AFFDL-TM-77-58-FBE

AIR FORCE FLIGHT DYNAMICS LABORATORY
DIRECTOR OF SCIENCE & TECHNOLOGY
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE OHIO

APPLICATIONS OF THE EQUIVALENT INITIAL QUALITY METHOD

James L. Rudd

Technical Memorandum AFFDL-TM-77-58-FBE

July 1977

FOREWORD

This document was prepared by James L. Rudd of the Structural Integrity Branch, Structural Mechanics Division, Air Force Flight Dynamics Laboratory. The work was conducted in-house under Project 1367, "Structural Integrity for Military Aerospace Vehicles," Work Unit 13670336, "Life Analysis Methods," using data generated under the F/RF-4C/D, F-4E(S) and A-7D damage tolerance and life assessment programs. The author would like to thank Mr. James E. Littlefield (Vought Corporation), Mr. Terry D. Gray (AFFDL/FBE) and Mr. Howard A. Wood (AFFDL/FBE) for their contributions. The research was conducted from March 1977 through July 1977.

This technical memorandum has been reviewed and is approved for publication.

R.M. Bader

ROBERT M. BADER, Chief
Structural Integrity Branch
Structural Mechanics Division

TABLE OF CONTENTS

LIST OF TABLES

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Contd)

ABSTRACT

This paper describes a method, the Equivalent Initial Quality Method, of analytically representing fastener hole initial quality resulting from material and structural manufacturing and processing operations. The representation is accomplished by representing the imperfections which are either inherent in a material or introduced during the manufacturing of a structure with a fatigue crack of a particular size and shape. Such a representation allows the damage accumulation process to be considered as entirely crack growth with zero time to initiate a crack. The Equivalent Initial Quality Method can be used both to determine the operational limits (i.e., economic repair limit, inspection interval and fracture limit) of existing aircraft and in the design of new aircraft. Applications of the method are presented which include its use on the F/RF-4C/D, F-4E(S) and A-7D damage tolerance and life assessment programs. Potential applications and possible limitations of the Equivalent Initial Quality Method are discussed.

SECTION 1

INTRODUCTION

For existing Air Force aircraft, damage tolerance and life assessment programs have been and are currently being conducted. The objectives of these programs are to define operational limits and to provide any necessary modification or operational usage options. The operational limits involved include: (1) an economic repair limit which specifies the opportune time for repairs and modifications before such repairs and modifications become too expensive; (2) an inspection interval which provides the opportune time for detecting damage by NDI techniques before the damage reaches critical proportions; and (3) a fracture limit which provides the time at which an aircraft failure potential is believed to exist if no inspection and/or repair is accomplished. These analytically predicted operational limits are based upon the assumption that initial flaws exist in the airframe due to material and structural manufacturing and processing operations.

For future Air Force aircraft, the current Air Force structural integrity policy, MIL-STD-1530A,[1] requires the aircraft to be designed to be durable. In particular, Military Specification MIL-A-8866B[2] contains the airplane durability design requirements which require an analysis to be conducted to demonstrate that the economic life of the airframe is in excess of the design service life when subjected to the design service loads spectra and the design chemical/thermal environment spectra. The economic life is defined as the time required for widespread damage to occur which is uneconomical to repair and, if not repaired, could cause functional problems affecting operational readiness. One of the factors which must be accounted for in the analysis is the initial quality resulting from material and structural manufacturing and processing operations.

Also for future Air Force aircraft, the current Air Force structural integrity policy requires the aircraft to be designed to be damage tolerant. Military Specification MIL-A-83444[3] contains the airplane damage tolerance design requirements which require safety of flight structure to meet certain residual strength and crack growth analyses requirements based on the design concept and degree of inspectability of the structure. This specification requires initial flaws to be assumed to exist as a result of material and structural manufacturing and processing operations. Small imperfections equivalent to a .005 inch radius corner flaw resulting from these operations shall be assumed to exist in each hole of each element of the structure, with the exception that the most critical hole of each element shall contain a flaw of length greater than .005 inch. The .005 inch radius corner flaw provides the basis for the fastener policy requirements and the continuing damage and remaining structure damage assumptions. However, if the contractor has developed initial quality on fastener holes (e.g., by fractographic studies), these data may be substituted to the procuring activity for review and serve as a basis for negotiating a size different than the specified .005 inch radius corner flaw.

In order to analytically predict operational limits for existing aircraft and to satisfy the durability and damage tolerance design requirements for future aircraft, the initial quality of the aircraft must be quantified. One method of quantification is to represent the initial quality with an analytically equivalent fatigue crack of a particular size and shape. Such a method is the Equivalent Initial Quality Method described in the following section. This method provides the capability to consider the damage accumulation process as entirely crack growth with zero time to initiate a crack. This approach is supported by past experience with tests of structures under simulated flight loading which indicate that the time to initiation of cracks from most structural details (e.g., sharp corners, holes, etc.) is relatively short and that the majority of the life (i.e., 95%) is spent growing the cracks to failure.[4]

SECTION 2

DESCRIPTION OF EQUIVALENT INITIAL QUALITY METHOD

The Equivalent Initial Quality Method shall be described for fastener holes, the most prevalent source of cracking in aircraft structure.[5] For the purposes of this paper, quality shall be defined as a measure of the condition of the structure relative to imperfections, flaws, defects, or discrepancies which are either inherent in the material or introduced during manufacturing of the structure. Some of the parameters which can contribute to the initial quality of fastener holes are illustrated in Figure 1. One method of analytically accounting for the initial quality of a fastener hole is to represent the initial quality with an analytically equivalent fatigue crack of a particular size and shape, such as the corner crack depicted in Figure 1. If an analytical initial quality representation is performed for each of a number of fastener holes, an equivalent initial quality statistical distribution can be obtained which represents the initial quality of the fastener holes produced by the particular material and structural manufacturing and processing operations involved.

The analytical initial quality representation, defined as the equivalent initial quality, can be obtained in the following manner. Consider a piece of structure with a fastener hole containing the defect of characteristic dimension ℓ, schematically illustrated in Figure 2. This defect results in fatigue crack initiation and propagation when subjected to some known load history. Upon failure of

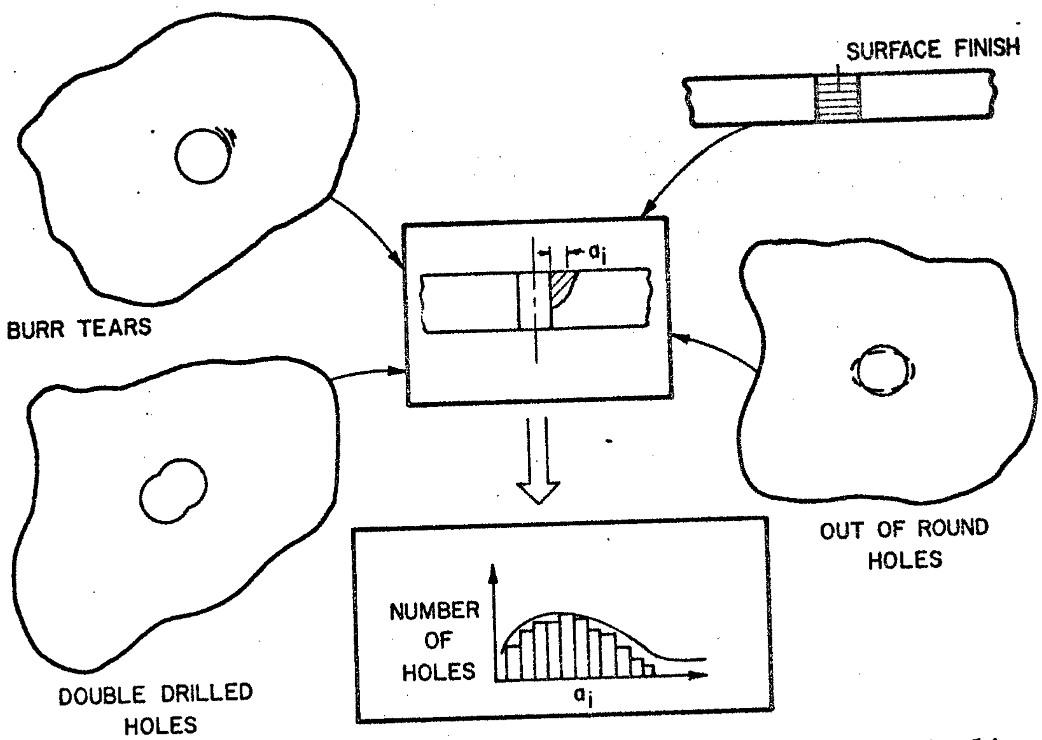


Figure 1. Parameters Which Affect Fastener Hole Initial Quality

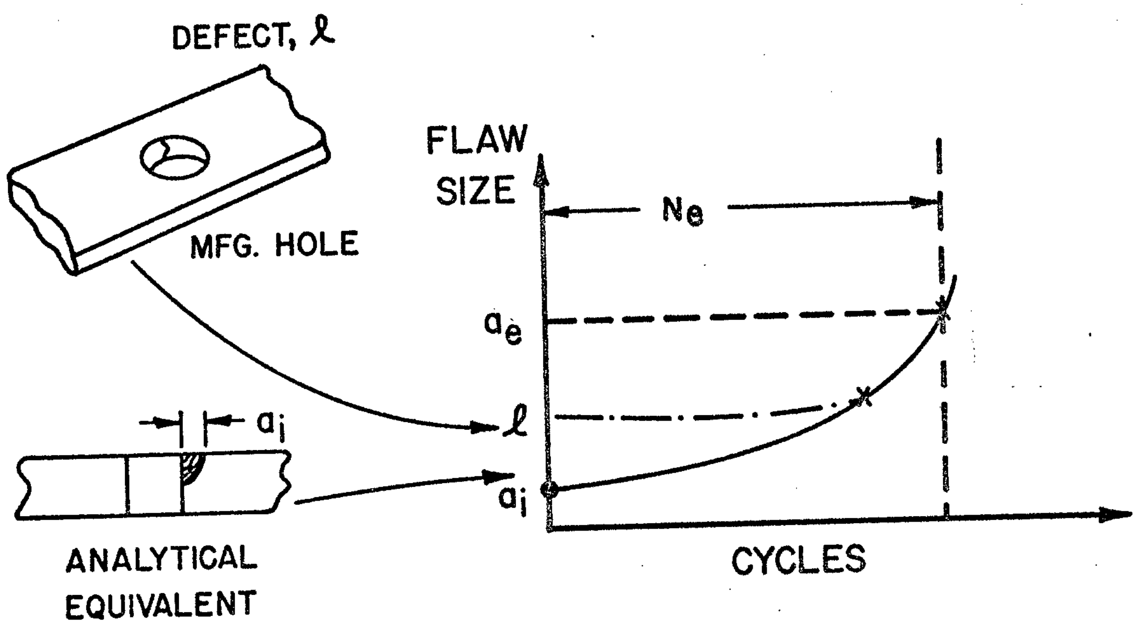


Figure 2. Definition of Equivalent Initial Quality

the structure, a fractographic examination of the fracture surface is performed (e.g., Figure 3) to obtain as much of the crack growth curve as possible. Analytical crack propagation analyses are performed until there is good agreement between the analytical prediction and the fractographic test data for the portion of the crack growth curve of interest. For example, if the Equivalent Initial Quality Method is to be used to obtain the initial crack size to be used in economic life predictions, then it may be desirable to obtain good agreement between the analytical prediction and the fractographic test data for crack sizes up to .03 inch (a_e = .03 inch in Figure 2), allowing removal of cracks by reaming the fastener hole to the next nominal hole size. Similarly, if the Equivalent Initial Quality Method is to be used to obtain the initial crack size to be used in establishing inspection intervals or fracture limits, then it may be desirable to obtain good agreement between the analytical prediction and the fractographic test data at failure ($a_e = a_c$). The initial crack size (crack size when the load history is first applied), a_i, of the analytical crack growth curve which correlates best with the fractographic test data is defined as the equivalent initial quality. Hence, a_i is said to be the analytical equivalent of the actual defect of characteristic dimension ℓ if each results in a preselected crack size a_e after N_e cycles of the same load history have been applied. This implies that fastener holes which contain actual crack sizes less than a_e after N_e cycles have been applied are of better quality than those which contain actual crack sizes equal to or greater than a_e after N_e cycles.

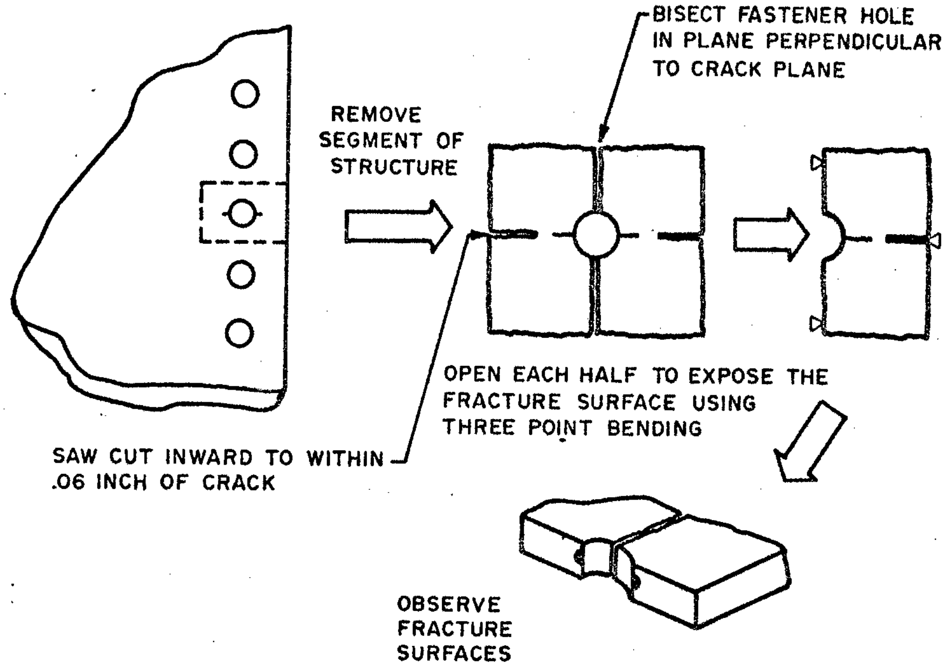


Figure 3. Method of Exposing Fracture Surfaces

Consider the following example which illustrates the use of the Equivalent Initial Quality Method. A photograph of the failure area of a full-scale fatigue test of an A-7A wing was available. The wing had been subjected to a ten level, blocked low-high stress spectrum. Fractographic measurements were taken from the photograph (Figure 4), making it possible to generate a large portion of the crack growth curve.(6) Crack propagation analyses were performed using the computer routine EFFGRO and the Wheeler Retardation Model until the analytical crack growth curve correlated well with the fractographic test data.(7) This correlation is presented in Figure 5 which indicates that the manufacturing quality of the test hardware at the failure location was equivalent to an initial crack of length a_i = 0.00109 inch. This excellent correlation of the analytical crack growth prediction with the fractographic test data indicates that the Equivalent Initial Quality Method was quite successful for this particular problem.

