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CRACK PROPAGATION IN ELECTRON BEAM WELDED Ti-6Al-4V

UNDER SPECTRUM LOADING CONDITIONS

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Center cracked panels of electron beam welded Ti-6Al-4V were tested under truncated F-14 aircraft wing fatigue spectra in an effort to simulate propagation of fatigue cracks from "spike" flaws at weld intersect holes. Predictions of fatigue lives of F-14 wing carrythrough sections for various initial defect sizes were made using fracture mechanics methods.

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SUMMARY

INTRODUCTION

This test program was undertaken as a result of the premature failure of an F-14A aircraft wing carrythrough section test article. The failure occurred by fatigue crack propagation originating at a "spike" defect at a weld intersection point. As shown schematically in Figure 1, a "spike" defect occurs unavoidably when two electron beam welds intersect. After the first weld is completed, the second weld is made with parameters set for thickness T1. When the electron beam encounters the preexisting weld, it "sees" an effective thickness of T2, the entire length of the first weld. A defect occurs at the limit of penetration of the beam. Standard procedure in fabrication of the wing carrythrough section is to remove the defect with a drilled hole. In the case of the fatigue test article, an incompletely removed defect at the surface of the hole escaped detection and served as the failure origin.

It was thought that similar defects might be present in production aircraft. This program was a study of the propagation rates of simulated electron beam weld flaws in Ti-6Al-4V under F-14 wing spectrum loads in order to predict safe lives of aircraft and to set revised inspection standards.

SUMMARY OF RESULTS

The results of the study using 4G and 5G loading spectra gave a total life expectancy of 2928 - 5328 and 1370 - 2250 flight hours respectively. These data included both initiation of cracks from machined notches and their propagation to failure. The data spread indicates the scatter band for the experimental results.

Residual life remaining after the first detection of fatigue cracking was 902 - 1528 and 50 - 810 flight hours respectively for the 4G and 5G spectrum load tests.

Fracture mechanics prediction methods were used to predict fatigue life expectancy of F-14 aircraft wing carrythrough sections (Table 6) for various size defects at weld intersect holes.

CONCLUSIONS

1. Crack propagation controlled fatigue lives of structures under variable amplitude loading conditions can be predicted from fracture mechanics specimen tests, but the reliability of the method utilized herein is limited at present by the requirement for successive graphical interpretations.

2. Weld intersect hole defects 0.354 inch (9.0 mm) and larger for 1/4 inch (6.35 mm) diameter holes, 0.079 inch (2.0 mm) and larger for 1/2 inch (12.7 mm) diameter holes, and 0.039 inch (1.0 mm) and larger for 1.0 inch (25.4 mm) diameter holes cannot be considered to have a safe life expectancy for even one Scheduled Depot Level Maintenance (SDLM) interval for the F-14 aircraft.

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EXPERIMENTAL PROCEDURES

A. MATERIAL

The base material for the test program was annealed Ti-6Al-4V in the form of two rolled plates each 1-3/4 x 12 x 52 inches (44.5 x 293 x 1346 mm). Chemical composition and tensile properties reported by the supplier are listed in Tables I and II. The plates were electron beam welded to form a 1-3/4 x 24 x 52 inch (44.5 x 586 x 1346 mm) panel by Grumman Aerospace Corporation (GAC) utilizing the same welding and inspection procedures used for thick-section welds on F-14 aircraft wing carrythrough sections. Inspection of the weld disclosed one burst type defect approximately 7/16 inch (11.1 mm) in diameter. The defect was drilled out so that all test specimens would be assured of being initially defect free.

TABLE I

CHEMICAL ANALYSIS (WEIGHT PERCENT) OF Ti-6Al-4V PLATE MATERIAL

Heat No.	Al	V	C	Fe	H_2	O_2	N_2	Ti
K-6795	6.2	4.2	0.026	0.20	0.002	0.17	0.009	Bal.

TABLE II

TENSILE PROPERTIES OF Ti-6Al-4V PLATE MATERIAL

Orientation	0.2% Offset Yield Strength ksi (MPa)	Tensile Strength ksi (MPa)	Elongation %
Longitudinal	137 (944)	145 (1000)	13
Transverse	130 (896)	139 (958)	12

B. SPECIMEN LAYOUT AND CONFIGURATION

Figure 2 shows the test specimen layout for the welded panel. The hole where the above mentioned burst defect was removed is indicated on the figure near the finish end of the weld.

The specimen configuration used (Figure 3) was a center notched panel with grip ends designed for clamping rather than the more common pin loading arrangement. Clamped ends were required in order to prevent rotation of the specimen under compressive loads and to avoid pin-slamming during transition between tension and compression.

The 1/2 inch (12.7 mm) center hole with crack starter notches was designed to simulate an incompletely removed "spike" defect at a weld intersection hole. The center portion of each specimen was swabbed with Keller's Reagent after surface machining was complete to locate the holes and notches on the weld centerline.

