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STRUCTURAL AREA INSPECTION FREQUENCY EVALUATION (SAIFE)

Volume II. Description of Simulation Logic

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APRIL 1978
FINAL REPORT

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16. Abstract

To assist in the evaluation of proposed structural inspection programs for commercial jet transport aircraft, a logic was developed to simulate structural defects, failures, and inspections. This logic was incorporated in a computer program entitled Structural Area Inspection Frequency Evaluation (SAIFE). With the objective of quantifying the evaluation process currently used to establish and modify inspection intervals, SAIFE accounts for the following factors: (1) aircraft design analysis; (2) fatigue testing; (3) production, service, and corrosion defects; (4) probability of crack or corrosion detection; and (5) aircraft modification economics. As a five-volume document, this report covers the initial contract effort plus a subsequent parametric analysis as follows: Volume I (entitled Executive Summary) presents the SAIFE logic and documents the methodology for the decision-making processes in the simulation logic. Volume II details the SAIFE simulation logic, presents the background data for the analytical functions and decision-making processes, and includes data for a typical simulation problem. Volume III (entitled Demonstration Input, Inspection Survey, and MRR Data) presents data tabulations derived from historical trends and design input data for a SAIFE demonstration problem. As the user's manual for the SAIFE computer program, Volume IV (entitled Software Documentation and User's Manual) contains detailed computer logic flow diagrams and a complete listing of the program which is written in SIMSCRIPT II.5. Volume V (entitled Results of Model Demonstration) presents the results of the program application to a hypothetical aircraft and compares these results with the service experience of operational aircraft.

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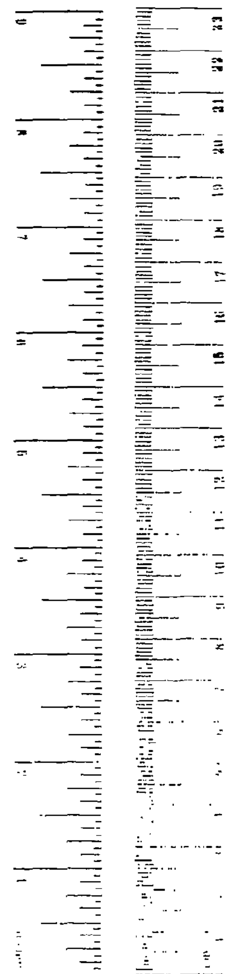
METRIC CONVERSION FACTORS

Approximate Conversions to Metric Measures

Symbol	When You Know	Multiply by	To Find	Symbol
	LENGTH			
in	inches	*2.5	centimeters	cm
ft	feet	30	centimeters	cm
yd	yards	0.9	meters	m
mi	miles	1.6	kilometers	km
	AREA			
in^2	square inches	6.5	square centimeters	cm^2
ft^2	square feet	0.09	square meters	m^2
yd^2	square yards	0.8	square meters	m^2
mi^2	square miles	2.6	square kilometers	km^2
	acres	0.4	hectares	ha
	MASS (weight)			
oz	ounces	28	grams	g
lb	pounds	0.45	kilograms	kg
	short tons (2000 lb)	0.9	tonnes	t
	VOLUME			
tsp	teaspoons	5	milliliters	ml
Tbsp	tablespoons	15	milliliters	ml
fl oz	fluid ounces	30	milliliters	ml
c	cups	0.24	liters	l
pt	pints	0.47	liters	l
qt	quarts	0.95	liters	l
gal	gallons	3.8	liters	l
ft^3	cubic feet	0.03	cubic meters	m^3
yd^3	cubic yards	0.76	cubic meters	m^3
	TEMPERATURE (exact)			
°F	Fahrenheit temperature	5/9 (after subtracting 32)	Celsius temperature	°C



Approximate Conversions from Metric Measures

Symbol	When You Know	Multiply by	To Find	Symbol
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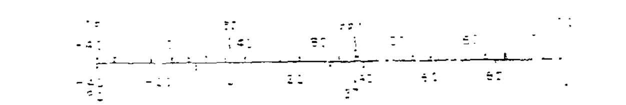
mm	millimeters	0.04	inches	in
cm	centimeters	0.4	inches	in
m	meters	3.3	feet	ft
m	meters	1.1	yards	yd
km	kilometers	0.6	miles	mi
	AREA			
cm^2	square centimeters	0.16	square inches	in^2
m^2	square meters	1.2	square yards	yd^2
km^2	square kilometers	0.4	square miles	mi^2
ha	hectares (10,000 m^2)	2.5	acres	
	MASS (weight)			
g	grams	0.035	ounces	oz
kg	kilograms	2.2	pounds	lb
t	tonnes (1000 kg)	1.1	short tons	
	VOLUME			
ml	milliliters	0.03	fluid ounces	fl oz
l	liters	2.1	pints	pt
l	liters	1.06	quarts	qt
l	liters	0.26	gallons	gal
m^3	cubic meters	35	cubic feet	ft^3
m^3	cubic meters	1.3	cubic yards	yd^3
	TEMPERATURE (exact)			
°C	Celsius temperature	9/5 (then add 32)	Fahrenheit temperature	°F



PREFACE

Technology Incorporated prepared this second volume of a five-volume report to document the simulation logic for Structural Area Inspection Frequency Evaluation in accordance with Article II, paragraph B of Contract DOT-FA74WA-3493. (Volume II along with Volume I completes the requirements of Phases I and II of the contract.) The effort is sponsored by the Aircraft Safety and Noise Abatement Division, Systems Research and Development Service of the Federal Aviation Administration.

The principal Technology Incorporated personnel engaged on this program were Mr. Carter J. Dinkeloo, project engineer, who served as principal investigator; Mr. Martin S. Moran, research engineer, who developed the model for the SAIFE computer program; and Mr. Ronald I. Rockafellow, program manager.

The contract monitors for the FAA were Messrs. Herbert Spicer and Charles Troha of the Aircraft Safety and Noise Abatement Division. The technical monitor was Mr. Arnold E. Anderjaska of the Flight Standards Division.

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I. INTRODUCTION

It is the mutual goal of the FAA, airframe manufacturers, and air carriers to constantly improve the structural integrity and inspection efficiency of civil aircraft. The good safety record of U.S. air carriers indicates that the current process of establishing and modifying structural inspection programs has been successful. However, with the increasing size and complexity of second- and third-generation transport aircraft, there is a need to quantify more precisely the current subjective evaluation process which relies heavily on reliability analyses of the new design and on operational experience of similar aircraft.

Because of the extreme complexity of the evaluation process, a computer simulation of all critical aircraft service life aspects was judged the most rational means for quantifying the process more exactly. As a five-volume document, this report documents the resultant Structural Area Inspection Frequency Evaluation (SAIFE) simulation logic. SAIFE accounts for the following factors: (1) aircraft design analysis; (2) component and full-scale fatigue testing; (3) production, service, and corrosion defects; (4) probability of crack or corrosion detection; and (5) aircraft modification economics. It treats these factors in a logical sequence that realistically represents the procedure currently used to establish and modify inspection intervals. SAIFE is designed to provide a repeatable method for evaluating proposed inspection programs. However, it is not intended to supplant the Maintenance Review Board or the air carrier use of the Standard Operations Specification - Aircraft Maintenance.

In addition to presenting the SAIFE logic applicable to initial demonstration, this report documents the research conducted to establish the quantitative functions required for decision logic in the simulation. Some of the documentation for these functions, such as fatigue life scatter, are taken from work conducted in other studies. Other functions, such as the probability of defect detection, are the result of work conducted as part of this contract. Whatever the source, all analytical information is referenced throughout the report. The logic applicable to the parametric study is given in Volume IV, Book 2 of this report.

Figure 1 illustrates the data sources and analytical functions that are integrated into the SAIFE logic. As Volume II, this volume presents the detailed simulation logic incorporated in SAIFE and all the background data required for the analytical functions and decision-making processes. It also includes much of the data required for a typical simulation program.

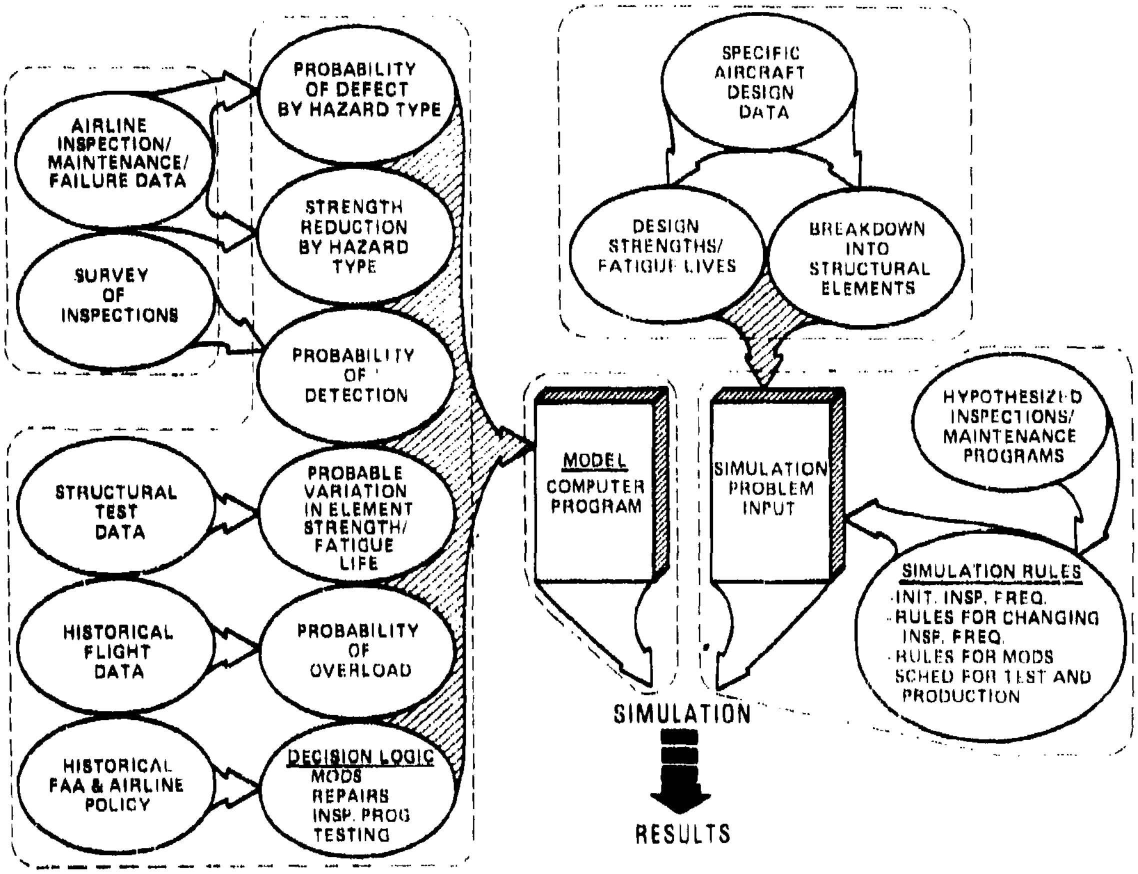


Figure 1. Approach to SAIFE Simulation Problem

II. DISCUSSION OF DETAILED SIMULATION LOGIC

The eight blocks in Figure 2 represent the major aspects of the SAIFE simulation logic which was originally

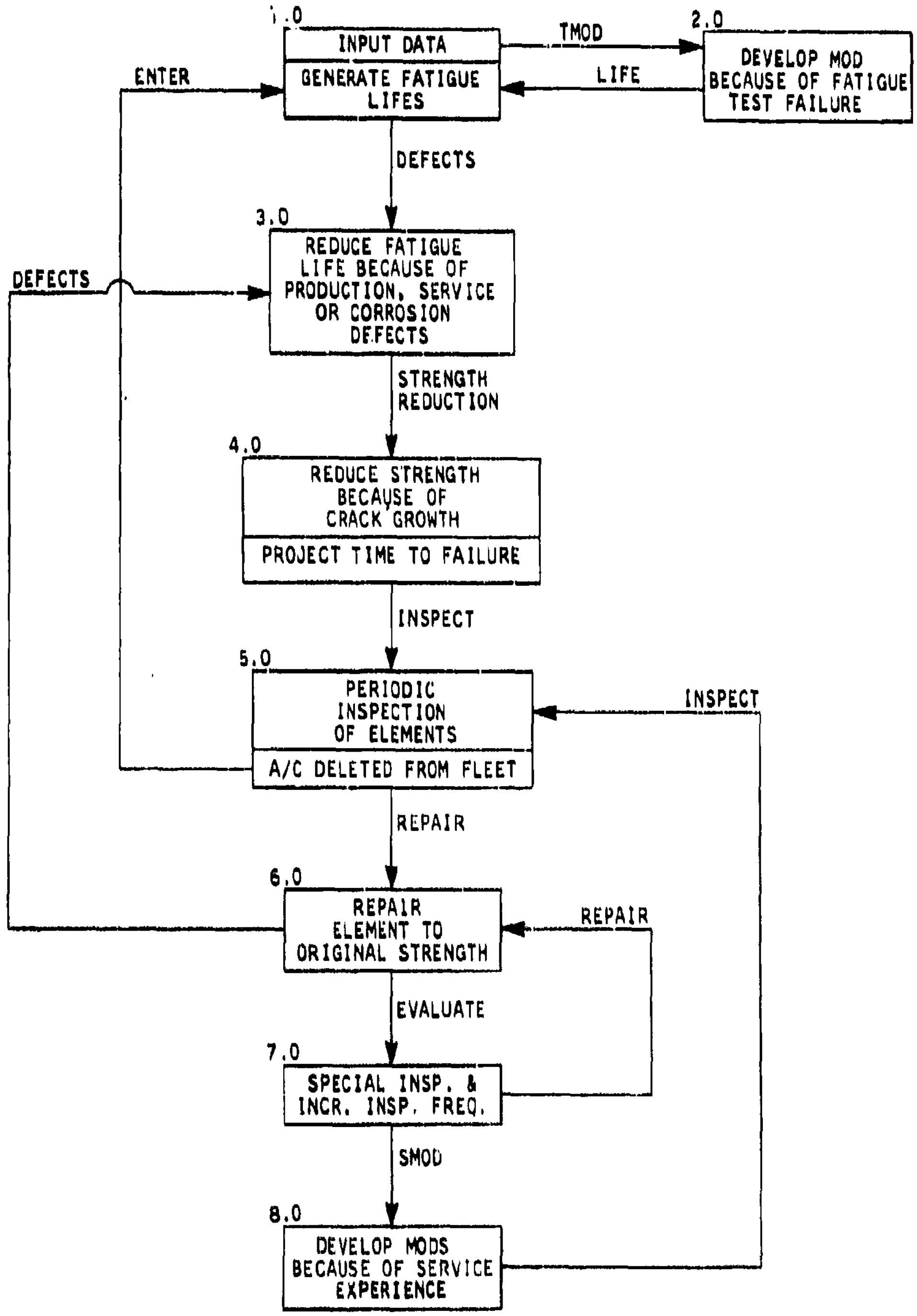


Figure 2. Flow Diagram Showing Major Aspects of SAIFE Logic

developed by Anderjaska in Reference 1. Block 1.0 accepts input data for the aircraft fleet and for each structural element in these aircraft. After determining whether element modifications are required because of the fatigue test results in Block 2.0, Block 1.0 assigns a fatigue life to each element in each aircraft. Block 3.0 determines whether production, service, or corrosion defects will occur; if it is determined that such defects will occur, Block 3.0 predicts the times when they will occur. After comparing the flight loads with the strength of each element, Block 4.0 predicts the time to structural failure for each aircraft. Block 5.0 conducts the periodic inspections of each element. If a structural failure has occurred, Block 5.0 deletes the corresponding aircraft from the fleet. However, if an element has a defect that is detected, Block 6.0 repairs the element. Depending on the magnitude of the detected defects, special inspections and increased inspection frequencies may be called for in Block 7.0 and modifications may be instituted in Block 8.0. When all the aircraft have been deleted from the fleet either through retirement from service or as a result of a structural failure, the simulation is complete.

As detailed in the following sections, each block is broken down into several subblocks which contain the functions and decision logic required for the simulation. Each subblock for a particular block breakdown is represented by the unit number identifying the block in Figure 2 and by a decimal number denoting the subblock sequence; for example, the subblock represented by 2.3 correlates with Block 2.0 in Figure 2 and is third in the subblock sequence. By design, the subsection numbering in this section correlates with this numbering system.

