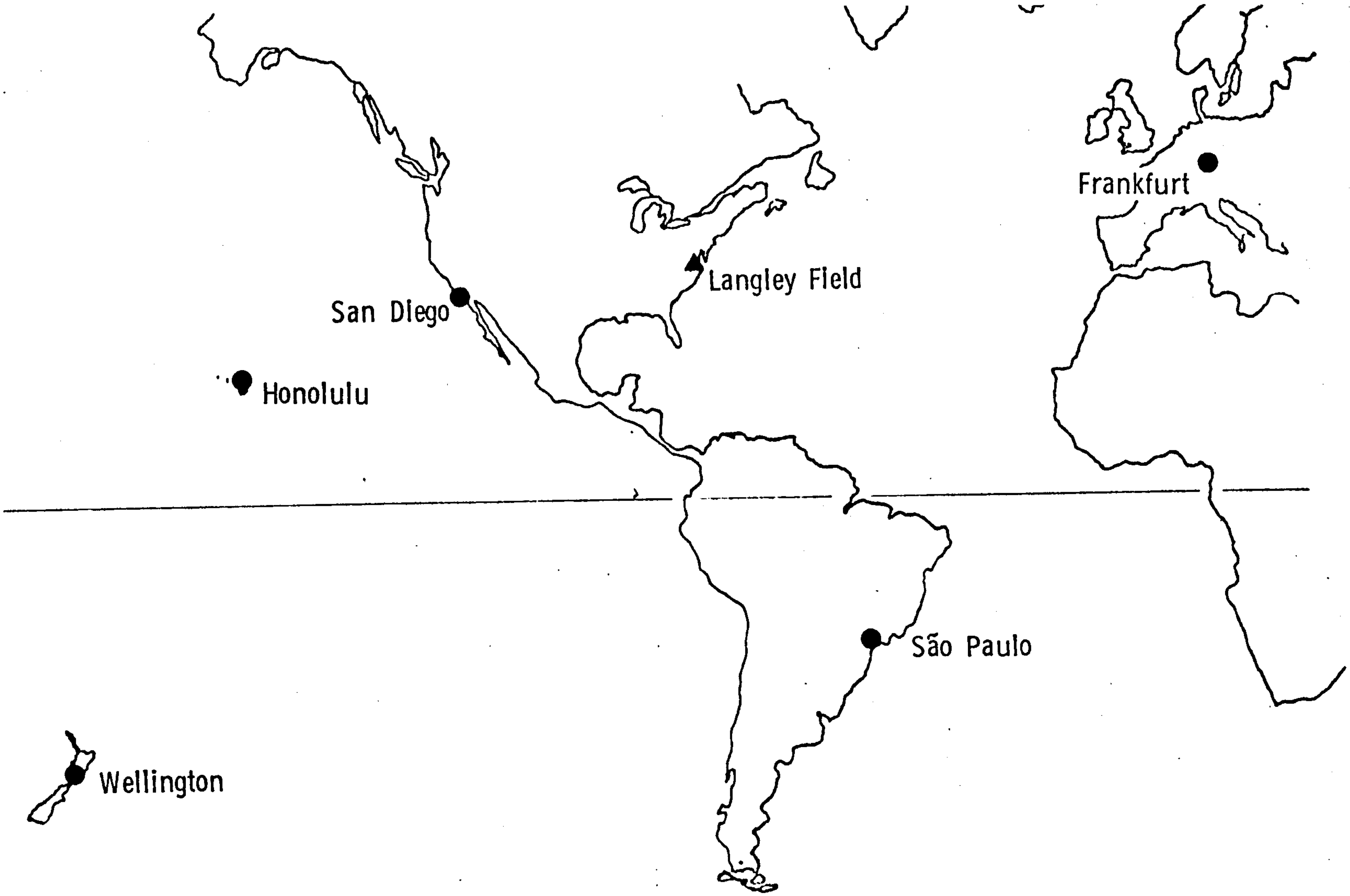

Figure 2. Worldwide Distribution of Environmental Exposure Racks

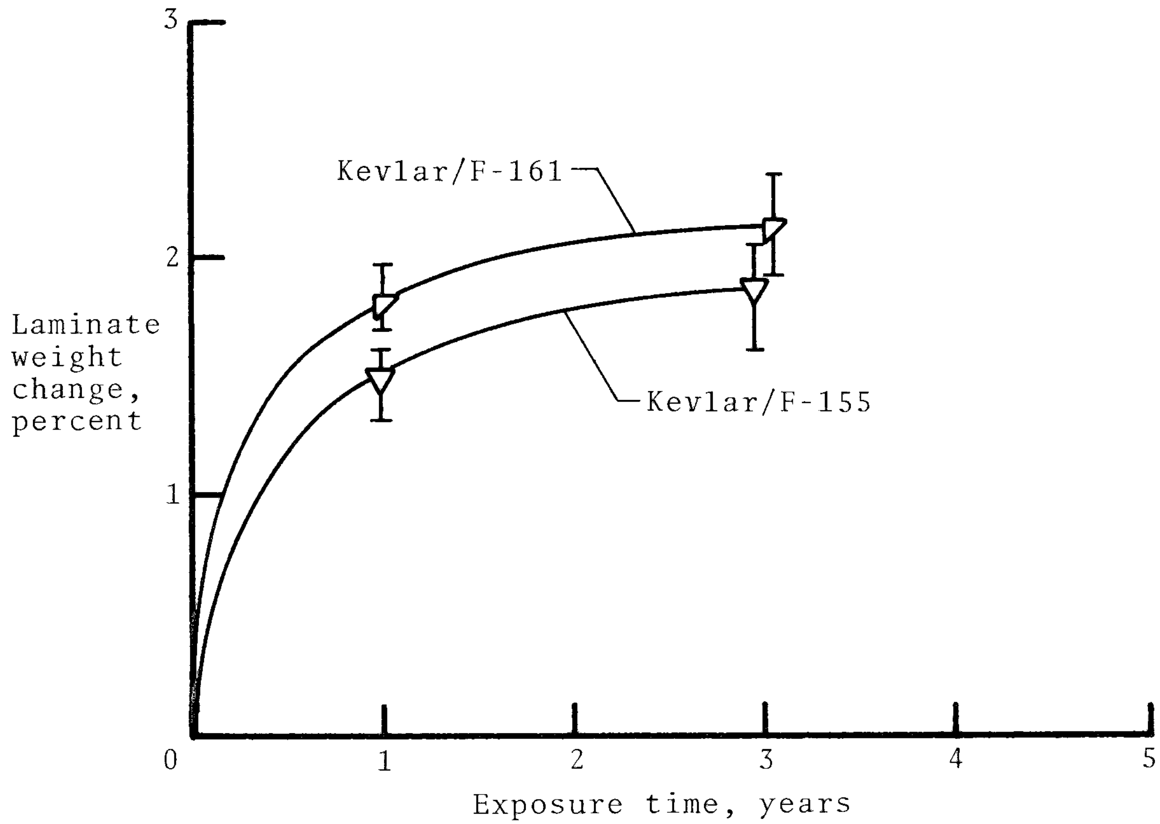

Figure 3. Moisture Pickup for Flexure Coupons After Worldwide Exposures

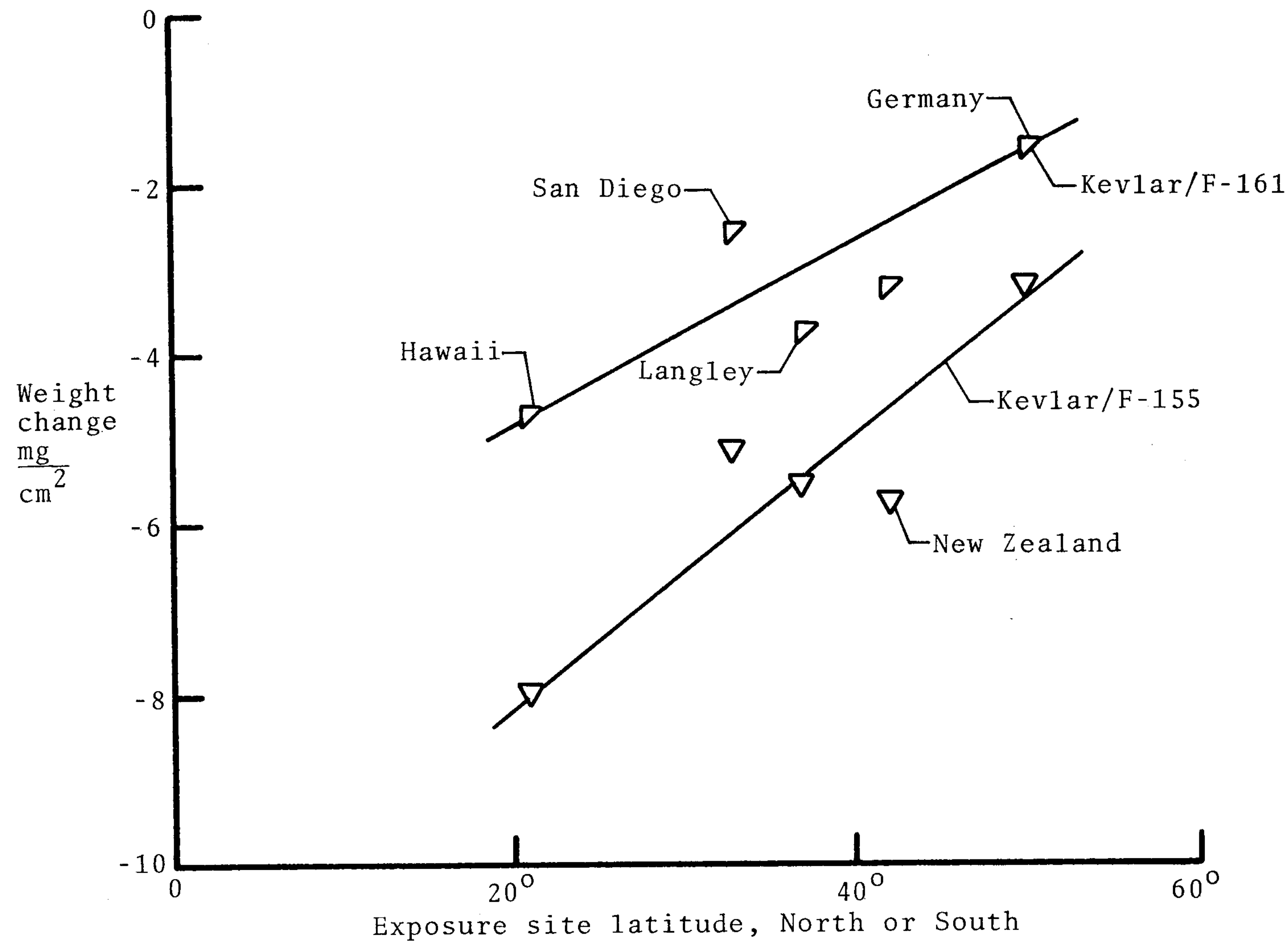

Figure 4. Weight Loss from 3 Years Outdoors UV

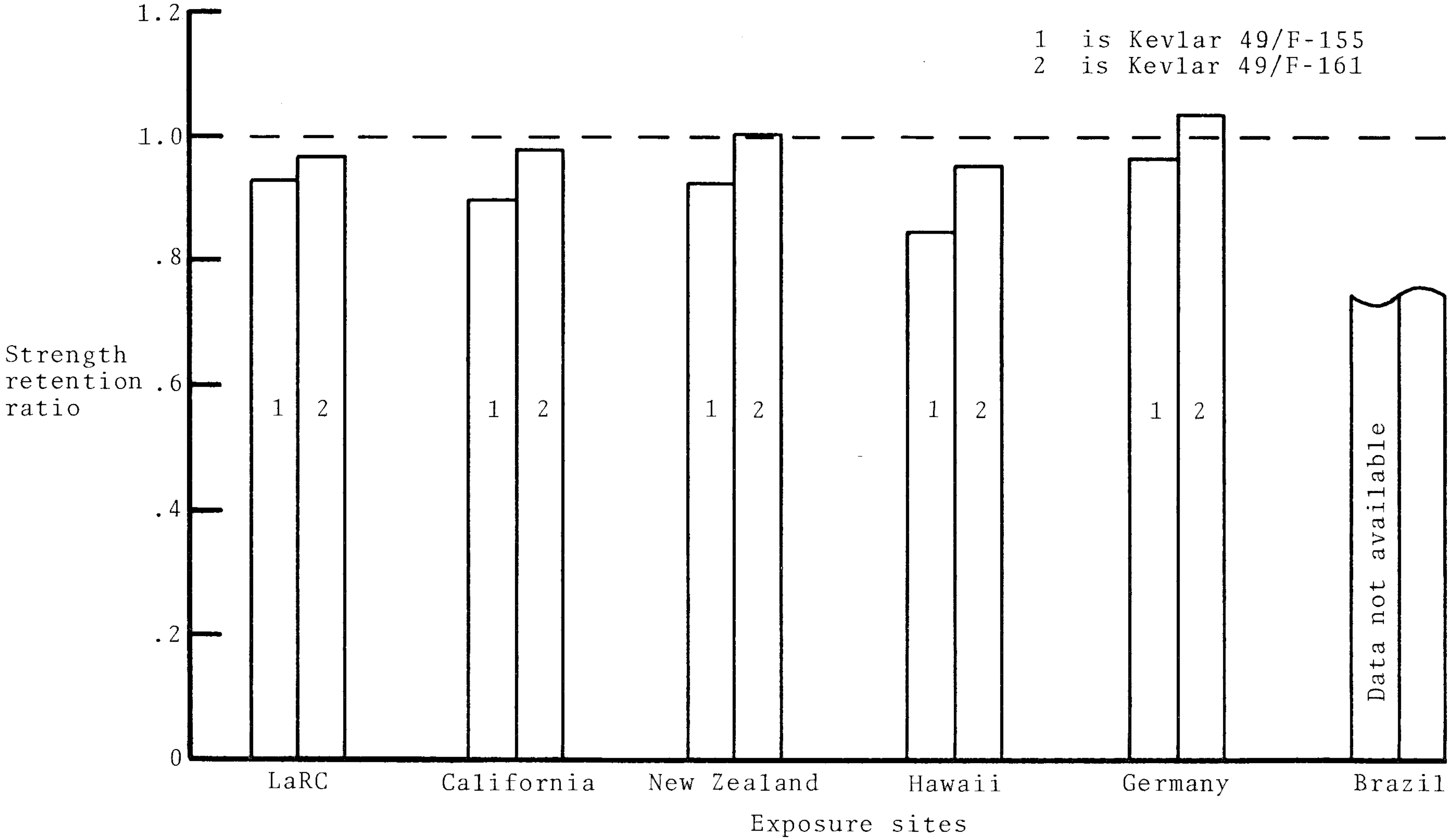

Figure 5. RT Flexure Strength Retention of Kevlar/Epoxy After 1 Year Outdoor Ground-Based Exposure