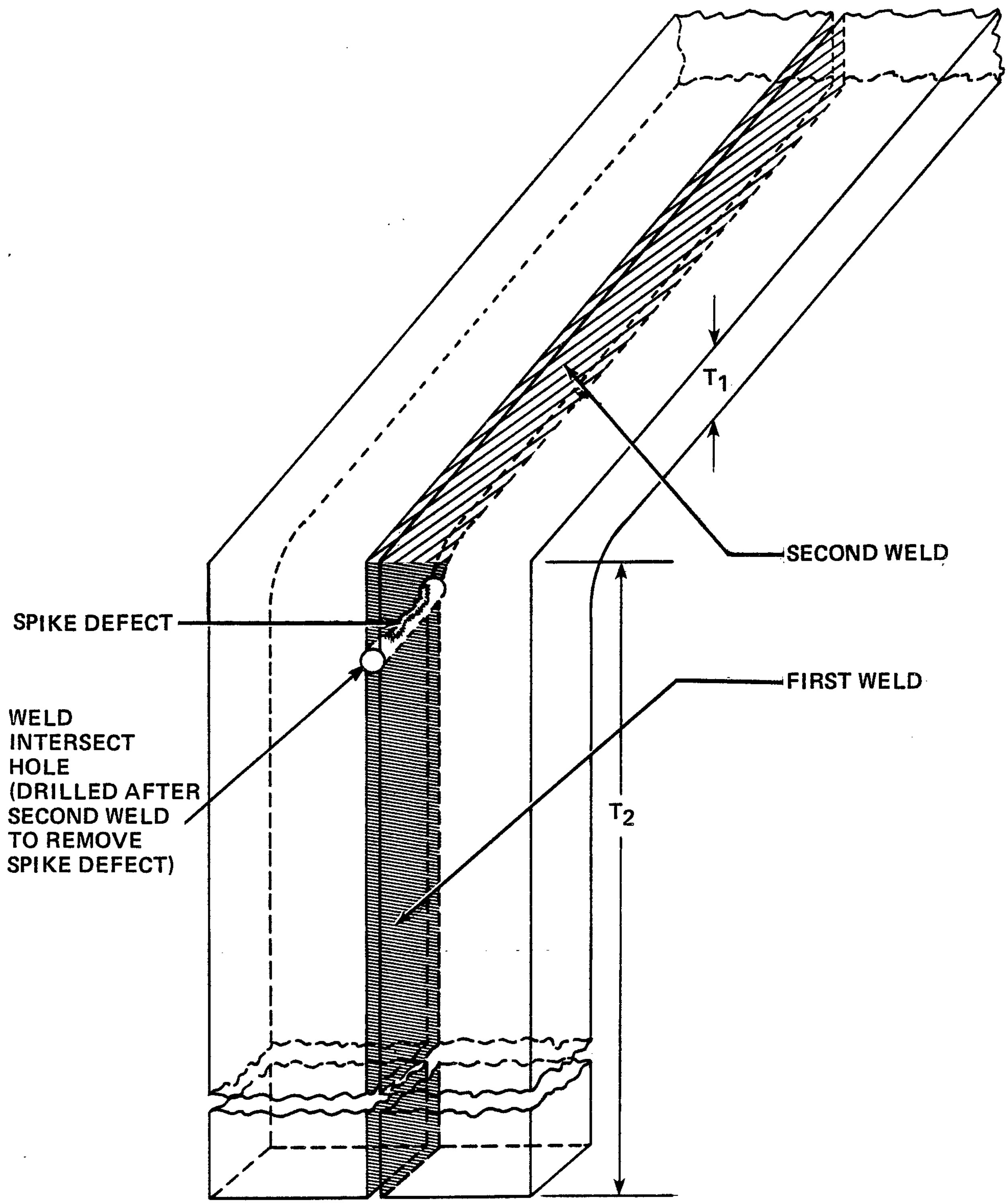

Figure 1. Spike Defects and Weld Intersect Holes

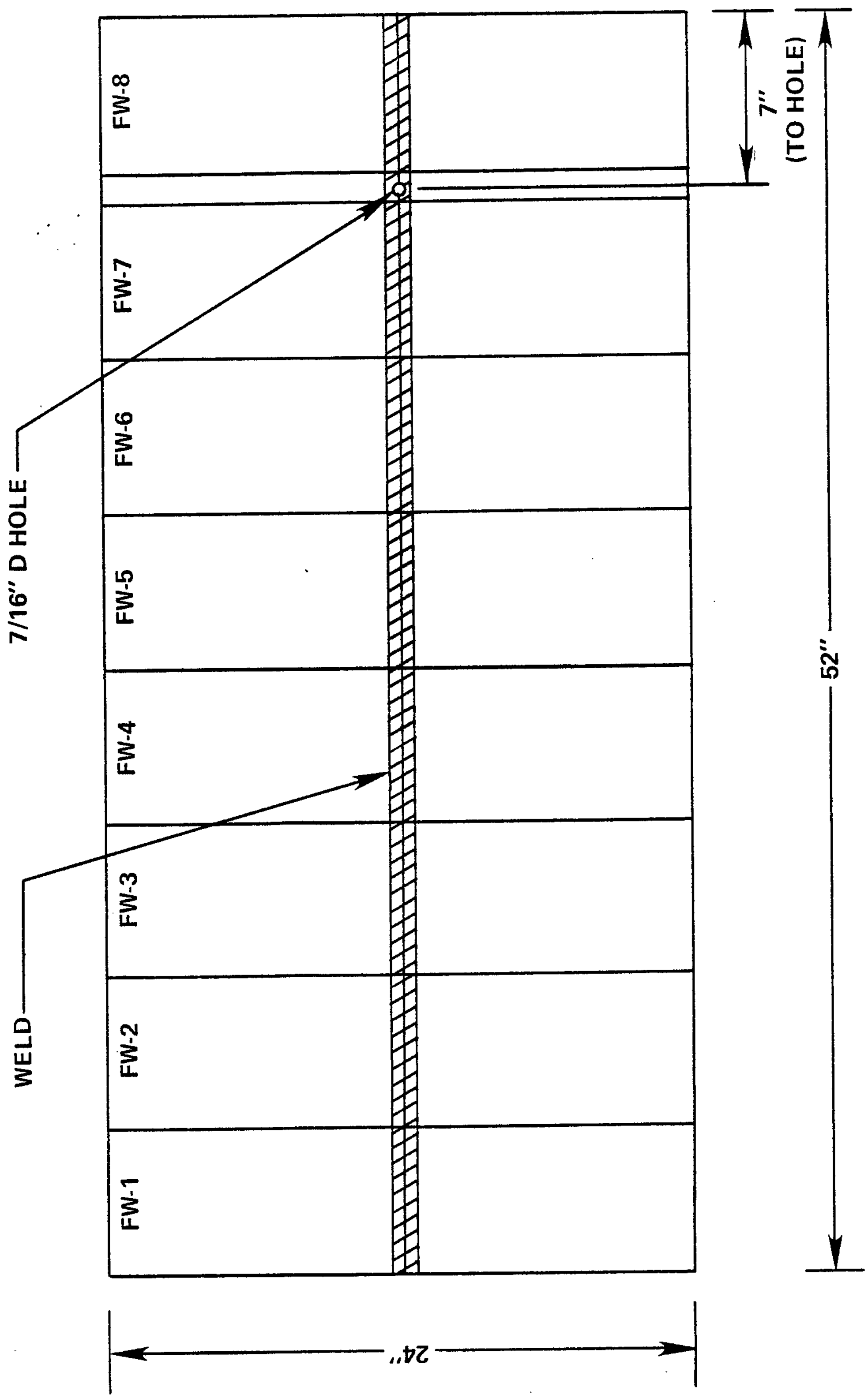

Figure 2. Titanium Electron Beam Weld Specimen Layout Diagram

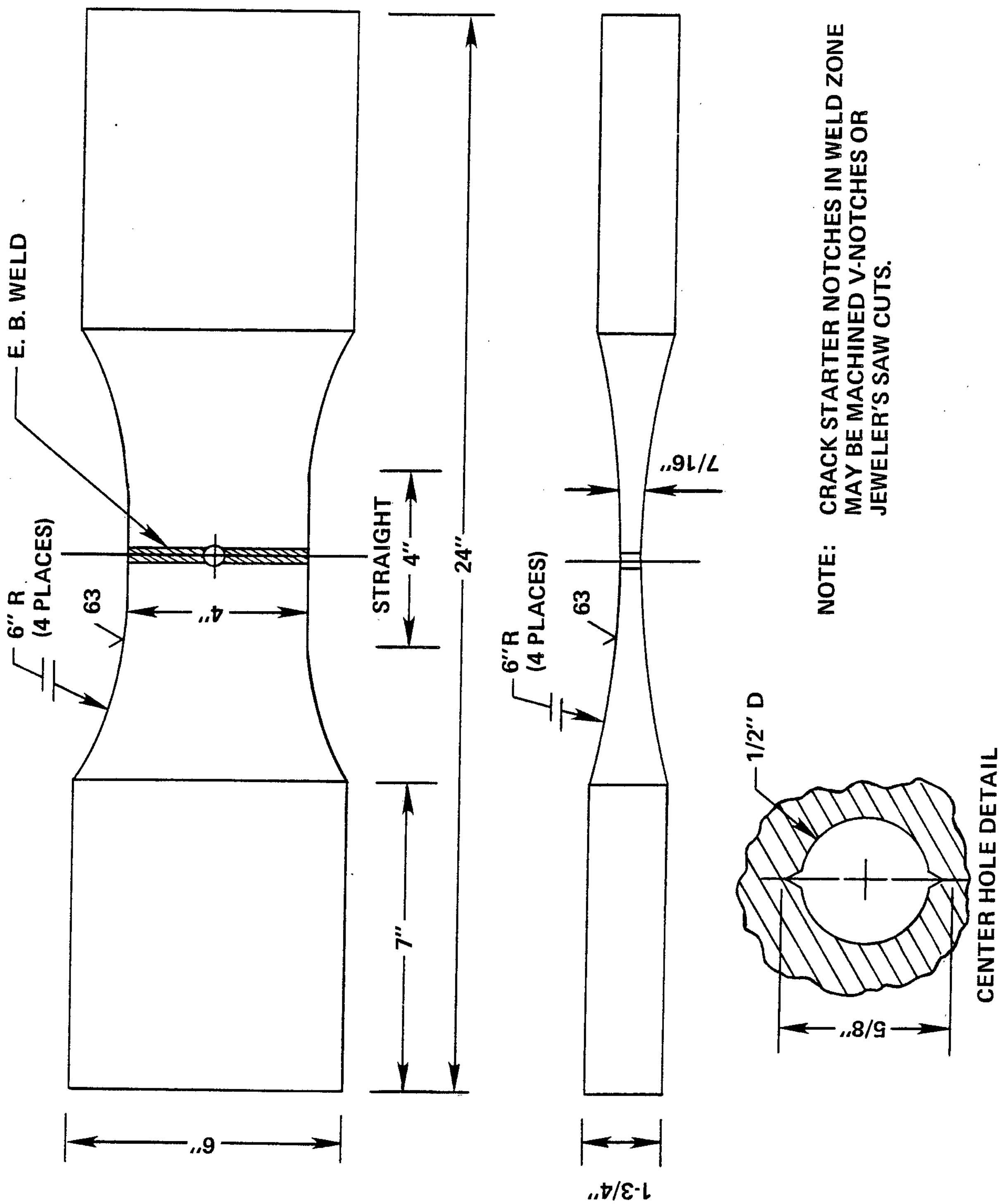

Figure 3. Titanium Flawed Electron Beam Weld Fatigue Specimen

C. TEST METHOD

1. Load Spectra

F-14 wing fatigue spectra truncated to 4G and 5G stress levels were provided by GAC (reference (a)) based on a limit stress of 38 ksi (262 MPa). Stress levels together with number of occurrences per 200 flight hours are listed in Tables III and IV. The sequence of application of the stresses was pseudo-randomized* within a 200 flight hour block, and corresponding loads calculated based on the gross area of the specimen (1.75 in.^{2}) (0.00113 m^2) were programmed into a tape controlled EMR Instruments Model 1641 Profiler. The profiler provided command signal and both slope and end point error detection for a Budd Type UEH closed loop servo-hydraulic test machine of 400,000 pound (1.79928 MN) capacity. Loading was sinusoidal at a frequency of 1 to 2 hz.

TABLE III

WELD INTERSECT HOLE STRESS SPECTRUM TRUNCATION TO 4G

Stress Level	Maximum Stress		Minimum Stress		No. of Occurrences 200 Flight Hr. Block
	psi	(kPa)	psi	(kPa)	
1	11,230	(77,428)	-370	(-2,551)	7
2	11,895	(82,013)	-6,670	(-45,988)	1
3	12,313	(84,895)	-4,817	(-33,212)	3
4	12,356	(85,192)	-2,964	(-20,436)	14
5	12,422	(85,647)	3,242	(22,353)	886
6	12,446	(85,812)	-1,204	(-8,301)	284
7	19,874	(137,026)	-6,080	(-41,920)	1
8	20,890	(144,032)	-3,040	(-20,960)	23
9	21,330	(147,065)	4,560	(31,440)	353
10	22,462	(154,870)	-2,128	(-14,672)	30
11	23,634	(162,951)	1,604	(11,059)	3
12	23,897	(164,764)	11,177	(77,063)	432
13	24,378	(168,080)	4,231	(29,172)	8
14	24,378	(168,080)	-2,128	(-14,672)	8

2. Flaw Propagation Measurements

As the specimens were cycled under spectrum loading conditions, observations were made with an optical micrometer to detect crack initiation at the tips of the notches. Crack length measurements after initiation were made at the end of each 200 flight hour block until the specimen failed. The surfaces of the specimens in the area of observation were lightly grit blasted to reduce glare.

* The application sequence was not entirely random in that the number of applications of the lesser occurring loads was fixed.

TABLE IV

WELD INTERSECT HOLE STRESS SPECTRUM TRUNCATION TO 5G

Stress Level	Maximum Stress psi	Maximum Stress (kPa)	Minimum Stress psi	Minimum Stress (kPa)	No. of Occurrences 200 Flight Hrs. Block
1	11,230	(77,428)	-370	(-2,551)	7
2	11,895	(82,613)	-6,670	(-45,988)	1
3	12,813	(88,343)	-4,817	(-33,213)	3
4	12,986	(89,535)	-2,964	(-20,436)	14
5	13,166	(90,776)	-1,204	(-8,301)	284