1. Detailed Description of Block 1.0, Input Data/Generate Fatigue Lives (see Figure 3)

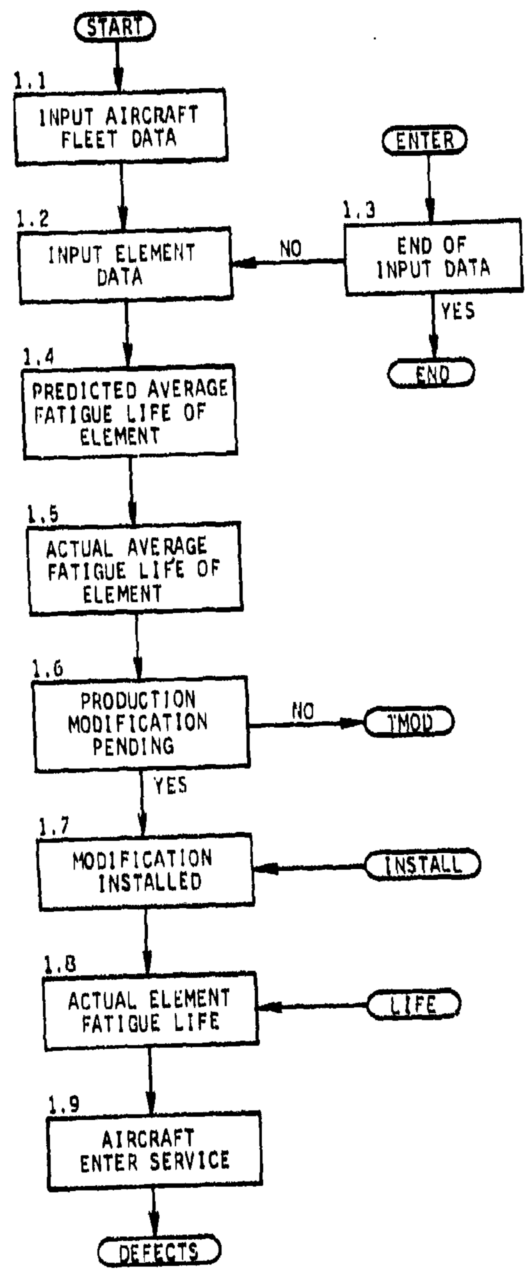


Figure 3. Detailed Flow Diagram - Input Data/Generate Fatigue Lives

1.1 Input Aircraft Fleet Data

This block introduces the following input data which is constant for all elements throughout the simulation.

(1) Aircraft type (two-element array).

(2) The size of fleet to be simulated.

(3) Service life of the aircraft (flight hours).

(4) Time at which aircraft production begins (flight hours after start of simulation).

(5) Initial aircraft production (flight hours between aircraft).

(6) Second aircraft production rate (flight hours between aircraft).

(7) Time at which second production rate takes effect (flight hours after start of production).

(8) Time at which fatigue test is started (flight hours after start of simulation).

(9) Fatigue test acceleration factor.

(10) Corrosion area growth rate (square inches per flight hour).

(11) Percentage of fatigue life at which inspection frequency is increased because of a fatigue test failure.

(12) Factor to reduce fatigue life when corrosion occurs in a stress concentration.

(13) Factor to reduce fatigue life when corrosion occurs outside a stress concentration.

(14) Mean (μ_R) and standard deviation (σ_R) for the log-normal distribution of the ratio of the actual average fatigue life to the predicted average fatigue life.

(15) Equation constants (two required) for gust and maneuver load distribution.

(16) Initial inspection intervals (four required) (flight hours).

(17) Cost of each level of inspection (four required) (dollars).

(18) D-level sampling percentage.

(19) A "YES" or "NO" switch used to implement the long list output option.

(20) Percentage of critical crack length at which an internal crack becomes external.

A total of 29 input constants are required for this data block.

1.2 <u>Input Element Data</u>

This block introduces the following input data which is unique to each element (therefore, each element must have a complete set of data specific to the given element):

(1) Element identification (four-element array).

(2) Predicted average fatigue life (flight hours).

(3) Actual average fatigue life (flight hours).

(4) Average slow crack growth rate (inches per flight hour).

(5) Average fast crack growth rate (inches per flight hour).

(6) Crack length to structural failure (inches).

(7) Critical crack length (inches).

(8) Fail-safe crack length (inches).

(9) Probability of production defect (occurrences per aircraft).

(10) Corrosion resistance rating.

(11) Service damage occurrence rate (occurrences per flight hour).

(12) Lead time to implement modifications (equivalent hours).

(13) Factor by which inspection intervals are decreased because of fatigue test failure (applied to "C" and "D" level inspections only).

(14) Factor by which inspection intervals are decreased because of unfavorable service experience (applies to "C" and "D" level inspections only).

(15) Factor by which inspection intervals are increased because of favorable service experience (applies to "C" and "D" level inspections only).

(16) Probability of fatigue crack initiating internally.

(17) Probability of corrosion initiating internally.

(18) Lowest internal inspection level.

(19) Lowest external inspection level.

(20) Repair costs for each level of inspection (four constants required) (dollars).

(21) Decision variable indicating whether or not modifications are to be fatigue-tested (yes or no).

(22) First modification tooling cost (dollars).

(23) Additional modification tooling cost (dollars).

(24) First modification installation cost (dollars).

(25) Additional modification installation cost (dollars).

(26) Repair cost for defect found during special inspection (dollars).

(27) Probability of existing corrosion being in a stress concentration.

(28) Initial corrosion occurrence rate (occurrences per flight hour).

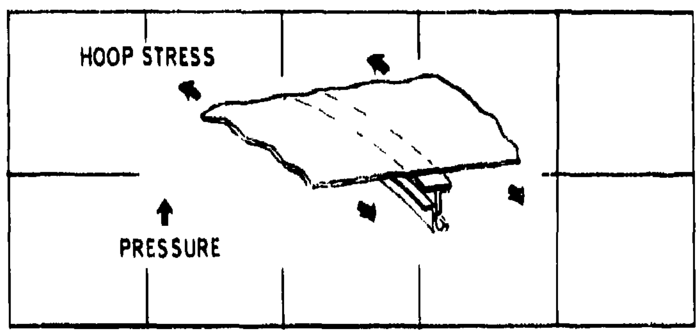
(29) Second corrosion occurrence rate (occurrences per flight hour).

(30) Airframe time at which second rate takes effect (flight hours).

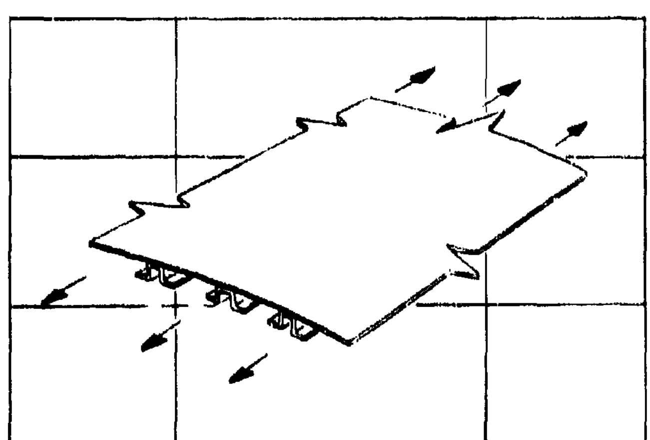
A total of 36 input constants are required for this data block.

Each element to be evaluated by the simulation is identified by three groups of alpha characters and by one group of numeric characters. The alpha characters define the basic element type and the general location on the aircraft, while the numeric characters define the specific location of the element by identifying the wing or fuselage station number. For example, an element identified as "FUS-MFR-TOP-400" would be a frame located in the fuselage crown with the attaching structure extending from station 390 to station 410.

The simulation is designed to handle as many individual elements in each aircraft as is necessary. Accordingly, the size of each wing or fuselage element depends only on naturally occurring design points such as rib or frame spacing. Therefore, the element identified at fuselage station 400 includes all structure and attaching parts between fuselage stations 390 and 410. In this example the fuselage frame element would also include all the attached skin as shown in Figure 4. This figure also shows a typical wing stringer element with attaching structure. The basic element types and the number of individual elements in each basic type are listed in Table 1. This table is applicable to a typical narrow-body aircraft and is used throughout this report to analyze service history data from narrow-body aircraft. The identification system is the same as that used to process the MRR/SDR historical data (Volume III). It offers a great deal of flexibility in laying out the elements on any particular aircraft and permits an easy comparison of the simulation output and the historical information.



Fuselage Frame Element



Wing Stringer Element

Figure 4. Typical Airframe Elements Used in the SAIFE Simulation

TABLE 1. ELEMENT DEFINITION AND DISTRIBUTION FOR TYPICAL NARROW-BODY JET TRANSPORT

Basic Type		No. Elements in Each Aircraft
WNG (wing):		
STR (stringer)	- FWD (forward)	33
	- CEN (center)	33
	- AFT (aft)	33
RIB (rib)	- FWD (forward)	33
	- CEN (center)	33
	- AFT (aft)	33
SPR (spar)	- FWD (forward)	33
	- CEN (center)	33
	- AFT (aft)	33
ACC (access)	- FRM (frame)	50
WSC (wing center section):		
SWB (spanwise beam)	- FWD (forward)	7
	- CEN (center)	7
	- AFT (aft)	7
RIB (rib)	- FWD (forward)	7
	- CEN (center)	7
	- AFT (aft)	7
STR (stringer)	- FWD (forward)	7
	- CEN (center)	7
	- AFT (aft)	7
FUS (fuselage):		
MFR (main frame)	- TOP (top)	60
	- SID (side)	60
	- BOT (bottom)	60
STR (stringer)	- TOP (top)	60
	- SID (side)	60
	- BOT (bottom)	60
FLR (floor)	- BEM (beam)	60
KEL (keel)	- BEM (beam)	60
WIN (window)	- FRM (frame)	50
DOR (door)	- FRM (frame)	10
PRS (pressure)	- WEB (web)	60

1.3 End of Data

This function simply monitors the input data for the end of data card. When the appropriate end of data code is encountered, the simulation is concluded.

1.4 Predicted Average Fatigue Life

A program input, this variable is the average fatigue life analytically predicted for each element design.

As used in this report, fatigue life is defined as the accumulated operational time when a crack initiates. SAIFE uses the following three types of fatigue lives which are used throughout this report:

(1) Predicted average fatigue life - the fleet average fatigue life for an element determined from the manufacturer's design analysis.

(2) Actual average fatigue life - the fleet average fatigue life for an element determined from extensive service experience.

(3) Actual element fatigue life - the fatigue life of an element on an individual aircraft in the fleet.

The primary source of data for this variable is the manufacturer's design analysis. When such data is not available directly from the manufacturer, it can be obtained in a less detailed format from a Maintenance Review Board report or a Fatigue Integrity Program report. It is also possible to calculate the fatigue life from service experience by using the method presented in Reference 2. If none of these sources are available, the average fatigue life must be approximated on the basis of the design service life for the current aircraft or from a previous aircraft of the same manufacturer.

1.5 Actual Average Fatigue Life

Since fatigue phenomena are not completely defined, the fatigue life prediction analysis should be performed statistically. Since in practice a statistical approach is not used, the actual fatigue life of a structure of a given design will usually differ from that analytically predicted. The probability of the actual life being greater or less than that predicted was studied by the Royal Aircraft Establishment (Reference 3). In Reference 3, fatigue lives based on full-scale structural fatigue tests on the wings of British military and civil aircraft were compared with those based on average fatigue performance in laboratory tests of typical aircraft joints. In each type of fatigue life derivation, the estimated life was based on average fatigue performance and, as necessary, on Miner's linear cumulative damage method. As shown in Figure 5, the results of this comparison indicate that the actual life may frequently be overestimated if the calculated fatigue performance is not statistically interpreted. To determine the distribution shown in Figure 5, a computer program was used to fit the following statistical (log-normal) distribution to the data:

$$R = \frac{\text{Actual Life}}{\text{Predicted Life}} \tag{1}$$

$$\mu_R = 0.841$$

$$\sigma_R = 0.695$$

The parameters μ_R (distribution mean) and σ_R (distribution standard deviation) are input variables. These parameters enable SAIFE to account for improvements in fatigue analysis techniques. An example of the relationship resulting from improved analysis techniques is shown in Figure 5, where μ_R = 1.000 and σ_R = 0.695.

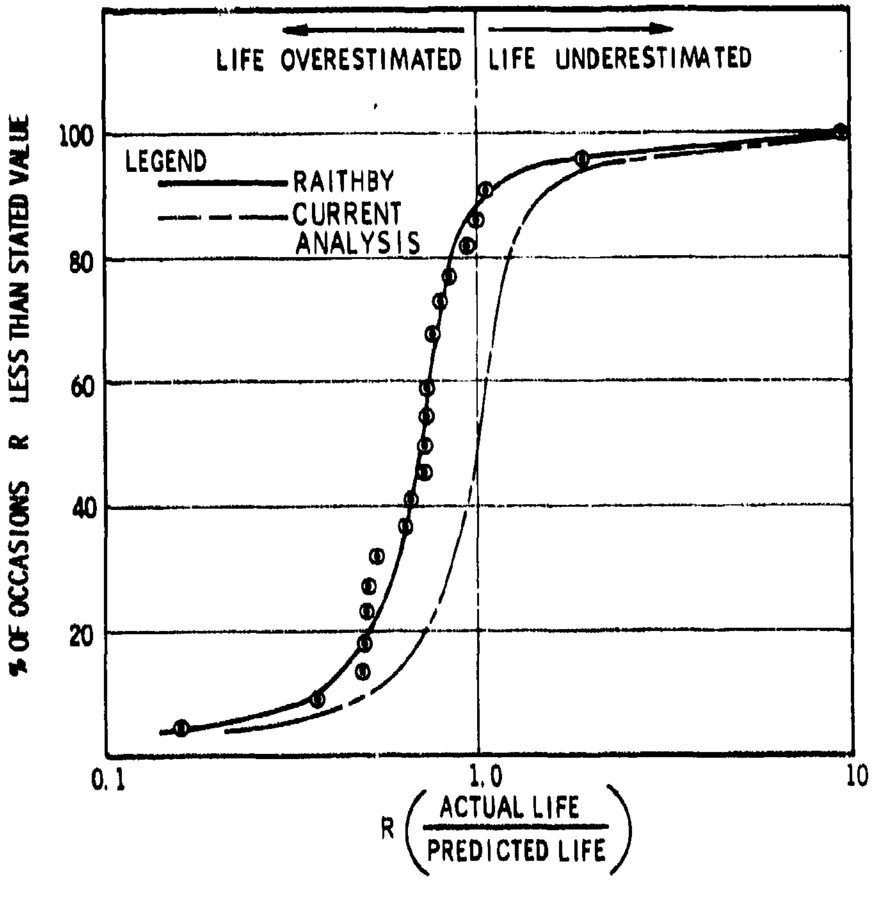


Figure 5. Comparison of Predicted and Actual Fatigue Life

With the application of Monte Carlo computer techniques, a log-normally distributed correction factor for the predicted average fatigue life (Block 1.4, Figure 3) is generated for each element design. This correction factor (R) yields the actual average fatigue life (Block 1.4, Figure 3) and is calculated for each element. The same statistical model is used for all the elements making up an aircraft model.

1.6 Production Modification Pending

When each aircraft enters service, this function determines whether previous simulation logic has instituted a modification because of operational experience on aircraft already in service or because of a previous fatigue test failure. If a modification is pending, the logic proceeds to Block 1.7 to determine whether the modification is available for installation. If a modification is not pending, the logic goes to the routine TMOD. It is assumed that if previous unfavorable operational experience has caused the development of an element modification, a subsequent fatigue test failure will not cause further modification of that element.

1.7 Modification Installed

This function indicates whether or not a prescribed structural modification has been installed in an aircraft before it enters service. The modification has been installed if

$$X4 * C2 > X8 + C11 + C12 \tag{2}$$

where X4 = production number of aircraft

C2 = production rate of aircraft

X8 = time between the delivery of the first aircraft and the time of the decision to develop a structural modification for a given element

C11 = lead time to develop a structural modification for a given element type

C12 = lead time from the development to the production-line incorporation of the structural modification for a given element type

Items C2, C11, and C12 are program inputs. Item C2 depends on the aircraft type being considered, while C11 and C12 depend on the particular element being considered. Any data available only in terms of calendar days must be converted to the equivalent simulation time (flight hours).

$$\text{Calendar days} * 8.2 = \text{equivalent simulation time} \qquad (3)$$

This conversion is based on the average yearly aircraft utilization of 3000 flight hours per 365 days.

The modification will be installed on aircraft that are already in service when the aircraft are out of service for repairs or for normally scheduled overhaul inspections if it has been determined that retrofit modifications are economically feasible or required for safety-of-flight reasons.

The validity of this function depends on the accuracy of C11 and C12. The best estimation of lead times should come from the aircraft manufacturer or from the air carrier if the carrier performs the modification. The lead time for a critical item that precedes the modification may be provided by a vendor or independent supplier.

1.8 Actual Element Fatigue Life

If identical fatigue tests are performed on several nominally identical test specimens, the resulting fatigue lives will not be identical. This basic fatigue life scatter is a function of the material properties, manufacturing quality, and process variations of the test specimens.

Also, no two aircraft within the same fleet experience identical load spectra. This load environment variation introduces additional fatigue life scatter among like structural elements within the same fleet.