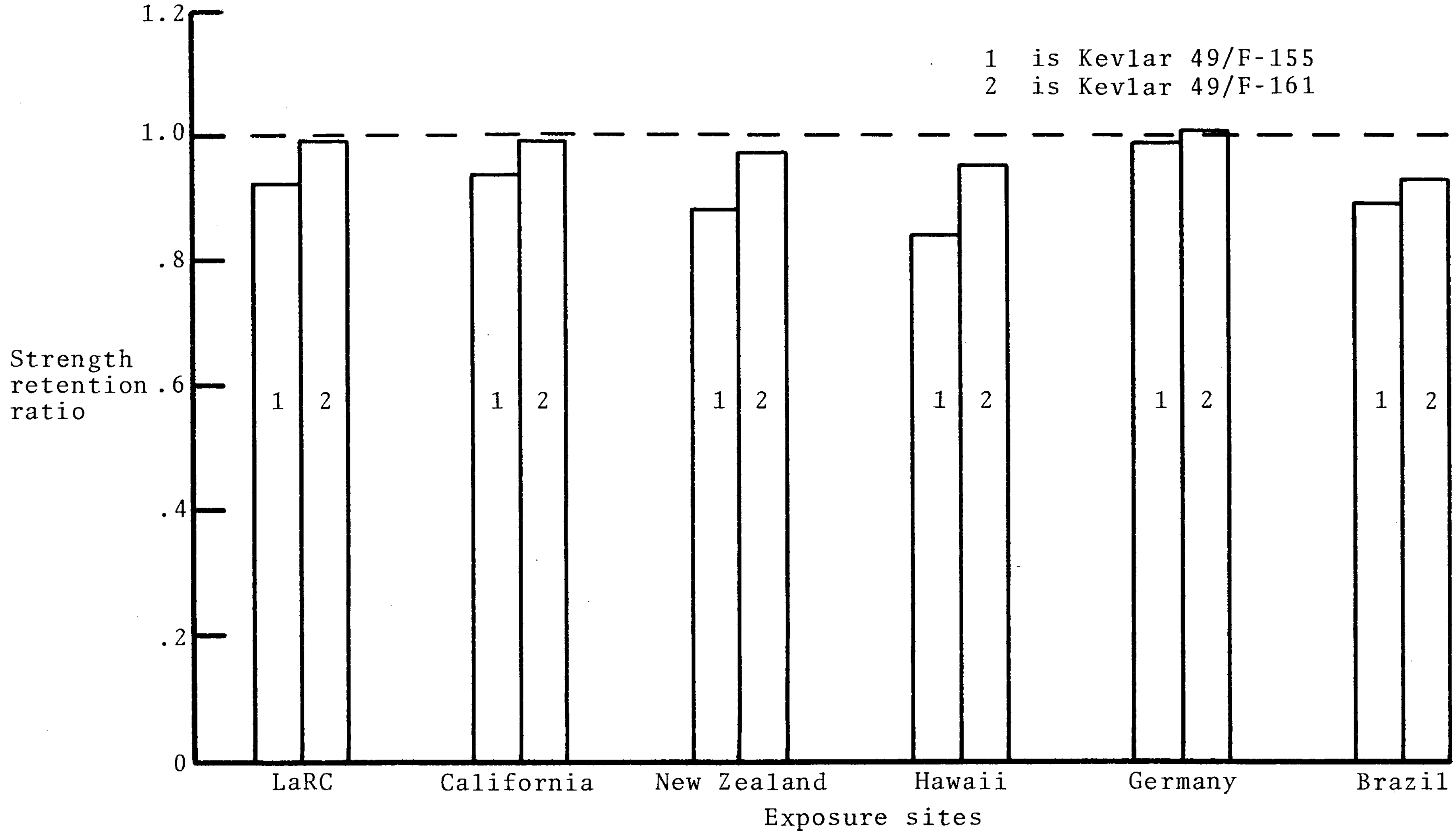

Figure 6. RT Flexure Strength Retention of Kevlar/Epoxy After 3 Years Outdoor Ground-Based Exposure

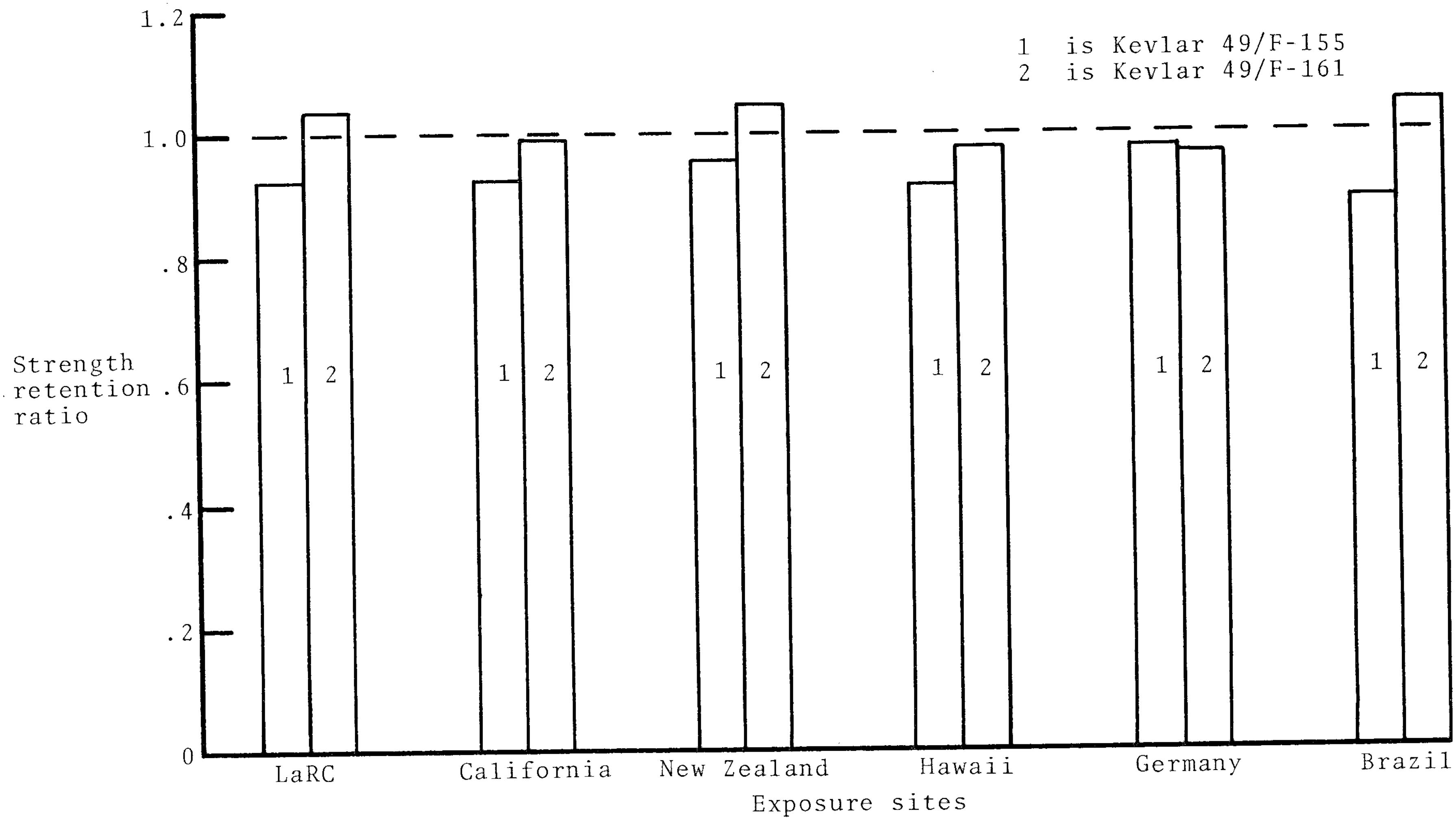

Figure 7. RT Shear Strength Retention of Kevlar/Epoxy After 1 Year Outdoor Ground-Based Exposure

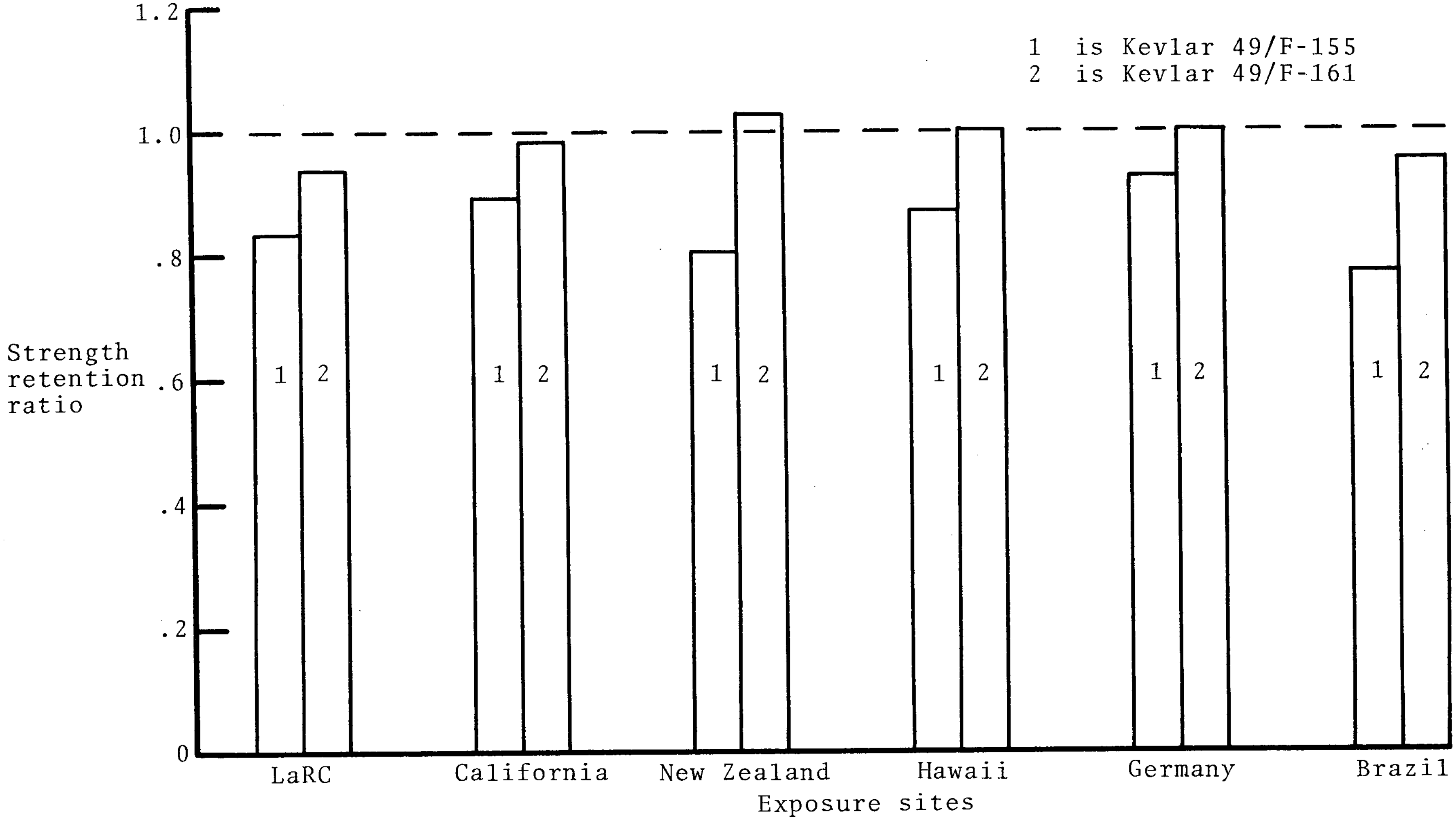

Figure 8. RT Shear Strength Retention of Kevlar/Epoxy After 3 Years Outdoor Ground-Based Exposure