6	13,222	(91,163)	3,242	(22,353)	886
7	19,874	(137,026)	-6,080	(-41,920)	1
8	21,850	(150,651)	-3,040	(-20,960)	21
9	22,480	(154,994)	4,560	(31,440)	323
10	23,442	(161,627)	-2,128	(-14,672)	37
11	23,634	(162,951)	1,604	(11,059)	3
12	24,227	(167,039)	11,177	(77,063)	432
13	29,032	(200,169)	4,560	(31,440)	38
14	29,032	(200,169	-2,668	(-18,395)	3

DATA ANALYSIS AND RESULTS

Three specimens, FW-3, FW-5 and FW-7, were tested under the 4G spectrum and four, FW-2, FW-4, FW-6 and FW-8, under the 5G spectrum. Specimen FW-1 was subjected to an accidental high load and broke before any crack propagation measurements could be made. The raw data from each test consisted of a series of recorded crack length measurements with corresponding elapsed blocks or flight hours; these data are shown in Table V.

TABLE V

FATIGUE CRACK LENGTH MEASUREMENTS FOR
Ti-6Al-4V ELECTRON BEAM WELD SPECIMENS

Specimen No.	Spectrum	Flight Hours	Total Crack Length (2 l) inch	(mm)
FW-3	4G	1800	0.6213	(15,781)(1)
		2000	0.6494	(16,496)
		2200	0.6418	(16,301)
		2400	0.6719	(17,066)
		2600	0.7115	(18,073)
		2800	0.7487	(19,017)

Specimen failed at 2928 flight hours;
Failure stress: 23,897 psi (164,764 kPa)

TABLE V (cont'd)

Specimen No.	Spectrum	Flight	Total Crack Length (2 1) inch	(mm)
FW-5	4G	3600	0.6380	(16,205) (1)
		3800	0.7267	(18,457)
		4000	0.7514	(19,086)
		4200	0.7815	(19,851)
		4400	0.8531	(21,669)
	Specimen failed at 4502 flight hours; Failure stress: 23,897 psi (164,764 kPa)			
FW-7	4G	3600	0.6396	(16,246) (1)
		3800	0.6500	(16,509) (2)
		4000	0.6579	(16,710) (2)
		4200	0.6615	(16,802) (2)
		4400	0.6774	(17,207) (2)
		4600	0.6839	(17,371) (2)
		4800	0.7088	(18,004)
		5000	0.7453	(18,931)
		5200	0.8117	(20,616)
	Specimen failed at 5328 flight hours; Failure stress: 23,897 psi (164,764 kPa)			
FW-2	5G	1000	0.6251	(15,877) (1)
		1200	0.6394	(16,240)
		1400	0.6449	(16,380)
		1600	0.6626	(16,830)
		1800	0.6784	(17,231)
		2000	0.7528	(19,120)
	Specimen failed at 2020 flight hours; Failure stress: unknown			
FW-4	5G	2000	0.6373	(16,187) (1)
		2200	0.6551	(16,639)
		2250	0.7322	(18,599)
	Specimen failed at 2250 flight hours; Failure stress: 22,480 psi (154,994 kPa)			
FW-6	5G	1200	0.650	(16.5) (3)
	Specimen failed at 1370 flight hours; Failure stress: 29,032 psi (200,169 kPa)			
FW-8	5G	1000 + (4)	0.6798	(17,266)
	Specimen failed at 1162 + (4) flight hours; Failure stress: unknown			

NOTES:
(1) Starter notch length; no crack.
(2) Cracked only one side of starter notch.
(3) Starter notch length; measured after failure.
(4) No crack was observed after 1000 flight hours. Specimen was then subjected to 12,782 cycles between 286 psi (1972 kPa) minimum stress and 17,420 psi (120,107 kPa) maximum stress to initiate crack before 5G spectrum cycling was resumed.