Clearly then, the fatigue life of aircraft structures must be treated as a stochastic variable whose frequency distribution reflects both the basic fatigue scatter and the load environment variation. Because, in the analysis of aircraft structures, only a small sample can be tested and the failure of even a single structure may be catastrophic, the expected time to first failure is a much more significant measure than the mean time to failure. A method for the estimate of the expected time to first failure is outlined by Freudenthal (Reference 4). In his paper, Freudenthal shows that the cumulative distribution function has the form of the two-parameter Weibull:

$$F(t) = 1 - \exp[-(t/\theta)^b] \qquad (4)$$

where t = time to crack initiation
θ = characteristic value
b = shape parameter

By pooling the results from various sources (Figure 6), Freudenthal makes an estimate of fatigue life scatter based on the representation of test data by the log-normal distribution. He concludes that a value of the standard deviation $\sigma(\log_{10}N) = 0.15 - 0.20$ is representative of most results in the long-life range ($N > 10^6$ cycles). These values are consistent with the work done by Abelkis (Reference 5) in which he concludes that a value of $\sigma = 0.14$ would describe the basic fatigue life scatter and that a value of $\sigma = 0.20$ would account for basic scatter and the additional scatter introduced by load environment variation. Using the actual average fatigue life ($\bar{t}$) generated by Block 1.5 and assuming a value for σ, the values of θ and b can be determined from the following relationship:

$$b = \pi/(2.303\ \sigma\ \sqrt{6}) \qquad (5)$$

$$\theta = \bar{t}/\Gamma(1 + 1/b) \qquad (6)$$

where Γ = the gamma function

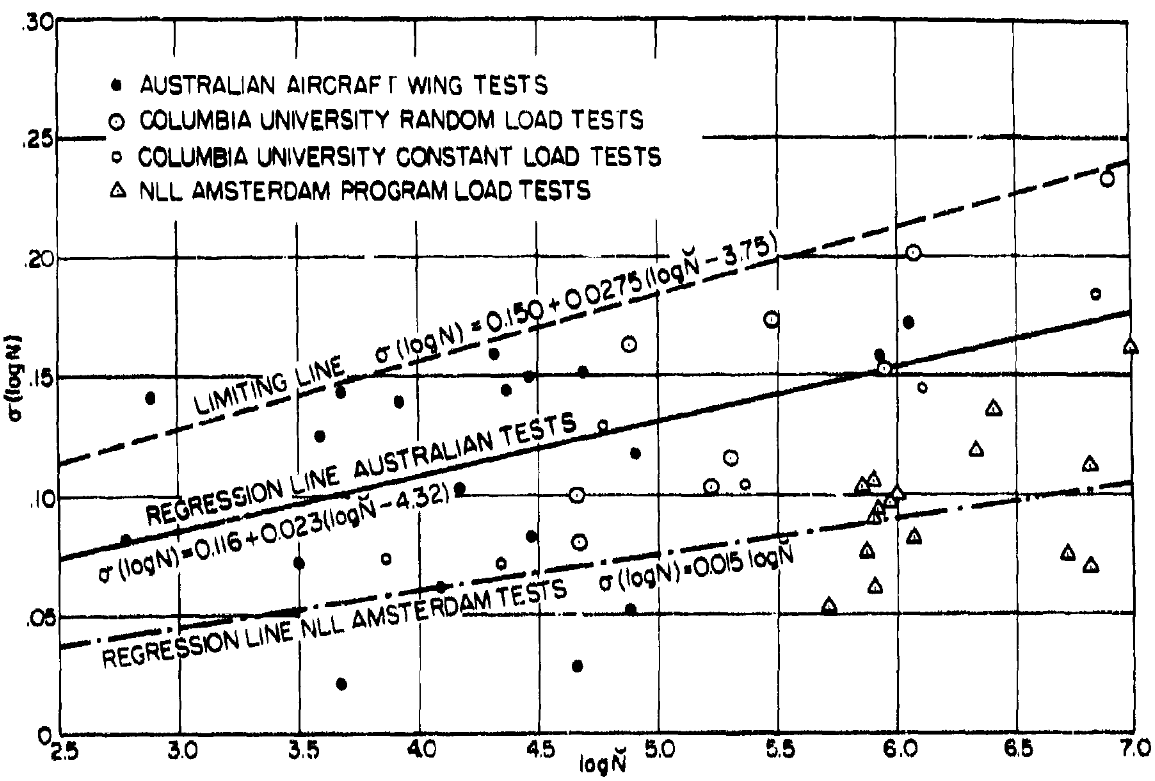


Figure 6. Relation Between $\delta = \sigma(\log_{10} N)$ and $\log \check{N}$

The Monte Carlo technique can now be applied to generate a two-parameter Weibull distribution of fatigue lives by replacing $[1-F(t)]$ in Equation (4) with a random number for each aircraft in the simulated fleet:

$$t = \theta[\ln(1/RN)]^{1/b} \tag{7}$$

where RN = a uniformly distributed random number.

The simulation program allows any structural element to have as many as three fatigue cracks at the same time. The times to subsequent crack initiations are arrived at by truncating the distribution at the previous fatigue life and drawing the time to the next crack initiation from the new conditional distribution. Let $F_1(t)$ be the cumulative distribution function of time to crack initiation. Make a random draw from $F_1(t)$ which yields t_1 (time to first crack initiation). Now truncate the lower end of the distribution at t_1 and make a random draw from the new conditional distribution $F_2(t)$ to yield t_2 (time to second crack initiation) where

$$F_2(t) = F_1(t|t > t_1) = \frac{F_1(t) - F_1(t_1)}{1 - F_1(t_1)} \tag{8}$$

and $F_1(t|t > t_1)$ = probability of crack initiation before time t with the condition that crack initiation occurs after time t_1

$F_1(t)$ = probability of crack initiation before time t

$F_1(t_1)$ = probability of crack initiation before time t_1

$1 - F_1(t_1)$ = probability of no crack initiation before time t_1

Now truncate the lower end of $F_2(t)$ at t_2 and make a random draw from the new conditional distribution $F_3(t)$ to yield t_3 (time to third crack initiation) where

$$F_3(t) = F_2(t|t > t_2) = \frac{F_2(t) - F_2(t_2)}{1 - F_2(t_2)} \tag{9}$$

How the three probability density functions $f_1(t)$, $f_2(t)$, and $f_3(t)$ are related can be determined from their definitions. From probability theory,

$$f_1(t) = \frac{dF_1(t)}{dt} \tag{10}$$

Likewise,

$$f_2(t) = \frac{dF_2(t)}{dt} = \frac{d}{dt} \quad \frac{F_1(t) - F_1(t_1)}{1 - F_1(t_1)} \tag{11}$$

or

$$f_2(t) = \frac{f_1(t)}{1 - F_1(t_1)} \tag{12}$$

Similarly,

$$f_3(t) = \frac{f_2(t)}{1-F_2(t_2)} = \frac{f_1(t)}{[1-F_1(t_1)][1-F_2(t_2)]} \tag{13}$$

If we let $K_1 = 1/(1-F_1(t_1))$ and $K_2 = 1/[1-F_1(t_1)][1-F_2(t_2)]$, we can then write $f_2(t) = K_1 f_1(t)$, $f_3(t) = K_2 f_1(t)$. Thus $f_2(t)$ and $f_3(t)$ can be formed from $f_1(t)$ by truncating the lower end at t_1 and t_2 and by multiplying all values of $f_1(t)$ by the constants K_1 and K_2, respectively. This process is demonstrated in Figure 7.

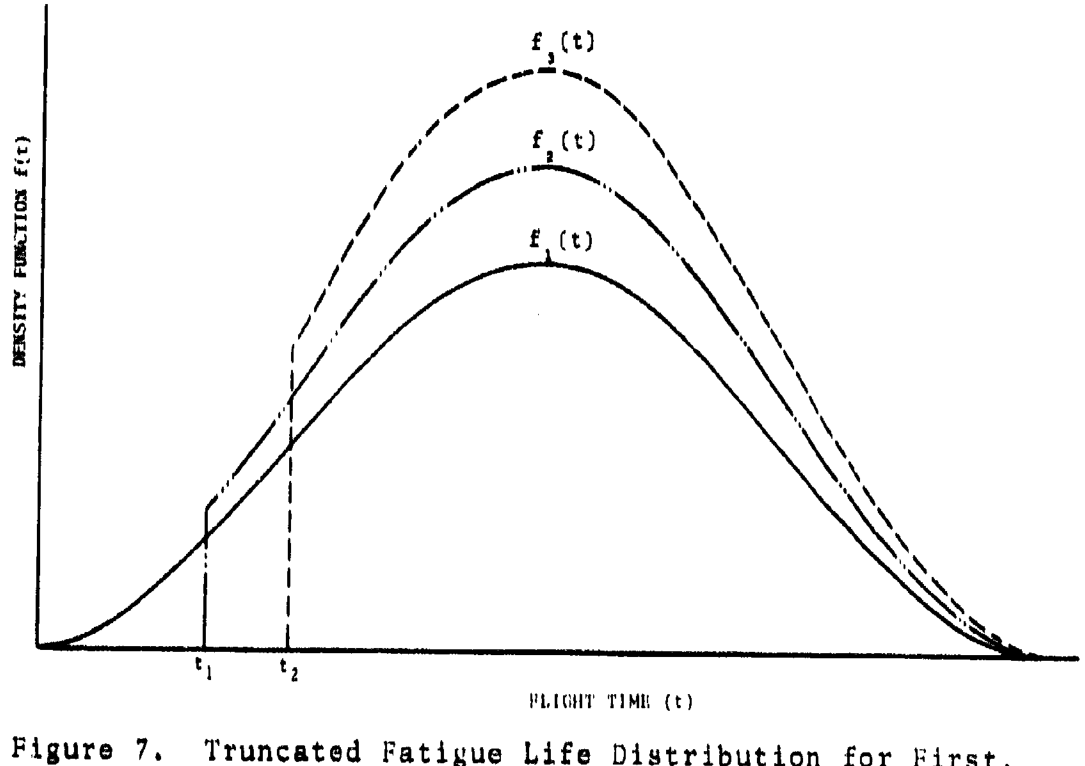


Figure 7. Truncated Fatigue Life Distribution for First, Second, and Third Cracks in an Individual Element

In Figure 8, the cumulative distribution of time to crack initiation is shown for each of the three cracks on the individual elements. Constructed from the results of a sample simulation problem, these distributions show a distinct Weibull form with progressively higher characteristic values (θ). This is the expected result from the method discussed above.

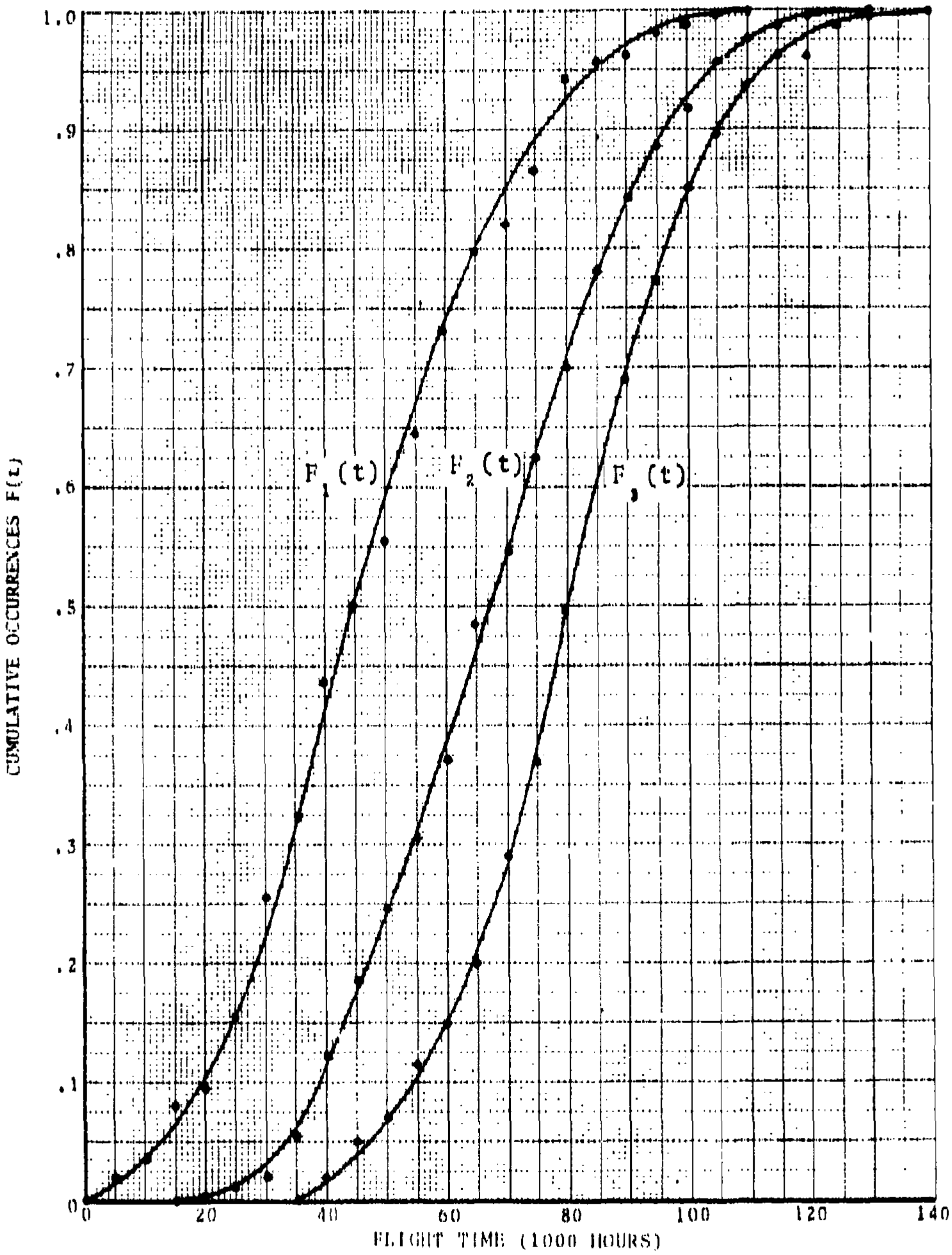


Figure 8. Cumulative Distribution of Fatigue Lives for First, Second, and Third Cracks

1.9 Aircraft Enters Service

This function causes aircraft to enter service at a production rate prescribed in the input. The program logic has the facility to change the production rate once at some time during production. The time of this change and the new production rate are also prescribed in the input. Aircraft continue to enter service until the fleet size defined in the input is attained.

2. Detailed Description of Block 2.0, Develop Modification Because of Fatigue Test Failure (see Figure 9)

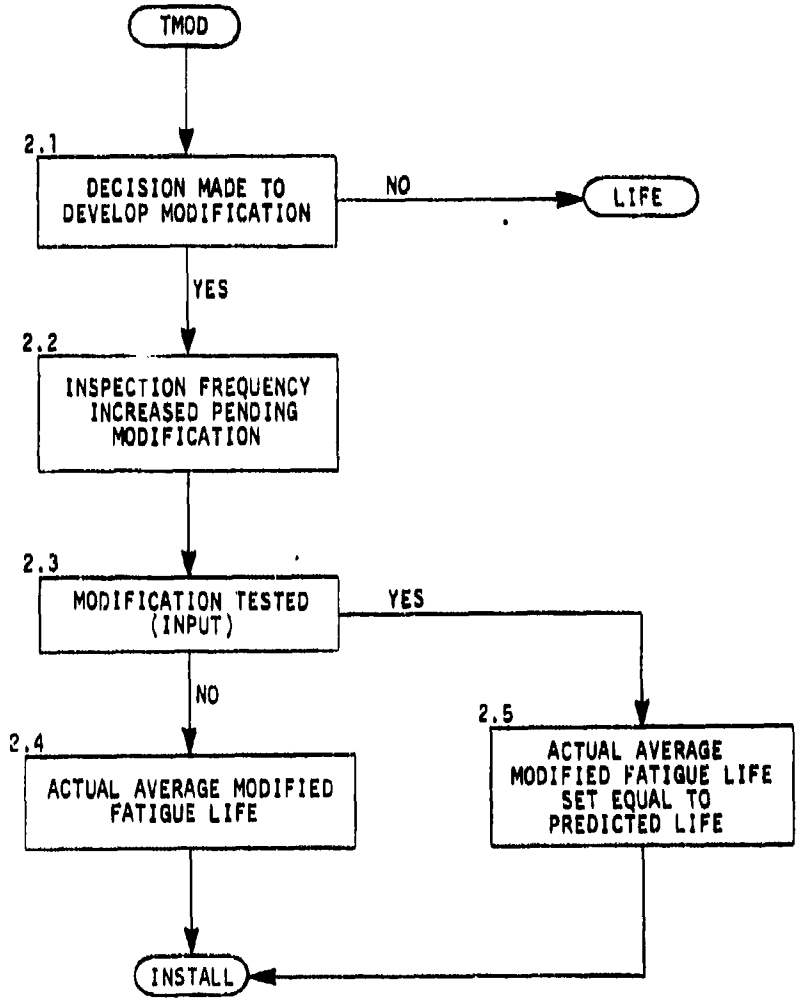


Figure 9. Detailed Flow Diagram - Develop Modification Because of Fatigue Test Failure

2.1 Decision Made to Develop Modification

This function determines whether a modification to an element is required so that the aircraft type may reach its predicted service life. The decision is based on the number of equivalent flight hours attained during fatigue testing. For airframe elements, a goal of two times the service life is commonly used. If this goal is achieved, the fatigue test may be discontinued or it may be continued to determine what additional margin of safety is present.

To determine whether the fatigue test goal has been achieved, the hours of testing are multiplied as follows by a fatigue test acceleration factor to arrive at the equivalent flight hours:

$$\text{Test Hours} * \text{Fatigue Test Acceleration Factor} = \text{Flight Hours} \quad (14)$$

The criteria for developing a modification are then as follows:

(1) If

$$\text{flight hours} \geq 2 * \text{service life} \quad (15)$$

a modification is not developed.

(2) If

$$\text{flight hours} < 2 * \text{service life} \quad (16)$$

a modification is developed and it is installed at production when it becomes available.

Because of the significantly higher cost of installing a modification on an aircraft already in service, retrofit modifications are not installed unless the fatigue test failure occurred in less than one service life. If the fatigue test failure occurred in less than one service life, the modification is required for safety-of-flight and is installed on all aircraft.

2.2 Inspection Frequency Increased Pending Modification

When it has been determined that a modification must be installed, there is a lead time required to design and fabricate the modification and to await the aircraft's being scheduled for an out-of-service period. During this lead time, it may be necessary to increase the frequencies of the lowest level close internal and close external inspections. The decision to increase the inspection frequency is based on the assumption that the fatigue test specimen represents an average of all elements and that a scatter factor is required to account for all the elements in a typical fatigue life distribution. Therefore, when the flight hours on any particular aircraft reach some percentage of the fatigue test failure life, either that aircraft must be modified or the inspection frequency must be increased until the modification is installed. The percentage of fatigue test failure life at which the inspection frequency is increased is an input parameter; in addition, the factor which increases the inspection frequency for both the close external and the close internal inspection is also an input parameter since the required change depends on the element being considered.