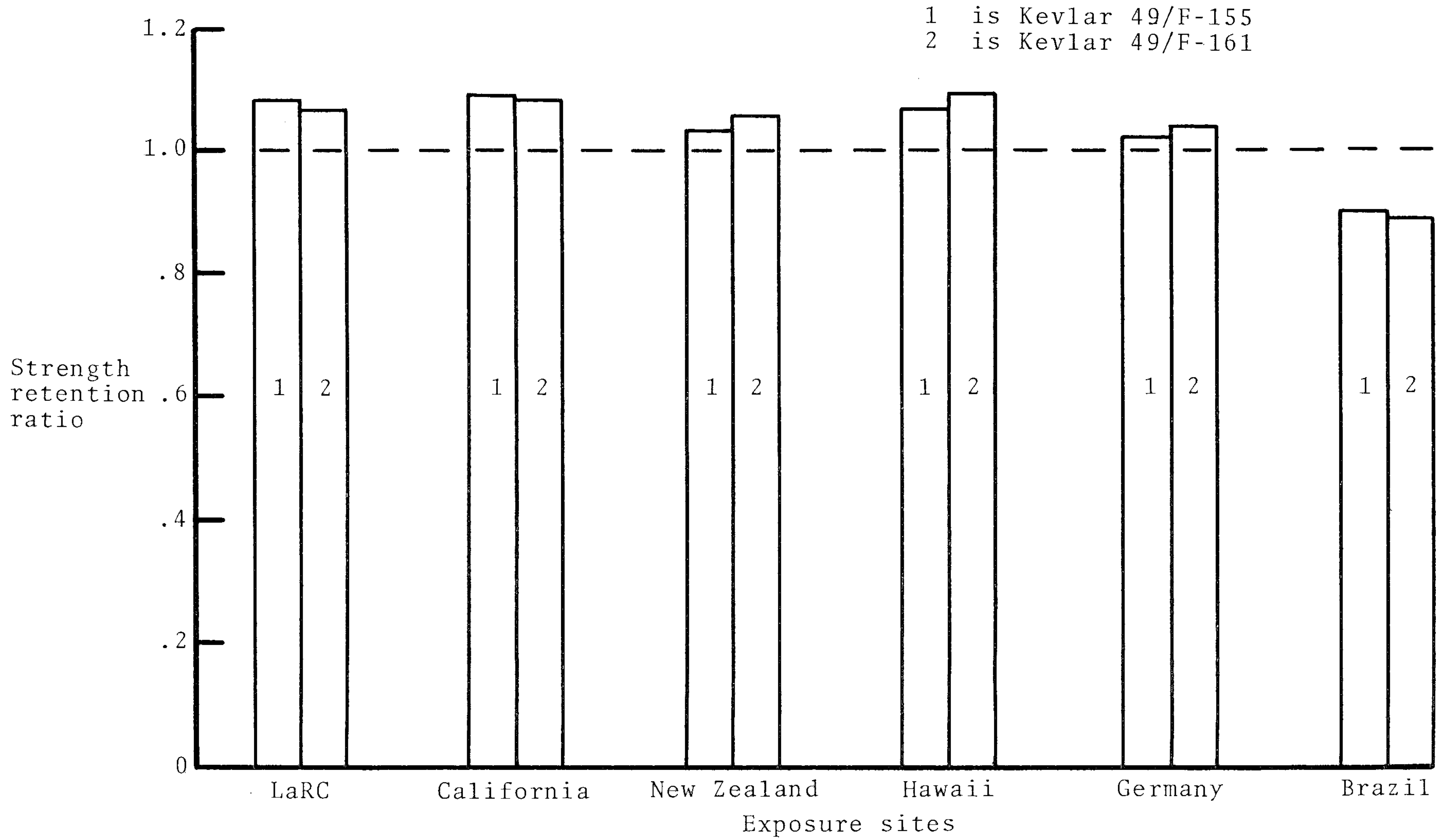

Figure 9. RT Compressive Strength Retention of Kevlar/Epoxy After 1 Year Outdoor Ground-Based Exposure

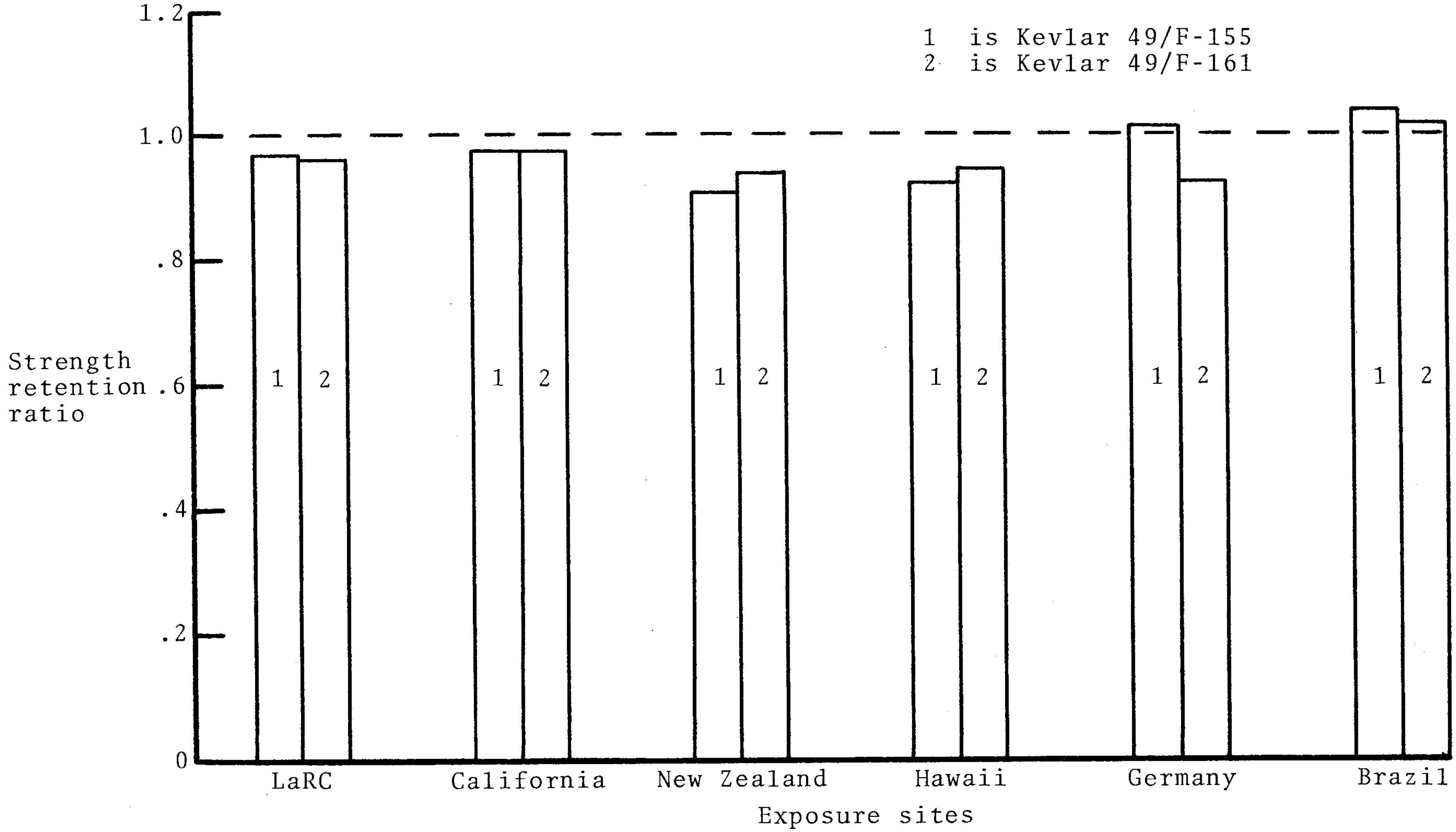

Figure 10. RT Compressive Strength Retention of Kevlar/Epoxy After 3 Years Outdoor Ground-Based Exposure

SECTION 5

SUMMARY OF RESULTS AND CONCLUSIONS

After four years of service and 32,000 flight hours, the Kevlar-49 panels are free of significant damage or defects; and Kevlar-49/epoxy appears to provide service life and structural performance for lightly loaded secondary structures equivalent to fiberglass/epoxy.

The minor damage that has been observed appears related to two factors which are independent of the substitution of Kevlar-49 for fiberglass. These are ground handling damage which appears to have caused the minor cracks in the wing-body fairings; and installation problems which appear to be responsible for some of the fastener hole deformation and fraying. Kevlar-49 appears resistant to damage propagation, as indicated by the absence of crack growth, and the absence of any general occurrence of hole deformation. Also, the limited degree of impact damage indicates Kevlar-49 is at least equivalent to fiberglass in impact resistance. The only condition which occurs on Kevlar-49 parts that is not also seen on fiberglass is fastener hole fraying, and this appears to be primarily the result of the original drilling operation. This condition has no apparent adverse effect on part performance.

The Kevlar-49 panels are for the most part free of skin-core debonds, and have been completely free of visible delaminations within the Kevlar-49 laminates. Two areas of particular concern with Kevlar-49 were the moisture pick-up in the fiber, and the relatively poor fiber-resin interface bond. The absence of any visible delaminations in any of the fairings is an indication that these are not serious problems for Kevlar-49 in lightly loaded parts. There is also no evidence of deleterious effects on the Kevlar-49 parts from exposure to the service environment, moisture, or aircraft fluids.

The serviceability of the reworked TWA panels to date indicates that standard fiberglass repair materials and procedures can be used for repair of Kevlar-49 parts, thus requiring no modification of airline maintenance procedures. In those cases where minor defects have been noted which may be related to the repairs, the problem appears to have been in selection of a less effective standard repair than could have been used.

The concurrent NASA evaluation of Kevlar-49/epoxy coupons under ground exposure conditions indicates that the material withstands extended exposures to moisture and ultraviolet without significant degradation of mechanical properties. This is for a wide range of climatic conditions including severe hot/wet and cold/wet climates, and for exposure periods of three years.

The test coupons were unpainted and the combined effects of moisture and ultraviolet (UV) produced a net weight loss. Estimates of moisture pickup agreed reasonably well with laboratory data from other sources. The UV weight loss is significant, but these effects will be prevented in service by painting. The results indicate that Kevlar-49 parts will be able to withstand any UV exposure resulting from paint loss.

The lowest retention of mechanical properties observed after three years has been 76% with most values well above 80%. The 350°F curing F-161 system has somewhat greater retention of properties in nearly all cases than the 250°F curing F-155. Between the one year and three year exposure periods, slight further reductions in compression and in F-155 shear have been noted, but the total reduction is still not significant. These retention values, in summary, indicate that no significant degradation of the Kevlar-49/epoxy system is occurring under real-time outdoor exposure.

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3. Stone, R.H.; Flight Service Evaluation of Kevlar-49/Epoxy Composite Panels in Wide-Bodied Commercial Transport Aircraft - Second Annual Flight Service Report. NASA CR-132733, October 1975.

4. Stone, R.H.; Flight Service Evaluation of Kevlar-49/Epoxy Composite Panels in Wide-Bodied Commercial Transport Aircraft - Third Annual Flight Service Report. NASA CR-145141, March 1977.

5. Pride, Richard A.; and Dow, Marvin B: Environmental Exposures of Advanced Composites. Third conference on Fibrous Composites in Flight Vehicle Design - Part I, NASA TMX - 3377, 1976, pp. 455-462.

APPENDIX I

DETAIL OBSERVATIONS OF KEVLAR-49 FAIRING PANELS - EASTERN AIR LINES AIRCRAFT N314EA (SERIAL NO. 1022), MAY 1977

Two of the six Kevlar-49 fairings were removed for weighing and inspection of fastener holes and the inner surface. These were the left-hand wing-body sandwich fairing and the left-hand underwing fillet panel. The right-hand wing-body fairing and underwing fillet panels were inspected in place on the aircraft. Detail observations on these parts are outlined below:

LEFT-HAND WING-BODY FAIRING (P/N 1515599-109)

1) The panel weight was 6.92kg. (15 1/4 lbs.) The original weight was 7.03kg (15 1/2 lbs.)

2) A small area 1.9 cm by 0.6 cm. (3/4 inch by 1/4 inch) was noted in the lower aft area of the exterior surface with loss of both paint and flame spray, but no apparent damage to the Kevlar-49 surface.

3) The tape patch noted in the 1975 and 1976 inspections was still in place, unchanged in appearance since the 1976 inspection. (Figure 11)

4) A delaminated area 1.9 cm. by 19.1 cm. (3/4 inch by 7 1/2 inch) was observed on the inner surface unchanged in appearance or extent since it was observed in the 1975 inspection.

5) Slight fraying of the fastener holes was visible from the exterior on all four edges, with more fraying noted on the bottom edge. Heavier fraying was noted on three holes, one on the top edge and two on the aft edge. (Figure 12)

6) A slight convexity was noted around the fastener holes on the bottom edge, inner surface.

LEFT-HAND WING-BODY FAIRING (P/N 1515599-109) (Cont.)