In instances where the load at failure was known, final crack lengths were calculated based on an assumed fracture toughness (K_{Ic}) of 38 ksi $\sqrt{in.}$ (41.8 MPa $\sqrt{m}$) for the electron beam weld. This assumption was based on earlier in-house testing of similar welds (reference (b)). No measurement could be made of the final crack length on the fractured specimens to verify the accuracy of the calculated crack lengths. As one can see in Figure 4, the similarity in appearance between the spectrum fatigue crack area and the fast fracture area made it difficult to accurately identify a line of demarkation by optical means.

Several analytical operations were performed to convert the data into a form useful for predicting weld defect fatigue behavior in F-14 aircraft. The elapsed flight hours corresponding to each crack length measurement were converted to flight hours remaining before failure, or life expectancies. The total crack lengths (2 1) were then plotted against the logarithms of the life expectancies so that the range of data points could be reasonably bounded by straight lines. The crack length vs. life expectancy curves are hyperbola-like and not amenable to graphical interpretation with linear coordinates. These plots are shown in Figures 5 and 6. From bounds of the data, ranges of life expectancies for any crack length could be inferred.

It was then necessary to provide a conversion from the geometry of the test specimen to that of a weld intersect hole in the wing carrythrough section. This was accomplished by correlating stress intensity factors (K_I). A stress intensity factor solution by Newman and Tada (reference (c)) was employed for the test specimen. This solution is designed for a finite width panel with a symmetrically cracked center hole under uniform tension and is illustrated in Figure 7. In the figure, the curve for c/b = .125, the hole radius to panel half-width ratio for the specimen, was constructed by interpolation.

The Bowie solution for a singly cracked hole (reference (d)), specified for uniaxial stress, was employed to simulate a hole in a weld intersection with an incompletely removed "spike" defect. This solution is illustrated in Figure 8. "Normalized" stress intensity factors were calculated for a specimen geometry over a range of total crack lengths (2 1) and for 1/4 inch (6.35 mm), 1/2 inch (12.7 mm) and 1 inch (25.4 mm) diameter weld intersection holes over a range of single crack lengths (a). For convenience unity was used for the stress term (σ) in all calculations, and all dimensions including crack lengths were in millimeters. Thus, the numerical results obtained for stress intensity factors (K_I) were comparable to each other, but had no relationship to actual stress intensity factors encountered by the specimens or the airplanes. Results of these calculations are plotted in Figures 9 and 10.

From Figures 9 and 10 it was then possible to find for any value of the specimen total crack length (2 1) the corresponding weld intersect hole defect size (a) that would result in the same stress intensity level (K_I) for the same applied stress (σ). This correlation, plotted in Figure 11, was used in conjunction with the data bounds on Figures 5 and 6 to determine range of life expectancies for various sizes of defects associated with the three hole sizes mentioned above.