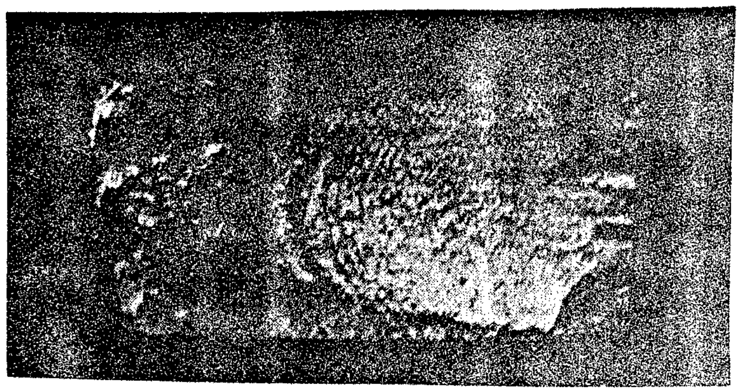


Figure 4. A-7A Wing Fatigue Test Fracture Surface

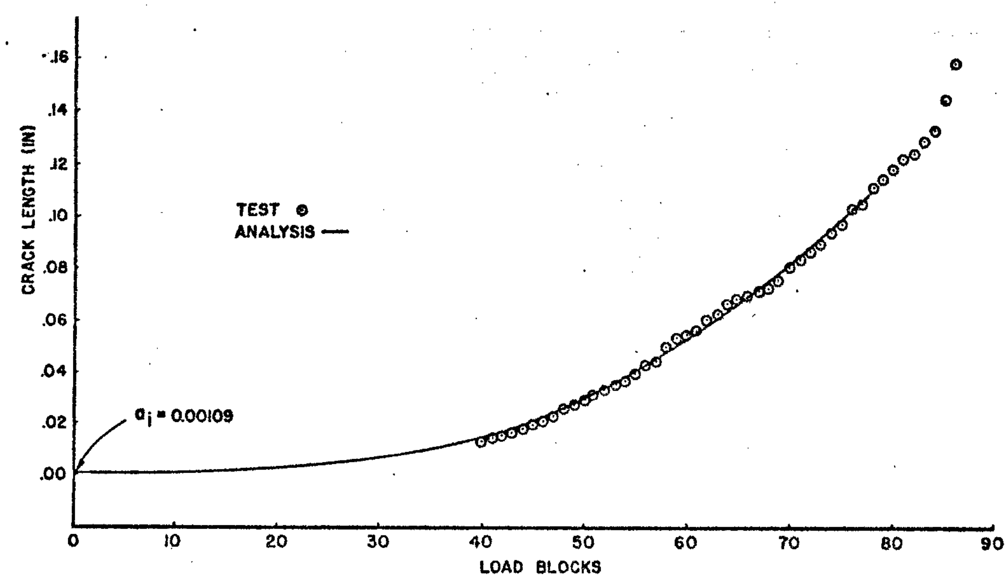


Figure 5. Equivalent Initial Quality Results for A-7A Wing Fatigue Test Failure

SECTION 3

APPLICATIONS

The first known application of the Equivalent Initial Quality Method occurred during the F/RF-4C/D[8] damage tolerance and life assessment program and the method was subsequently used on similar programs for the F-4E(S)[9] and A-7D[10] aircraft. The objectives of each program were to define the operational limits (i.e., economic repair limit, inspection interval, and fracture limit) of the particular aircraft involved and to provide modification and/or operational usage options to meet 8,000 flight hours, should the predicted operational limits of the aircraft be less than 8,000 flight hours. In order to accomplish the above objectives, an assessment of the initial quality of the structure resulting from material and structural manufacturing and processing operations was necessary. This assessment was accomplished using the Equivalent Initial Quality Method.

3.1 F/RF-4C/D INITIAL QUALITY ASSESSMENT

The F/RF-4C/D damage tolerance and life assessment program was conducted from June 1973 to June 1974. The program included an initial quality assessment which was accomplished using the fastener holes contained in the inner wing lower torque box skins of a F-4B/J full-scale fatigue test article. The F-4B/J was manufactured in 1966, between the F-4C and F-4D and is thus representative of the F-4C and F-4D aircraft. The fatigue test article was subjected to a ten level, blocked low-high

Three methods were used to determine the F/RF-4C/D equivalent initial quality. These methods involved: (1) a combined crack propagation analysis and fractographic test data; (2) a combined crack propagation analysis and measured final (11,800 hours) crack depth and shape; and (3) a combined crack propagation analysis, calculated crack depth at 8,000 hours and estimated crack shape based on actual crack shapes in adjacent fastener holes. The objective of each method was to establish the actual or equivalent initial crack size at time zero (start of the test). A total of 104 cracks from fastener holes in the F-4B/J lower torque box skins were used to establish the F/RF-4C/D equivalent initial quality. The 104 cracks from fastener holes included: (1) 21 cracks with fractographic traces from 11,800 hours; (2) 49 cracks with measured crack depths at 11,800 hours; and (3) 34 cracks with calculated crack depths at 8,000 hours. The number of holes and the areas of the lower torque box skins that were analyzed are shown in Figure 6.

3.1.1 Cracks with Fractographic Traces from 11,800 Hours

Fractographic analyses were performed for 21 of the largest cracks from fastener holes in the F-4B/J inner wing lower torque box skins. These fractographic analyses involved the use of scanning electron and optical microscope techniques to obtain as much of the crack growth curve as possible. Analytical crack propagation analyses were then performed until there was good agreement between the analytical prediction and the fractographic test data. Retardation due to peak loads

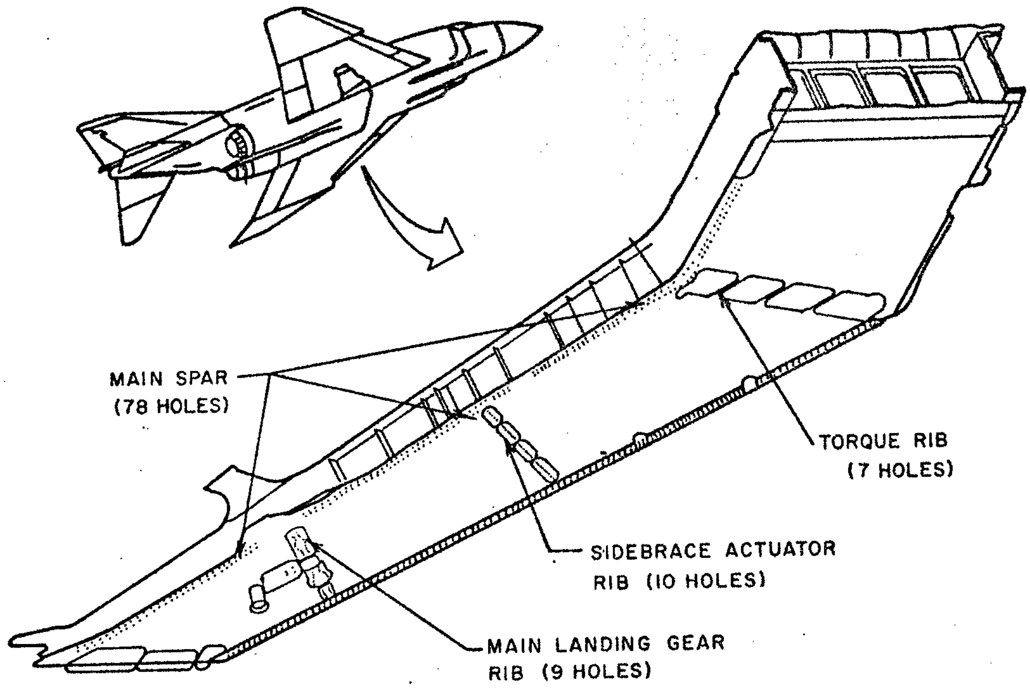


Figure 6. F-4B/J Lower Torque Box Skin

was accounted for based on the Wheeler Method.[11] The Wheeler retardation factor, m, used in the final prediction was chosen to be that m which provided the best correlation between the analytical prediction and the fractographic test data. The flaw shape parameter, Q, used in the predictions was determined from a photograph of the actual crack shape. The initial crack depth of the crack growth prediction which correlated best with the fractographic test data was taken as the equivalent initial quality.

Figure 7 presents a typical example of the determination of the equivalent initial quality using the above method. For this particular example, the retardation factor and flaw shape parameter were found to be 2.05 and 1.97, respectively. This resulted in an equivalent initial quality of .0029 inch. A similar type of analysis was performed for the other fractographic traces with the corresponding crack growth parameters shown in Table 2.

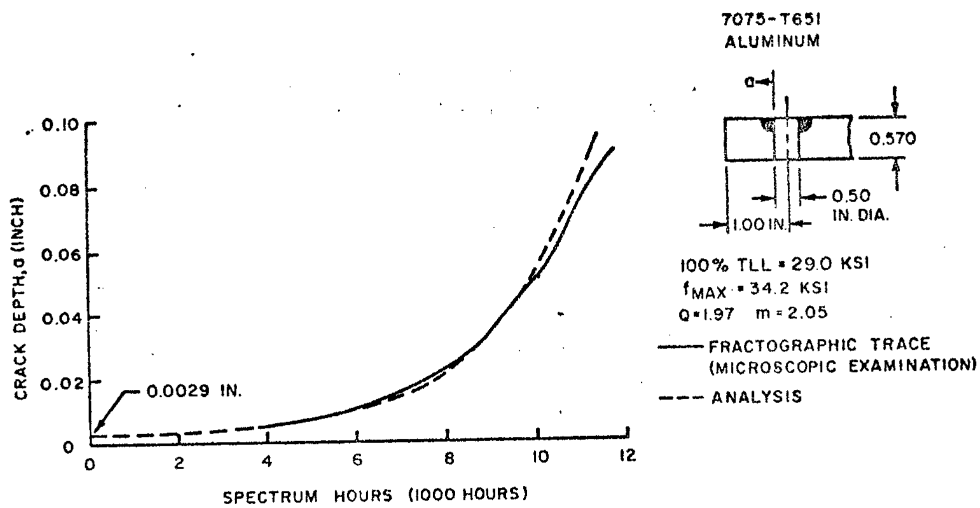


Figure 7. Typical Equivalent Initial Crack Depth from Fractographic Trace for F/RF-4C/D Aircraft

HOLE NUMBER	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	"BEST FIT" RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
60L	S	.250	31.5	16.0	1.73	3.8	.032	.00036
120L	SW	.156	32.4	0	2.41	1.5	.087	.00250
209L	DW	.250	29.0	0	1.97	2.1	.098	.00290
221L	SW	.125	32.0	0	1.32	1.7	.023	.00090
322L	SW	.156	29.0	0	2.40	2.0	.103	.00470
336L	S	.125	32.0	0	1.92	2.1	.032	.00251
505R	S	.125	32.0	0	1.48	2.3	.035	.00150
517R	DW	.125	30.0	0	2.41	1.8	.064	.00360
2143L	S	.125	24.0	0	1.58	2.2	.042	.00540
2144L	S	.125	24.0	0	1.93	1.0	.038	.00370
2166L	SW	.125	26.9	7.7	1.60	2.9	.063	.00510
2198L	S	.125	26.9	7.7	1.97	1.7	.065	.00220
2199L	S	.125	26.9	7.7	1.97	2.3	.058	.00501
2300R	SW	.125	27.0	0	2.28	1.3	.040	.00356
2302R	SW	.125	27.0	0	2.13	1.4	.072	.00370
2309L	D	.125	24.0	0	1.96	2.3	.030	.00670
2400L	S	.125	32.0	0	2.41	1.0	.135	.00240
2404R	S	.125	32.0	0	1.80	2.5	.040	.00254
2410R	D	.125	32.0	0	2.01	2.7	.037	.00351
2424R	S	.125	32.0	0	2.40	2.0	.034	.00272
2425L	D	.125	32.0	0	2.41	2.2	.048	.00383

NOTES:

1. Flaw Types: Single, S
   Double, D
   Width Correction, W

2. Fastener Hole Numbers Per Reference 12.

Table 2. Equivalent Initial Crack Depths from Fractographic Traces for F/RF-4C/D Aircraft

3.1.2 Cracks with Measured Crack Depths at 11,800 Hours

Actual crack depths for 49 cracked fastener holes in the F-4B/J lower torque box skins were determined at failure (11,800 hours) using optical microscope measurements. The flaw shape parameter, Q, was determined for each crack from a photograph of the actual crack shape. An average Wheeler retardation factor, m, was used in the predictions which was based on the peak spectrum stress, f_{max}, and the flaw shape parameter, Q. The average retardation factor was obtained from Figure 8, which is based on crack growth from element fatigue tests, full-scale fatigue tests, and several different spectra. The material in these tests was 7075-T651 aluminum. The spectra included one block spectrum and more than 20 different flight-by-flight spectra.[13] The initial crack depth of the crack growth prediction which resulted in the measured crack depth after 11,800 hours was taken as the equivalent initial quality.

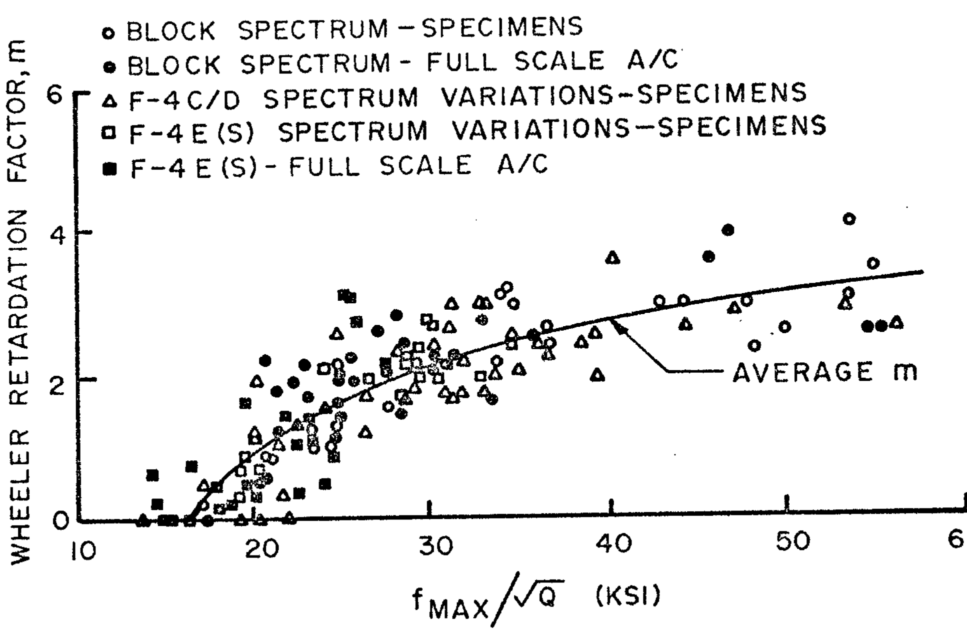


Figure 8. Average Wheeler Retardation Factor, m

Figure 9 presents a typical example of the determination of the equivalent initial quality using the above method. For this particular example, the retardation factor and flaw shape parameter were found to be 1.891 and 1.97, respectively. This resulted in an equivalent initial quality of .0017 inch. A similar type of analysis was performed for the other 48 fastener holes with the resulting equivalent initial crack depths and the corresponding crack growth parameters presented in Table 3.