7) Considerable paint chipping and re-painting was noted around the fastener holes. (Figure 13)

8) No evidence of Skydrol contamination.

LEFT-HAND UNDERWING FILLET (P/N 1545328-109)

1) The panel weight was 1.02kg. (2 1/4 lbs.)

2) Paint is missing, with the Kevlar-49 surface exposed in extensive areas of the upper fillet. (Figure 14)

3) Fraying of fastener holes is visible from the exterior, but more noticeable viewed from the inner surface. Several elongated holes were noted which were the same ones observed in the 1975 inspection. No increase in elongation was noted. Fraying was more pronounced on the elongated holes. (Figures 15, 16, 17)

4) A slight gouged spot was noted on the upper forward area of the inner surface.

RIGHT-HAND WING-BODY FAIRING (P/N 1515599-110)

1) The following cracks observed in previous inspections were unchanged in appearance or extent:

- 1.3 cm. (1/2 in.) crack at forward edge between 5th and 6th holes. No associated delamination. (Figure 18)
- 0.6 cm. (1/4 in.) crack near exact center (may or may not be into skin) with scratch through paint to flame spray extending upward 20.3 cm. (8 in.). No delamination.
- 0.3 cm. (1/8 in.) ding lower center - definitely into skin. No delamination.
- 0.8 cm. (5/16 in.) crack in lower forward area - with slight associated delamination. 1.6 cm. (5/8 in.)
- 0.3 cm. (1/8 in.) crack aft center - no associated delamination.

RIGHT-HAND WING-BODY FAIRING (P/N 1515599-110) (Cont.)

2) An additional crack 0.8 cm. (5/16 in.) was noted in the upper forward area. This crack may have been only in the paint.

3) Fasteners were in alignment on all edges. Considerable paint loss occurred along the edges, but flame spray was intact.

RIGHT-HAND WING-BODY FILLET (P/N 1545328-110)

1) All fasteners were in line with no evidence of installation problems.

2) Slight paint loss noted in lower section.

LEFT-HAND AFT ENGINE FAIRING (P/N 1538592-129)

(Inspected by EAL Maintenance Engr. 9-15-77)

1) A puncture was observed on the outer surface in the upper aft area. This was a triangular area, 1.9 cm. (0.75 in.) on three sides. The area was covered with tape to prevent moisture entry.

2) The inner surface was free of defects or damage.

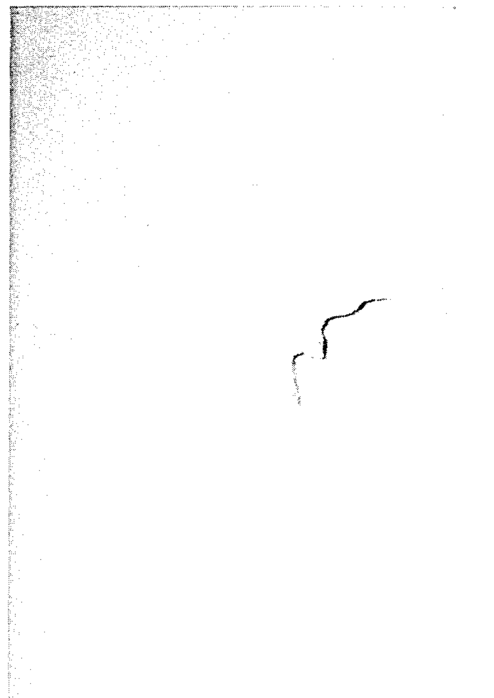

Figure 11. Eastern LH Wing-Body Fairing - External Tape Patch

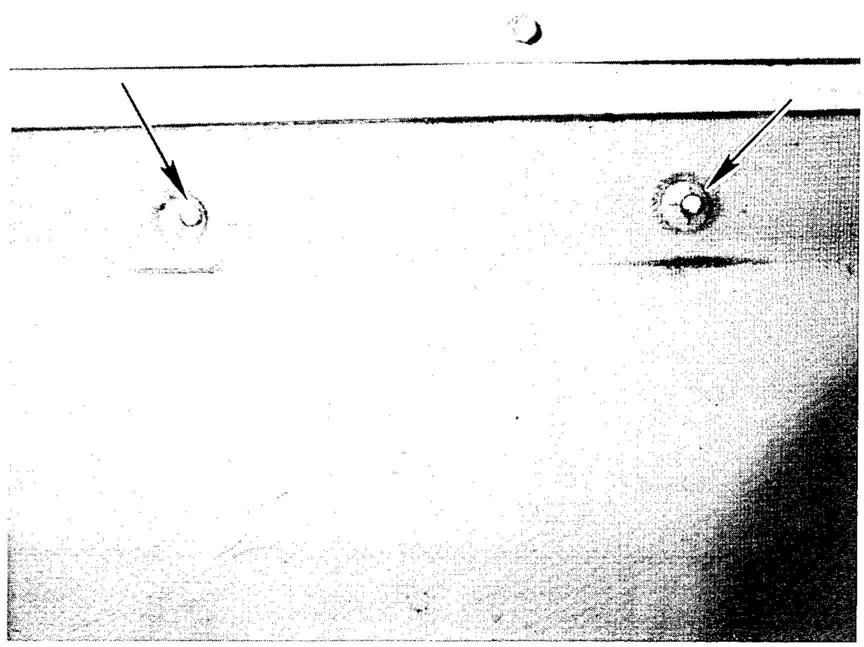

Figure 12. Eastern LH Wing-Body Fairing - Frayed Fastener Holes, Lower Aft Edge on Inner Surface

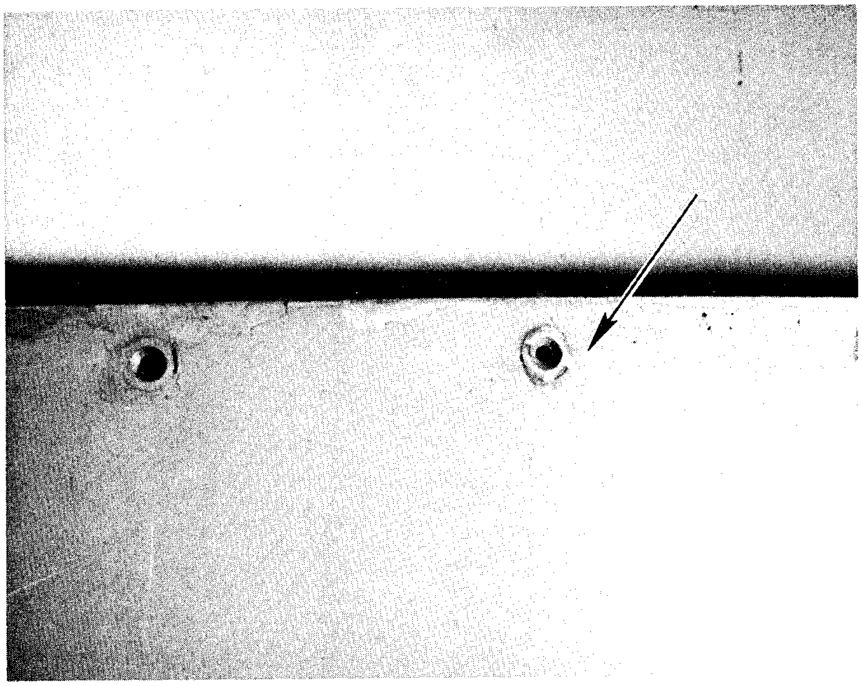

Figure 13. Eastern LH Wing-Body Fairing - Paint Loss Areas Around Fastener Holes, Exterior

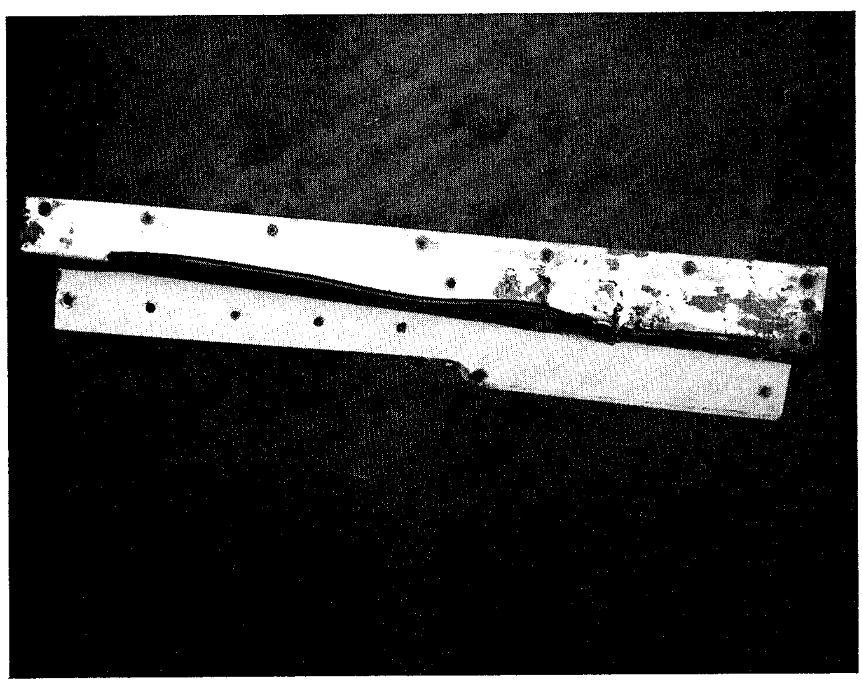

Figure 14. Eastern LH Underwing Fillet - Exterior Surface

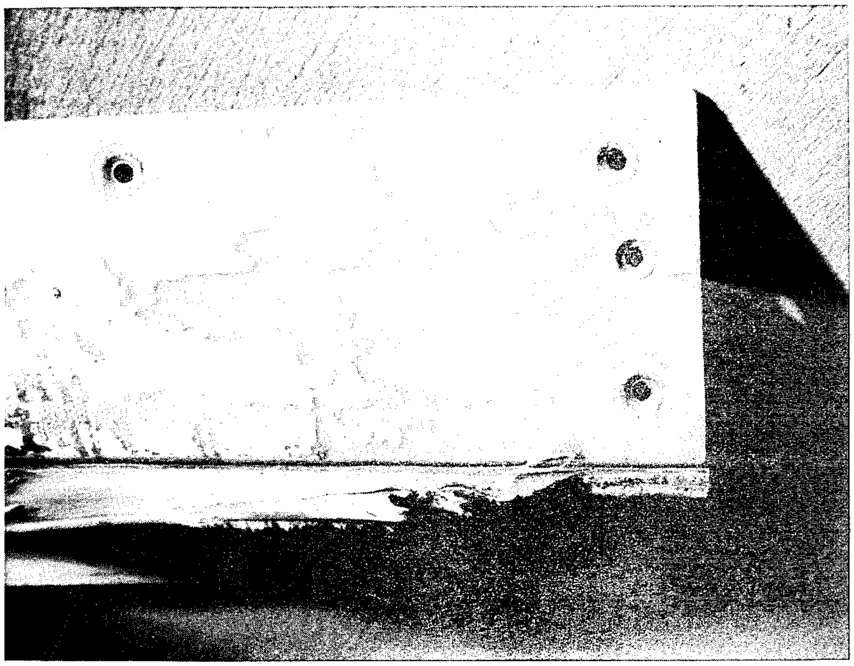

Figure 15. Eastern LH Underwing Fillet - Fastener Holes, Upper Aft Corner, from Exterior

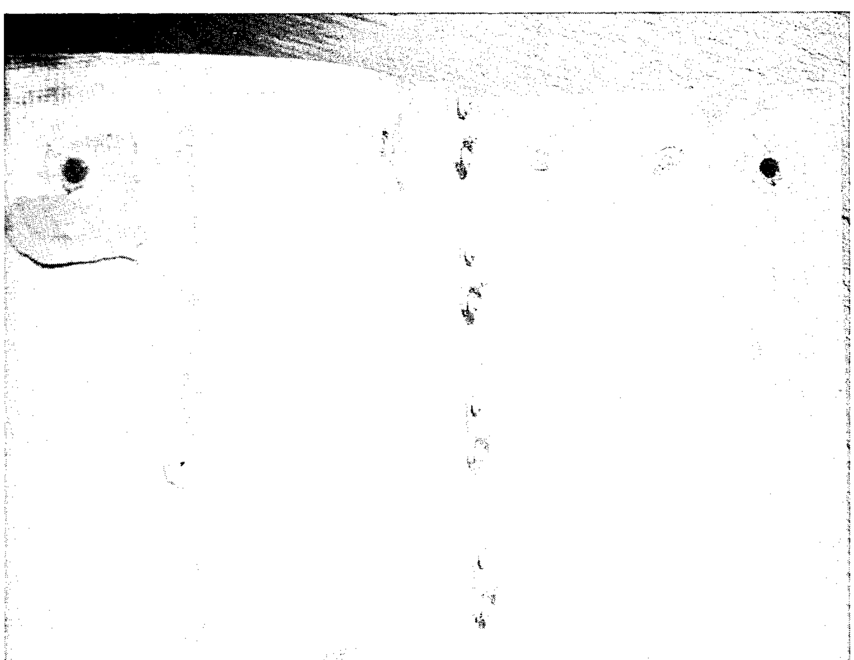

Figure 16. Eastern LH Underwing Fillet - Fastener Holes, Upper Aft Corner, from Inner Surface