2.3 Modification Tested

When Block 2.1 indicates that a modification is required, it is assumed that the modification is designed to the element's original predicted average fatigue life. This function indicates whether a modification is fatigue tested before its incorporation into the fleet. It is an input parameter for each element type.

2.4 Actual Average Modified Fatigue Life

This function repeats the logic established in Block 1.5. It is based on the assumption that the design analysis of the modification is similar to that of the original element but has an increased probability of being accurate. Therefore, the analysis for the modification results in an actual average fatigue life closer to the required life because of experience gained during operational usage. This increased probability of being accurate is accounted for in SAIFE by decreasing the standard deviation and increasing the mean of the log-normally distributed correction factor discussed in Section 1.5. The standard deviation is decreased by 15% and the mean is increased by 15% of the quantity (1.0 - mean).

Monte Carlo techniques are again used along with the distribution established in Figure 5 to determine the actual average fatigue life of the modified element.

2.5 Actual Average Modified Fatigue Life Set Equal to Predicted Life

If the result of Block 2.3 was to fatigue test the modified element, then it is assumed that the modified element will attain its predicted life or be redesigned and retested until it does. Therefore, the average life predicted by Block 1.4 becomes the actual average fatigue life of the modified element. The logic then returns to Block 1.8 where a fatigue life is assigned to each modified element as the modification is installed on each aircraft. As was done for the original element, the individual fatigue lives are determined by using a Monte Carlo technique.

3. <u>Detailed Description of Block 3.0, Reduce Fatigue Life Because of Production, Service, or Corrosion Defects (see Figure 10)</u>

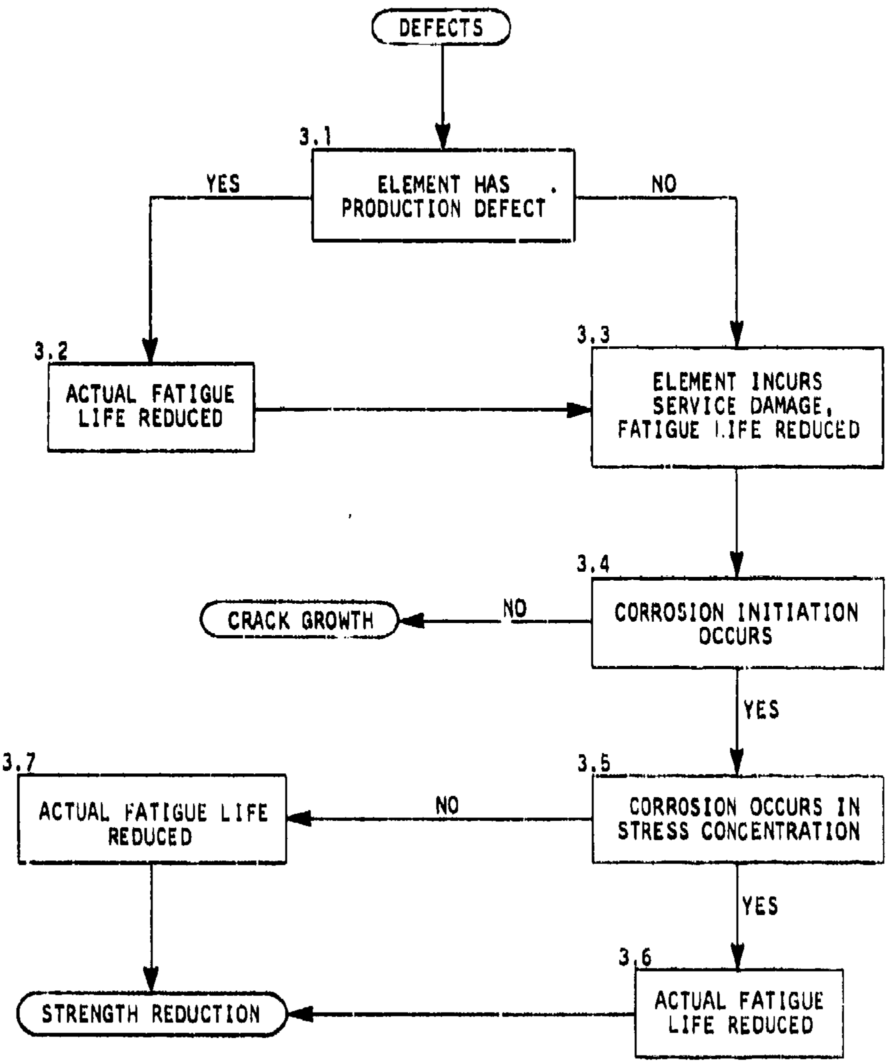


Figure 10. Detailed Flow Diagram - Reduce Fatigue Life Because of Production, Service, or Corrosion Defects

3.1 <u>Element Has Production Defect</u>

As one-time occurrences, production defects result in structural damage only when they initiate the progressive fatigue failure mechanism. Typical production errors include surface irregularities, such as burrs, nicks, and gouges; incorrect dimensions and dimensional tolerances;

improper surface finish and heat treatment; and missing or improperly installed fasteners and shims. The probability of a given element having a production defect before entering service is assumed to be constant for all elements of a particular structural type (spars, frames, etc.) regardless of individual aircraft or the aircraft type.

Data establishing the production defect probability for each structural element type were obtained from the Mechanical Reliability Reports (MRR's) and Service Difficulty Reports (SDR's) in conjunction with the service bulletins issued periodically by the airframe manufacturers and Airworthiness Directives issued by the FAA. Because of the limited regulatory requirements imposed on the air carriers in documenting the nature of a structural defect, the description of a defect occurrence in MRR/SDR data is generally less thorough than the description of a potential structural failure in the pertinent service bulletin. A reported fatigue crack is classified as having been induced by a production defect only if a recognizable production error either is explicitly stated in the MRR/SDR data to be the cause of the failure or is implied to be the cause by reference to the applicable service bulletin or Airworthiness Directive.

From a review of the MRR/SDR's submitted to the FAA during the period 1963 to 1973, a total of 59 fatigue failures were identified which could be attributed to production defects with almost all of these failures resulting from missing or improperly installed fasteners and shims. The number of production defects found in each type of structural element throughout the fleet, along with the average number of individual elements in each aircraft type, are presented in Table 2. The MRR/SDR's, and consequently Table 2, consider only production defects that result in cracks or corrosion. Those defects that do not affect the aircraft structural integrity are not considered to be significant in the SAIFE logic.

The probability that an individual structural element enters service with a production defect is equal to the ratio of the number of production defect occurrences identified in that element type to the total number of elements of that type within the fleet. The equation for determining the probability of a production defect occurring is

$$P_P = \frac{\text{No. of Defects}}{1406 * \text{No. of Individual Elements}} \qquad (17)$$

For the purposes of these calculations, it is assumed that the fleet size is 1406 aircraft, the largest number of pertinent aircraft registered to certified route air carriers in any given year during the period 1963 to 1973.

These data are from Reference 6, the FAA Statistical Handbook, 1973. As shown in Table 2, the largest number of production defects were found in the fuselage stringers, although the fuselage main frames have the greatest probability of entering service with such a defect. Since it is assumed that any given element has a small finite probability of a production defect occurrence, the smallest calculated probability (1.19×10^{-5}) is assigned to those structural elements in which no production defects were identified during the review of MRR/SDR data.

TABLE 2. PROBABILITY OF A PRODUCTION DEFECT OCCURRING

Element Type	No. of Production Defects	Ave. No. Elements Per A/C	Probability of Production Defect in Indiv. Element
Fuselage			
Door frame	0	10	1.19×10^{-5}*
Window frame	0	50	1.19×10^{-5}*
Main frame	17	180	6.72×10^{-5}
Floor beam	1	60	1.19×10^{-5}
Keel beam	2	60	2.37×10^{-5}
Pressure web	4	60	4.74×10^{-5}
Stringer	24	180	9.48×10^{-5}
Wing			
Access frame	0	50	1.19×10^{-5}*
Rib	2	100	1.42×10^{-5}
Spar	3	100	2.13×10^{-5}
Stringers	3	100	2.13×10^{-5}
Wing Center Section			
Rib	1	21	3.39×10^{-5}
Spanwise beam	0	21	1.19×10^{-5}*
Stringer	1	21	3.39×10^{-5}

* estimated

By using Monte Carlo techniques, each individual structural element is tested at the time an aircraft enters service to determine whether a production defect is present. If the uniformly distributed random number drawn is less than or equal to the appropriate production defect occurrence probability, the element is said to have a production defect; otherwise, the element is assumed to be free of such defects.

3.2 Actual Fatigue Life Reduced

As previously stated, a production defect results in structural damage only when it initiates the progressive fatigue failure mechanism. Fatigue crack initiation resulting from a production defect generally occurs early in the life of an aircraft. For the purposes of this simulation, it is assumed that the effect of a production defect is to lower the actual fatigue life of the element.

To determine the fatigue life of a production-damaged element, the distribution of fatigue crack occurrences resulting from production defects with respect to time of crack initiation was determined. With the use of two assumptions regarding crack growth rate and crack size at detection, this distribution may be approximated from the available MRR/SDR data.

Of the 59 fatigue cracks attributed to production defects, only 30 are documented with both the measured crack length and the aircraft flight hours. Since for three fatigue cracks of known length no detection time was reported, a time was assigned to each on the basis of the aircraft's year of manufacture, the date of the crack detection, and an assumed yearly average of 3000 flight hours. Since the crack length was not reported for the remaining 26 fatigue defects, it was assumed that the distribution of lengths for these 26 cracks was the same as that reported for the 33 cracks.

For each of the 59 fatigue cracks initiated by production defects, the crack length is plotted as a function of the crack detection time in Figure 11. Then a crack growth rate was postulated by constructing a line between the origin and one of the data points such that parallel growth rate lines passing through each of the other data points yield no negative times to crack initiation.

Assuming that this crack growth rate is constant for all fatigue failures initiated by production defects, the time to crack initiation can be computed from the reported crack detection time and crack length by the following equation:

$$t_o = t_{DET} + (\log 0.1 - \log \ell_{CR})\, K_{CR} \qquad (18)$$

where t_o = time to crack initiation (flight hours)

t_{DET} = crack detection time (flight hours)

ℓ_{CR} = crack length at detection (inches)

K_{CR} = crack growth constant

K_{CR} = 1575 for production damaged elements

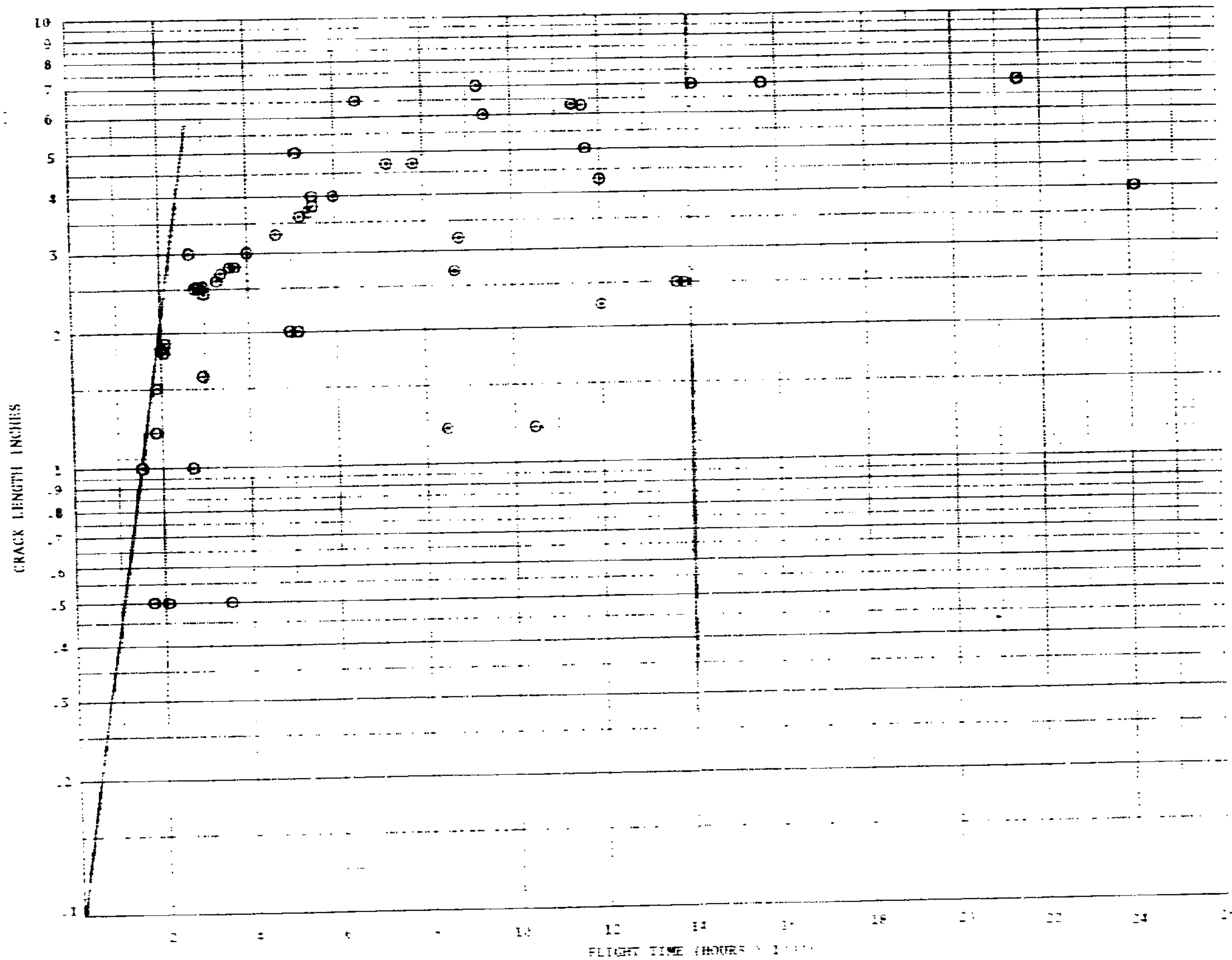


Figure 11. Time of Defect Detection for Production-Damaged Elements vs. Length of Crack

Figure 12 is a histogram of times to crack initiation determined by applying Equation (18) to each of the 59 fatigue cracks initiated by production defects. Several standard statistical models were fitted to the data in Figure 12, with the Weibull distribution offering the best fit. Thus, whenever the simulation program determines that a particular element has a production defect, the time to first crack initiation is drawn from the above Weibull distribution rather than from the original Weibull distribution discussed in Section 1.8. Times to second and third crack initiations are still drawn from the Weibull distribution in Section 1.8.

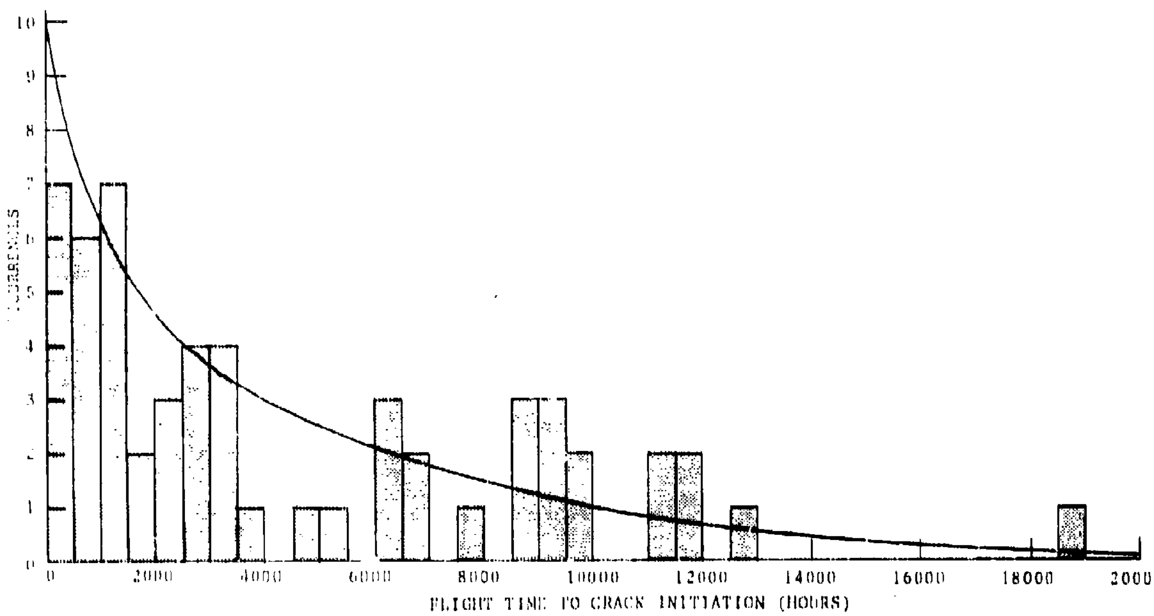


Figure 12. Histogram of Crack Occurrences on Production-Damaged Elements

3.3 Element Incurs Service Damage, Fatigue Life Reduced

Like production defects, service defects result in structural degradation because they initiate the progressive fatigue failure mechanism. Typical examples of service damage include defects occurring during normal ground service operations, such as tears and dents in the aircraft's skin or cargo floor, and defects occurring during regular maintenance operations, such as damage to parts during installation or removal. Data available from MRR/SDR's show that the service damage occurrence rate is constant over the life of the aircraft.