3.1.3 Cracks with Calculated Crack Depths at 8,000 Hours

Calculated crack depths for 34 fastener holes in the F-4B/J lower torque box skins were determined that had eddy current crack indications

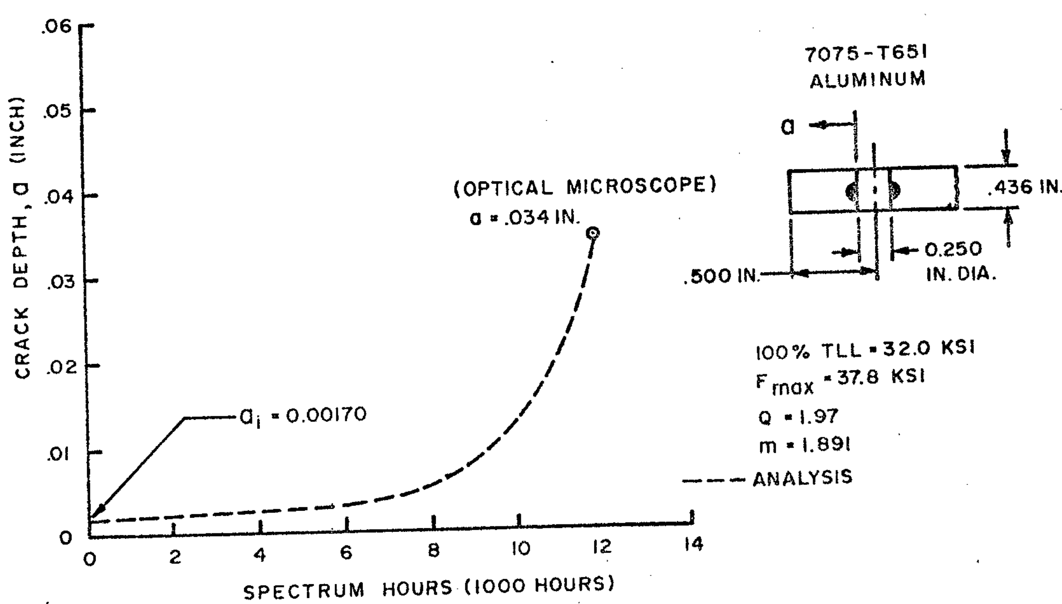


Figure 9. Typical Equivalent Initial Crack Depth from Measured Crack for F/RF-4C/D Aircraft

HOLE NUMBER	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	AVERAGE RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)[1]
49L	SW	.125	32.0	0	2.41	1.649	.013	.00210
49.1L	SW	.125	32.0	0	1.92	1.920	.008	.00153
51L	SW	.156	32.4	0	2.41	1.680	.048	.00231
51R	SW	.156	32.4	0	1.15	2.451	.006	.00074
52R	SW	.156	32.4	0	1.96	1.925	.004	.00140
53R	DW	.156	32.4	0	2.41	1.680	.020	.00200
54L	SW	.156	31.5	0	2.41	1.609	.017	.00221
58R	SW	.250	31.5	11.3	1.27	2.510	.003	.00007
60R	S	.250	31.5	16.0	1.68	3.800	.010	.00005
101L	DW	.156	31.5	0	2.41	1.609	.020	.00213
103L	S	.156	31.5	0	2.41	1.609	.007	.00218
104L	DW	.156	31.5	0	2.41	1.609	.022	.00215
104R	SW	.156	31.5	0	2.41	1.609	.011	.00210
105L	SW	.156	31.5	0	2.41	1.609	.012	.00212
106L	DW	.156	31.5	0	2.41	1.609	.030	.00220
107L	DW	.156	31.5	0	2.41	1.609	.015	.00200
119R	DW	.156	32.4	0	2.41	1.680	.016	.00194
120R	SW	.156	32.4	0	2.41	1.680	.016	.00204
200R	DW	.125	29.0	0	2.41	1.388	.034	.00274
205R	SW	.156	29.0	0	1.62	1.890	.009	.00159
206R	SW	.156	29.0	0	2.03	1.617	.013	.00212
207R	SW	.156	29.0	0	2.07	1.592	.020	.00228
209R	S	.250	29.0	0	1.97	1.655	.020	.00238
213R	SW	.125	32.0	4.2	1.79	2.105	.005	.00093
218L	SW	.125	32.0	4.2	1.97	2.009	.014	.00118
222R	DW	.125	32.0	0	2.41	1.649	.018	.00202
230L	DW	.125	30.0	0	2.41	1.481	.025	.00247
235R	DW	.125	30.0	0	2.30	1.543	.019	.00228
305L	S	.125	31.5	0	2.30	1.668	.018	.00238
315R	S	.125	31.5	0	2.41	1.609	.034	.00272
341L	S	.125	32.0	0	1.97	1.891	.020	.00198
508R	DW	.125	32.0	0	1.97	1.891	.034	.00170
509R	S	.125	32.0	0	1.97	1.891	.046	.00218
511R	SW	.125	32.0	0	2.41	1.649	.015	.00210
516R	DW	.125	32.0	0	2.01	1.868	.023	.00168
519R	DW	.125	32.0	0	2.41	1.649	.027	.00211
2135L	S	.125	26.9	7.7	1.81	1.836	.059	.00190
2211L	S	.125	24.0	0	1.75	1.305	.050	.00385
2301L	DW	.125	27.0	0	2.41	1.181	.009	.00278
2301R	SW	.125	27.0	0	2.41	1.181	.013	.00308
2302L	SW	.125	27.0	0	2.41	1.181	.003	.00225
2305L	D	.125	27.0	0	2.41	1.181	.022	.00343
2307L	S	.125	27.0	0	2.12	1.367	.049	.00368
2310L	S	.125	24.0	0	1.46	1.553	.017	.00260
2311R	SW	.125	24.0	0	2.41	.807	.025	.00430
2400R	S	.125	32.0	0	2.41	1.649	.030	.00260
2408L	S	.125	32.0	0	2.05	1.845	.021	.00202
2409R	S	.125	32.0	0	2.22	1.750	.023	.00226
2428R	D	.125	32.0	0	2.41	1.649	.037	.00250

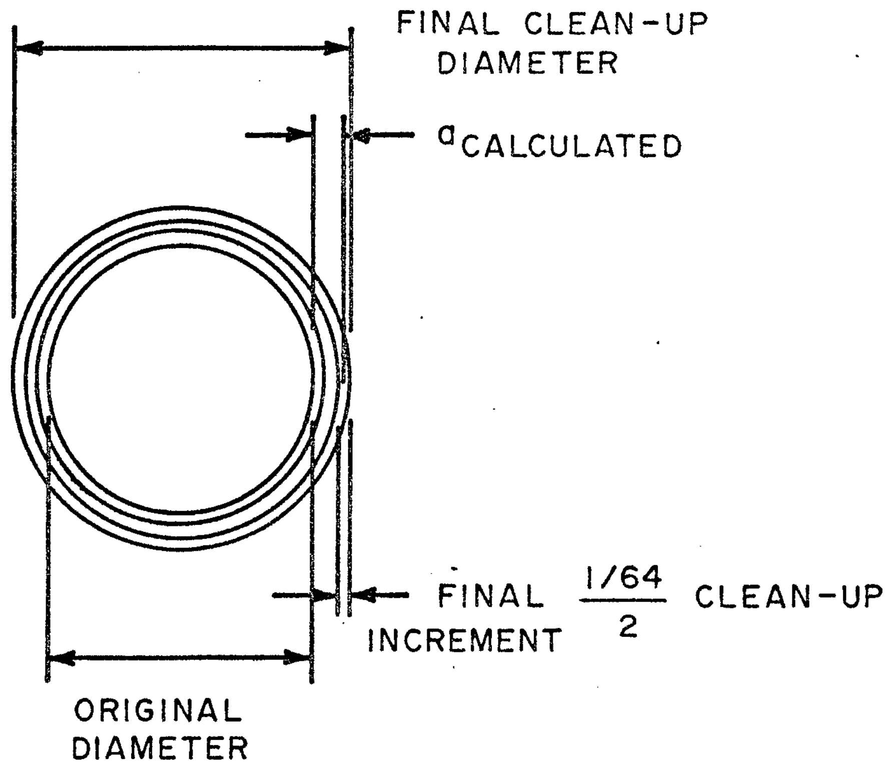
NOTES: 1. Single, S
Double, D
Width Correction, W

2. Fastener Hole Numbers Per Reference 12.

Table 3. Equivalent Initial Crack Depths from Measured Cracks for F/RF-4C/D Aircraft

at 8,000 hours during the F-4B/J full-scale fatigue test. These fastener holes were reamed to remove the cracks and oversize fasteners were installed prior to further testing. Since the actual cracks were destroyed at 8,000 hours, the crack depths were estimated based on the data resulting from the reaming and eddy current information. Cracks were cleaned up in 1/64 inch diametrical increments, until there was no eddy current indication. A calculated crack depth at 8,000 hours was determined from Figure 10. Actual crack shapes at 11,800 hours in adjacent fastener holes were utilized to determine an average flaw shape parameter, Q, for each analysis location. An average Wheeler retardation factor, m, was obtained from Figure 8 and was used in the predictions. The initial crack depth of the crack growth prediction which resulted in the calculated crack depth after 8,000 hours was taken as the equivalent initial quality.

Figure 11 presents a typical example of the determination of the equivalent initial quality using the above method. For this particular example, the retardation factor and flaw shape parameter were found to be 2.014 and 1.96, respectively. This resulted in an initial quality of .00198 inch. A similar type of analysis was performed for the other 33 fastener holes with the resulting equivalent initial crack depths and the corresponding crack growth parameters presented in Table 4.



$$a_{CALCULATED} = \frac{\text{FINAL DIA.} - \frac{1/64}{2} - \text{ORIGINAL DIA.}}{2}$$

Figure 10. Calculated Crack Depth from Reamed Fastener Hole

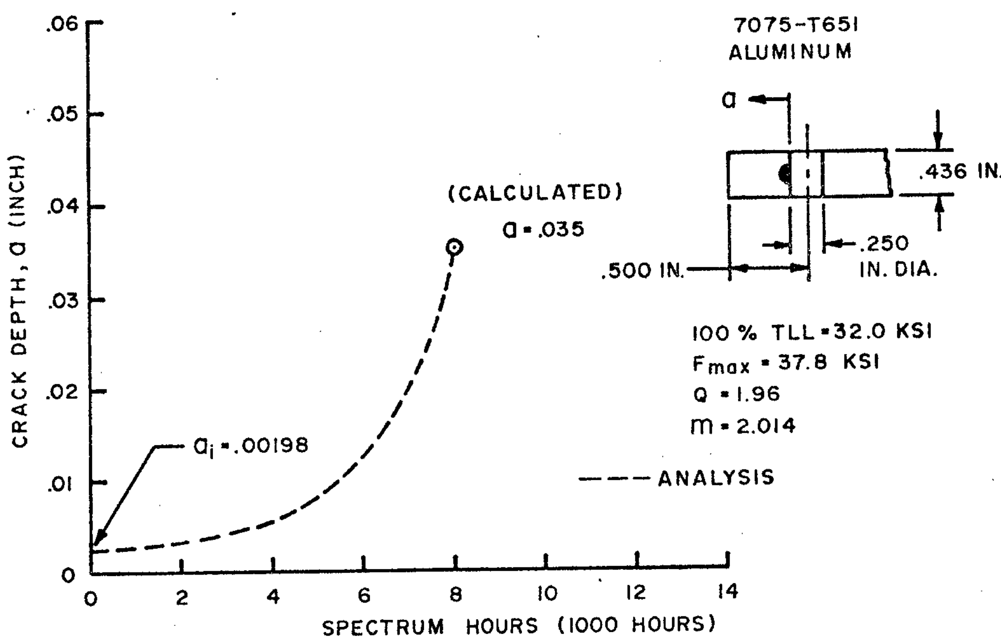


Figure 11. Typical Equivalent Initial Crack Depth from Calculated Crack for F/RF-4C/D Aircraft

HOLE NUMBER	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	AVERAGE RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
50L	DW	.125	32.0	0	1.96	1.897	.019	.00211
50R	SW	.125	32.0	0	1.96	1.897	.043	.00272
55R	S	.156	31.5	0	2.03	1.820	.005	.00025
57L	S	.250	31.5	11.3	1.68	2.291	.082	.00068
57R	SW	.250	31.5	11.3	1.68	2.291	.012	.00025
102L	DW	.156	31.5	0	2.41	1.609	.005	.00210
102R	S	.156	31.5	0	2.41	1.609	.005	.00240
107R	S	.156	31.5	0	2.41	1.609	.027	.00342
111R	S	.156	31.5	0	2.41	1.609	.027	.00342
201R	S	.125	29.0	0	2.03	1.617	.004	.00247
206L	S	.156	29.0	0	2.03	1.617	.012	.00284
207L	DW	.156	29.0	0	2.03	1.617	.036	.00299
212L	SW	.125	32.0	4.2	1.96	2.014	.043	.00210
214L	DW	.125	32.0	4.2	1.96	2.014	.043	.00189
214R	SW	.125	32.0	4.2	1.96	2.014	.035	.00198
215L	DW	.125	32.0	4.2	1.96	2.014	.019	.00158
216L	DW	.125	32.0	4.2	1.96	2.014	.051	.00200
216R	DW	.125	32.0	4.2	1.96	2.014	.035	.00178
217L	DW	.125	32.0	4.2	1.96	2.014	.019	.00158
217R	DW	.125	32.0	4.2	1.96	2.014	.089	.00272
218R	DW	.125	32.0	4.2	1.96	2.014	.058	.00210
219R	DW	.125	32.0	4.2	1.96	2.014	.035	.00178
220R	SW	.125	32.0	4.2	1.96	2.014	.043	.00210
220.5L	DW	.125	32.0	4.2	1.96	2.014	.089	.00272
220.5R	DW	.125	32.0	4.2	1.96	2.014	.019	.00158
232R	SW	.125	30.0	0	2.31	1.537	.051	.00400
321L	S	.156	29.0	0	2.03	1.617	.114	.00662
324L	S	.156	29.0	0	2.03	1.617	.012	.00286
326L	S	.156	29.0	0	2.03	1.617	.027	.00330
338R	D	.125	32.0	0	1.96	1.897	.012	.00210
340R	SW	.125	32.0	0	1.96	1.897	.012	.00202
342R	DW	.125	32.0	0	1.96	1.897	.012	.00194
379R	S	.125	32.4	0	2.17	1.808	.004	.00195
390R	S	.125	32.4	0	2.17	1.808	.012	.00246

NOTES: 1. Flaw Types: Single, S; Double, D; Width Correction, W
2. Fastener Hole Numbers Per Reference 12

Table 4. Equivalent Initial Crack Depths From Calculated Cracks for F/RF-4C/D Aircraft

3.1.4 F/RF-4C/D Distribution of Initial Flaws

A F/RF-4C/D equivalent initial crack depth statistical distribution was obtained from the individual equivalent initial crack depths established for each fastener hole. This equivalent initial crack depth statistical distribution is representative of the initial quality of the fastener holes produced by the particular material and structural manufacturing and processing operations involved. A total population of 104 equivalent initial crack depths was ordered as shown in the histogram of

Figure 12. This histogram was used to obtain the equivalent initial crack depth statistical distribution presented in Figure 13. It was found that a Johnson Su distribution matched most closely with the equivalent initial crack depth data.

In order to utilize a finite sample size for fleet aircraft projections, a 95% confidence distribution was established as shown in Figure 13. This distribution can be used in conjunction with a knowledge of the number of fastener holes per structural component which can be reworked economically to select the initial crack size used to determine the economic life of the structural component. For example, assume that it is determined that one out of one thousand fastener holes can be reworked economically. Hence, 99.9% of the flaws are less than the equivalent initial crack depth of interest. Figure 13 indicates that with 95% confidence, 99.9% of the flaws have an equivalent depth less than 0.01 inch, or that one out of one thousand flaws have an equivalent depth equal to or greater than 0.01 inch. Hence, the initial crack depth to be used in the economic life predictions is 0.01 inch.