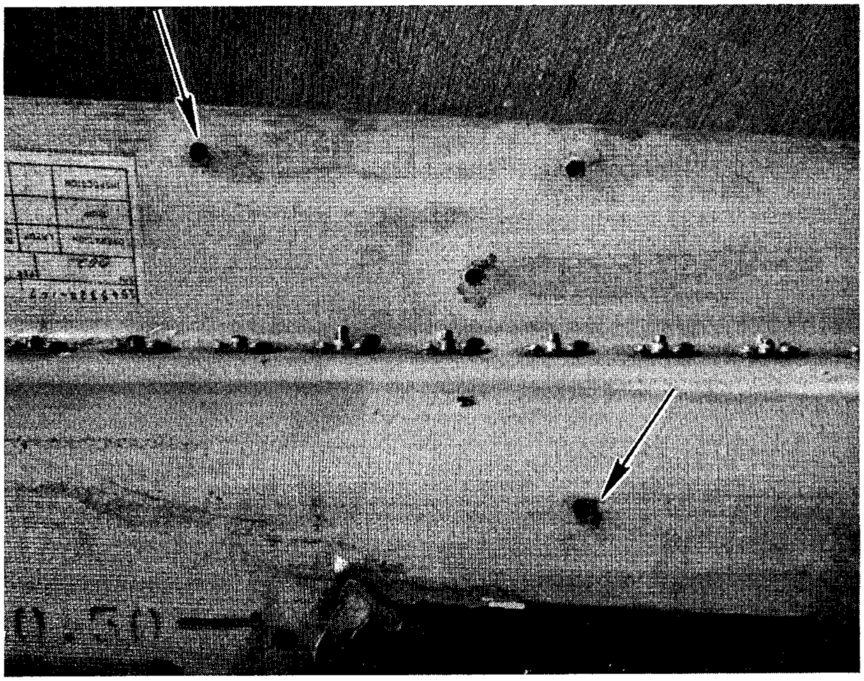

Figure 17. Eastern LH Underwing Fillet - Frayed Fastener Holes from Inner Surface

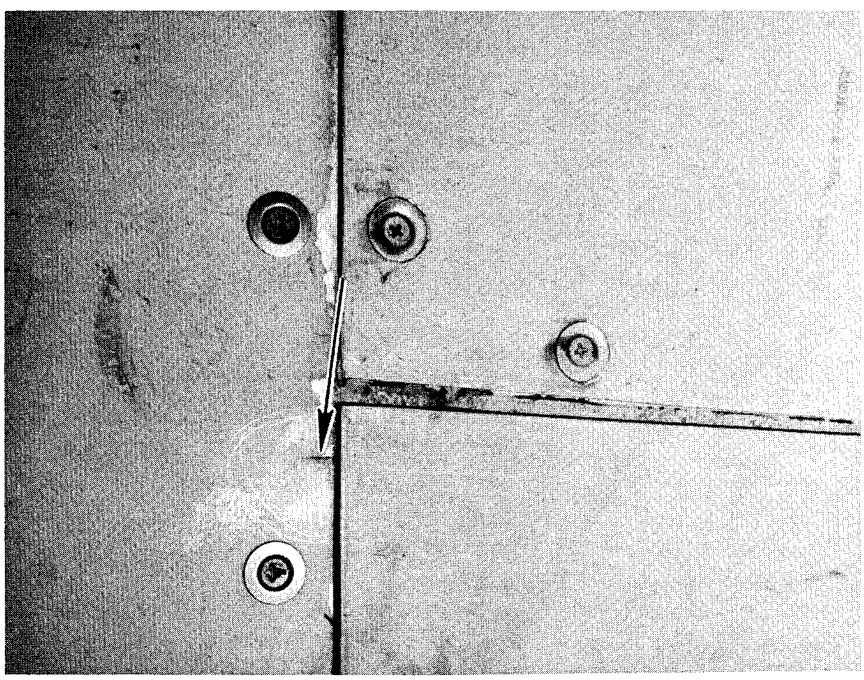

Figure 18. Eastern RH Wing-Body Fairing - Crack in Forward Edge 0.3 cm. (1/8 in.) Length

APPENDIX II

DETAIL OBSERVATIONS OF KEVLAR-49 FAIRING PANELS-AIR CANADA AIRCRAFT CF-TNB-502 (SERIAL NO. 1021), SEPTEMBER 1977

Five of the six Kevlar-49 fairings were removed for weighing and inspection of fastener holes and the inner surface. These were both wing-body fairings and both underwing fillet panels, and the left-hand aft engine fairing. Detail observations on these parts are outlined below:

LEFT-HAND WING-BODY FAIRING (P/N 1515599-109)

1) The panel weight was 697 kg (15 3/8 lbs.). Previous panel weight measured in 1976 was 672 kg (14.81 lbs.).

2) A 3.2 cm. (1 1/4 in.) crack was noted in the upper, aft area. (Figure 19). This crack has not grown or delaminated further since it was first observed in 1975. No other cracks were observed, and a 1.3 cm. (1/2 in.) crack observed in 1976 was apparently only in the original paint layer and had been repainted.

3) Fastener holes appeared slightly frayed as viewed from exterior and inner surfaces, with a greater degree of fraying noted on the bottom edge holes as viewed from the inner surface. (Figure 20). Slight hole elongation was observed on all fastener holes on the bottom edge, plus two other holes on the forward and top edges.

4) Extensive paint loss was observed on edges and upper forward area, but no assocated damage in the part. (Figure 21).

5) No defects or damage were observed on the inner surface.

RIGHT-HAND WING-BODY FAIRING (P/N 1515599-110)

1) The panel weight was 7.14 kg (15 3/4 lbs.). Original panel weight was 7.03 kg (15 1/2 lbs.).

RIGHT-HAND WINGBODY FAIRING (P/N 1515599-110) (Cont.)

2) Two small cracks were observed: an 0.6 cm (1/4 in.) crack in the center, forward area first noted in 1975, and a 0.3 cm (1/8 in.) crack first noted in 1976. They had not grown or propagated since first observed, and still had no associated delamination.

3) Two disbonded areas were observed on the inner surface in the upper center area; one area was irregularly shaped 12.7 cm (5 in.) long varying from 0.6 cm (1/4 in.) to 1.3 cm (1/2 in.) width; and a nearby area 2.5 cm (1 in.) by 1.3 cm (1/2 in.). (Figure 22). An area 26.7 cm (10 1/2 in.) by 3.2 cm (1 1/4 in.) slightly below the delaminated areas appeared to have been sanded. The sanding appeared to be on an overlay which may have been a repair patch. A third disbond was noted in the lower, aft area 10.2 cm (4 in.) long varying from 1.3 cm (1/2 in.) to 2.2 cm (7/8 in.) wide. The inner surface of this panel had not previously been inspected.

4) Slight fraying of fastener holes was observed on the upper, aft, and forward edges as viewed from either surface. A greater degree of fraying was observed on the bottom edge fastener holes. Several fastener holes were elongated to a slight degree on the bottom and aft edge. (Figure 23). One hole in the lower, forward, corner was elongated to 1.1 cm (7/16 in.) maximum dimension from the original 0.5 cm (3/16 in.) diameter. All lower edge holes had a slight convex deformation of the laminates around them with markings from the fastener heads.

LEFT-HAND UNDERWING FILLET (P/N 1545328-109)

1) The panel weight was 0.85 kg (1 7/8 lbs.). Previous weight in 1976 was 0.6 kg (1.31 lbs.).

LEFT-HAND UNDERWING FILLET (P/N 1545328-109) (Cont.)

2) No surface damage or defects were noted on either surface but considerable paint loss was observed in the upper exterior surface with the Kevlar-49 surface exposed. (Figure 24).

3) Slight fraying was observed on all fastener holes. Hole deformation was noted on four holes in the lower area, and seven holes in the upper area. (Figure 25). Four of these were only slight deformation, but the others had maximum dimensions of 0.6 cm (1/4 in.), and 1.1 cm (7/16 in.) in some cases from the original 0.5 cm (3/16 in.) diameter. This represented some increase in the incidence of hole deformation over that observed in 1976, specifically three of the holes in the lower area which were noted to have a slight deformation in 1977, and three of the holes in the upper area observed to have deformation up to 1.1 cm (7/16 in.) in 1977.

RIGHT-HAND UNDERWING FILLET (P/N 1545328-110)

1) The panel weight was 0.85 kg (1 7/8 lbs.).

2) No damage or defects were noted on either surface, but considerable paint loss was observed on the upper area with the Kevlar-49 surface exposed. (Figure 26).

3) Slight fraying was observed around all fastener holes. Hole deformation was noted on eight holes in the upper area, four to a slight degree and the others deformed to a maximum dimension of 1.1 cm (7/16 in.) (Figure 27) from the original 0.5 cm (3/16) diameter. One hole on the lower edge was slightly elongated. This represented an increase in the observed number of elongated holes since their

last inspection in 1975. The additional elongations were all on the top edge including three of the holes which showed slight elongation, and three of the holes were elongated to 1.1 cm (7/16 in.).

LEFT-HAND AFT ENGINE FAIRING (P/N 1538592-129)

1) The panel weight was 2.89 kg (6 3/8 lb.).

2) No surface damage or defects were noted. (Figure 28).

3) Extensive fraying of fastener holes was observed on all edges as viewed from both surfaces. (Figure 29). Loose Kevlar-49 fibers were noted inside holes. The intercostal holes through the core area were more frayed than holes in the edge laminate. (Figure 30). Elongation was noted only to a very slight degree on three holes along the bottom edge.

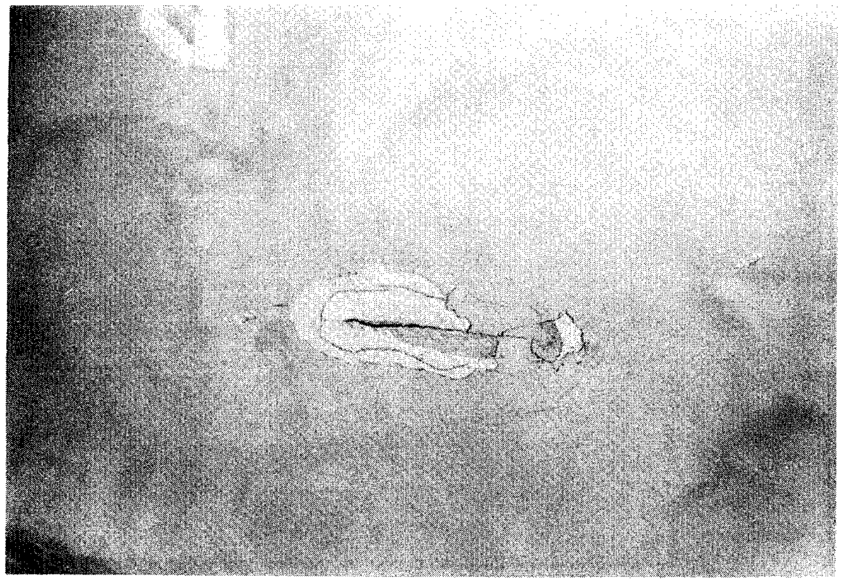

Figure 19. Air Canada LH Wing-Body Fairing - Crack, Exterior Surface, 3.2 cm (1-1/4 in.) Length

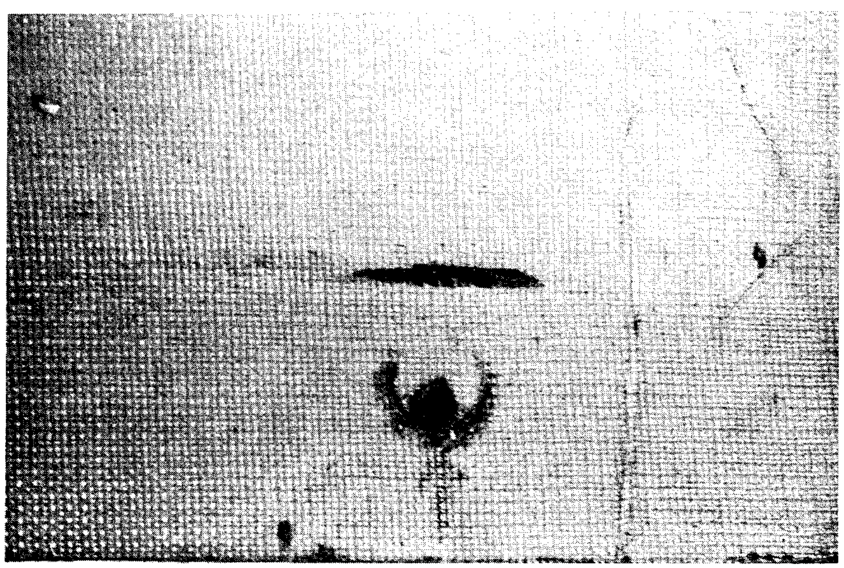

Figure 20. Air Canada LH Wing-Body Fairing - Close-Up of Fastener Holes on Bottom Edge, Inner Surface

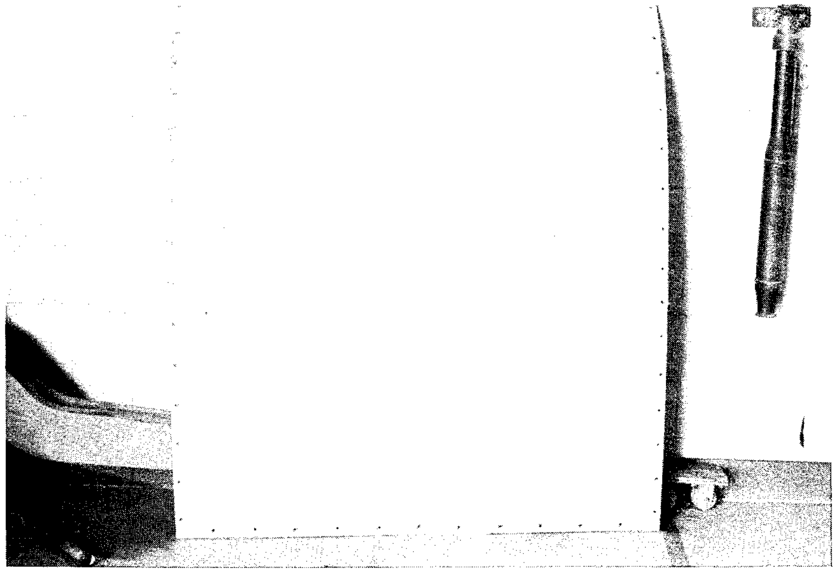

Figure 21. Air Canada LH Wing-Body Fairing - Exterior

Figure 22. Air Canada RH Wing-Body Fairing - Outline of Disbond Areas on Inner Surface - 12.7 cm (5 in.) by 1.3 cm (1/2 in.); and 2.5 cm (1 in.) by 1.3 cm (1/2 in.)