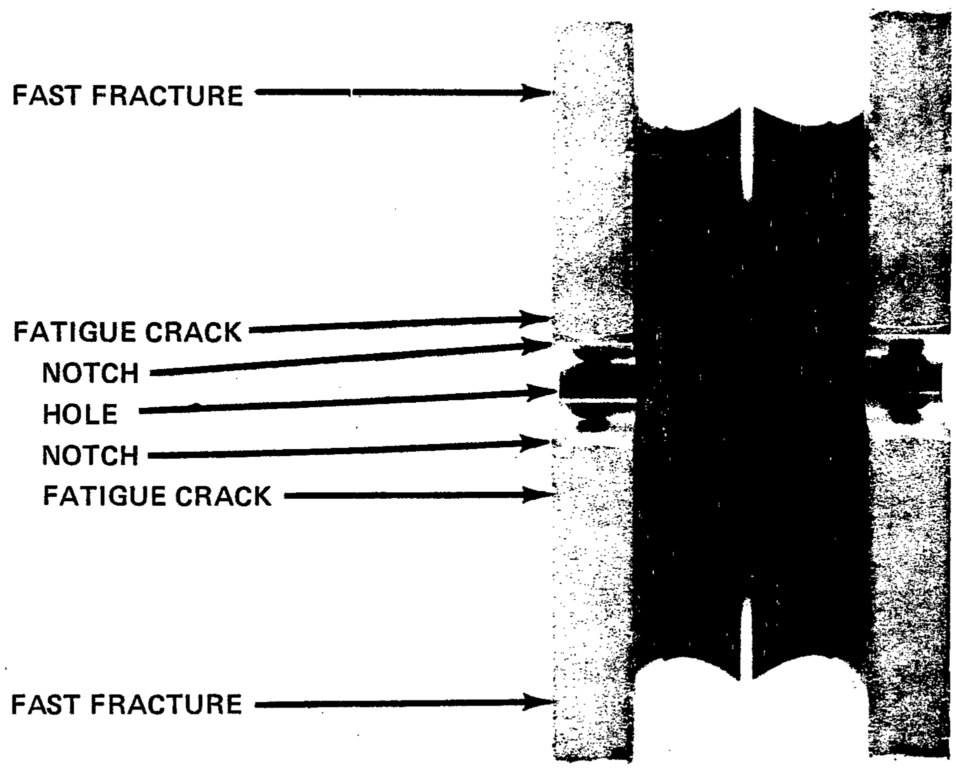

Figure 4. Specimen Fracture Surface

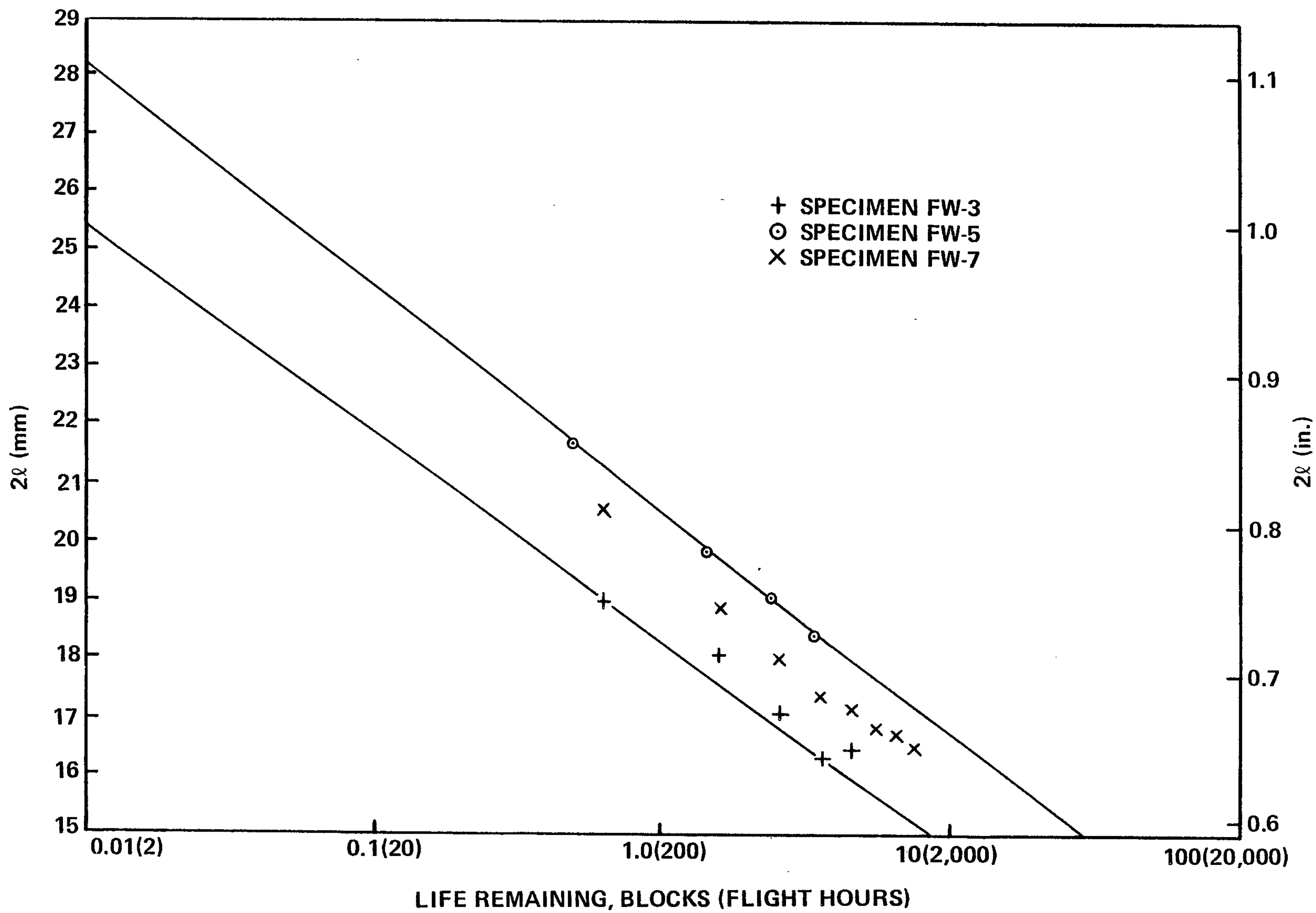

Figure 5. Total Crack Length vs. Fatigue Life Expectancy for Center Cracked Specimens under 4G Spectrum Loading

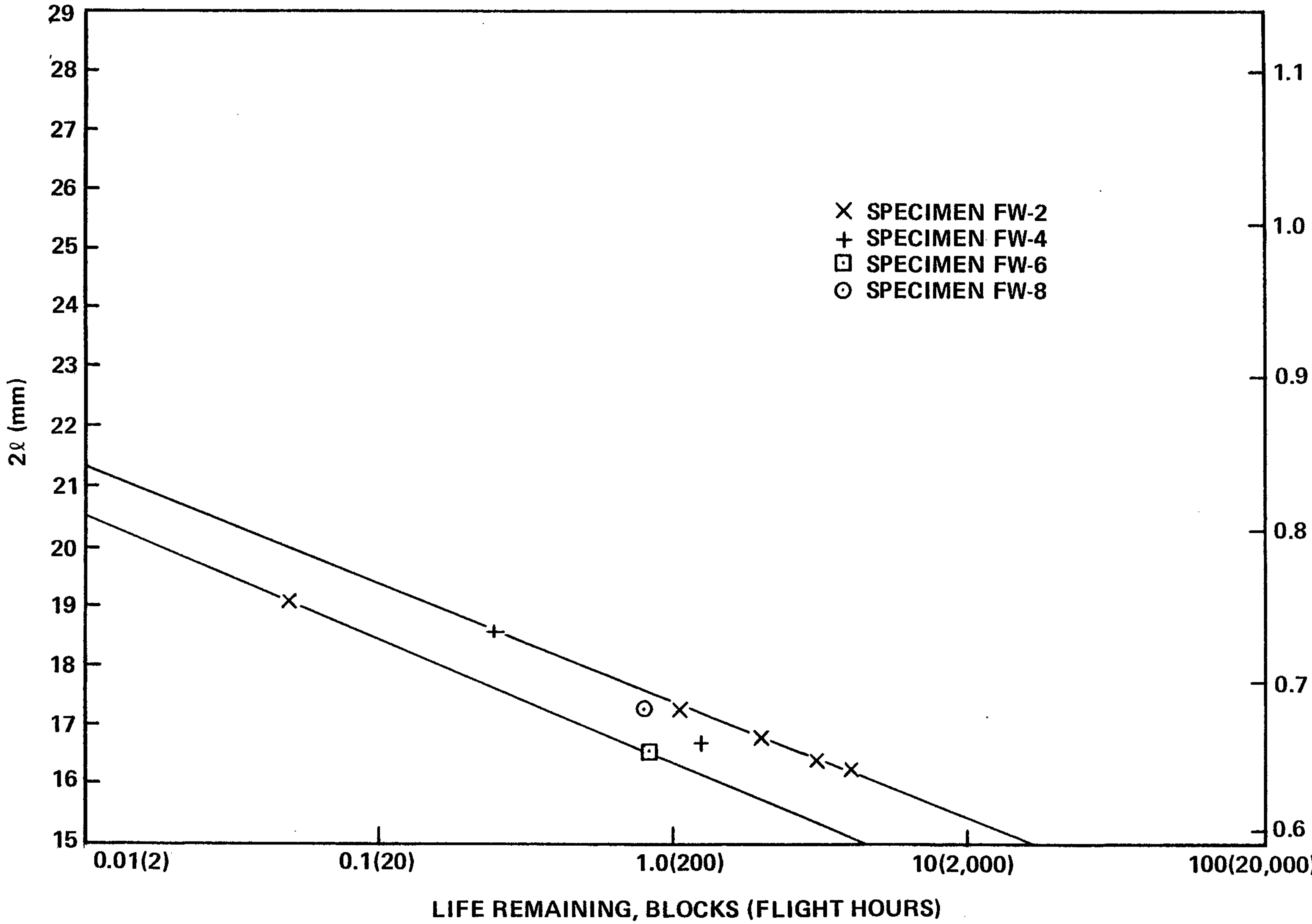

Figure 6. Total Crack Length vs. Fatigue Life Expectancy for Center Cracked Specimens under 5G Spectrum Loading