3.2 F-4E(S) INITIAL QUALITY ASSESSMENT

The F-4E(S) damage tolerance and life assessment program was conducted from August 1974 to May 1975. The program included an initial quality assessment which was accomplished using the fastener holes contained in two full-scale fatigue test articles. The fastener holes consisted of F-4E(S) inner and outer wing lower torque box skin reamed

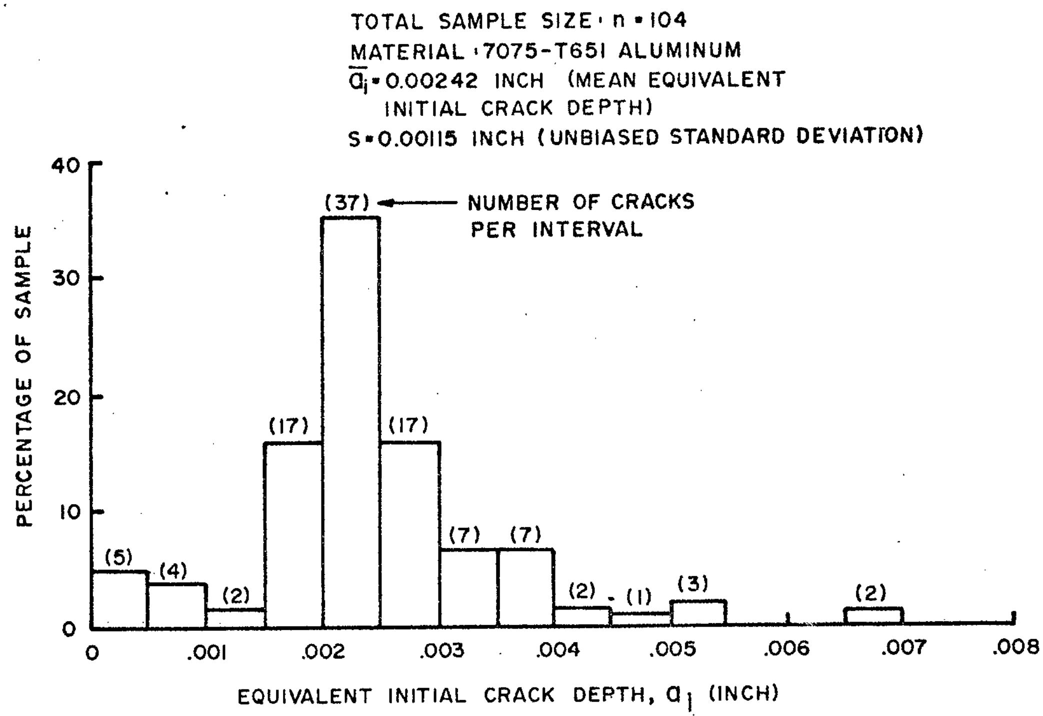


Figure 12. Equivalent Initial Crack Depth Histogram for F/RF-4C/D Aircraft

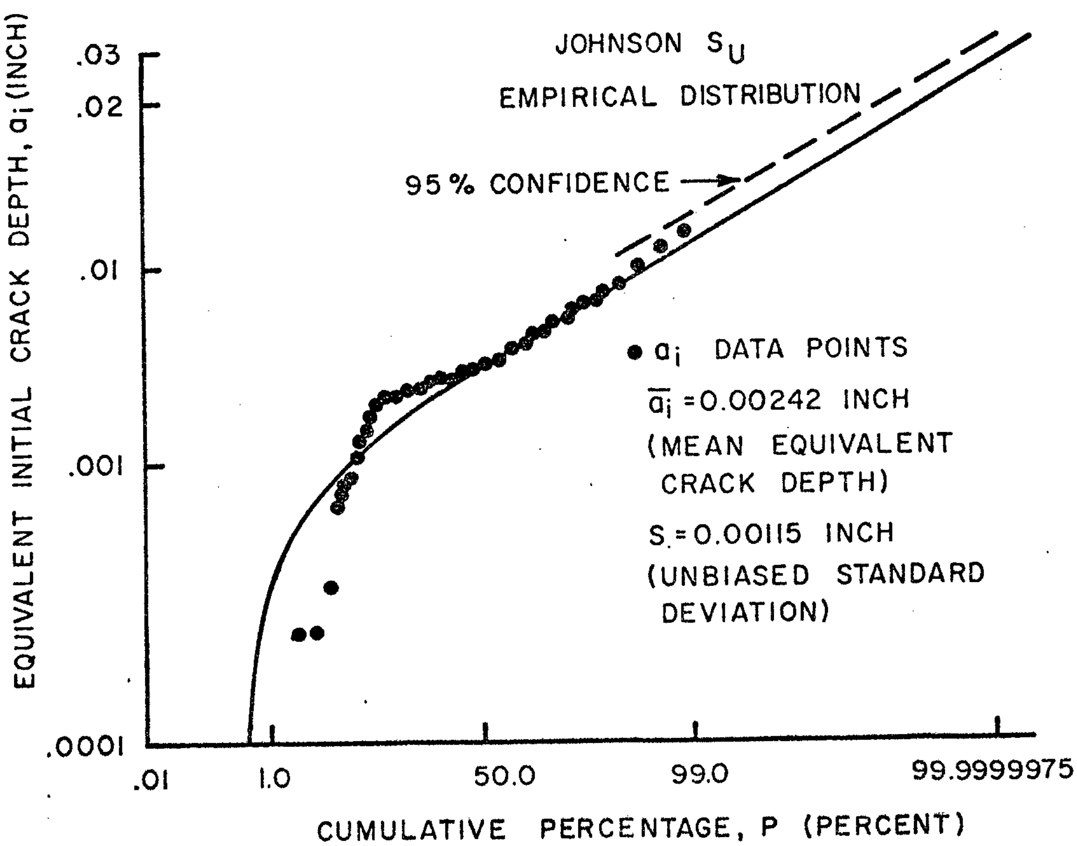


Figure 13. Equivalent Initial Crack Depth Distribution for F/RF-4C/D Aircraft

fastener holes, F-4E(S) fuselage and inner wing trailing edge aileron closure skin drilled fastener holes, and F-4B/J inner wing lower torque box skin reamed fastener holes. The F-4B/J initial quality results were presented in the previous section for the F/RF-4C/D aircraft. The F-4E(S) full-scale fatigue test article was fatigue tested for 10,000 hours to the F-4E(S)-4 points test spectrum.(14) The material for the F-4E(S) inner and outer wing lower torque box skins was 7075-T651 aluminum while the material for the F-4E(S) fuselage and inner wing trailing edge aileron closure skins was 7178-T6 aluminum. A fatigue test strain survey and a finite element computer program were utilized to determine the stress levels at the locations being analyzed. Reamed fastener holes inspected during the F-4E(S) full-scale fatigue test article teardown at 10,000 hours included all lower inner wing torque box skin fastener holes and all primary outer wing lower torque box skin fastener holes. Drilled fastener holes inspected during the teardown included all primary fuselage and inner wing trailing edge fastener holes and all inner wing lower torque box integral stiffener fastener holes. All cracks that were detected by NDT techniques were exposed and evaluated. In addition, fastener holes that were likely candidates for cracks were bisected and examined by optical microscope even if no NDT crack indications existed. Metallurgical investigations of the flaw origins revealed that the flaws were the result of various mechanical sources.

Two methods were used to determine the equivalent initial quality for the F-4E(S) full-scale fatigue test article. These methods involved

(1) a combined crack propagation analysis and fractographic test data and (2) a combined crack propagation analysis and measured final (10,000 hours) crack depth and shape. The objective of each method was to establish the actual or equivalent initial crack size at time zero (start of the test).

For the reamed fastener holes, a total of 104 cracks in the F-4B/J inner wing lower torque box skins and a total of 51 cracks in the F-4E(S) inner and outer wing lower torque box skins were used to establish the F-4E(S) reamed hole equivalent initial quality. The 51 F-4E(S) cracks included 25 cracks with fractographic traces from 10,000 hours and 26 cracks with measured crack depths at 10,000 hours. For the drilled fastener holes, a total of 75 cracks in the F-4E(S) center fuselage and the inner wing trailing edge aileron closure skins were used to establish the F-4E(S) drilled hole equivalent initial quality. The 75 cracks included 34 cracks with fractographic traces from 10,000 hours and 41 cracks with measured crack depths at 10,000 hours. A summary of the data base for the drilled and reamed equivalent initial crack depth distributions is shown in Table 5.

3.2.1 Cracks with Fractographic Traces from 10,000 Hours

Fractographic analyses were performed for 25 of the largest cracks from reamed fastener holes in the F-4E(S) inner and outer wing lower torque box skins and for 34 of the largest cracks from drilled fastener

Full Scale Test Article	REAMED Fractographic Traces	Measured Crack Depths	Calculated Crack Depths	TOTAL	DRILLED Fractographic Traces	Measured Crack Depths	TOTAL
F-4B/J	21	49	34	104	--	--	--
F-4E(S)	25	26	--	51	34	41	75
	46	75	34	155	34	41	75

155 Reamed Fastener Holes
75 Drilled Fastener Holes

230 Total Equivalent Initial Crack Depths

Table 5. Equivalent Initial Crack Depth Data Base for F-4E(S) Aircraft

holes in the F-4E(S) center fuselage and aileron closure skins. These fractographic analyses involved the use of scanning electron and optical microscope techniques to obtain as much of the crack growth curve as possible. Analytical crack propagation analyses were then performed until there was good agreement between the analytical prediction and the fractographic test data. Retardation due to peak loads was accounted for based on the Wheeler Method. The retardation factor, m, used in the final prediction was chosen to be that m which provided the best correlation between the analytical prediction and the fractographic test data. The flaw shape parameter, Q, used in the predictions was determined from a photograph of the actual crack shape. The initial crack depth of the crack growth prediction which correlated best with the fractographic test data was taken as the equivalent initial quality.

Figure 14 presents a typical example of the determination of the equivalent initial quality using the above method. For this particular example, the retardation factor and flaw shape parameter were found to

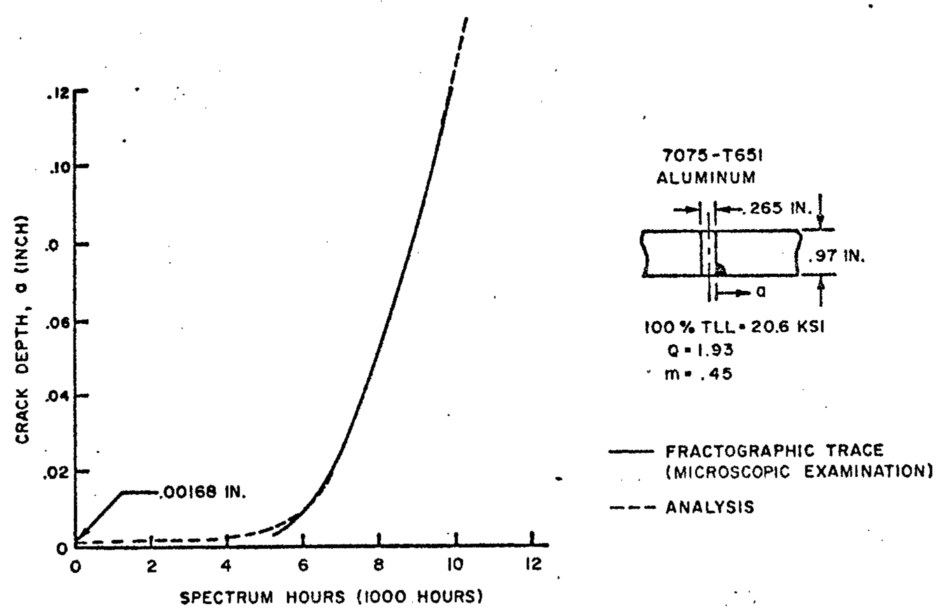


Figure 14. Typical Equivalent Initial Crack Depth from Fractographic Trace for F-4E(S) Aircraft

be .45 and 1.93, respectively. This resulted in an equivalent initial quality of .00168 inch. A similar analysis was performed on other fractographic traces with corresponding crack growth parameters as shown in Table 6 for reamed fastener holes and in Table 7 for drilled fastener holes.

3.2.2 Cracks with Measured Crack Depths at 10,000 Hours

Actual crack depths for 26 cracked reamed fastener holes in the F-4E(S) inner and outer wing lower torque box skins and 41 cracked drilled fastener holes in the F-4E(S) center fuselage and aileron closure skin were determined at 10,000 hours using optical microscope measurements. The flaw shape parameter, Q, was determined for each crack from a

HOLE NUMBER	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS 100% TLL (KSI)	BEARING STRESS 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	"BEST FIT" RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
128R	SW	.265	24.4	21.8	1.93	6.00	.0399	.00305
129R	SW	.265	24.4	21.8	1.93	6.00	.0362	.00296
130R	SW	.265	24.4	21.8	1.93	2.20	.1776	.00181
131R	SW	.265	24.4	21.8	1.93	4.50	.0366	.00206
144R	S	.172	21.6	12.5	1.72	1.10	.1081	.00209
336R	S	.140	22.9	6.0	1.93	1.70	.0376	.00331
339R	S	.140	22.6	5.7	1.84	.55	.1088	.00284
346R	S	.140	21.8	2.6	1.93	.45	.0487	.00379
378R	S	.140	21.9	10.0	1.68	1.50	.0544	.00223
409R	S	.172	21.2	6.4	2.43	.80	.0408	.00376
2100R	SW	.132	20.6	17.1	1.93	.45	.1205	.00168
2102L	SW	.132	17.3	15.8	1.93	.90	.0670	.00279
2106R	SW	.132	16.3	16.2	1.93	.00	.0425	.00213
2107L	SW	.132	16.2	15.8	1.93	.30	.0947	.00295
2110R	SW	.132	13.3	15.1	1.93	.10	.0570	.00407
2206R	SW	.132	13.0	12.8	1.93	.00	.0345	.00444
2300L	SW	.125	21.6	2.9	2.42	.70	.0363	.00645
2308L	SW	.125	20.2	2.1	1.93	.00	.0985	.00682
2312L	SW	.125	20.2	6.5	1.93	.00	.0726	.00491
2320L	SW	.125	19.7	.9	1.93	.00	.0450	.00616
2580L	SW	.125	11.6	40.0	2.42	2.40	.0486	.00227
2580R	SW	.125	11.6	40.0	1.93	1.96	.0611	.00140
5022R	SW	.125	13.3	0.0	1.71	.20	.0586	.00140
5040R	DW	.125	18.3	0.0	1.97	.70	.1312	.00110
5060R	SW	.125	13.1	0.0	1.80	.50	.0522	.00170

NOTES: 1. Flaw Types: Single, S
Double, D
Width Correction, W

2. Fastener Hole Numbers Per Reference 14.

Table 6. Equivalent Initial Crack Depths for Reamed Holes from Fractographic Traces for F-4E(S) Aircraft