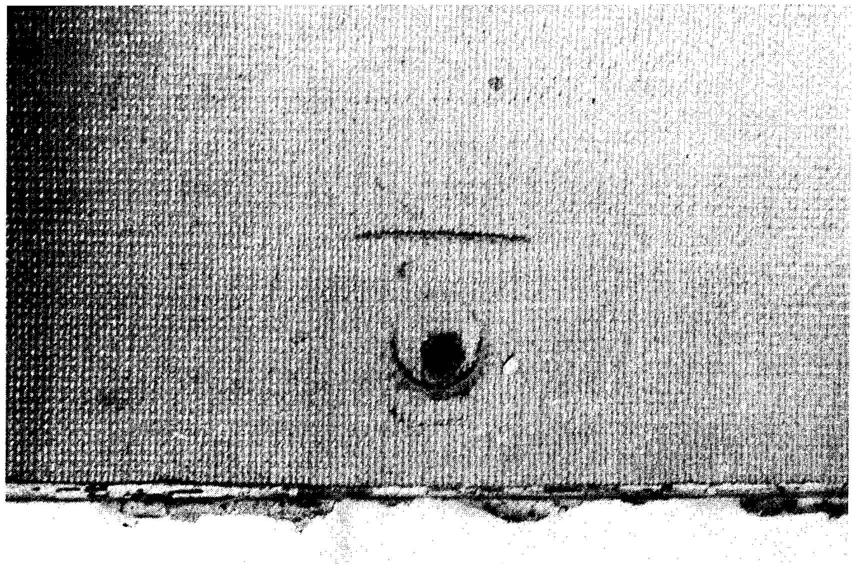

Figure 23. Air Canada RH Wing-Body Fairing - Close-Up of Fastener Holes, Bottom Edge, from Inner Surface

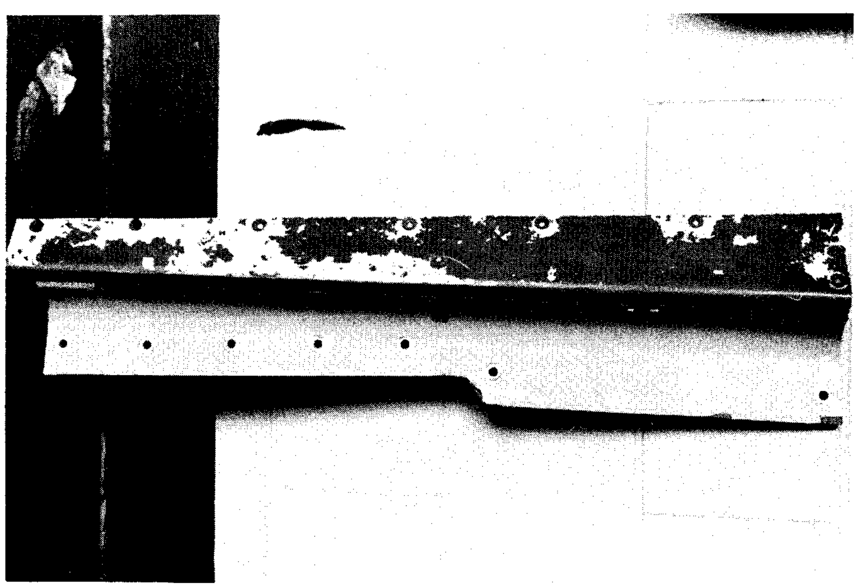

Figure 24. Air Canada LH Underwing Fillet - Exterior Surface

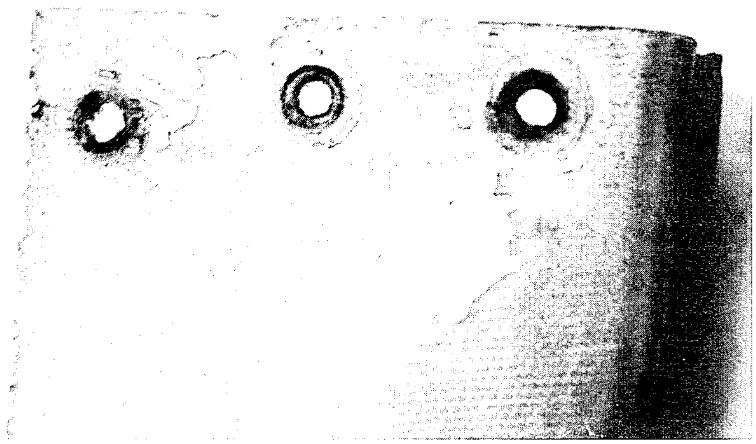

Figure 25. Air Canada LH Underwing Fillet - Fastener Holes, Upper Aft Area, from Exterior

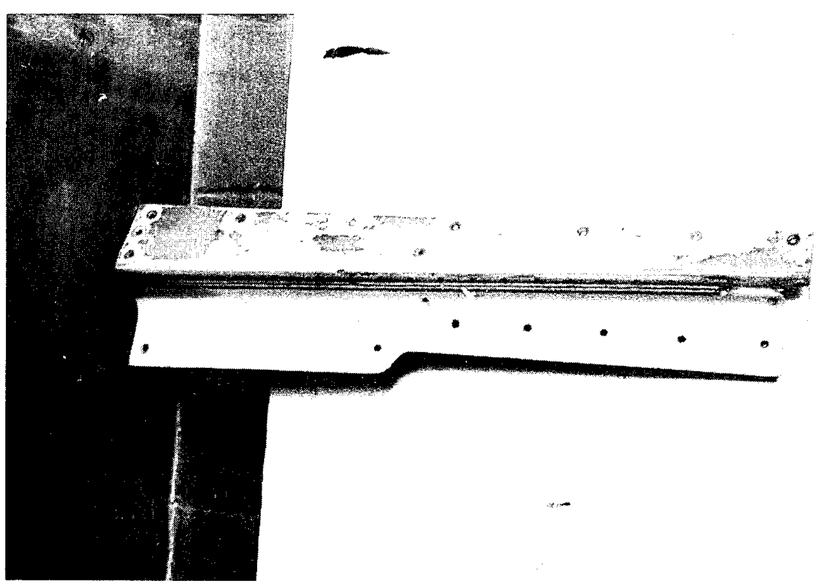

Figure 26. Air Canada RH Underwing Fillet - Exterior Surface

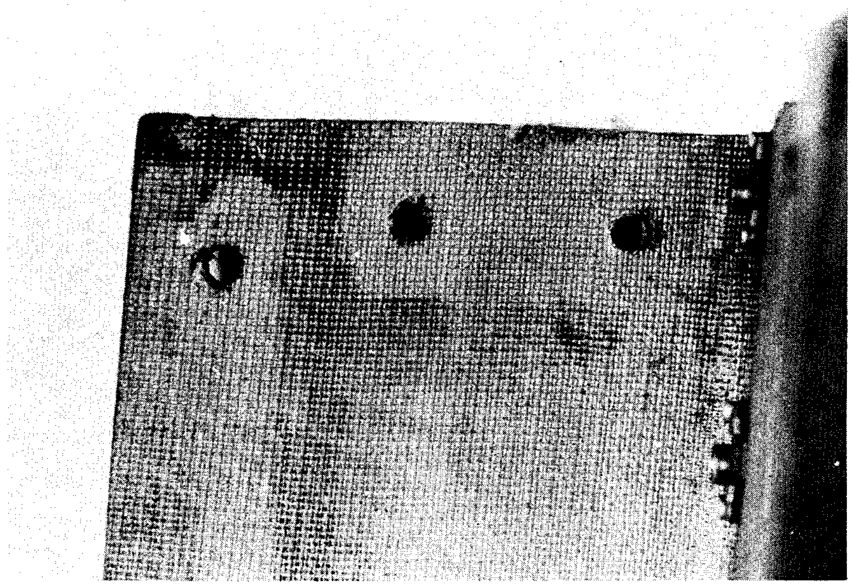

Figure 27. Air Canada RH Underwing Fillet - Fastener Holes, Upper Aft Area, from Inner Surface

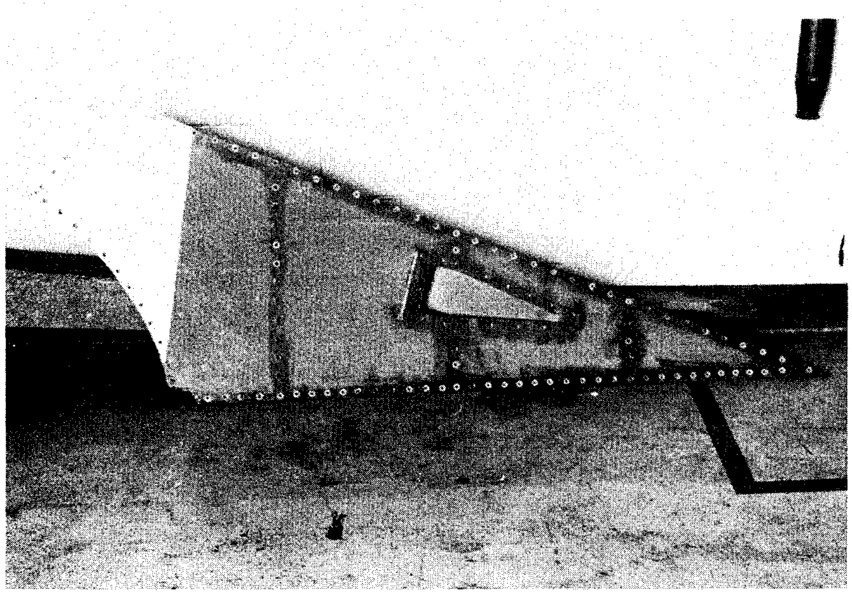

Figure 28. Air Canada LH Aft Engine Fairing - Exterior Surface

Figure 29. Air Canada LH Aft Engine Fairing - Frayed Fastener Holes, Bottom Edge, from Inner Surface

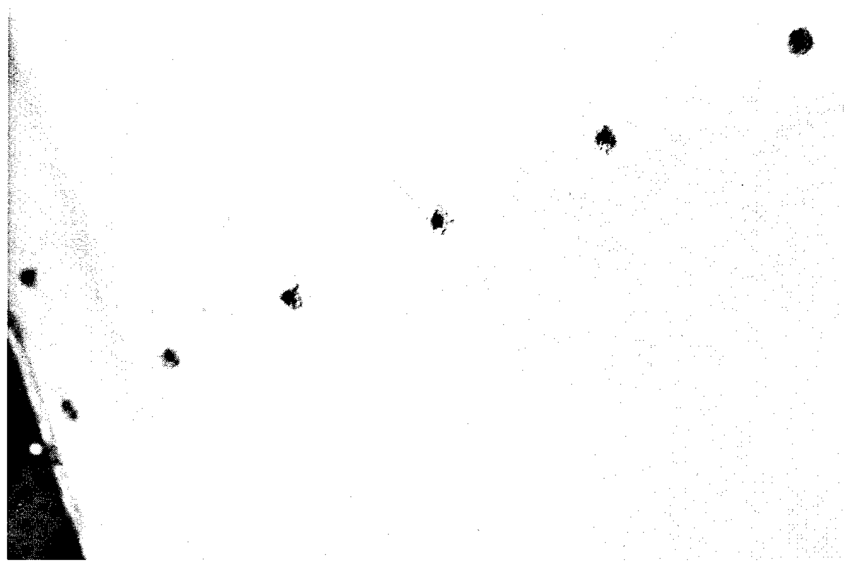

Figure 30. Air Canada LH Aft Engine Fairing - Frayed Intercostal Holes Through Core Area

APPENDIX III

DETAIL OBSERVATIONS OF KEVLAR-49 FAIRING PANELS-TWA AIRCRAFT N31030 (SERIAL NO 1111), NOVEMBER 1977

Three of the six Kevlar-49 fairings were removed for weighing and inspection of fastener holes and the inner surface. They were the left-hand wing-body fairing, the left-hand underwing fillet, and the right-hand aft engine fairing. The other three panels were inspected in place on the aircraft. Detail observations on these parts are outlined below. Panel weights could not be determined, because of the gross inaccuracy of the only available scales.