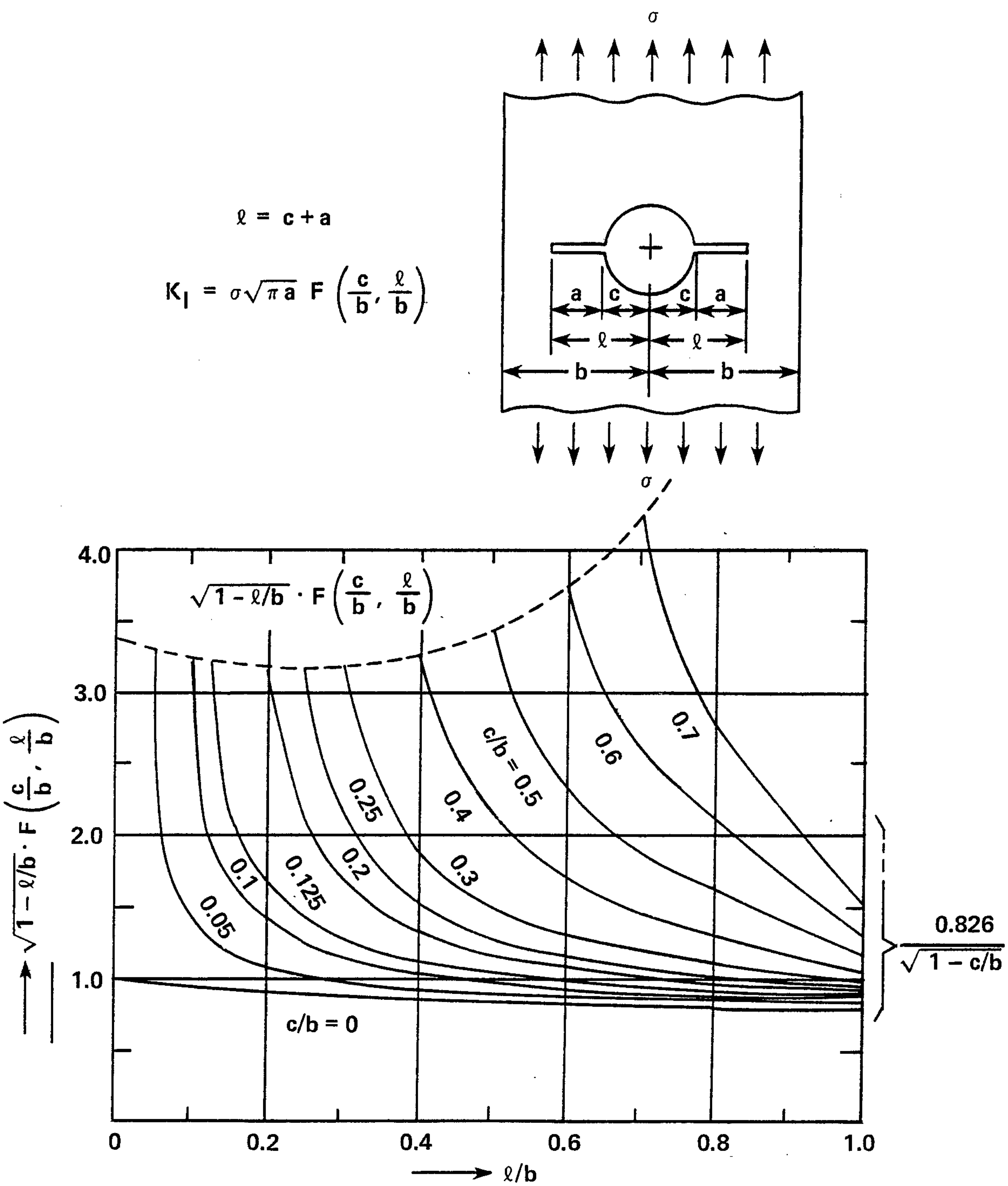

Figure 7. Stress Intensity Solution for Electron Beam Weld Specimens (reference (c))

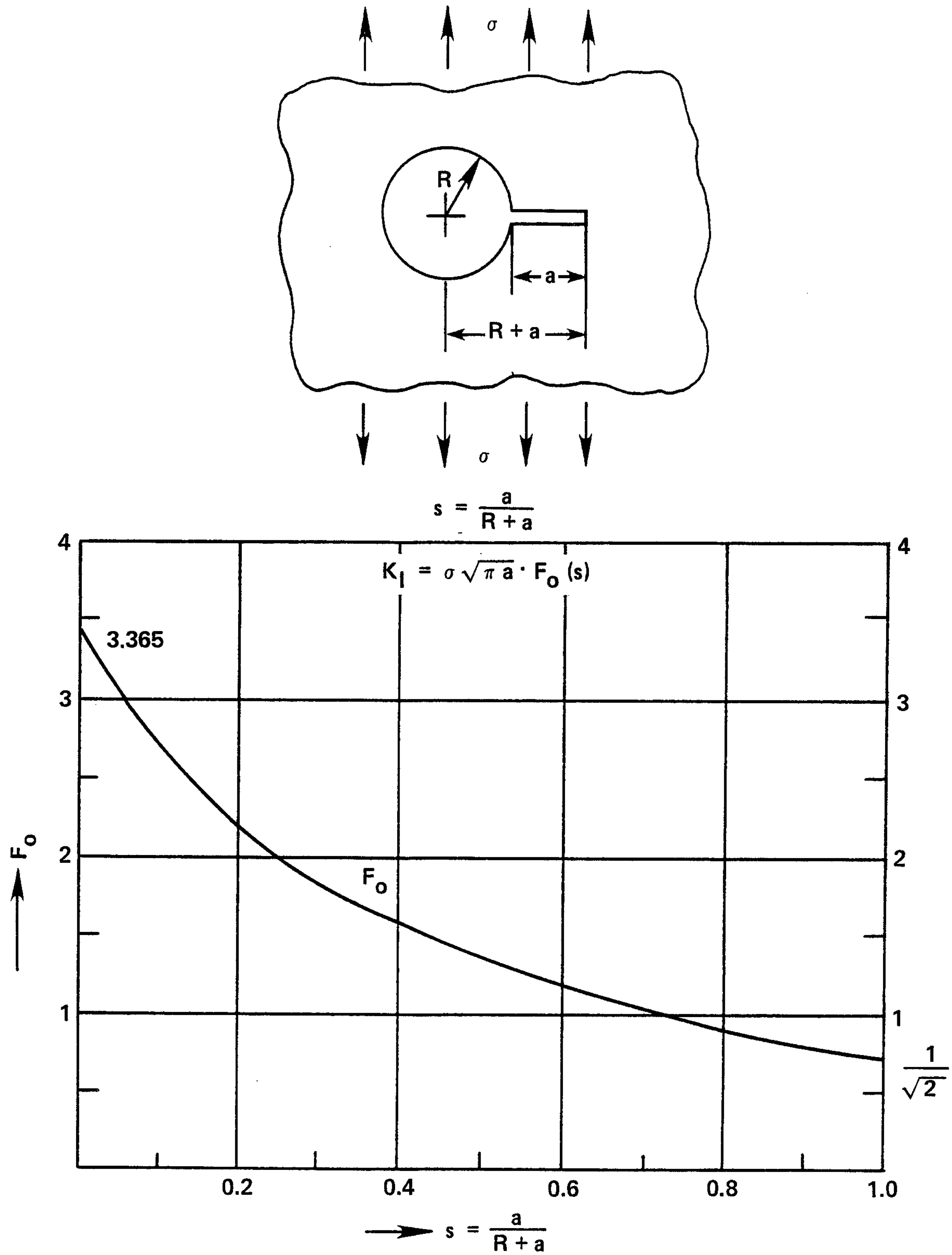

Figure 8. Stress Intensity Solution for Weld Intersect Hole with "Spike" Defect (reference (d))