FUSELAGE OR WING STATION OF HOLE	FUSELAGE OR WING LOCATION	FLAW TYPE	HOLE RADIUS r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER, Q	"BEST FIT" RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
FS 289.5	L/H Skin @ STR 1	DW	.095	12.0	36.0	1.91/1.00	.75	.2473	.00110
FS 291	L/H Skin @ STR 1	DW	.095	18.0	40.4	1.00	2.30	.6310	.00056
FS 359.7	L/H Skin @ STR 1	SW	.095	10.9	.0	1.91	.00	.1028	.00573
FS 267.5	R/H Skin @ STR 1	SW	.095	12.1	32.4	1.91/1.00	.00	.2330	.00222
FS 269.1	R/H Skin @ STR 1	SW	.095	12.1	29.0	1.91/1.00	.00/0.25	.1800	.00299
FS 283.0	R/H Skin @ STR 1	SW	.095	12.2	30.5	1.60/1.00	.00/0.80	.1728	.00168
FS 288.2	R/H Skin @ STR 1	DW	.095	12.0	33.0	1.70/1.00	.90/1.00	.1288	.00105
FS 368.0	L/H Skin @ STR 3	S	.095	11.0	28.7	1.00	.40	.2030	.00109
FS 307.0	R/H Skin @ STR 4	D	.095	18.5	14.4	1.50/1.00	.00	.4557	.00125
FS 254.0	L/H STR 1	DW	.100	18.0	20.0	1.97/1.00	.30/1.50	.3324	.00187
FS 266.7	L/H STR 1	S	.100	15.0	30.0	1.97/1.00	.00/0.50	.3032	.00241
FS 268.9	L/H STR 1	DW	.095	15.0	30.0	1.97/1.00	.50	.2934	.00141
FS 276.6	L/H STR 1	DW	.095	11.3	55.8	1.97/1.00	1.70	.3030	.00092
FS 283.8	L/H STR 1	DW	.100	12.1	40.3	1.97/1.00	1.68/1.00	.4164	.00130
FS 313.5	L/H STR 1	DW	.095	15.2	31.7	1.97/1.00	.30	.3205	.00144
FS 316.1	L/H STR 1	S	.078	15.2	31.7	1.97	.95	.0577	.00105
FS 261.6	R/H STR 1	DW	.100	13.8	24.1	1.97/1.00	.63/2.00	.2230	.00236
FS 268.9	R/H STR 1	DW	.095	12.1	40.0	1.97/1.00	.30/2.17	.3157	.00155
FS 271.1	R/H STR 1	SW	.100	13.9	37.4	1.97/1.00	.50/1.00	.2058	.00143
FS 275.5	R/H STR 1	DW	.095	11.3	45.0	1.97/1.00	2.00	.3147	.00129
FS 280.0	R/H STR 1	DW	.095	11.7	40.0	1.97/1.00	1.70	.3038	.00257
FS 290.5	R/H STR 1	DW	.095	15.0	38.0	1.97/1.00	2.00	.3007	.00125
FS 271.2	L/H STR 3	SW	.095	22.4	38.1	1.22/1.00	2.39/2.80	.2818	.00061
FS 269.8	R/H STR 3	S	.100	14.0	23.6	1.20/1.00	.00	.2756	.00184
FS 289.0	R/H STR 3	D	.125	15.0	25.0	1.97	.79	.2721	.00143
FS 304.0	R/H STR 4	S	.128	18.5	14.4	1.97/1.00	.50/0.00	.4900	.00234
FS 304.0	R/H STR 4	DW	.050	18.5	.0	1.20/1.00	.75/1.00	.3087	.00454
FS 330.0	L/H STR 5	DW	.095	17.0	19.5	1.97/1.00	.80	.1370	.00148
FS 271.0	L/H STR 6	S	.095	15.0	35.0	1.78/1.00	.00	.2915	.00175
FS 249	R/H Up. Long	DW	.160	11.9	.0	1.91	.00	.3350	.00206
XA 147.0	L/H AIL CL Skin	D	.125	16.2	.0	1.00	.50	.3338	.00350
XA 147.0	L/H AIL CL Skin	D	.078	16.0	40.0	1.00	2.04	.3460	.00146
XA 133.0	R/H AIL CL Skin	D	.125	16.2	.0	1.00	.90	.1450	.00215
XA 147.0	R/H AIL CL Skin	D	.125	16.2	.0	1.00	.25	.3297	.00289

NOTES: 1. Flaw Types: Single, S; Double, D; Width Correction, W

Table 7. Equivalent Initial Crack Depths for Drilled Holes from Fractographic Traces for F-4E(S) Aircraft

photograph of the actual crack shape. An average Wheeler retardation factor (Figure 8) was used in the predictions which was based on the peak spectrum stress, f_{max}, and the flaw shape parameter, Q. The initial crack depth of the crack growth prediction which resulted in the measured crack depth after 10,000 hours was taken as the equivalent initial quality.

Figure 15 presents a typical example of the determination of the equivalent initial quality using the above method. For this particular example, the retardation factor and flaw shape parameter were found to be 1.167 and 1.93, respectively. This resulted in an equivalent initial quality of .00197 inch. Tables 8 and 9 present the resulting equivalent initial crack depths and the corresponding crack growth parameters for the 26 cracked reamed fastener holes and the 41 cracked drilled fastener holes, respectively.

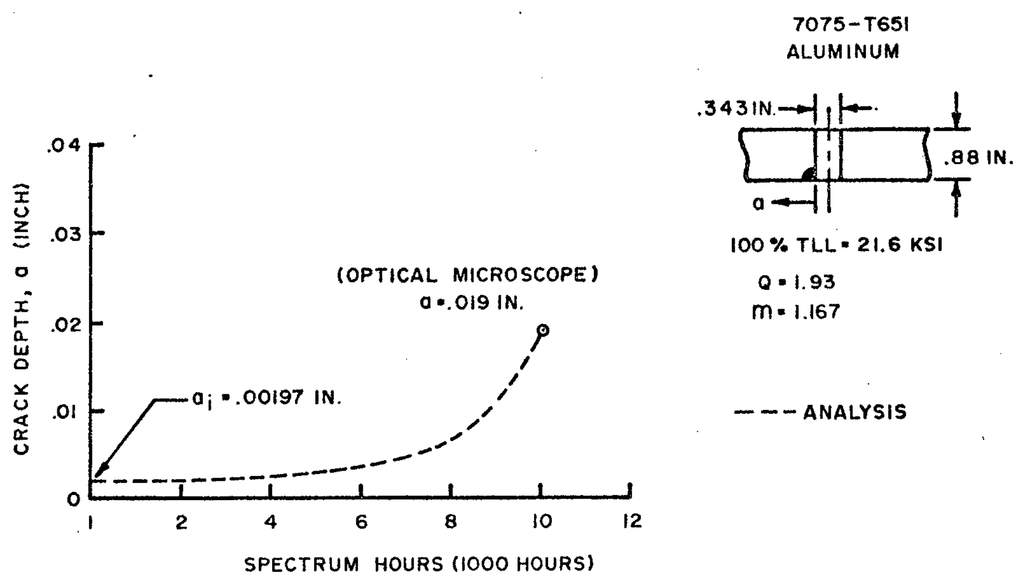


Figure 15. Typical Equivalent Initial Crack Depth from Measured Crack for F-4E(S) Aircraft

HOLE NUMBER	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	AVERAGE RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
129L	SW	.265	24.4	21.8	1.93	6.000	.051	.00360
130L	S	.265	24.4	21.8	1.93	1.918	.010	.00135
131L	SW	.265	24.4	21.8	1.93	4.500	.085	.00289
134R	S	.172	22.6	5.7	1.93	.787	.002	.00166
141R	S	.172	21.6	12.5	1.93	1.167	.011	.00189
143R	S	.172	21.6	12.5	2.18	.956	.004	.00193
145R	S	.172	21.6	12.5	1.93	1.167	.019	.00197
146R	S	.172	21.6	12.5	1.31	1.701	.005	.00124
272R	S	.156	21.2	6.4	1.93	.637	.006	.00205
333R	S	.140	20.6	17.1	1.23	1.906	.004	.00103
337R	S	.140	22.9	6.0	2.43	.400	.017	.00299
341L	S	.140	22.6	5.7	2.43	.332	.017	.00311
379R	S	.140	21.9	10.0	1.93	1.030	.008	.00193
2109L	SW	.132	13.3	15.1	1.65	.492	.022	.00270
2207R	SW	.132	13.0	12.8	1.79	.000	.022	.00353
2309L	SW	.125	20.2	2.1	2.43	.000	.009	.00476
2318L	SW	.125	20.2	6.5	2.43	.000	.019	.00535
2442L	S	.125	17.9	12.8	2.25	.413	.018	.00257
5019L	SW	.125	13.3	0.0	1.37	.000	.004	.00060
5019R	SW	.125	13.3	0.0	1.97	.000	.044	.00110
5037L	SW	.125	18.3	0.0	1.65	.000	.027	.00030
5037R	SW	.125	18.3	0.0	1.72	.000	.046	.00040
5040L	SW	.125	18.3	0.0	1.97	.000	.058	.00080
5067L	DW	.125	13.1	0.0	1.97	.000	.047	.00140
5239L	D	.125	19.5	23.9	1.18	2.320	.008	.00070
5239R	S	.125	19.5	23.9	1.79	1.950	.024	.00110

NOTES: 1. Flaw Types: Single, S; Double, D; Width Correction, W
2. Fastener Hole Numbers Per Reference 14

Table 8. Equivalent Initial Crack Depths for Reamed Holes from Measured Cracks for F-4E(S) Aircraft

FUSELAGE STATION OF HOLE	FUSELAGE OR WING LOCATION	FLAW TYPE	HOLE RADIUS, r (INCH)	GROSS STRESS, 100% TLL (KSI)	BEARING STRESS, 100% TLL (KSI)	FLAW SHAPE PARAMETER Q	AVERAGE RETARDATION FACTOR, m	MEASURED CRACK DEPTH, a (INCH)	EQUIVALENT INITIAL CRACK DEPTH, a_i (INCH)
FS 324.5	L/H Skin @ STR 1	SW	.095	11.4	36.2	1.91	.433	.041	.00108
FS 251.5	R/H Skin @ STR 1	SW	.095	12.6	29.7	1.00	1.090	.059	.00080
FS 259.5	R/H Skin @ STR 1	SW	.095	12.4	30.4	1.91/1.00	.176/1.123	.080	.00149
FS 266.8	R/H Skin @ STR 1	SW	.095	12.3	31.1	1.91/1.00	.225/1.156	.128	.00271
FS 268.3	R/H Skin @ STR 1	SW	.095	12.3	31.2	1.91/1.00	.235/1.163	.102	.00174
FS 268.2	L/H STR 1	SW	.100	15.0	30.0	1.97/1.00	.469/1.466	.161	.00162
FS 271.2	L/H STR 1	SW	.100	13.9	37.3	1.97/1.00	.916/1.767	.100	.00116
FS 271.9	L/H STR 1	SW	.100	13.9	37.3	1.31/1.00	1.558/1.767	.229	.00404
FS 273.5	L/H STR 1	SW	.100	12.5	43.0	1.97/1.00	1.142/1.907	.161	.00123
FS 275.0	L/H STR 1	SW	.100	11.5	45.0	1.97/1.00	1.164/1.903	.163	.00365
FS 279.3	L/H STR 1	SW	.100	11.7	46.0	1.85/1.00	1.273/1.884	.100	.00094
FS 302.9	L/H STR 1	SW	.125	13.7	32.5	1.97/1.00	.371/1.352	.256	.00173
FS 304.4	L/H STR 1	SW	.125	13.9	32.5	1.97/1.00	.401/1.379	.071	.00117
FS 304.6	L/H STR 1	SW	.100	13.9	32.5	2.19/1.00	.131/1.379	.044	.00114
FS 307.0	L/H STR 1	SW	.100	14.2	32.0	1.97	.408	.030	.00103
FS 314.4	L/H STR 1	DW	.100	15.0	32.0	1.97/1.00	.852	.025	.00093
FS 315.1	L/H STR 1	SW	.100	15.5	32.0	2.00/1.00	.551/1.554	.034	.00098
FS 315.5	L/H STR 1	SW	.083	15.5	32.0	1.97	.587	.042	.00094
FS 253.5	R/H STR 1	SW	.100	15.5	20.0	1.97	.000	.104	.00198
FS 256.6	R/H STR 1	SW	.100	17.0	23.0	1.97/1.00	.123/1.277	.093	.00136
FS 260.2	R/H STR 1	SW	.100	16.0	28.0	1.76	.710	.040	.00107
FS 260.8	R/H STR 1	SW	.100	16.0	29.5	2.00/1.00	.535/1.564	.142	.00132
FS 269.5	R/H STR 1	SW	.100	12.5	39.0	1.97	.861	.025	.00105
FS 270.3	R/H STR 1	SW	.100	12.5	38.5	1.97/1.00	.823/1.666	.156	.00214
FS 273.3	R/H STR 1	SW	.100	12.5	41.0	1.97/1.00	.500/1.00	.201	.00148
FS 274.0	R/H STR 1	SW	.100	12.5	42.5	1.97/1.00	.500/1.00	.204	.00133
FS 292.8	R/H STR 1	SW	.100	12.0	35.0	1.97/1.00	.339/1.283	.094	.00122
FS 262.6	L/H STR 3	SW	.095	14.2	22.1	1.97	.000	.077	.00211
FS 263.4	L/H STR 3	SW	.095	14.2	22.3	1.97/1.00	.000/0.763	.114	.00317
FS 270.2	L/H STR 3	SW	.095	18.2	30.8	1.97/1.00	.961/1.889	.211	.00158
FS 270.2	L/H STR 3	SW	.095	18.2	30.8	1.97/1.00	.961/1.889	.188	.00137
FS 272.9	L/H STR 3	SW	.095	18.2	30.8	1.97/1.00	.961/1.889	.185	.00136
FS 273.6	L/H STR 3	SW	.095	13.9	24.3	1.97/1.00	.000/0.882	.115	.00268
FS 275.0	L/H STR 3	SW	.095	13.9	24.6	1.97	.000/0.907	.110	.00245
FS 276.6	L/H STR 3	DW	.095	13.8	24.9	1.89	.154	.047	.00129
FS 282.4	L/H STR 3	SW	.095	13.7	26.2	1.55	.367	.043	.00104
FS 286.0	L/H STR 3	SW	.095	13.6	26.9	1.85	.000	.080	.00126
FS 303.0	R/H STR 3	SW	.095	13.2	29.8	1.53	.658	.034	.00095
FS 304.2	R/H STR 3	SW	.095	13.2	29.8	1.97/1.00	.031/1.090	.117	.00129
FS 299.0	R/H STR 4	SW	.128	17.4	14.4	1.97/1.00	.000/0.527	.091	.00173
FS 301.0	R/H STR 4	SW	.128	17.4	14.4	1.97/1.00	.000/0.527	.166	.00187

NOTES: Single, S; Double, D; Width Correction, W

Table 9. Equivalent Initial Crack Depths for Drilled Holes from Measured Cracks for F-4E(S) Aircraft

3.2.3 F-4E(S) Distributions of Initial Flaws

A F-4E(S) equivalent initial crack depth statistical distribution was obtained for reamed fastener holes as well as for drilled fastener holes. The distributions were obtained from the individual equivalent initial crack depths established for each fastener hole. These equivalent initial crack depth statistical distributions are representative of the initial quality of the fastener holes produced by the particular material and structural manufacturing and processing operations involved. The material for the reamed fastener holes was 7075-T651 aluminum while the material for the drilled fastener holes was 7178-T6 aluminum.

The total population of 155 initial flaws for reamed fastener holes consisted of 104 initial flaws from the F-4B/J full-scale fatigue test article, which was previously presented for the F/RF-4C/D aircraft, and 51 initial flaws from the F-4E(S) full-scale fatigue test article. The total population of 155 initial flaws was ordered as shown in the histogram of Figure 16. This histogram was used to obtain the equivalent initial crack depth statistical distribution presented in Figure 17. It was found that a Johnson Su distribution matched most closely with the equivalent initial crack depth data for the reamed fastener holes. Similarly, the total population of 75 initial flaws for drilled fastener holes resulted from the F-4E(S) full-scale fatigue test article and was ordered as shown in the histogram of Figure 18. This histogram was used to obtain the equivalent initial crack depth statistical distribution presented in Figure 19. It was found that a Johnson S_B distribution matched most closely with the equivalent initial crack depth data for the drilled fastener holes. In order to utilize the finite sample size distributions for fleet aircraft projections, 95% confidence distributions were also established for reamed and drilled fastener holes in Figures 17 and 19, respectively.