LEFT-HAND WING-BODY FAIRING (P/N 1515599-109)

1) A small crack 0.3 cm (1/8 in.) in length was observed in the center, aft area. This crack had not been previously observed.

2) The outer and inner surfaces showed no other defects or damage, although some paint loss was observed on the forward edge. (Figure 31).

3) Slight fraying of fastener holes was visible from the inner surface only. (Figure 32). Six holes were observed to have deformation with a maximum dimension of 0.55 cm (7/32 in.) to 0.6 cm (1/4 in.) from the nominal 0.5 cm (3/16 in.) diameter. These were randomly located on the top, forward, and bottom edges.

LEFT-HAND UNDERWING FILLET (P/N 1545328-109)

1) No damage or defects were observed on either the inner or outer surfaces. (Figure 33). A paint loss area on the upper surface had been repainted since the 1976 inspection.

2) Extensive fraying was visible from both the inner and outer surfaces on several fastener holes in the upper fillet sections, which were also slightly elongated. (Figure 34). These included three holes

LEFT-HAND UNDERWING FILLET (P/N 1545328-109)

along the upper edge, one of which was elongated to a maximum dimension in excess of 0.6 cm (1/4 in.) from the nominal 0.5 cm (3/16 in.) diameter. These three upper aft holes showed more extensive fraying than the other holes.

3) Two holes on the lower fillet edge had been filled and relocated when the panels were reinstalled on aircraft 1111. These filled areas were visible from both exterior and inner surfaces. (Figure 35). Both showed fraying visible from the exterior side only. One was elongated to 0.6 cm (1/4 in.), but the other was not elongated. The nonelongated hole showed considerably greater fraying. This hole was located at the lower, aft corner. A third hole on the upper aft edge was reported in the Second Annual Report to have been filled, but this proved to be in error.

4) Other holes in the lower fillet segment showed only slight fraying. Two of those were slightly elongated, while a third hole, located on the aft, lower edge was elongated to 0.8 cm (5/16 in.) maximum dimension.

RIGHT-HAND AFT ENGINE FAIRING (P/N 1544685-117)

1) A paint loss area observed in 1976 had been repainted. (Figure 36). The paint had a rough, porous appearance, and two possible damage areas were observed in the center of the painted area. One was 0.3 cm (1/8 in.) diameter, and the other was 0.5 cm (3/16 in.) diameter. There was no associated delamination.

2) The inner surface was free of defects or damage, with the vapor barrier coating applied during reinstallation in lieu of Tedlar Film showing no defective areas or disbonds. (Figure 37).

RIGHT-HAND AFT ENGINE FAIRING (P/N 1544685-117)

3) Very slight fraying of fastener holes was visible from the inner surface only. (Figure 38). Deformation was observed on a large number of holes, approximately 2/3 of the total. (Figure 39). The holes were typically enlongated from 0.55 cm (7/32 in.) to 0.7 cm (9/32 in.) maximum dimension from the nominal 0.5 cm (3/16 in.) diameter. Three of the intercostal holes were also elongated to 0.6 cm (1/4 in.). These were all on the aft intercostal.

   All of these fastener holes had been filled and relocated upon reinstallation. Some holes had been redrilled partially through the filler. The holes with the greatest deformation were through the filled area in some but not all cases.

RIGHT-HAND WING-BODY FAIRING (P/N 1515599-110)

1) A small 0.3 cm (1/8 in.) long crack had not grown since first observed in the 1976 inspection.

2) A small depressed area was observed in the center aft area. No associated disbond was detected.

3) A large teardrop shaped disbond area was observed about 11.4 cm (4 1/2 in.) by 2.5 cm (1 in.) in area. (Figure 40). The area showed a rather deep concave depression indicative of core crushing. Paint and flame spray appeared undamaged. This probably is the location of a depressed area observed in the 1976 inspection and of a repair made during panel reinstallation. This repair is described in the Second Annual Report, and consisted of damaged skin and core removal and replacement.

4) Another delamination was detected in the lower forward area, approximately 5.1 cm (2 in.) by 1.3 cm (1/2 in.) in area.

RIGHT-HAND WING-BODY FAIRING (P/N 1515599-110)

5) No evidence of fastener misalignment was observed, but some fasteners showed gaps between the fastener head and the panel.

RIGHT-HAND UNDERWING FILLET (P/N 1545328-110)

1) No damage or defects were visible on the exterior surface. (Figure 41). The lower aft edge had a bulged appearance indicating possible misalignment. No fasteners showed evidence of misalignment, however.

LEFT-HAND AFT ENGINE FAIRING (P/N 1538592-129)

1) No disbonds were detected. However only the lower aft area could be reached for coin tapping. No damage or defects were visible. No fastener misalignment was visible.

2) An area of paint loss and Flame spray loss had increased since the 1976 inspection, and Kevlar-49 surface was exposed to UV. (Figure 42). The exposed areas showed no signs of discoloration or other damage. Some of the exposed Kevlar-49 areas had been repainted.

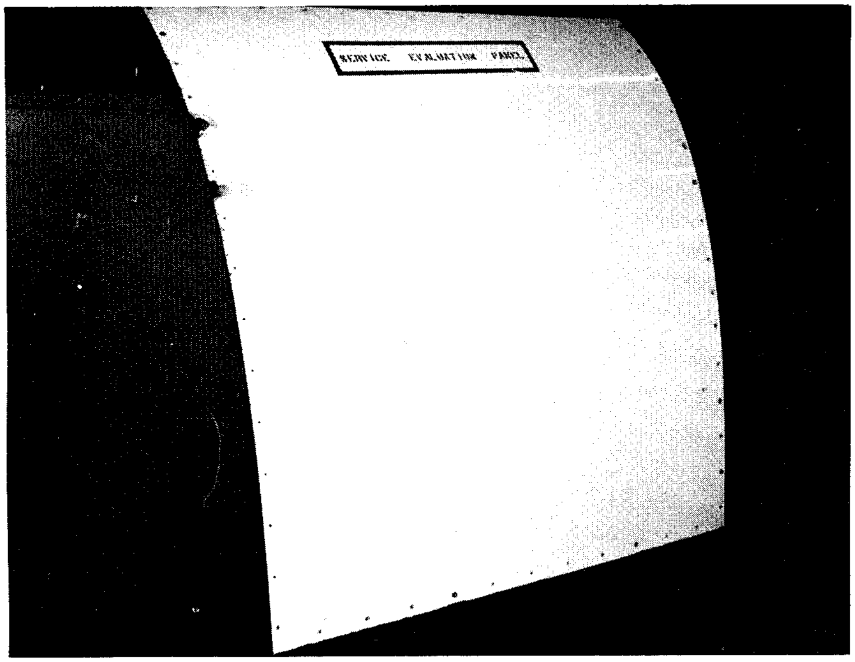

Figure 31. TWA LH Wing-Body Fairing - Exterior Surface

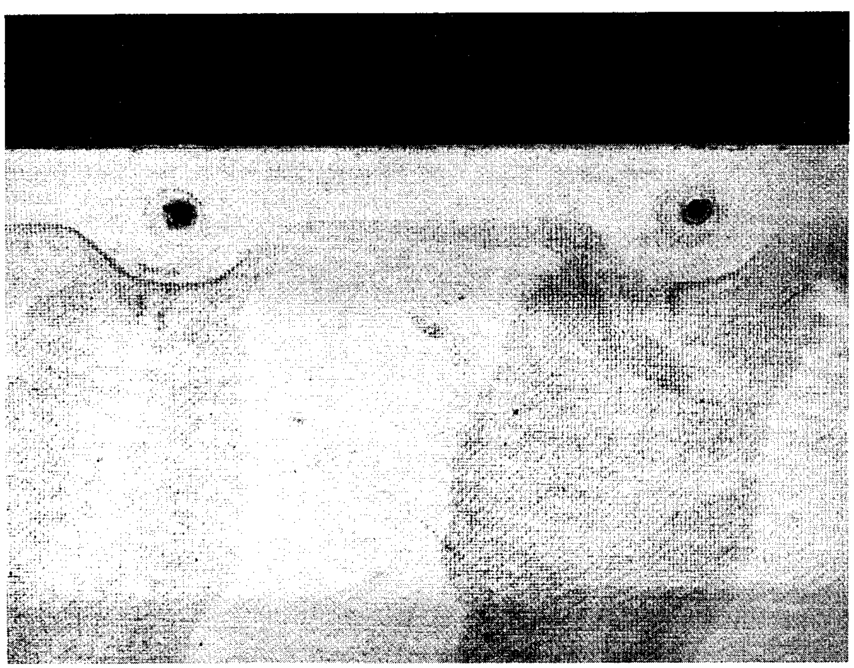

Figure 32. TWA LH Wing-Body Fairing - Typical Fastener Holes Viewed from Inner Surface

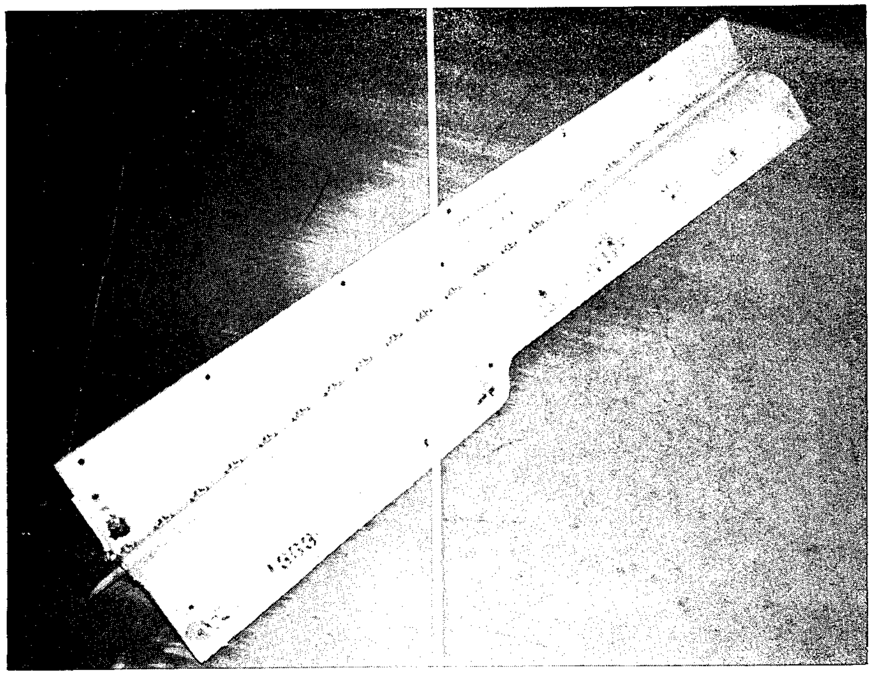

Figure 33. TWA LH Underwing Fillet - Inner Surface

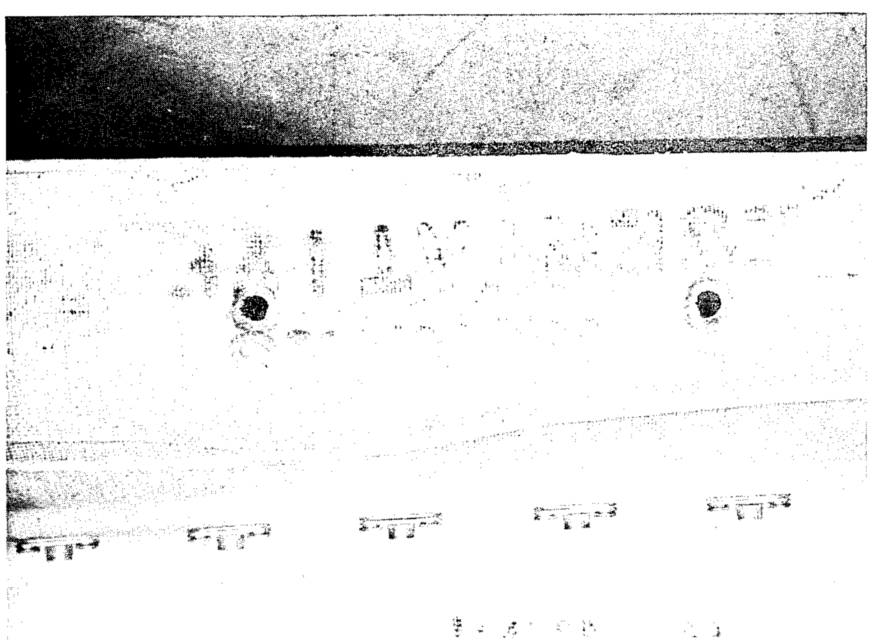

Figure 34. TWA LH Underwing Fillet - Fastener Holes on Upper Edge Viewed from Inner Surface