A fatigue crack is classified as having been induced by service damage only if the MRR/SDR report states the cause to be a recognizable service damage defect. Of the 61 crack defects identified that could be attributed to service damage, only 23 are documented with both the measured crack length and the aircraft flight hours. No crack detection time was reported for five fatigue cracks of known length, and the measured crack length was not reported for 32 other fatigue defects; for the one remaining occurrence, neither the crack length nor the crack detection time was recorded. As was done for production defects, a crack detection time, based on the aircraft's time in service and an assumed average number of flight hours per year, was assigned as required. In addition, it was assumed that the distribution of lengths for the 32 reports where crack length was not recorded was the same as that for the 23 cracks where the length was reported.

For each of the cracks initiated by service damage, the crack length is plotted in Figure 13 as a function of the crack detection time. A crack growth rate is then postulated such that no negative times to crack initiation result. Assuming that this crack growth rate is the same for all fatigue failures initiated by service damage, the time to crack initiation can be computed from the reported crack detection time and crack length by the following equation:

$$t_o = t_{DET} + (\log 0.1 - \log \ell_{CR})\, K_{CR} \qquad (19)$$

where K_{CR} = 1430 for service-damaged elements

Assuming that service damage and crack initiation occur almost simultaneously, the times determined from Equation (19) can be used to construct a histogram of times to service damage. As indicated in Figure 14, the service damage occurrence rate is independent of aircraft service time.

The probability of service damage occurring on an element of a given type is equal to the number of occurrences recorded in MRR/SDR's divided by the product of the number of flight hours in the data and the number of individual elements in each aircraft. The equation for calculating the service damage probability is

$$P_S = \frac{\text{No. of Defects} * K_S}{45{,}791{,}114 * \text{No. of Individual Elements}} \qquad (20)$$

where K_S = adjustment factor discussed below

45,791,114 = total flight hours in MRR/SDR data base

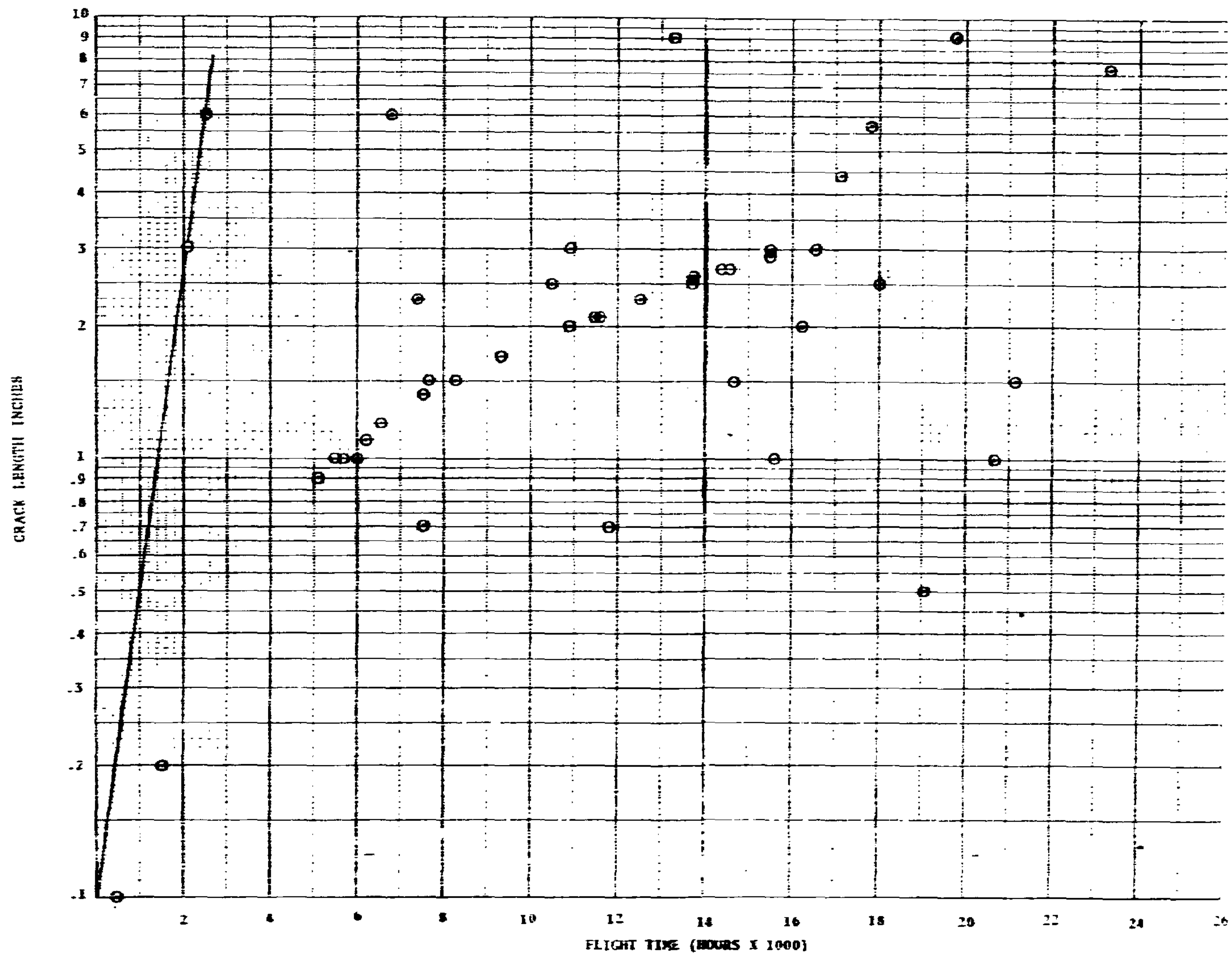


Figure 13. Time of Defect Detection for Service-Damaged Elements vs. Length of Crack

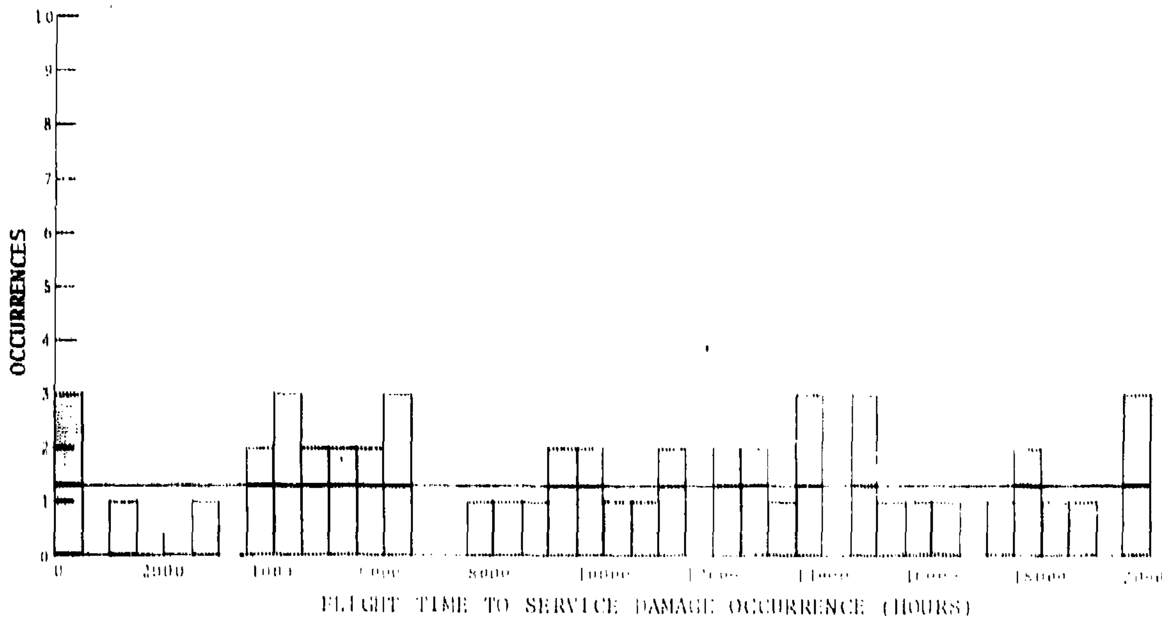


Figure 14. Histogram of Service Damage Occurrences

The appropriate occurrence rate for each structural element type along with the number of occurrences and the number of individual elements in each aircraft is given in Table 3. The service damage occurrence rate is adjusted by a factor of two because of the results of the Maintenance Inspectors Survey (Volume III) which indicates that service damage is twice as prevalent as actually reported. The difference between the survey results and service damage reported by MRR/SDR is attributed to the limited defect description in the MRR/SDR reporting format. Therefore, the MRR/SDR data are used to determine the relative rate of service damage occurrence between the basic element types, and the survey data are used to adjust these rates to what is felt to be more realistic occurrence rates.

No fatigue cracks attributable to service damage are reported for four element types. It is assumed that the wing center section ribs and stringers do not experience service damage. Although no service damage occurrences in the fuselage pressure webs or wing access frames were documented, it is assumed that individual elements of these types have a finite probability of experiencing service damage during their lives and that the occurrence rates for the foregoing are the same as those for the fuselage window frames and wing stringers, respectively.

Using Monte Carlo techniques, each individual structural element is tested when an aircraft enters service to determine whether service damage is incurred and, if so, at what time the resultant fatigue crack initiates. The governing equation which, in conjunction with the selected uniformly distributed random number, determines this time to crack initiation is

$$t = \frac{1}{\lambda} \ln(RN) \tag{21}$$

where λ is the appropriate service damage occurrence rate for the element type, and RN is the random number selected by Monte Carlo methods.

TABLE 3. SERVICE DAMAGE OCCURRENCE RATES

Element Type	No. of Service Damage Occurrences Before 11040 Flt Hr	Ave. No. Elements Per A/C	Adj. Service Damage Occurrence Rate (occ./flt.hr)
Fuselage			
Door frame	7	10	3.06×10^{-8}
Window frame	8	50	6.99×10^{-9}
Main frame	21	180	5.10×10^{-9}
Floor beam	1	60	2.43×10^{-10}
Keel beam	1	60	2.43×10^{-10}
Pressure web	0	60	2.43×10^{-10}*
Stringer	14	180	3.40×10^{-9}
Wing			
Access frame	0	50	1.31×10^{-9}*
Rib	1	100	4.36×10^{-10}
Spar	1	100	4.36×10^{-10}
Stringer	3	100	1.31×10^{-9}
Wing Center Section			
Rib	0	21	0
Spanwise beam	4	21	8.32×10^{-9}
Stringer	0	21	0

* estimated

3.4 Corrosion Initiation Occurs

Each individual structural element has a finite probability of experiencing corrosion during its service life. The corrosion occurrence rate for each element type is determined from the service experience documented in the MRR/SDR's. The first step in the formulation of the corrosion occurrence rate is a determination of the corrosion growth rate. For this determination, the corrosion occurrence data of the DC-9 wing center section stringers are analyzed by examining the reported corrosion depths as a function of the aircraft's accrued flight hours. Of the 52 reported corrosion occurrences on the DC-9, only 36 are documented with both the corrosion depth and aircraft life. Furthermore, in 15 of these occurrences, the corrosion initiated at fastener holes in the stringers. Since the corrosion depth growth rate may be related to the state of stress, the instances in which corrosion occurred in a stress concentration are excluded during the initial analysis.

For each of the remaining 21 corrosion occurrences, the reported corrosion depth is plotted as a function of the corrosion detection time (see Figure 15). A corrosion growth rate is then postulated by constructing a line between the origin and one of the data points such that parallel growth rate lines passing through each of the other data points yield no negative times to corrosion initiation. This is a somewhat conservative approach in that it assumes corrosion may initiate as soon as an aircraft enters service. The calculated rate is 1.09×10^{-5} inches per flight hour.

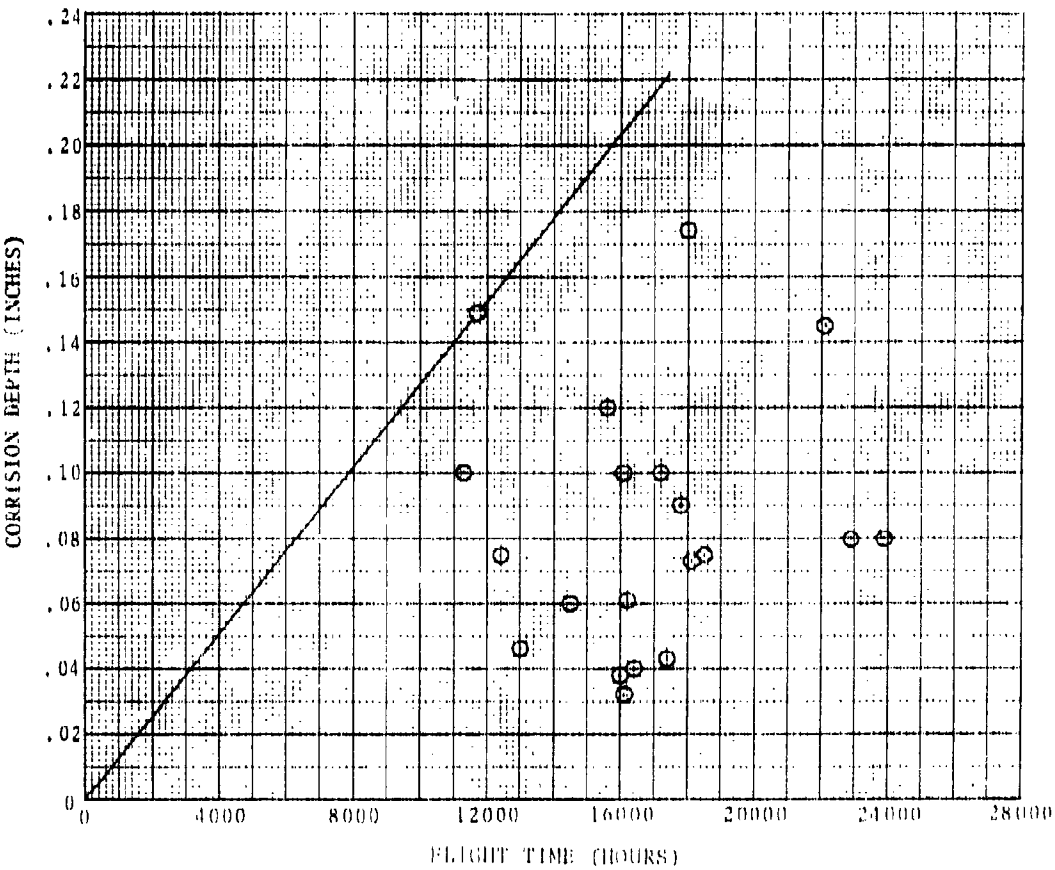


Figure 15. Time of Defect Detection vs. Depth of Corrosion for Corrosion-Damaged Elements

Now the time to corrosion initiation may be determined for each occurrence of known depth and detection time from the following equation:

$$t_o = t_{co} - \frac{d_{co}}{k_{co}} \tag{22}$$

where t_o = time to corrosion initiation (flight hours)

t_{co} = corrosion detection time (flight hours)

d_{co} = corrosion depth at detection (inches)

k_{co} = corrosion growth rate (inches per flight hour)

For those corrosion occurrences of unreported detection time, a value of t_{co} is assigned on the basis of the aircraft's time in service and an assumed average number of flight hours per year. For those corrosion occurrences of unrecorded depth, a value equal to the mean of the 21 previously plotted corrosion occurrences is assigned. The corrosion initiation time for each of the other reported corrosion occurrences throughout the fleet can now be calculated by using the same equation and growth rate.

It should be noted that the same corrosion depth growth rate is used to determine the time to corrosion initiation for all occurrences regardless of whether the corrosion is located in a stress concentration. This procedure was found acceptable after analyzing the data for corrosion coincident with stress concentrations. For the 15 fully documented occurrences in the DC-9 wing center section stringers, the previously established techniques were then applied to these data to postulate a new corrosion depth growth rate. This rate was found to be very close to 1.09×10^{-5} in. per flight hour. Since this rate was similar to that previously calculated, the original growth rate was chosen to conservatively define time to corrosion initiation for all occurrences.

For each structural element type, a curve of cumulative corrosion occurrences throughout the fleet is plotted as a function of time to corrosion initiation. Each plot exhibits either a constant corrosion occurrence rate or two constant but unequal rates. The occurrence rate during the initial service life of the aircraft is frequently lower because protective finishes and coatings are intact and have not deteriorated or been scratched and gouged. As the protective finishes break down, the rate increases. At the high-occurrence rates, the data shows that a time is reached when the number of high-time aircraft in the fleet decreases and accordingly the occurrence rate also decreases. This decrease is ignored in the simulation since all the aircraft in the fleet being simulated are flown to their retirement lives. These corrosion occurrence curves are shown in Figures 16 through 27. The occurrence rate for a fleet of 1406 aircraft is then derived from this data by using Equation (23). The fleet size data were obtained from the FAA statistical handbook of aviation for 1970 and 1973 (References 6 and 7).

$$P_C = \frac{\Delta \text{ Occurrences}}{\Delta \text{ Flight Hours} * 1406 * \text{No. of Individual Elements}} \quad (23)$$

3.5 Corrosion Occurs in Stress Concentration

The fatigue life reduction of a structural element resulting from corrosion damage depends upon the state of stress in the corroded region. For all elements of a particular structural type, the probability of corrosion occurring within a stress concentration is assumed to be constant over the life of the aircraft, regardless of individual aircraft or aircraft type.