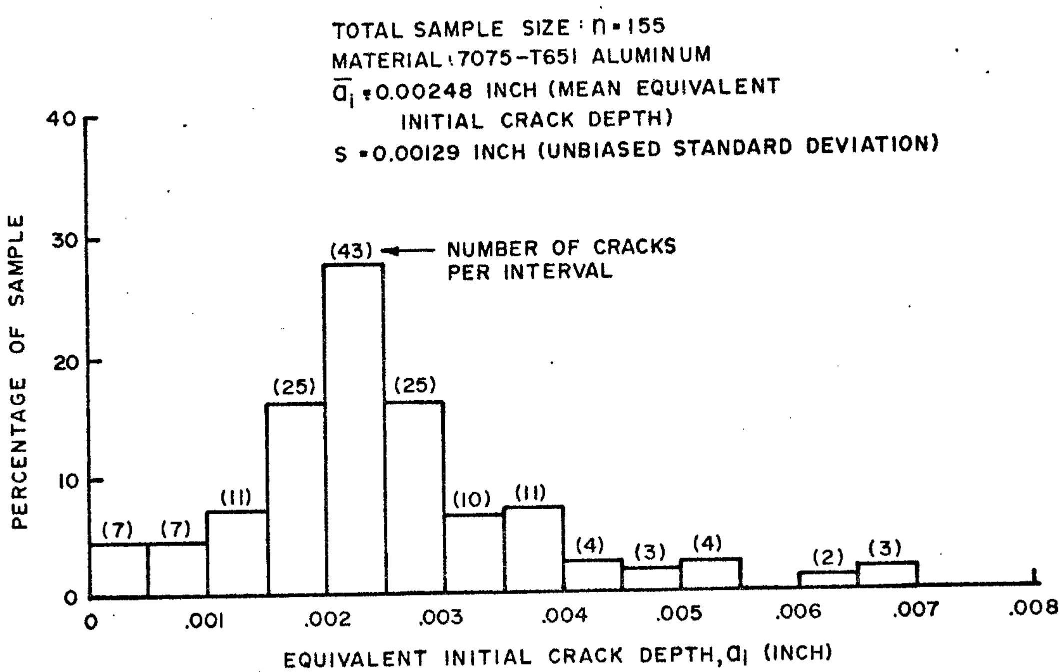


Figure 16. Equivalent Initial Crack Depth Histogram for Reamed Holes for F-4E(S) Aircraft

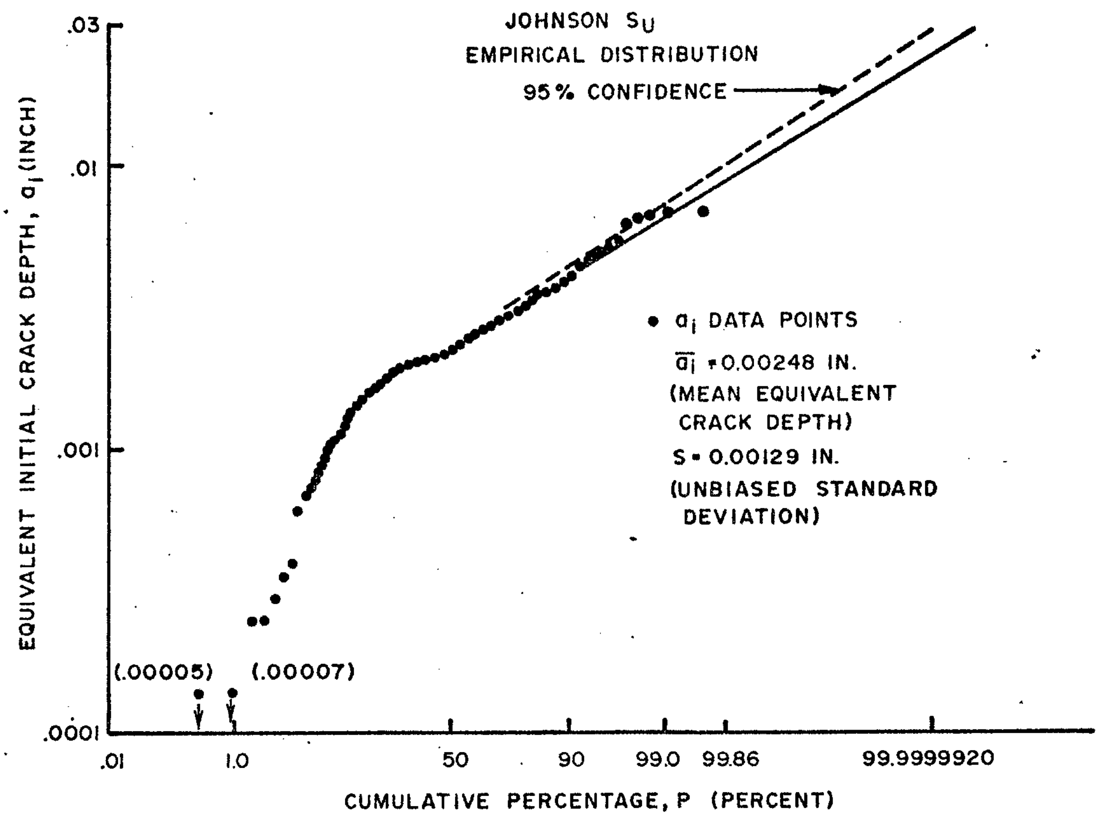


Figure 17. Equivalent Initial Crack Depth Distribution for Reamed Holes for F-4E(S) Aircraft

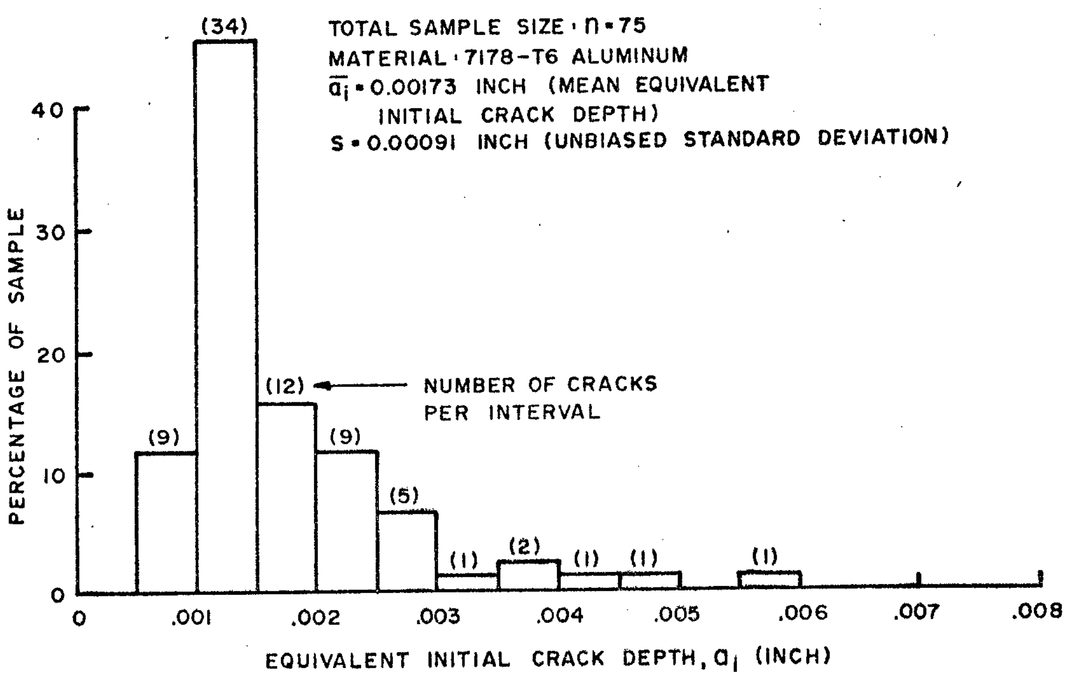


Figure 18. Equivalent Initial Crack Depth Histogram for Drilled Holes for F-4E(S) Aircraft

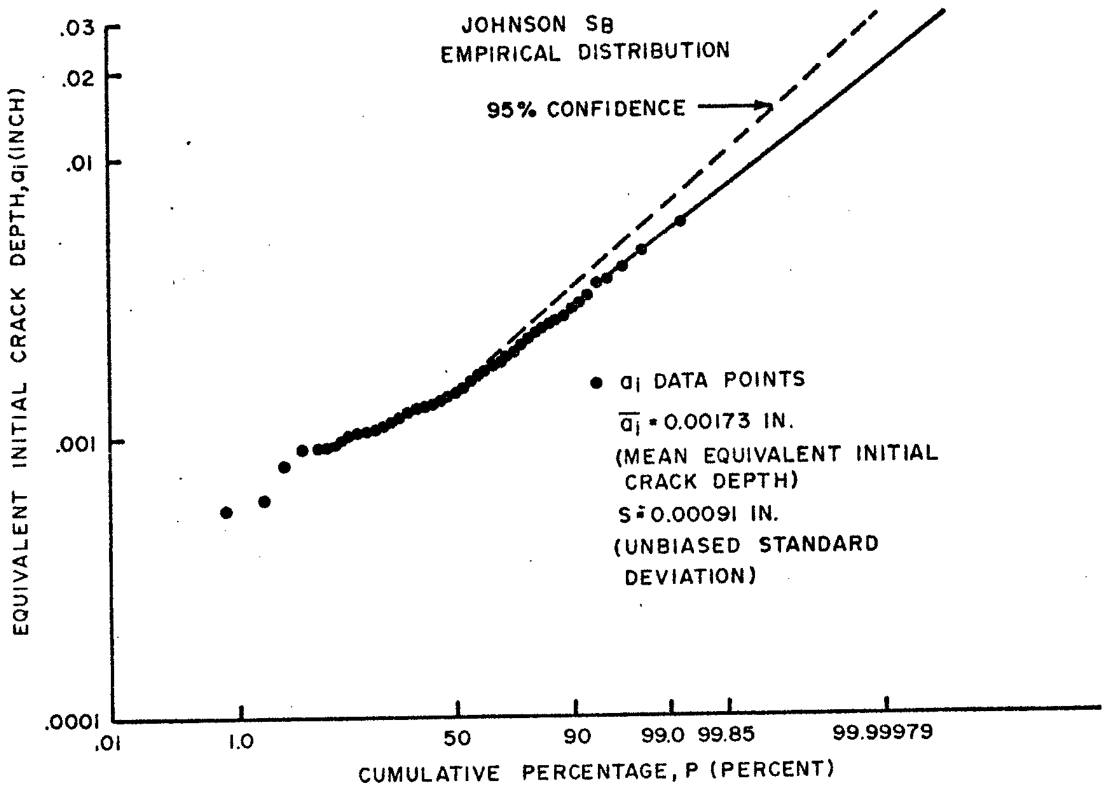


Figure 19. Equivalent Initial Crack Depth Distribution for Drilled Holes for F-4E(S) Aircraft

Figure 20 illustrates the effect of adding the equivalent initial crack depths of the 51 F-4E(S) cracked reamed fastener holes to the distribution of equivalent initial crack depths for 104 F-4B/J cracked reamed fastener holes used to represent the F/RF-4C/D initial quality. The addition of the 51 F-4E(S) initial flaws yields a greater probability of large equivalent initial crack depths. For example, the F/RF-4C/D equivalent initial crack depth (104 flaws) distribution indicates that 99.98% of the equivalent initial crack depths are less than .01 inch while the F-4E(S) equivalent initial crack depth (155 flaws) distribution indicates that 99.95% of the equivalent initial crack depths are less than .01 inch.

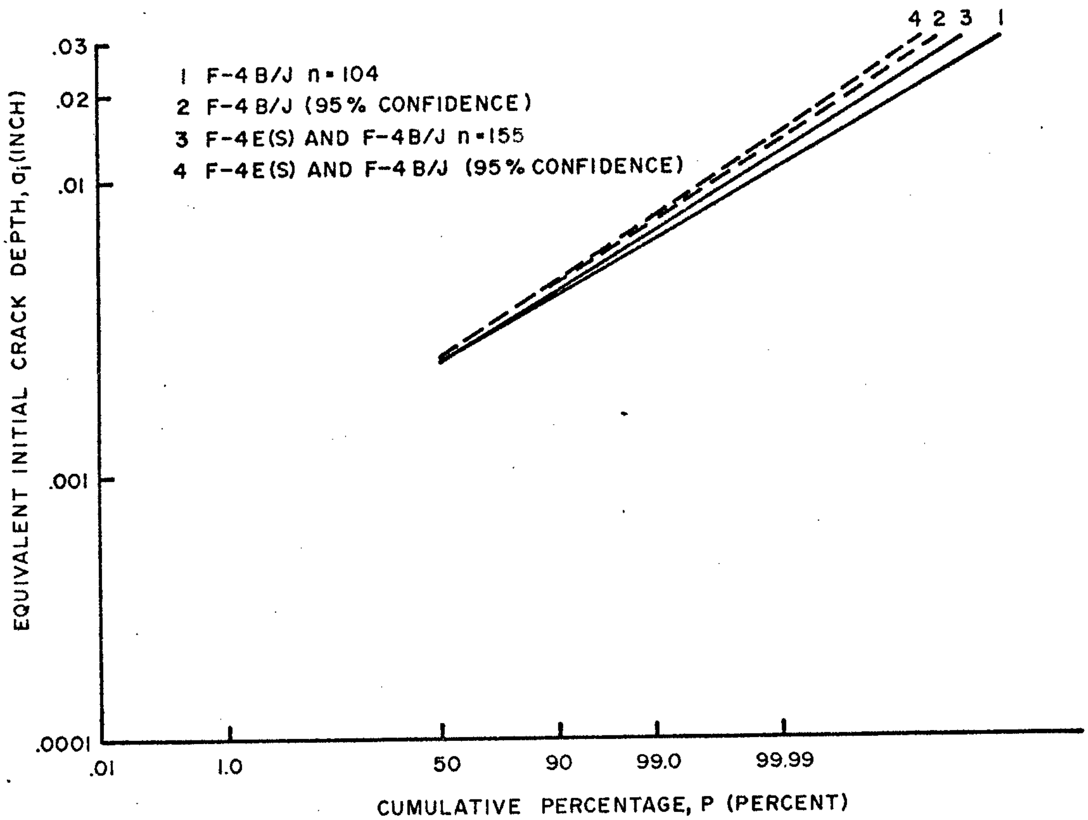


Figure 20. Comparison of Reamed Fastener Hole Distributions for F-4E(S) and F-4B/J Fatigue Test Articles

Figure 21 presents a comparison of the F-4E(S) equivalent initial crack depth distributions for reamed and drilled fastener holes. It can be seen that with 95% confidence, the reamed and drilled fastener holes have approximately the same cumulative percentage at .01 inch. However, a greater probability of occurrence exists for drilled fastener holes for equivalent initial crack depths greater than .01 inch.

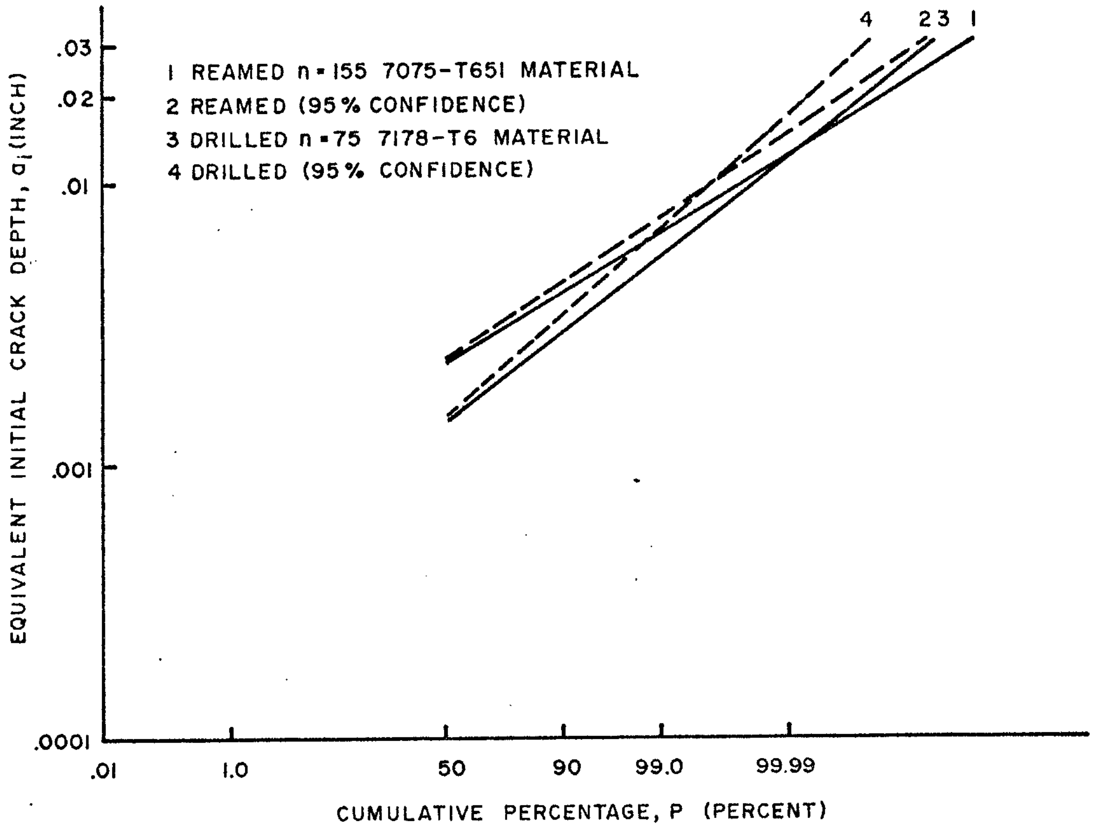


Figure 21. Comparison of Drilled and Reamed Fastener Hole Distributions for F-4E(S) Aircraft

3.3 A-7D INITIAL QUALITY ASSESSMENT

The A-7D damage tolerance and life assessment program was conducted from September 1974 to January 1977. The program included an initial quality assessment which was accomplished using the fastener holes contained in test specimens cut from the lower wing skin of an A-7D production airplane. The specimens were subjected to a known load history, resulting in fatigue crack initiation and propagation. The location of each specimen in the lower wing skin is illustrated in Figure 22. Each specimen was made of 7075-T6 aluminum and contained multiple holes. The geometric details for each specimen are presented in Table 10, indicating that the thickness ranged from approximately 3/16 inch to 1/4 inch and the nominal values of the width and hole diameter were 3 inches and 1/4 inch, respectively. The specimens contained two types of holes, countersunk holes (wet wing region) and straight shank holes (dry wing region).