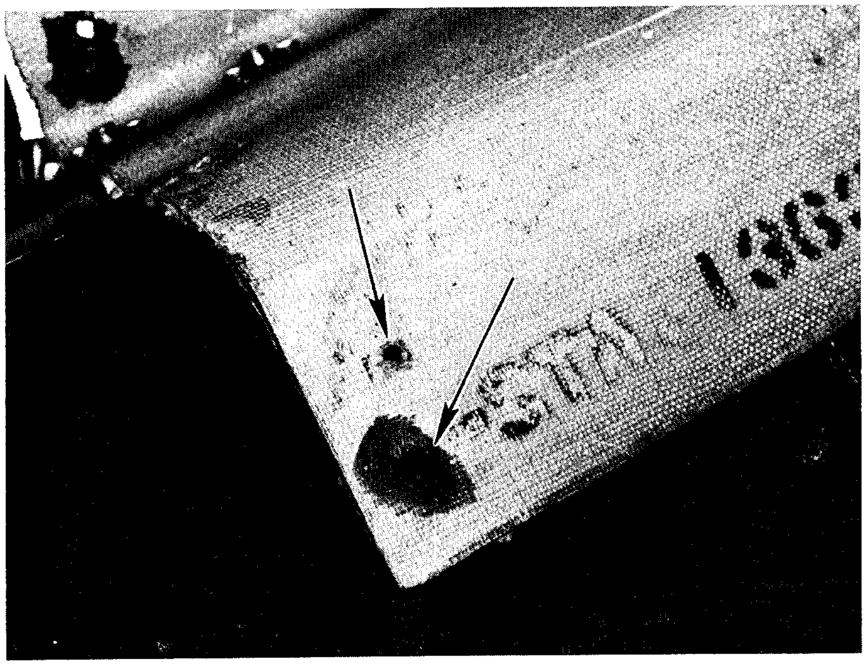

Figure 35. TWA LH Underwing Fillet - Filled and Relocated Fastener Holes

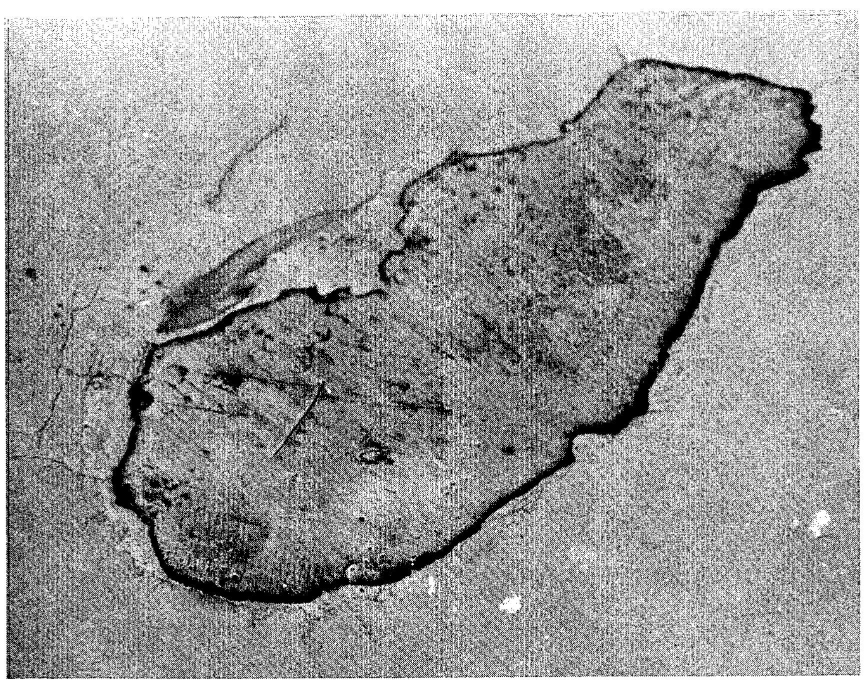

Figure 36. TWA RH Aft Engine Fairing - Repainted Area with Possible Skin Damage

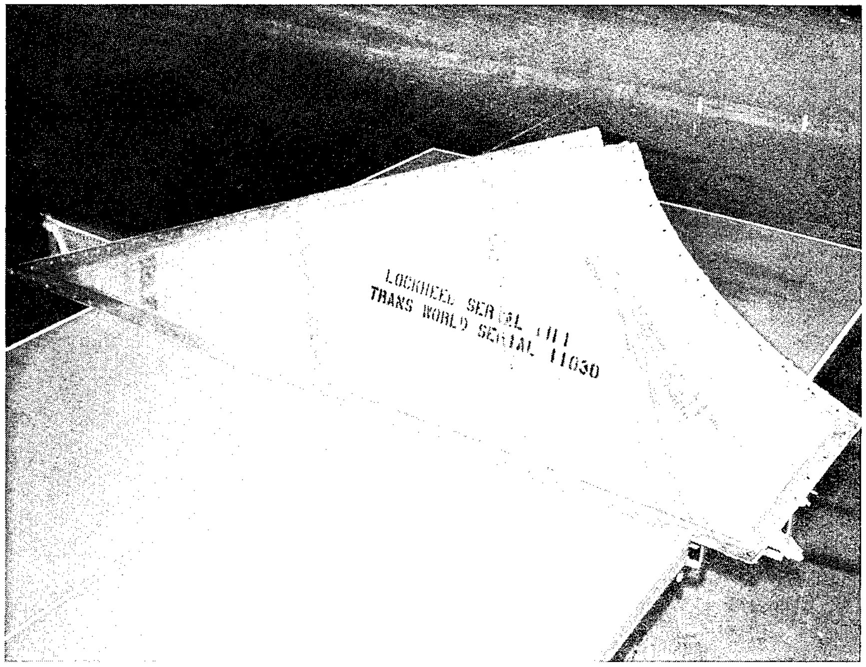

Figure 37. TWA RH Aft Engine Fairing - Inner Surface with Vapor Barrier Coating

Figure 38. TWA RH Aft Engine Fairing - Fastener Holes Viewed from Inner Surface

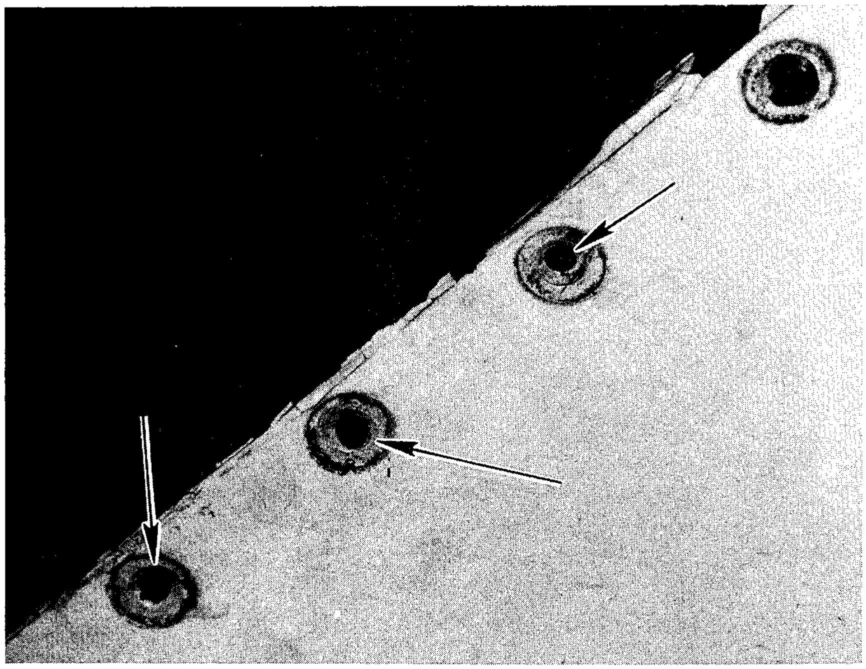

Figure 39. TWA RH Aft Engine Fairing - Fastener Holes Viewed from Exterior, Showing Elongation

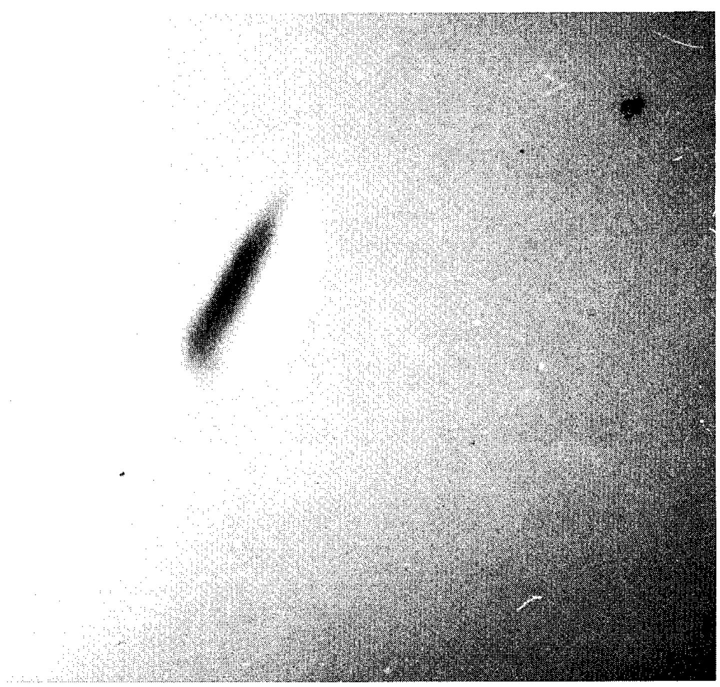

Figure 40. TWA RH Wing-Body Fairing - Concave Depression with Disbond, Exterior Surface

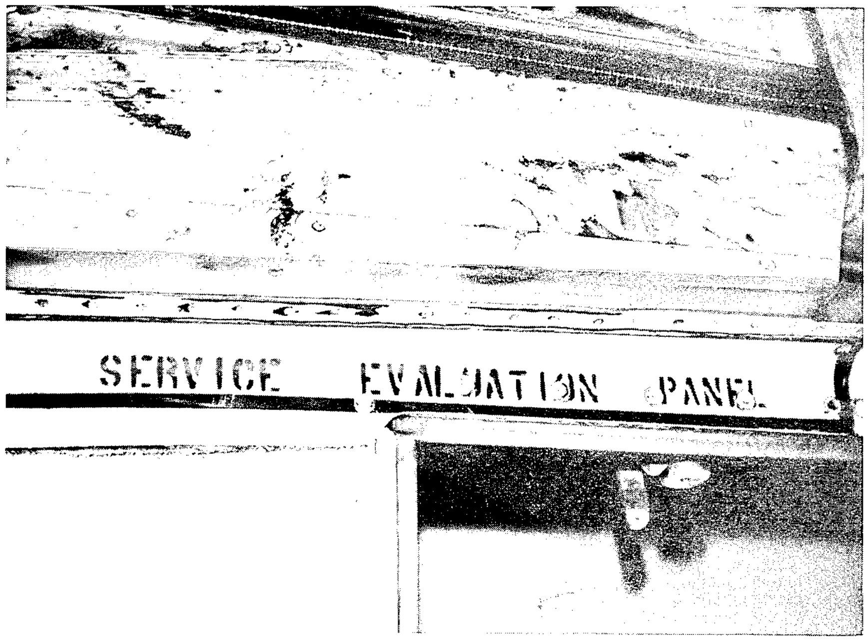

Figure 41. TWA RH Underwing Fillet - Exterior Surface on Aircraft

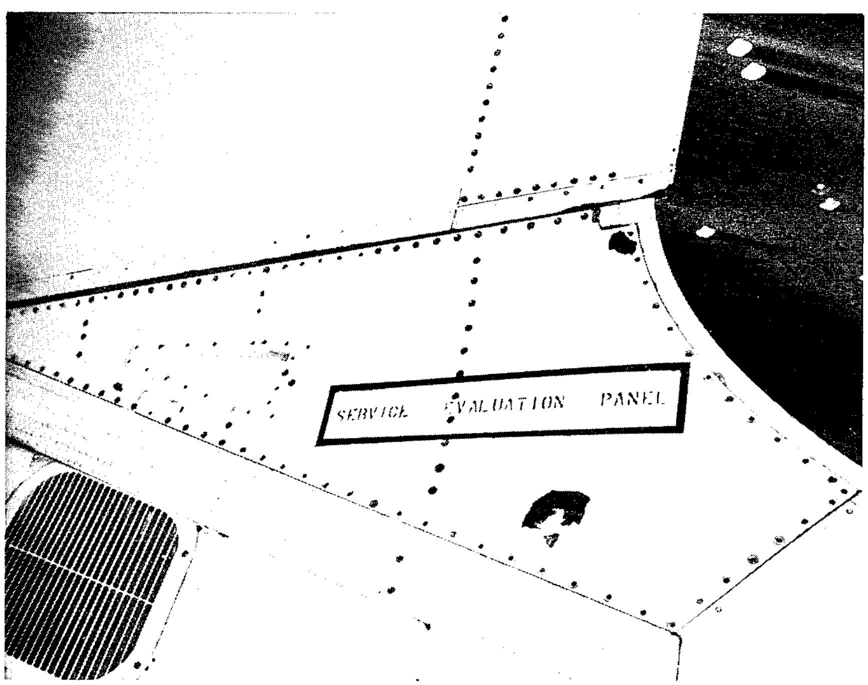

Figure 42. TWA LH Aft Engine Fairing - Exterior Surface on Aircraft Showing Paint Loss Areas