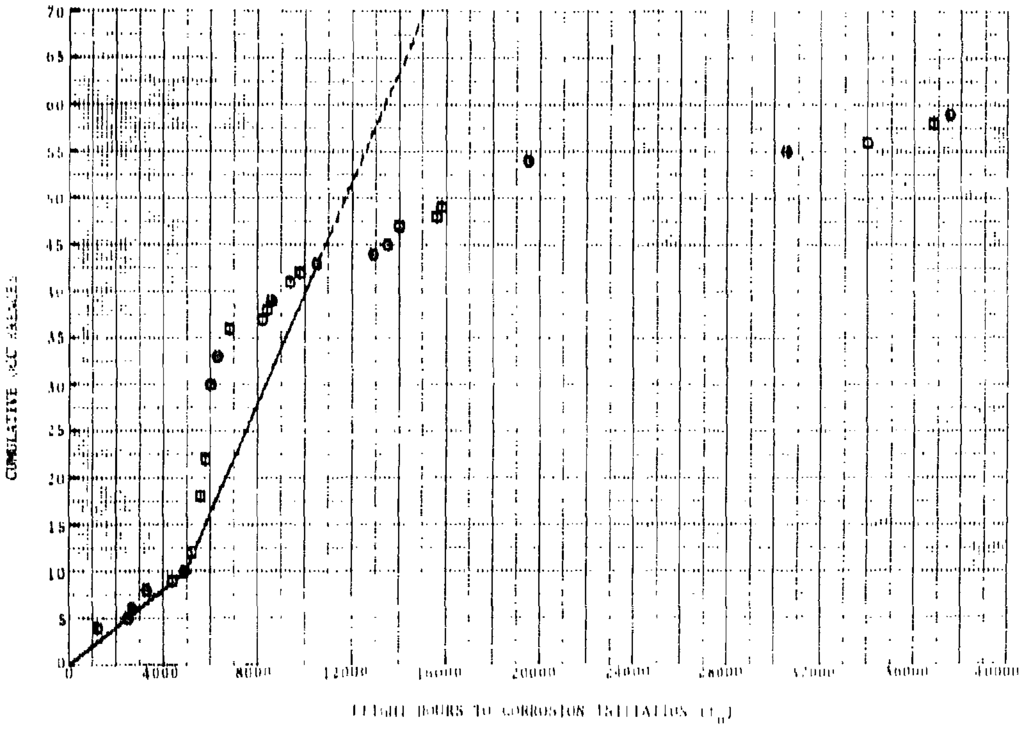


Figure 16. Corrosion Occurrence Rate for Fuselage Floor Beams

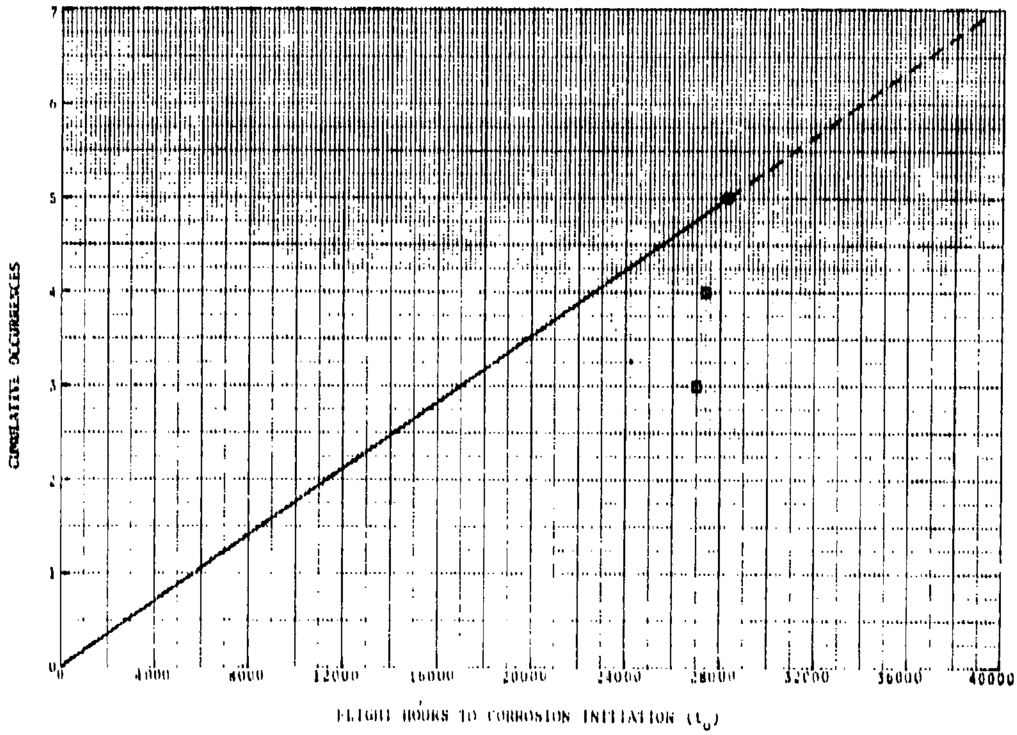


Figure 17. Corrosion Occurrence Rate for Fuselage Keel Beams

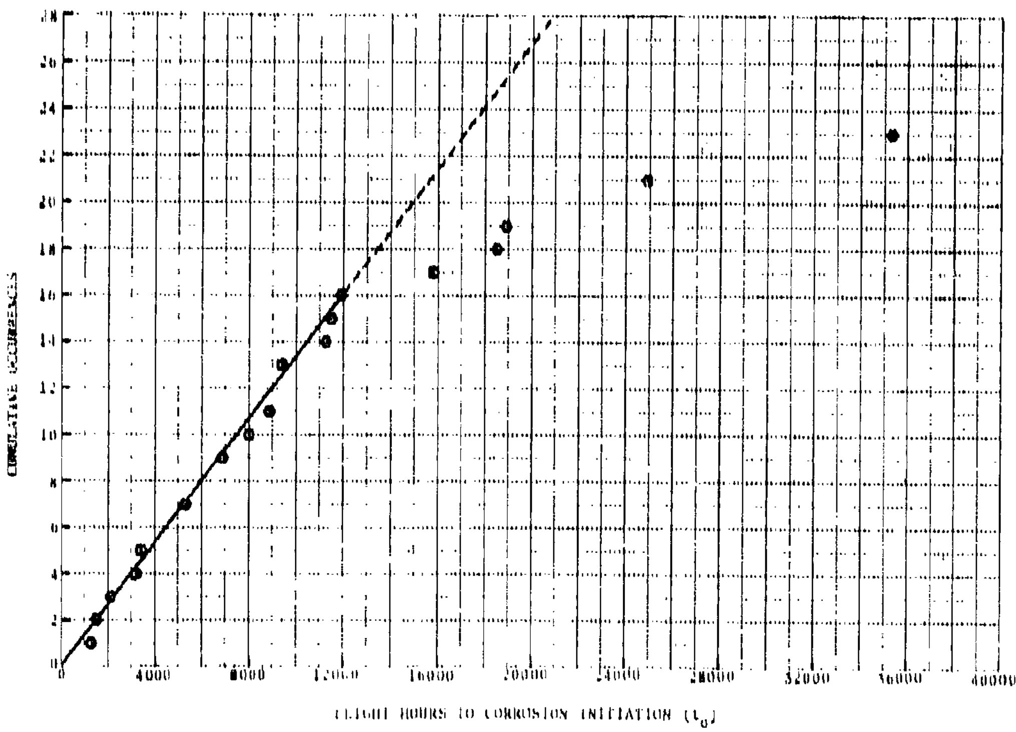


Figure 18. Corrosion Occurrence Rate for Fuselage Main Frames

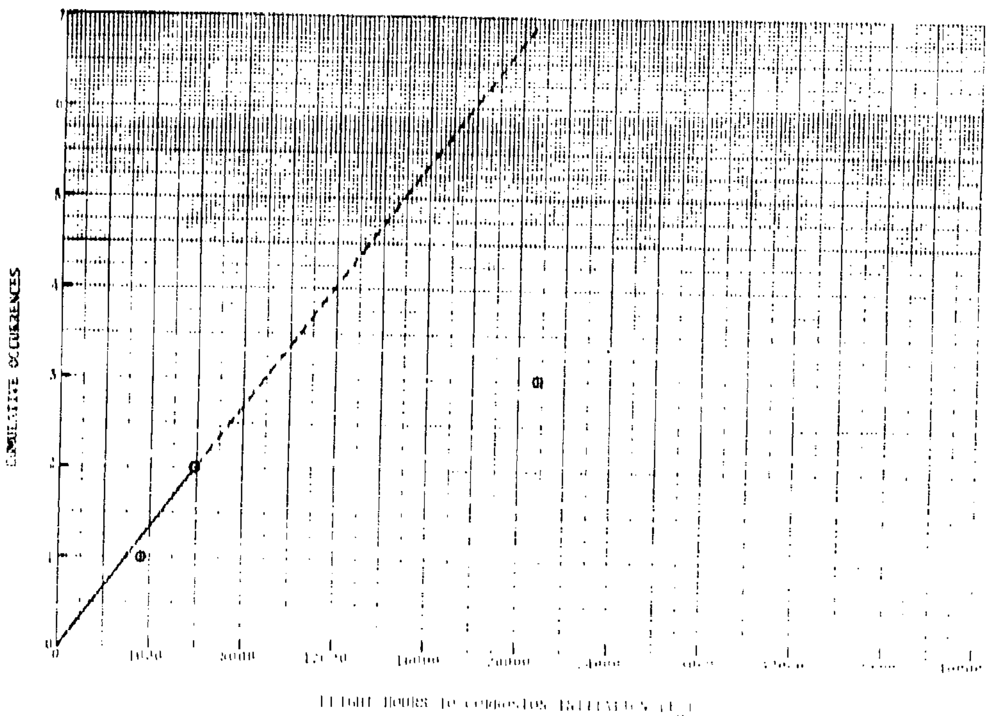


Figure 19. Corrosion Occurrence Rate for Fuselage Window Frames

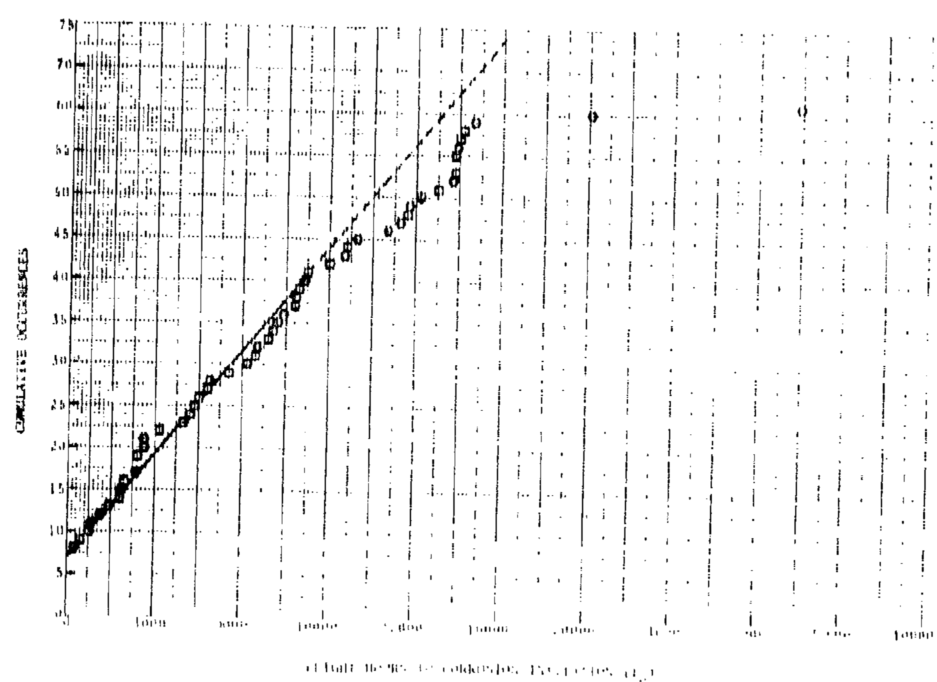


Figure 20. Corrosion Occurrence Rate for Fuselage Door Frames

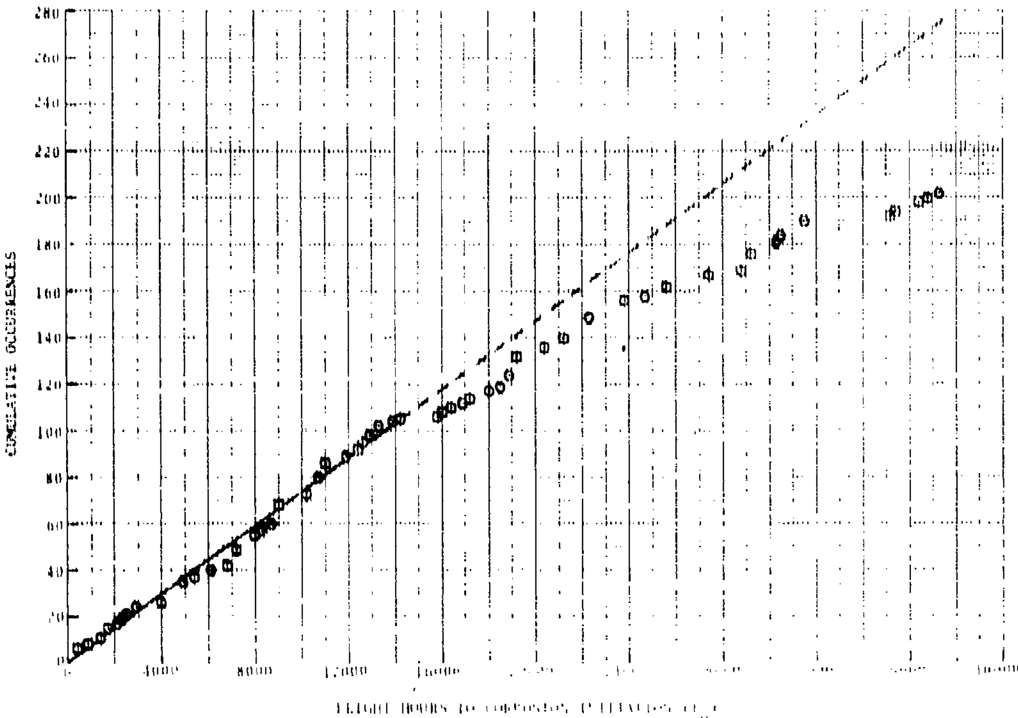


Figure 21. Corrosion Occurrence Rate for Fuselage Stringer (Bottom)

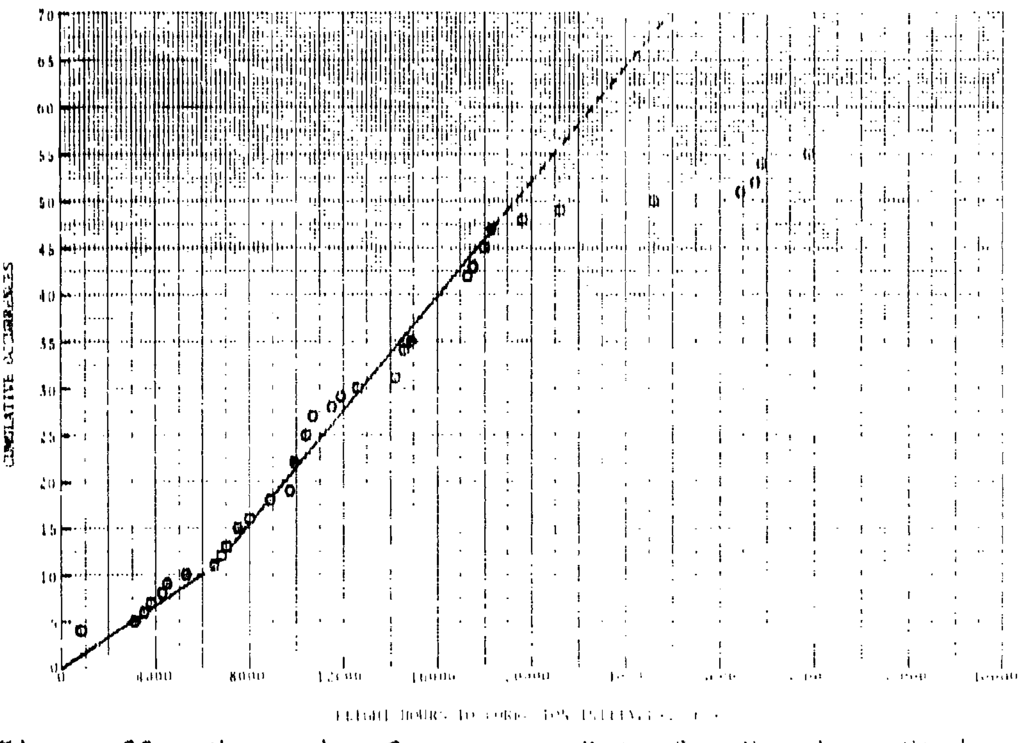


Figure 22. Corrosion Occurrence Rate for Fuselage Stringer (Side and Top)