The test specimens were subjected to a fatigue stress spectrum consisting of 5,000 cycles with a maximum stress of 20 ksi and a stress ratio of 0.1 followed by 100 cycles with a maximum stress of 30 ksi and a stress ratio of 0.1. The block spectrum was chosen because it produced test lives of reasonable length (less than 20 blocks) and fracture surfaces which were readily readable.

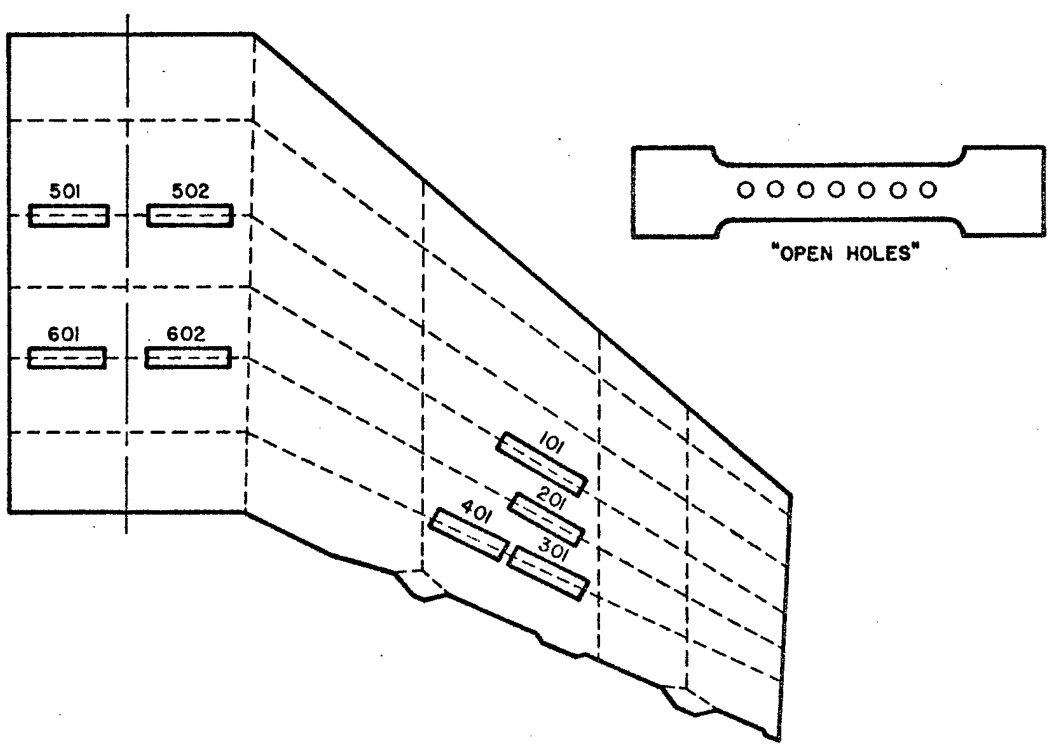
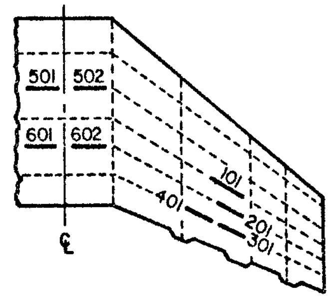
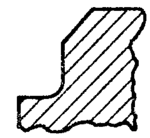


Figure 22. A-7D Quality Assessment Specimen Locations



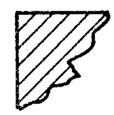
SPECIMEN	THICKNESS	WIDTH	HOLE DIAMETER
101	0.226	2.93	0.253(1)
201	0.226	2.93	0.253(1)
301	0.217	3.00	0.253(1)
401	0.231	3.00	0.253(1)
501	0.183	2.9	0.253(2)
502	0.176	3.00	0.253(2)
601	0.263	3.00	0.253(2)
602	0.264	3.00	0.253(2)

(1)



COUNTERSUNK HOLE

(2)



STRAIGHT SHANK HOLE

Table 10. Geometric Details of A-7D Quality Assessment Specimens

Table 11 contains a summary of the number of fastener holes involved, the number of flaws detected, the number of flaws fractographically examined, the crack length range at the time of specimen failure and the equivalent initial quality range. All but two of the 44 holes contained double flaws. One of these two holes contained one crack while no crack was detected in the other hole. This resulted in a total of 85 flaws, of which 44 were examined fractographically. The flaws were arbitrarily chosen for fractographic examination at magnifications ranging from 30X to 400X using a universal measuring microscope. The equivalent initial quality range for all the holes was found to be 0.00015 inch to 0.0022 inch. A statistical distribution of the A-7D equivalent initial quality was obtained which shall be presented later.

SPECIMEN	NO. HOLES	NO. FLAWS	af RANGE	FLAWS TRACKED	ai RANGE
101	7	14	0.05-0.75	14	0.0004-0.0022
201	6	12	<0.01-1.10	12	0.0004-0.0012
301	4	8	0.01-0.65	1	0.0003
401	3	6	0.02-0.50	6	0.0002-0.0014
501	8	14	0.00-0.60	1	0.0007
502	8	16	<0.01-0.62	1	0.0006
601	4	8	0.02-0.50	8	0.00015-0.0009
602	4	7	0.00-1.05	1	0.0006
TOTAL	44	85		44	

Table 11. A-7D Quality Assessment Test Results

The fractographic examinations revealed the origins of the flaws for both the straight shank holes and the countersunk holes as illustrated in Figure 23. This figure indicates that there is equal possibility of flaw occurrence along the bore of the hole for the straight shank hole while the most frequently occurring flaw location for the countersunk hole is at the inside radius of the small diameter portion of the hole. Typical flaw origins for each type of hole are shown on the fracture surfaces of Figure 24. Also illustrated in Figure 24 is the readability of the fracture surfaces for the selected stress spectrum, with the dark marking bands resulting from the application of the high load (maximum stress of 30 ksi) portion of the spectrum.

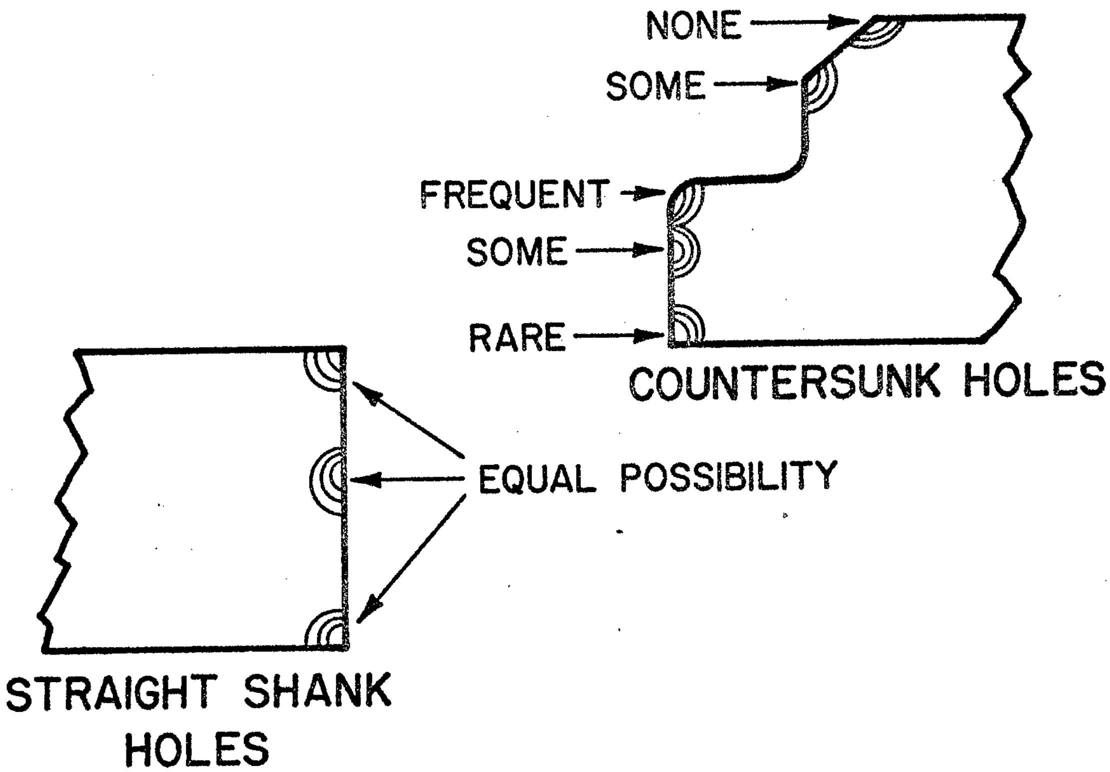
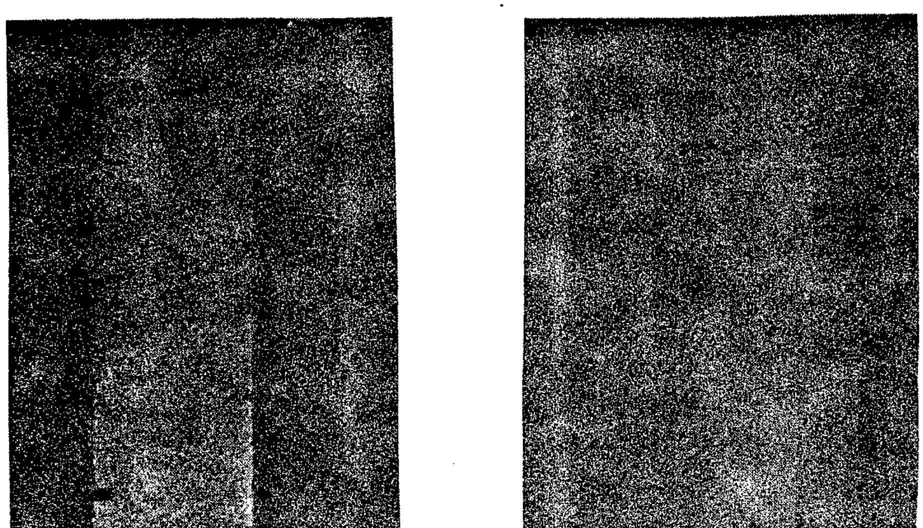


Figure 23. A-7D Initial Flaw Locations



Countersunk Hole   Straight Shank Hole

Figure 24. Fracture Surfaces for Countersunk and Straight Shank Holes

Metallurgical investigations of the A-7D flaw origins revealed that the flaws were the result of two different sources, anodize pitting and mechanical sources. The majority of the flaws (86.4%) initiated from anodize pits in the following manner (Figure 25). Insoluble microconstituents were exposed along the bore of the hole during the hole drilling operation. The anodizing ate away the microconstituents and caused pitting. The exposed pits were then filled with aluminum oxide, resulting in flaw initiation. The remaining flaws (13.6%) were due to the mechanical aspects of machining the holes. An example of flaw initiation from anodize pitting is presented in Figure 26.

Figure 27 presents the fractographic test results for a typical straight shank hole. The crack depth for this particular hole was determined to be .0139 inch at the time of failure of the specimen, which occurred during the twelfth block of loading. The selection of the stress spectrum to mark the fracture surfaces was indeed a good choice as it was possible to determine the crack depth at the beginning of the high load segment of each loading block. The crack depth at the

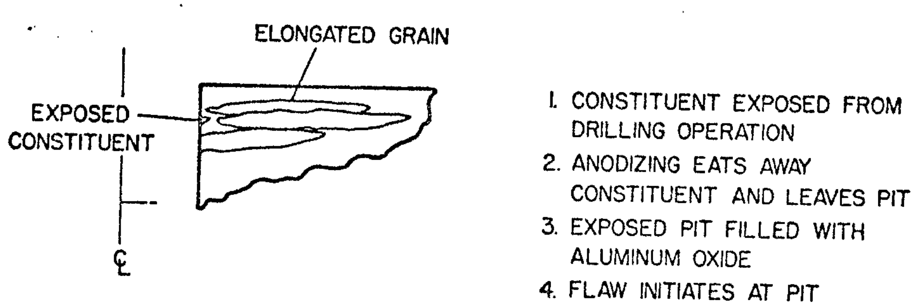
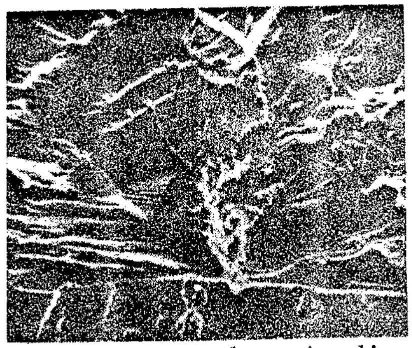
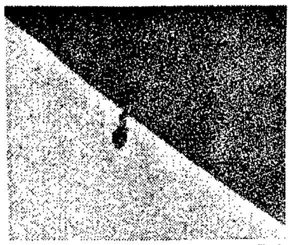


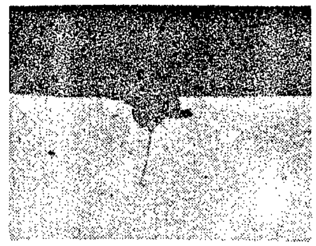
Figure 25. Cause of Flaw Initiation from Anodizing



Flaw Origin Located at Anodize Pit



Typical Anodize Pit



Fatigue Crack Initiation from Anodize Pit

Figure 26. Flaw Origin from Anodize Pit

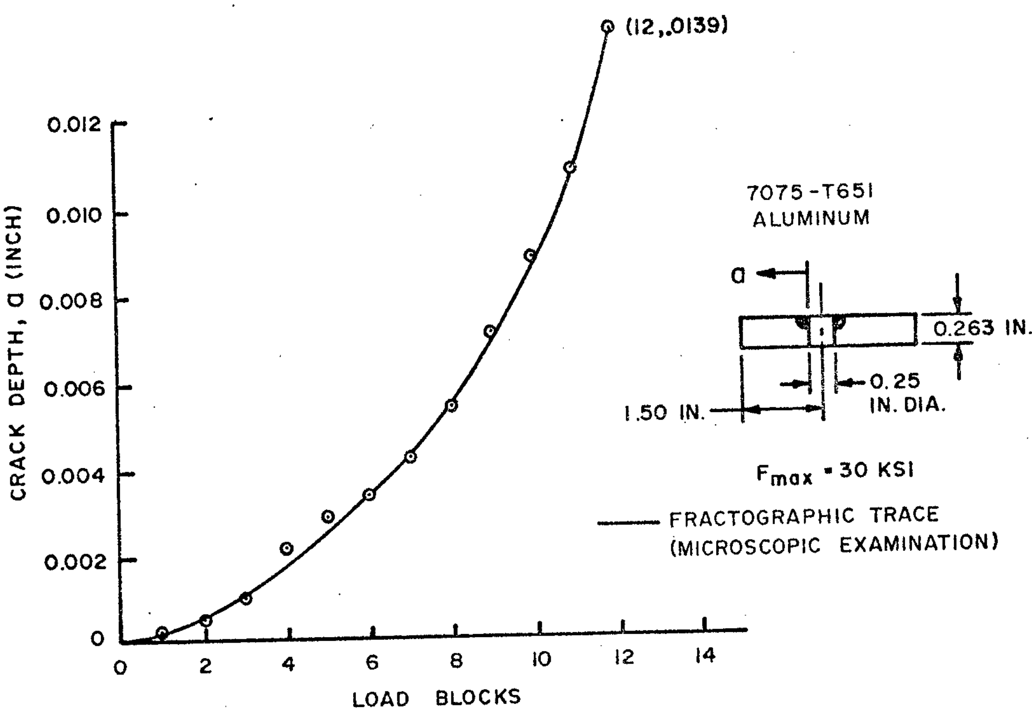


Figure 27. Typical Equivalent Initial Crack Depth for A-7D Aircraft

beginning of the high load segment of the first block of loading was taken as the equivalent initial quality. It was possible to track the crack back to the first load block for each of the flaws fractographically examined. Hence, the equivalent initial quality was established from fractography alone and no crack propagation analyses were required.