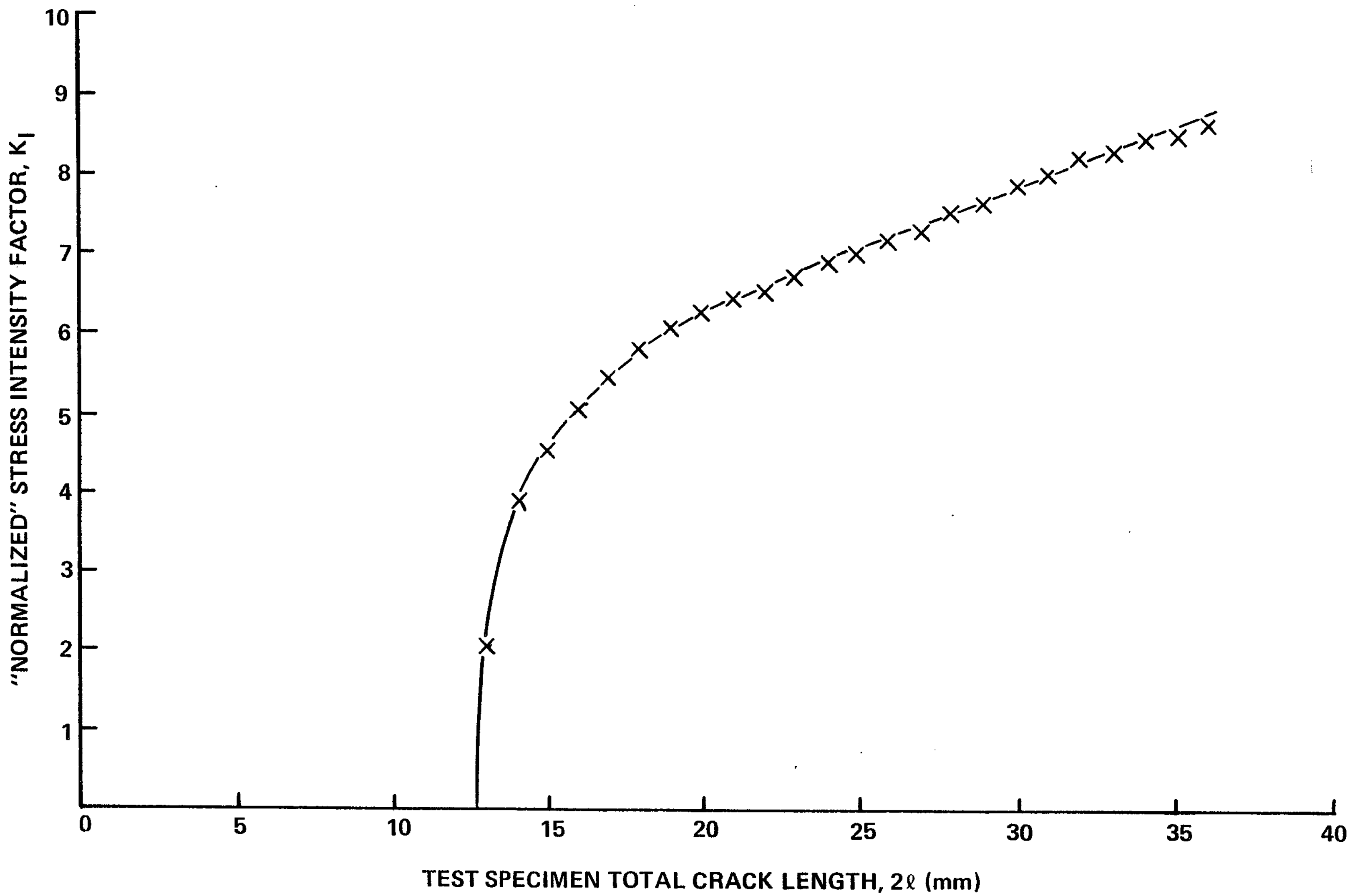

Figure 9. Stress Intensity vs. Crack Length Relationship for Electron Beam Weld Test Specimens

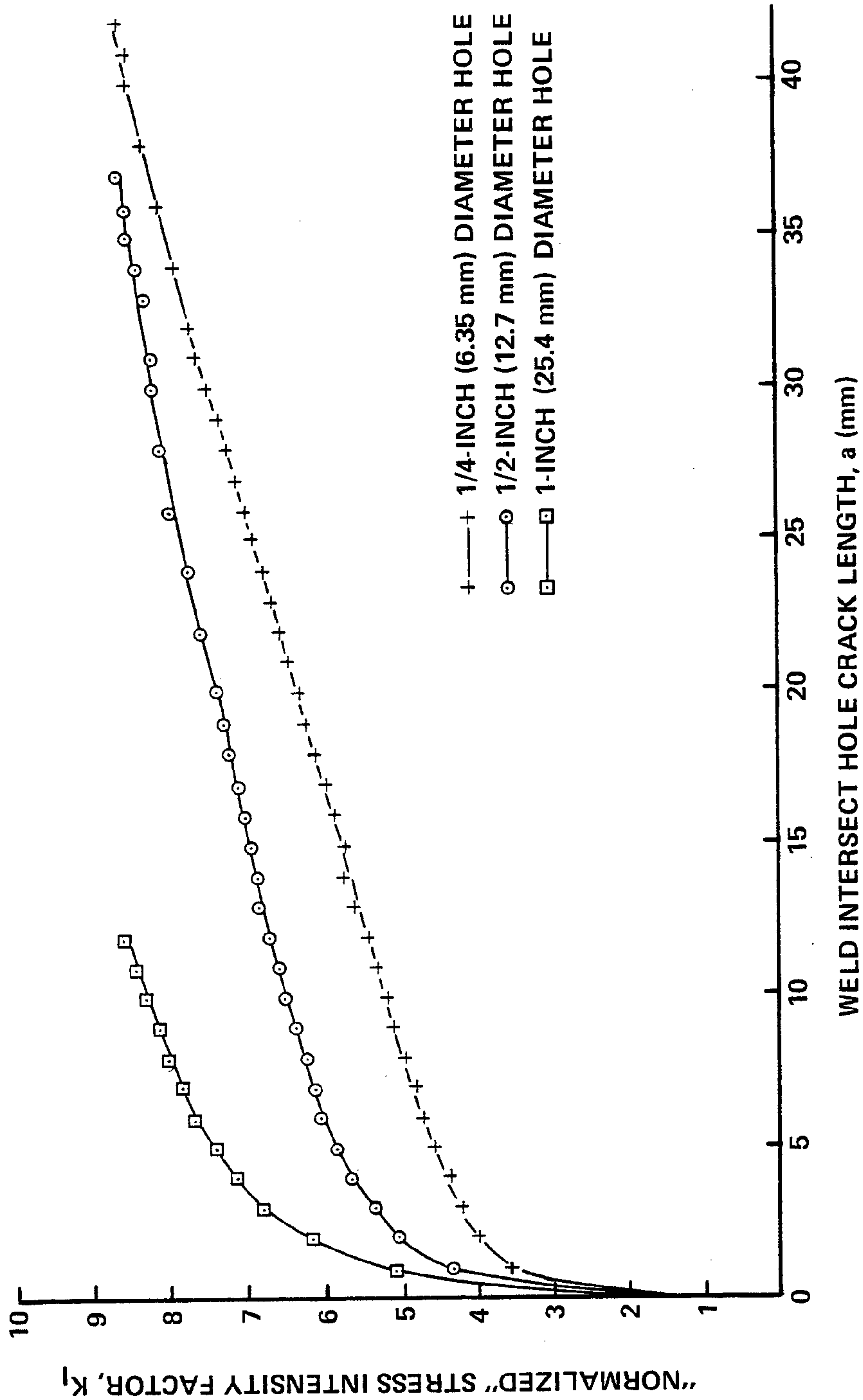

Figure 10. Stress Intensity vs. Crack Length Relationships for Weld Intersect Holes