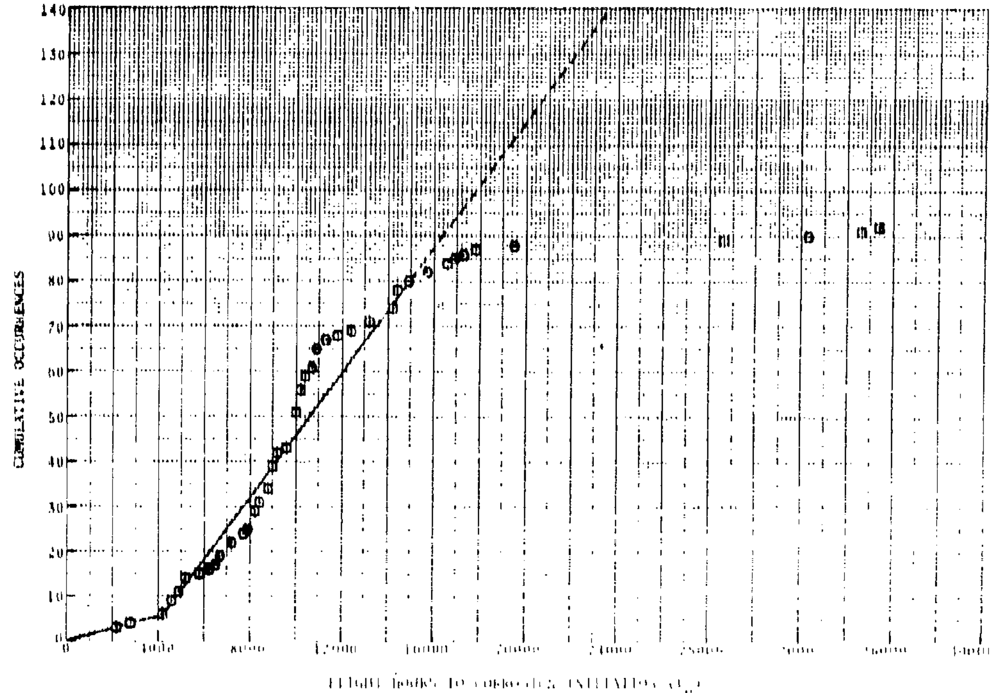


Figure 23. Corrosion Occurrence Rate for Wing Center Section Stringers

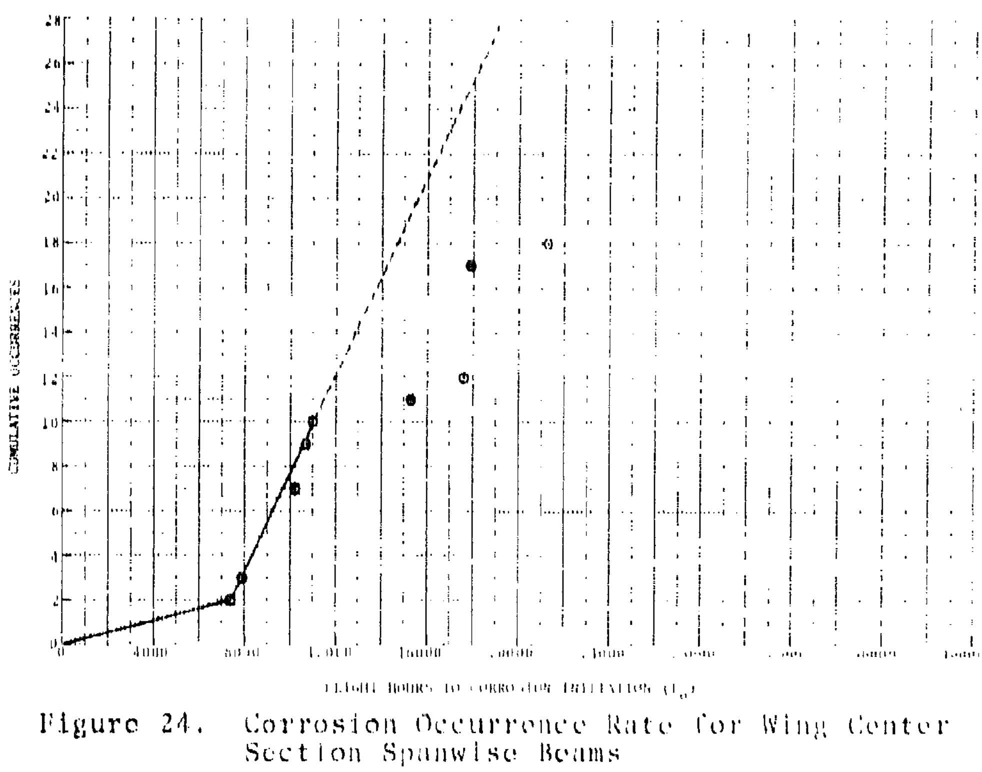


Figure 24. Corrosion Occurrence Rate for Wing Center Section Spanwise Beams

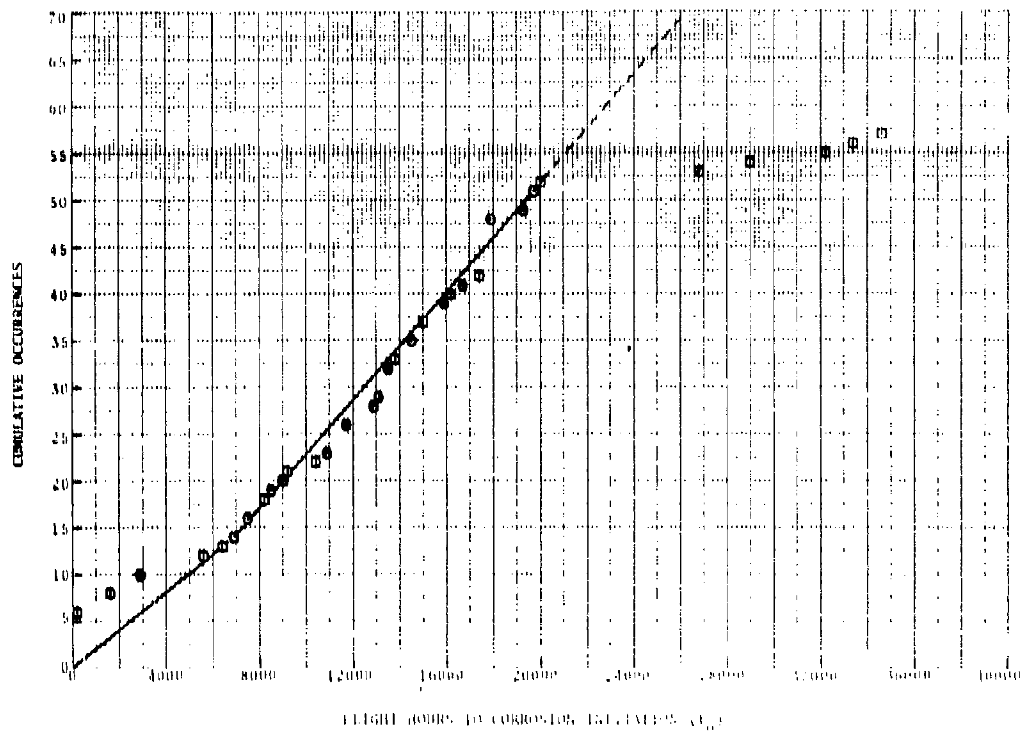


Figure 25. Corrosion Occurrence Rate for Wing Spars

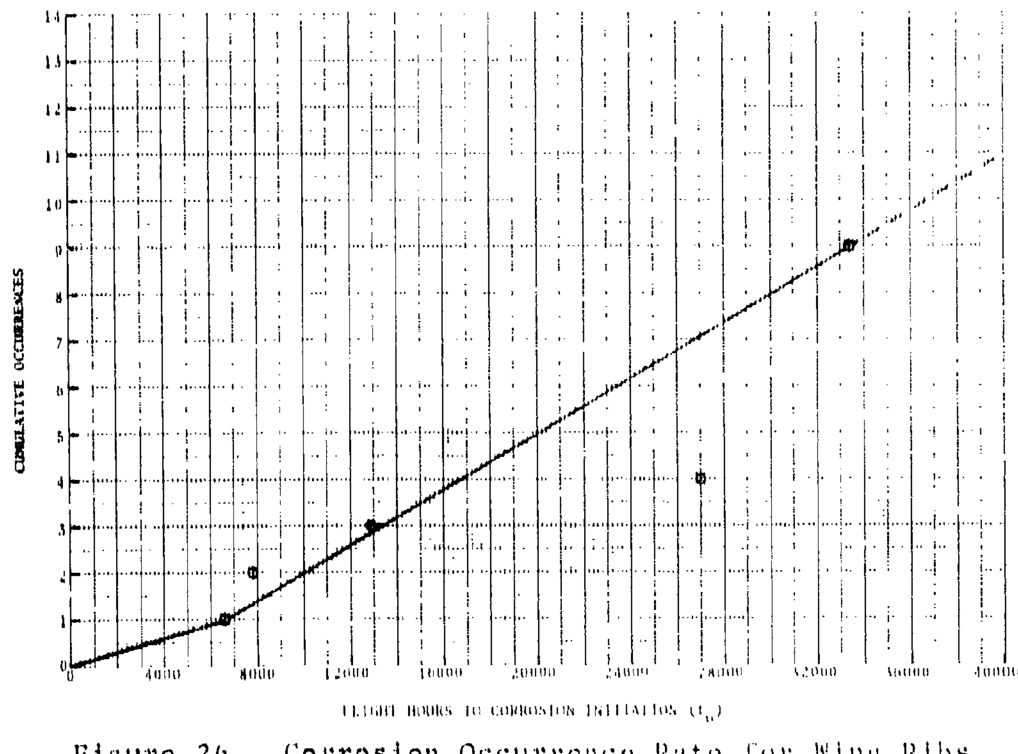


Figure 26. Corrosion Occurrence Rate for Wing Ribs

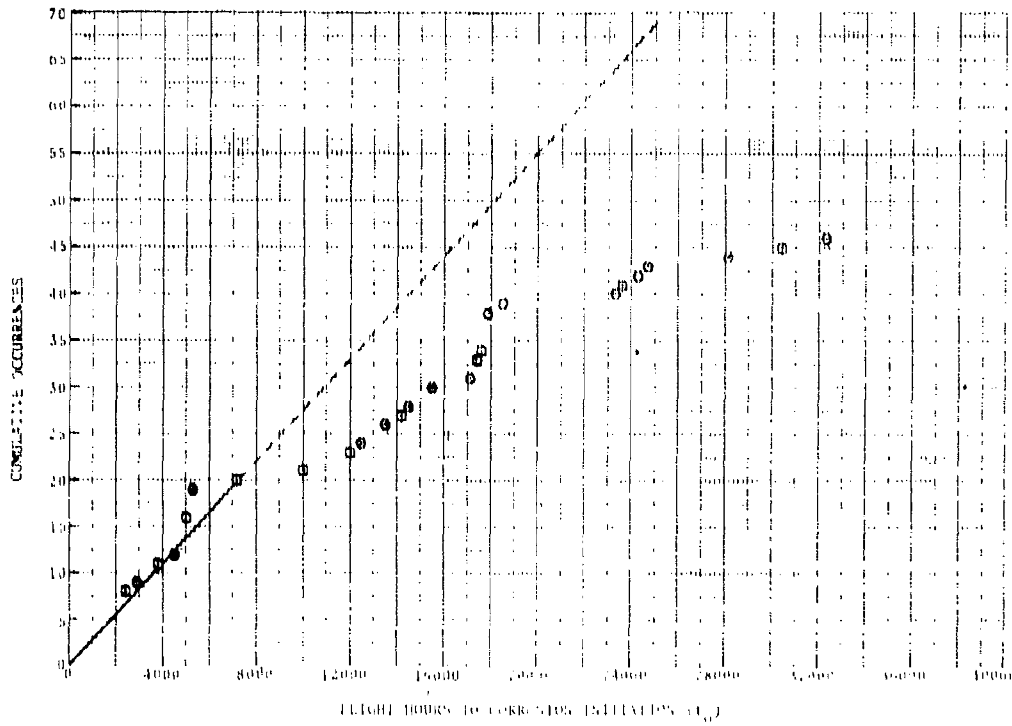


Figure 27. Corrosion Occurrence Rate for Wing Stringers

The corrosion occurrences documented in the MRR/SDR's are classified as being located within a stress concentration only if a recognizable design feature known to be a stress riser is coincident with the reported corrosion. The two types of stress riser identified in the available data are fastener holes and bend radii (or fillets).

Of the 583 documented corrosion occurrences, 75 were classified as being located in stress concentrations. The number of corrosion occurrences within stress concentrations found in each type of structural element as well as the total number of corrosion occurrences in elements of that type are presented in Table 4. The probability that corrosion exists in a stress concentration is equal to the ratio of the number of corrosion occurrences found in stress concentrations to the total number of corrosion occurrences identified in that structural element type within the fleet. Since it is assumed that a corrosion occurrence in any type of structural element has a finite probability of being located in a stress concentration, a probability must be assigned to those structural elements for which no occurrences of corrosion are reported. This assignation is made by assuming a probability equivalence between element types of generally similar design and structural function. For example, the probability of a corrosion occurrence being located in a stress concentration of either a door frame or a window frame is assumed to be the same. Probability values assigned in this manner are indicated by an asterisk in Table 4.

TABLE 4. PROBABILITY OF CORROSION OCCURRENCE IN A STRESS CONCENTRATION

Structural Element Type	No. of Corrosion Occurrences	No. of Occurrences in Stress Concen.	Probability of Corrosion Occur. in Stress Concen.
Fuselage			
Door frames	13	5	0.385
Main frames	23	3	0.130
Window frames	3	2	0.385*
Floor beams	59	1	0.017
Keel beams	5	0	0.017*
Pressure webs	1	0	0.136*
Stringers	257	35	0.136
Wing			
Access frames	0	0	0.385*
Ribs	9	1	0.111
Spars	57	3	0.053
Stringers	46	3	0.065
Wing Center Section			
Ribs	0	0	0.056*
Spanwise beams	18	1	0.056
Stringers	92	21	0.228

* estimated

Using Monte Carlo methods, each incident of corrosion is tested at the time it occurs to determine whether it is located in a stress concentration. If the uniformly distributed random number drawn is less than or equal to the appropriate probability of corrosion occurrence in a stress concentration, the corrosion is assumed to be located in a stress concentration; otherwise, the corrosion is assumed to occur in a uniform stress field.

3.6 Actual Fatigue Life Reduced Because of Corrosion in a Stress Concentration

The presence of corrosion on a structural element contributes to the potential failure of the element by reducing the original fatigue life of the element. Tests conducted on spar caps taken from HU-16 aircraft and documented in Reference 8 show that severe exfoliation corrosion reduces fatigue life significantly, but that surface pitting or very mild exfoliation has only a minor effect on fatigue life.

When it has been determined that corrosion has occurred in a stress concentration, the fatigue life of the element is reduced by a factor that is an input variable in Block 1.1. Reference 8 shows that when corrosion occurs in a stress concentration, it generally manifests itself as exfoliation. The reference indicates that fatigue tests conducted on HU-16 spar caps showed that severe exfoliation corrosion results in a fatigue life reduction of up to 70%. However, since the tests were conducted on specimens that had previously experienced service fatigue damage, it was felt that approximately 30% of the reduction was because of such damage. Therefore, 40% of the fatigue life reduction was attributed to exfoliation corrosion and the suggested simulation input for fatigue life reduction when corrosion occurs in a stress concentration is 0.40.

3.7 Actual Fatigue Life Reduced Because of Corrosion Outside a Stress Concentration

Corrosion outside a stress concentration also affects the fatigue life of the element, but the fatigue life reduction is less severe than when the corrosion is in a stress concentration. Reference 1 indicates that the fatigue life reduction because of corrosion outside a stress concentration is approximately one half of the fatigue life reduction when the corrosion is in a stress concentration. Therefore, the suggested simulation input for fatigue life reduction when corrosion is outside a stress concentration is 0.20.

4. Detailed Description of Block 4.0, Reduce Strength Because of Crack Growth/Project Time to Failure (see Figure 28)

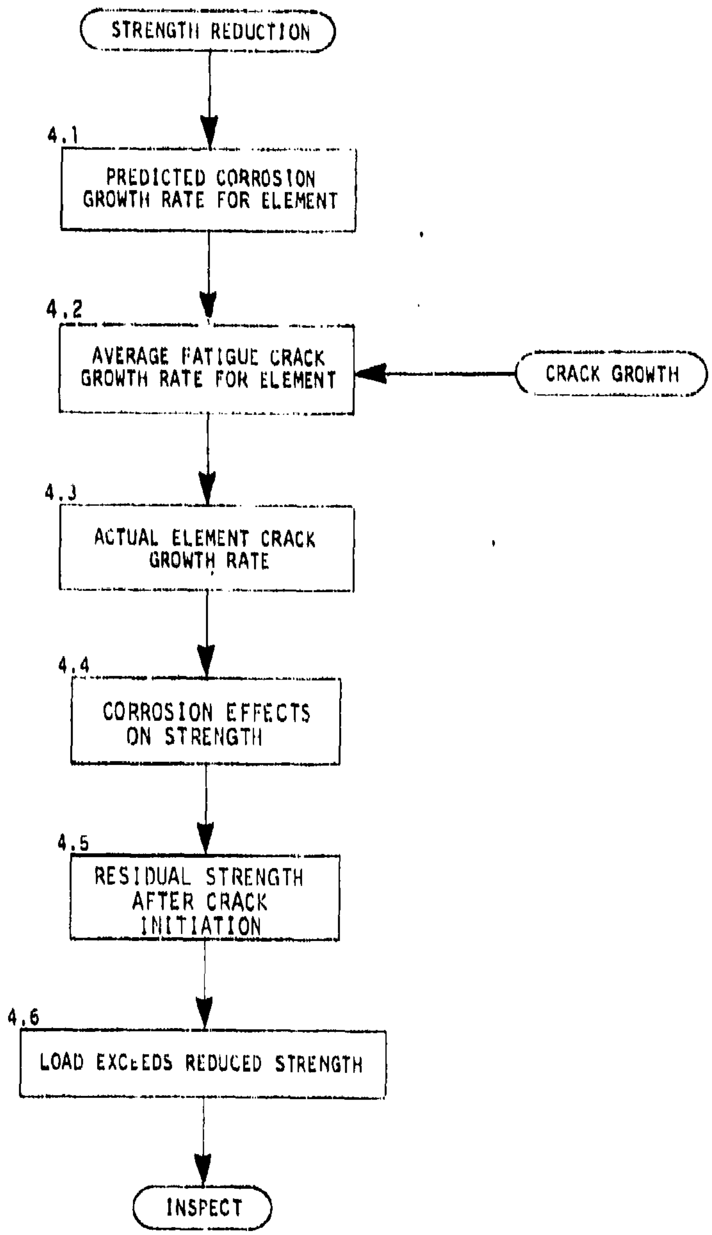


Figure 28. Detailed Flow Diagram - Reduce Strength Because of Crack/Corrosion Growth/Predict Time to Failure

4.1 Predicted Corrosion Growth Rate

If, as a result of Block 3.4, it has been determined that corrosion is present on an element, this function defines the rate at which the area and depth of the corrosion will grow. The growth rates presented are the result of an analysis of MRR/SDR data. Because of the small number of usable data points, the growth rates presented are determined from data for different element types and are assumed to represent a nominal condition. This nominal growth rate is modified by applying a correction factor to each element type based on the corrosion resistance rating (CRR) as determined by the MSG-2 conference convened for the certification of each aircraft type. The CRR has a range of 1 through 4 with 4 representing the elements that are most resistant to corrosion. The growth rate for each element then becomes

$$\text{Actual Rate} = \text{Nominal Rate} \times \text{Adjustment Factor} \tag{24}$$

where

CRR	Adjustment Factor
1	1.50
2	1.25
3	1.00
4	0.75

Nominal depth rate = 1.09×10^{-5} in./flt hr

Nominal area rate = 2.0×10^{-3} sq.in./flt hr

The corrosion depth growth rate was derived in Block 3.4 from the data presente. in Figure 15. The corrosion area growth rate was also determined from the MRR/SDR data.