An A-7D equivalent initial crack depth statistical distribution was obtained from the individual equivalent initial crack depths established for each fastener hole. This equivalent initial crack depth statistical distribution is representative of the initial quality of the fastener holes produced by the particular material and structural manufacturing and processing operations involved. A total population of 44 initial flaws was used to obtain the equivalent initial crack depth statistical distribution presented in Figure 28. It was found that a log normal distribution matched most closely with the equivalent initial crack depth data. In order to utilize a finite sample size for fleet aircraft projections, a 95% confidence distribution was also established and presented in Figure 28.

3.4 SUMMARY OF F/RF-4C/D, F-4E(S), AND A-7D INITIAL QUALITY ASSESSMENTS

The F/RF-4C/D and F-4E(S) equivalent initial quality was established using a combination of analytical crack propagation analyses, fractographic test data, measured final crack depths, and calculated crack depths. The A-7D equivalent initial quality was established using fractographic test data alone. A plot of the equivalent initial crack

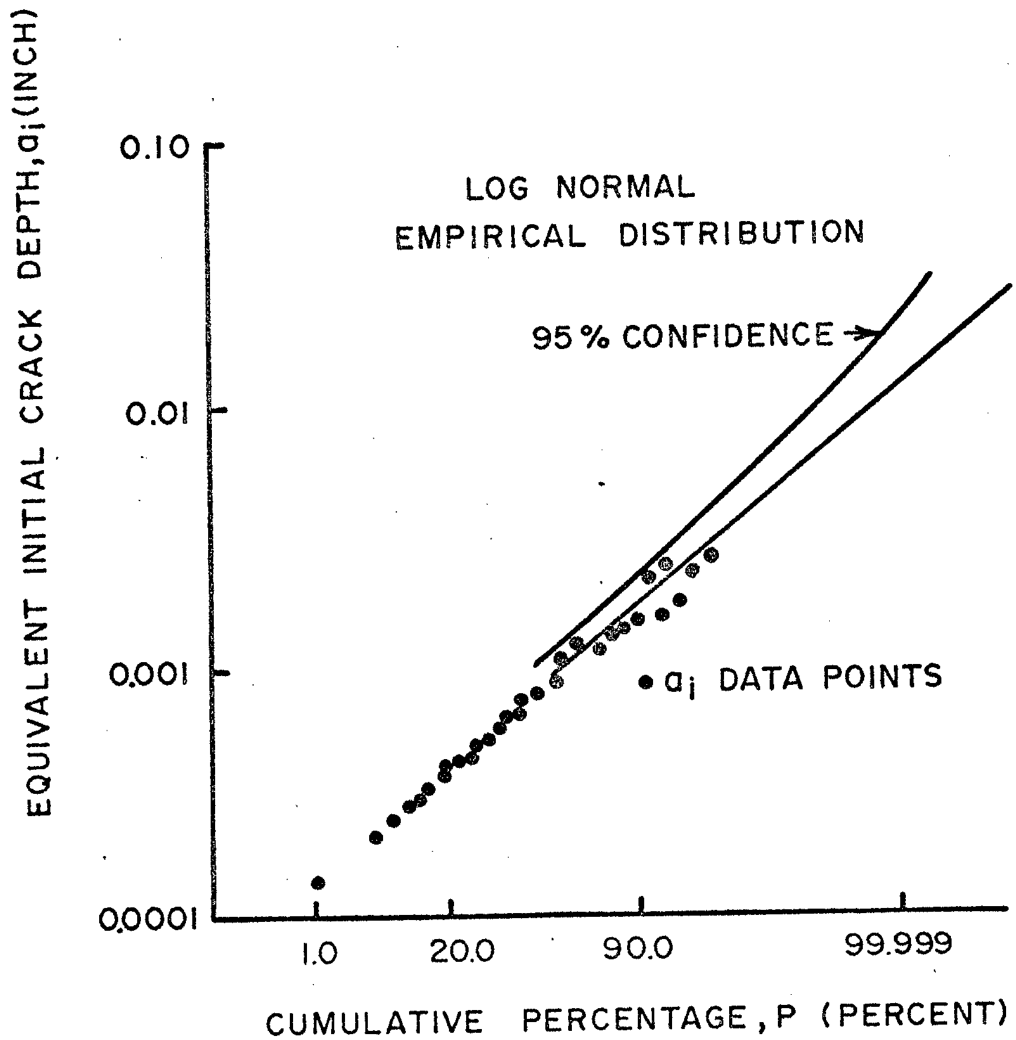


Figure 28. Equivalent Initial Crack Depth Distribution for A-7D Aircraft

depth distribution determined for each aircraft for 7075-T651 aluminum is presented in Figure 29.

Recalling that one of the objectives of each damage tolerance and life assessment program was to determine the operational limits of the particular aircraft involved, Table 12 presents the average initial crack size, the durability initial crack size (initial crack size used to establish economic repair limit), and the fracture initial crack size (initial crack size used to establish inspection interval and fracture limit) determined for each aircraft. The determination of these initial crack sizes was based upon a number of factors (e.g., equivalent initial crack depth distributions, MIL-A-8866B, MIL-A-83444, NDI capability, etc.).

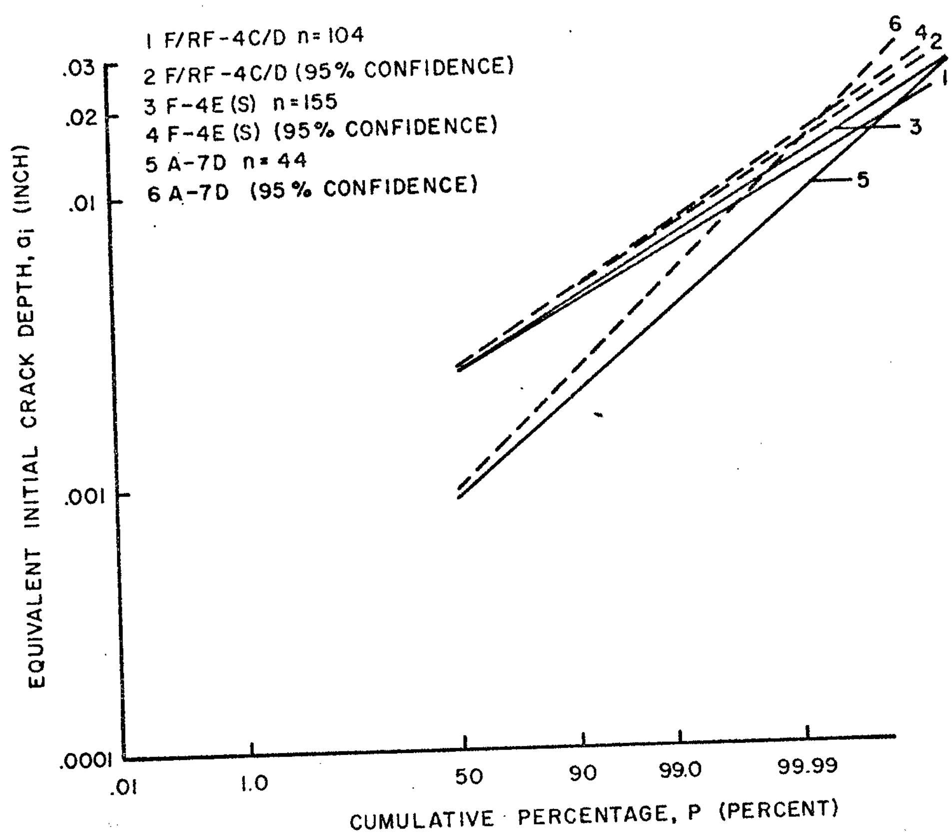


Figure 29. Equivalent Initial Crack Depth Distributions for F/RF-4C/D, F-4E(S) and A-7D Aircraft

	AVERAGE EQUIVALENT INITIAL CRACK	DURABILITY EQUIVALENT INITIAL CRACK	FRACTURE EQUIVALENT INITIAL CRACK
A-7D	 $a_i = .0008$	 $a_i = .0050$	 $a_i = .0500$
F/RF-4C/D	 $a_i = .0024$	 $a_i = .0100$	 $a_i = .0300$
F-4E(S)	 $a_i = .0025$	 $a_i = .0100$	 $a_i = .0300$

Table 12. Equivalent Initial Crack Depths for F/RF-4C/D, F-4E(S) and A-7D Aircraft

SECTION 4

CONCLUSIONS

The Equivalent Initial Quality Method has been used to assess the initial quality of such existing aircraft as the F/RF-4C/D, F-4E(S) and A-7D. The analytical representation of the initial quality resulting from material and structural manufacturing and processing operations was used in the determination of the operational limits (i.e., economic repair limit, inspection interval, and fracture limit) for these aircraft. The Equivalent Initial Quality Method could be used to evaluate the initial quality of other existing aircraft as well as be used in the design of new aircraft. For example, the method could be used to satisfy the safety and durability design requirements of Military Specifications MIL-A-83444 and MIL-A-8866B, respectively.

The Equivalent Initial Quality Method could potentially be used to determine the acceptability of a particular manufacturing or processing operation. For example, equivalent initial quality statistical distributions could be obtained for a number of different manufacturing or processing operations. Once an acceptable initial quality is established (e.g., acceptable initial quality to meet the economic life requirements, MIL-A-8866B), the acceptability of each manufacturing or processing operation could then be determined (Figure 30). Hence, the Equivalent Initial Quality Method could be used to evaluate any new manufacturing or prcessing operation. If a manufacturing or processing operation which produces fastener holes of an acceptable quality is used in production, the nondestructive inspection (NDI) requirements could be minimized, resulting in manufacturing cost savings.

While the Equivalent Initial Quality Method appears to have great potential for analytically representing the initial quality of fastener holes, it is felt that further research is required to reveal the limitations of the method. For example, studies are necessary to investigate the sensitivity of the method to type of damage, damage size and shape, stress level, material, load transfer, type of fastener, etc. Such studies are required to determine both the strengths and the limitations of the Equivalent Initial Quality Method.

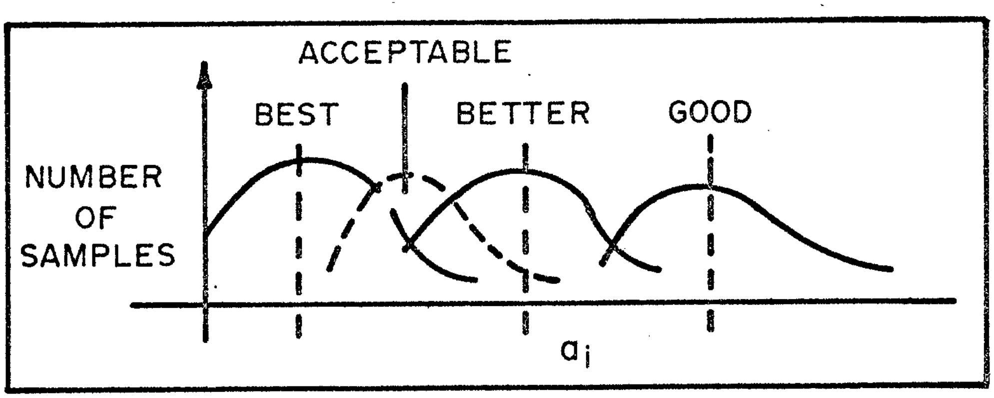


Figure 30. Acceptability of Manufacturing or Processing Operations

REFERENCES

1. Anon., "Aircraft Structural Integrity Program, Airplane Requirements," MIL-STD-1530A, December 1975, Air Force Aeronautical Systems Division.

2. Anon., "Airplane Strength and Rigidity Reliability Requirements, Repeated Loads and Fatigue," MIL-A-8866B, August 1975, Air Force Aeronautical Systems Division.

3. Anon., "Airplane Damage Tolerance Requirements," MIL-A-83444, July 1974, Air Force Aeronautical Systems Division.

4. Wood, H.A., Gallagher, J.P., Engle, R.M., and Potter, J.M., "Current Practice on Estimating Crack Growth Damage Accumulation with Specific Application to Structural Safety, Durability, and Reliability," AFFDL-TR-75-32, January 1976, Air Force Flight Dynamics Laboratory.

5. Gran, R.J., Orazio, F.D., Paris, P.C., Irwin, G.R., Hertzberg, R., "Investigation and Analysis Development of Early Life Aircraft Structural Failures," AFFDL-TR-70-149, March 1971, Air Force Flight Dynamics Laboratory.

6. Duval, R.E., "A-7A Fatigue Wing Crack Growth," Report No. 74-53452-172, December 1974, LTV Aerospace Corporation.

7. Gray, T.D., "An Equivalent Initial Flaw Analysis of the A-7A Wing Fatigue Test," Report No. 2-53410/ASIP/5DIR-10, May 1975, LTV Aerospace Corporation.

8. Anon., "Final Report, F/RF-4C/D Damage Tolerance and Life Assessment Study," Report No. MDC A2883, Vol I, June 1974, McDonnell Aircraft Company.

9. Anon., "Model F-4E Slatted Airplane Fatigue and Damage Tolerance Assessment," Report No. MDC A3390 Vol I, July 1975, McDonnell Aircraft Company.

10. Dumesnil, C.E., Gratke, S.D., Wilson, R.P., Martin, C.P., White, D.J., Hooks, O.L., "A-7D ASIP Part I, Damage Tolerance and Fatigue Assessment Program," Report No. 2-53440/7R-5928, Vol I, January 1977, Vought Corporation.

11. Wheeler, O.E., "Spectrum Loading and Crack Growth," ASME Transactions, Journal of Basic Engineering, Vol 94, March 1972, pp. 181-186.

12. Anon., "Results of Teardown Inspection of the F-4B/J Fatigue Test Aircraft for FSCP60 R1," McAIR Report No. F623, Vol VI, October 1973, McDonnell Aircraft Company.

13. Pinckert, R.E., "Damage Tolerance Assessment of F-4 Aircraft," Report No. McAIR 76-015, September 1976, McDonnell Aircraft Company.

14. Anon., "Results of Teardown Inspection of the F-4E Slatted Fatigue Test Aircraft for FSCP 52R1S1," Report No. MDC A2272 Vol IV, to be published, McDonnell Aircraft Company.