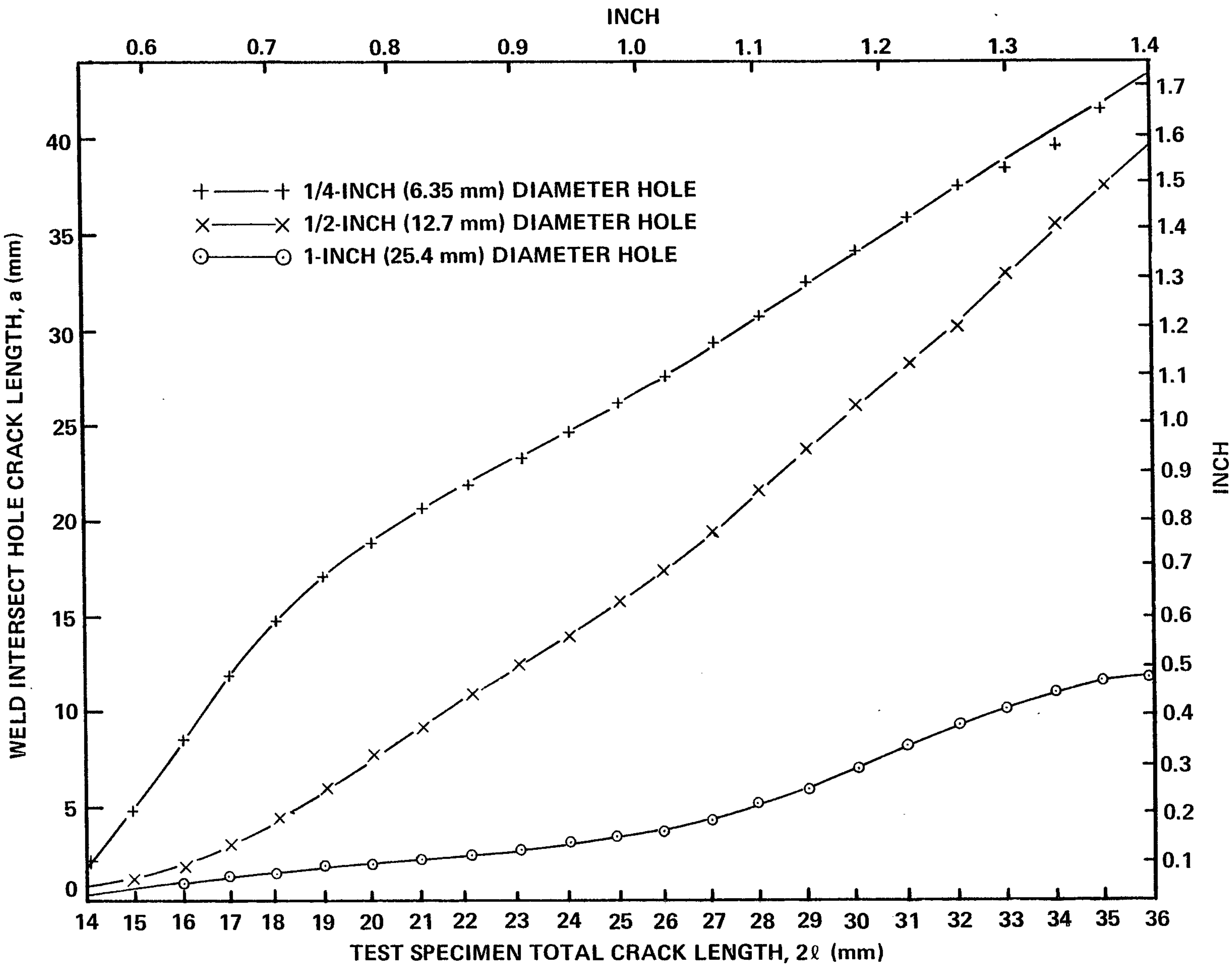

Figure 11. Relationship of Weld Intersect Hole Crack Length to Test Specimen Total Crack Length for Equivalent Stress Intensity Factors

Table VI summarizes the fatigue life expectancies thus determined. Since the mathematical solutions have been programmed for a Wang 600 calculator, it is also possible to determine life expectancies for other hole sizes and defect sizes.

TABLE VI

LIFE EXPECTANCIES FOR F-14 AIRCRAFT WING CARRYTHROUGH SECTIONS FOR VARIOUS SIZE DEFECTS AT WELD INTERSECT HOLES

Hole Size in.	Hole Size (mm)	Defect Size in.	Defect Size (mm)	Life Expectancies 4G Loading Flt. Hr.	Life Expectancies 5G Loading Flt. Hr.
0.25	(6.35)	0.039	(1.0)[1]	3940 - 12800	2780 - 11600
0.25	(6.35)	0.079	(2.0)[1]	2940 - 9600	1720 - 7000
0.25	(6.35)	0.118	(3.0)[1]	2240 - 7460	1080 - 4240
0.25	(6.35)	0.158	(4.0)[1]	1980 - 6660	900 - 3480
0.25	(6.35)	0.197	(5.0)[1]	1580 - 5360	610 - 2320
0.25	(6.35)	0.236	(6.0)[1]	1380 - 4700	480 - 1800
0.25	(6.35)	0.276	(7.0)[1]	1210 - 4200	394 - 1440
0.25	(6.35)	0.315	(8.0)[1]	1010 - 3560	286 - 1060
0.25	(6.35)	0.354	(9.0)	830 - 2960	260 - 940
0.25	(6.35)	0.394	(10.0)	690 - 2480	188 - 770
0.25	(6.35)	0.473	(12.0)	470 - 1740	98 - 328
0.25	(6.35)	0.551	(14.0)	300 - 1140	46 - 146
0.25	(6.35)	0.630	(16.0)	180 - 700	19 - 58
0.25	(6.35)	0.709	(18.0)	100 - 410	7 - 20
0.25	(6.35)	0.788	(20.0)	48 - 200	—
0.25	(6.35)	0.866	(22.0)	20 - 90	—
0.50	(12.7)	0.039	(1.0)[1]	2040 - 6900	940 - 3860
0.50	(12.7)	0.079	(2.0)	900 - 3150	290 - 1060
0.50	(12.7)	0.118	(3.0)	500 - 1840	108 - 370
0.50	(12.7)	0.158	(4.0)	300 - 1140	46 - 146
0.50	(12.7)	0.197	(5.0)	198 - 760	22 - 68
0.50	(12.7)	0.236	(6.0)	134 - 540	12 - 34
0.50	(12.7)	0.276	(7.0)	87 - 360	6 - 16
0.50	(12.7)	0.315	(8.0)	59 - 250	—
0.50	(12.7)	0.354	(9.0)	40 - 174	—
0.50	(12.7)	0.394	(10.0)	28 - 122	—
0.50	(12.7)	0.433	(11.0)	19 - 84	—
1.00	(25.4)	0.039	(1.0)	900 - 3150	290 - 1060
1.00	(25.4)	0.059	(1.5)	280 - 1070	40 - 130
1.00	(25.4)	0.079	(2.0)	90 - 380	6 - 16
1.00	(25.4)	0.098	(2.5)	29 - 128	—

Note: [1] Calculated by extrapolation of data

DISCUSSION OF RESULTS

The test and data analysis method described above can be applied in general to problems of fatigue crack propagation under variable amplitude loading (and consequently variable stress intensity amplitude). Its advantage is that it circumvents the uncertain procedure of integrating crack growth (da/dN) vs. stress intensity amplitude (ΔK) curves. Its primary limitation is that several successive graphical interpretations are required, the most uncertain of which is the determination of boundary limits for the crack length vs. life expectancy data.

In this test program there were at least two other factors also affecting the applicability of the results to aircraft. The first is the question of how closely the 4G and 5G load spectra duplicate the actual loading of the airframe in service. The second is the fact that fatigue cracks were substituted for actual weld flaws. Fatigue cracks undoubtedly have smaller tip radii than do spike defects. Therefore, in service, fatigue lives should be longer than those indicated. This is because of the added time required to initiate a fatigue crack at the tip of the defect. Such a time delay could be estimated from fatigue crack initiation data.

The practical significance of this test program is contained in the predicted life times shown in Table VI. At present the Scheduled Depot Level Maintenance (SDLM) interval for the F-14 aircraft is 30 months, which corresponds to 500 to 700 flight hours (reference (e)), and during maintenance the aircraft are not inspected specifically for defects or cracks in the wing carrythrough section. Clearly no defect size listed in Table VI that falls within the range of the data is tolerable under 5G loading, and most are not tolerable under 4G loading. Most of the lifetimes for small defects determined by extrapolation of the test data are tolerable, but these are subject to larger errors than those that fall within the range of the data. Thus, one must conclude that on the basis of the available data, an F-14 wing carrythrough section cannot tolerate defects at weld intersect holes as large as 0.079 inch (2.0 mm) for 1/2 inch (12.7 mm) diameter holes, 0.354 inch (9.0 mm) for 1/4 inch (6.35 mm) diameter holes, and 0.039 inch (1.0 mm) for 1 inch (25.4 mm) diameter holes.

A C K N O W L E D G E M E N T

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