4.2 Average Crack Growth Rate

The simulation program uses two constant crack growth rates to approximate the normally non-constant fatigue crack growth rate. These two constant rates are inputs for each element type. The first constant rate represents the crack growth from crack initiation until the critical crack length is reached. This is the slow growth rate period.

The second constant rate represents the fast growth rate and accounts for the crack growth from critical crack length to structural failure. Taken from References 9 and 10, Figures 29 through 36 are typical crack growth curves for selected components of a DC-8 and DC-10. Drawn over the crack growth curves in each of the figures are the two constant approximations used in the SAIFE program. For those very few elements (Figure 35) whose crack growth curves do not conform to the general shape discussed above, a single constant approximation is used. To determine the input parameters required for the simulation program, the slow growth portion of the crack growth curve must be extrapolated backward to zero crack length to determine the constant rate from crack initiation to critical crack length. The fast growth portion of the crack growth curve starts at the critical crack length and is applied until structural failure.

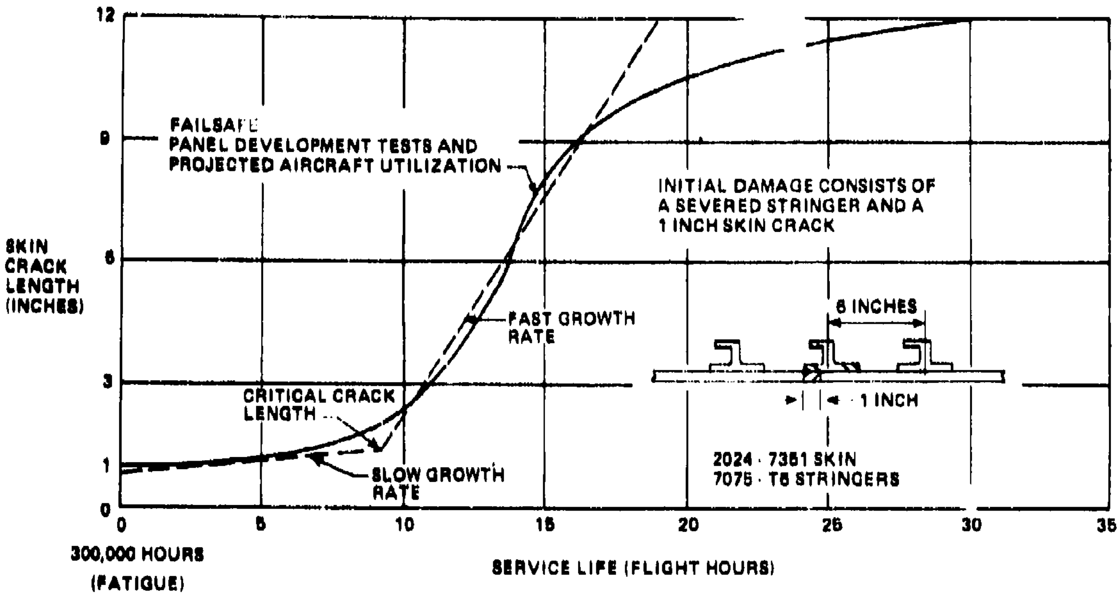


Figure 29. Two-Bay Longitudinal Crack Propagation Rate for Typical Lower Surface Wing Stringers

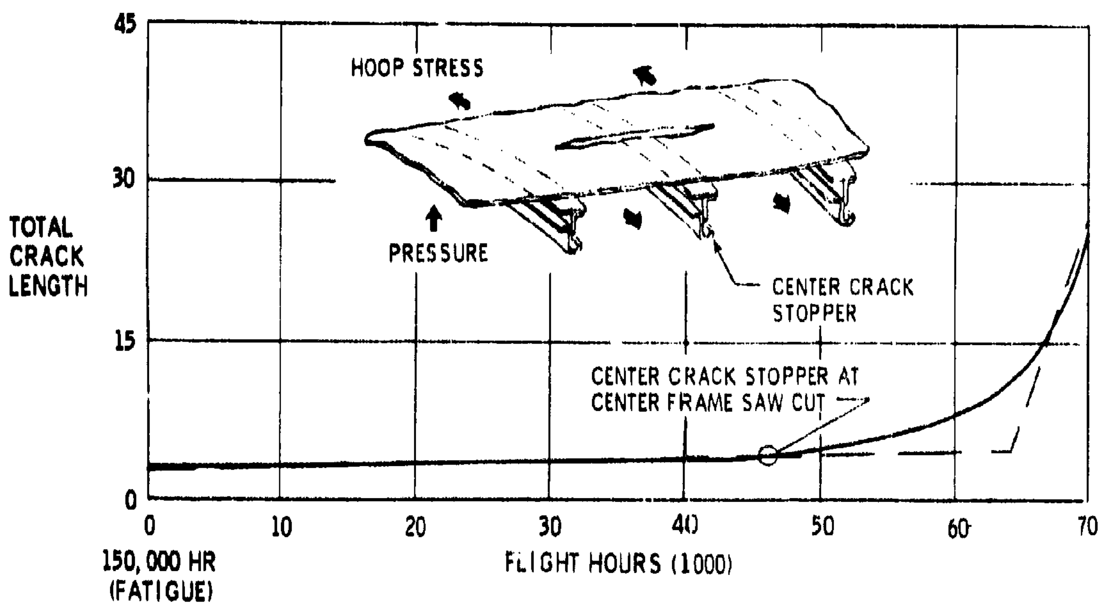


Figure 30. Crack Propagation Rate for Typical Fuselage Main Frames

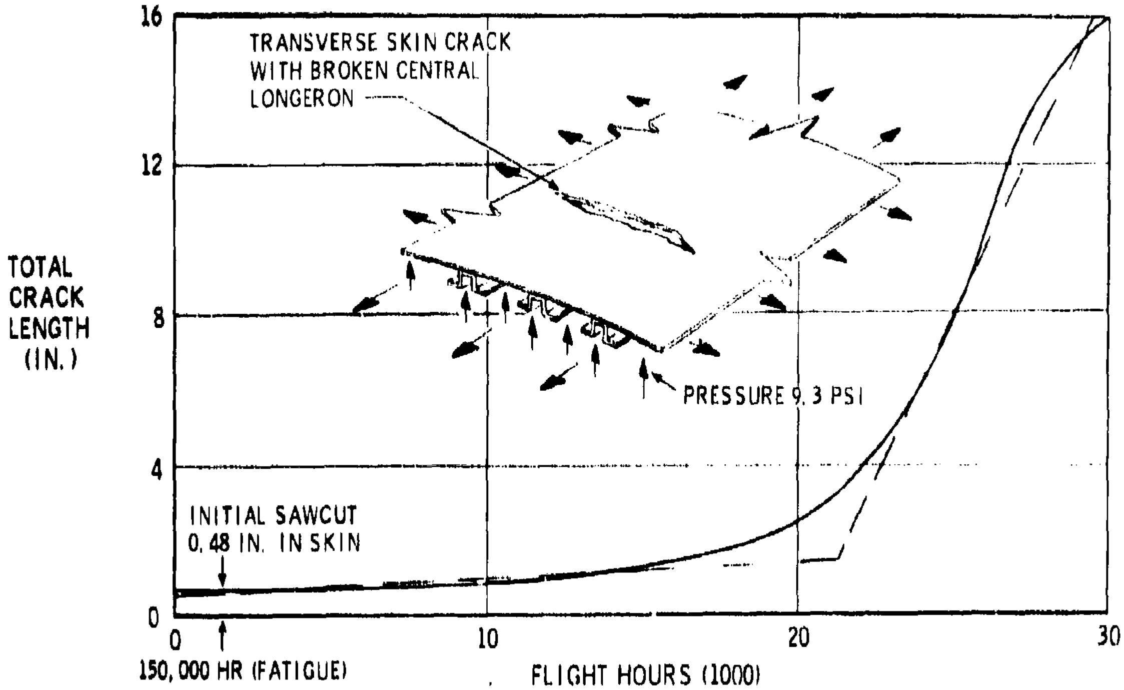


Figure 31. Crack Propagation Rate for Typical Fuselage Stringers

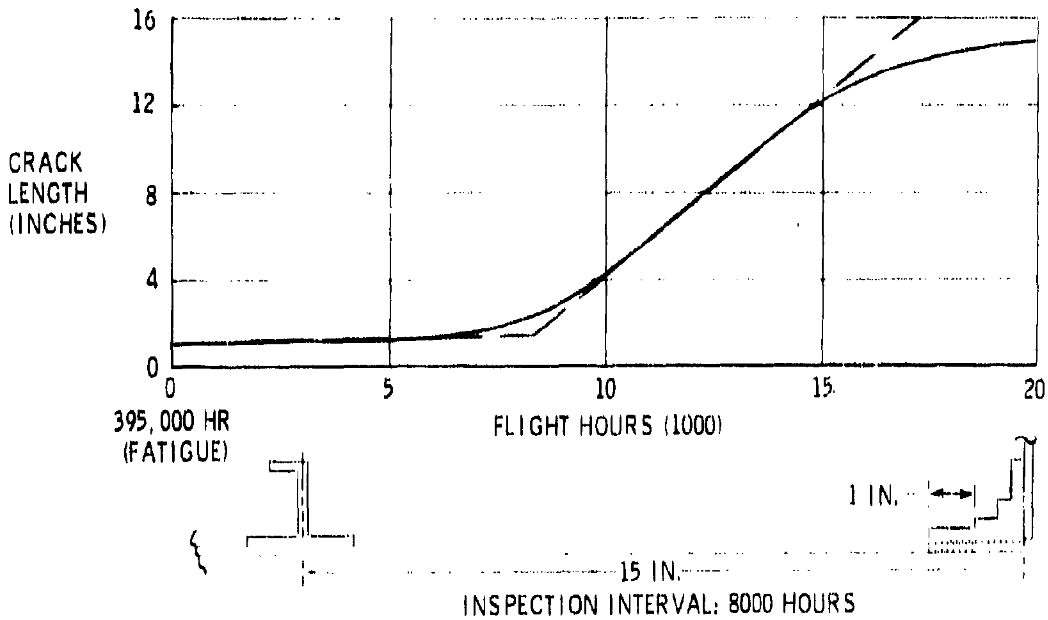


Figure 32. Crack Propagation Rate for Typical Lower Rear Wing Spars

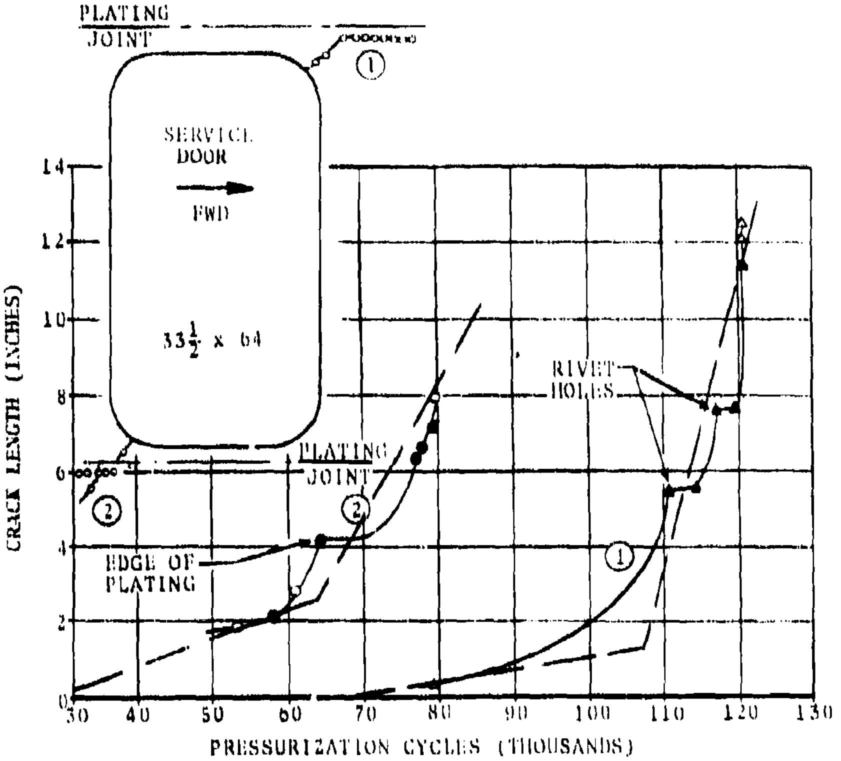


Figure 33. Crack Propagation Rate for Typical Service Door

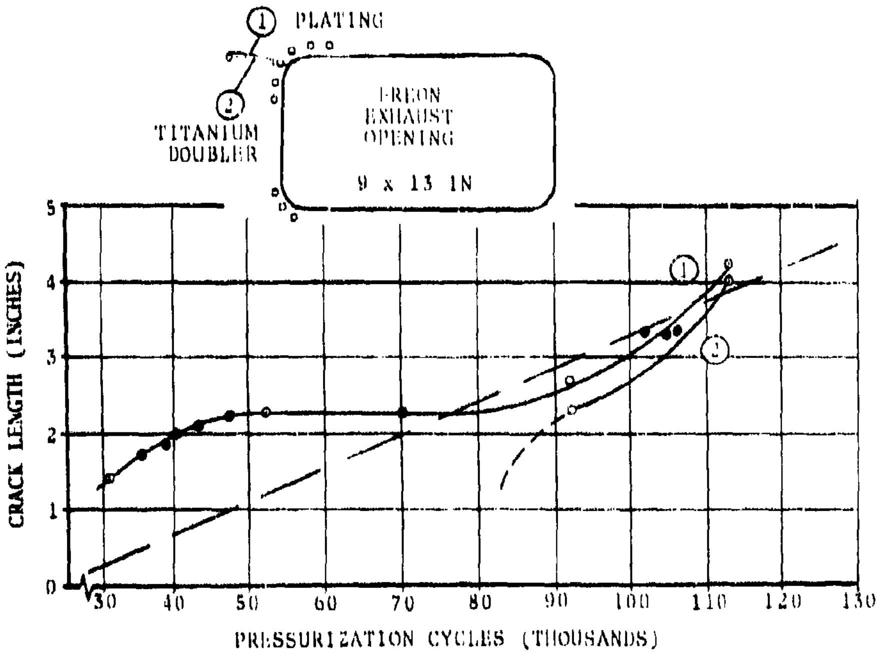


Figure 34. Crack Propagation Rate for Typical Rear Exhaust Opening

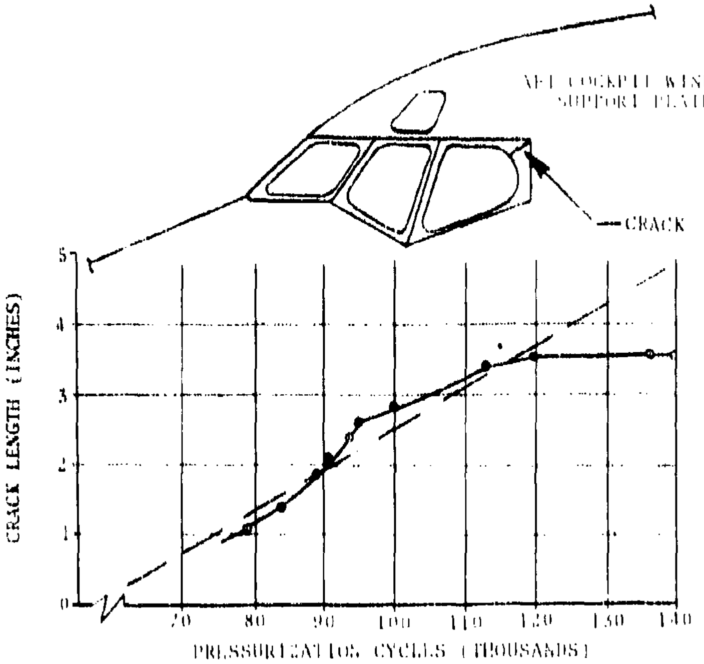


Figure 35. Crack Propagation Rate for Typical Cockpit Window

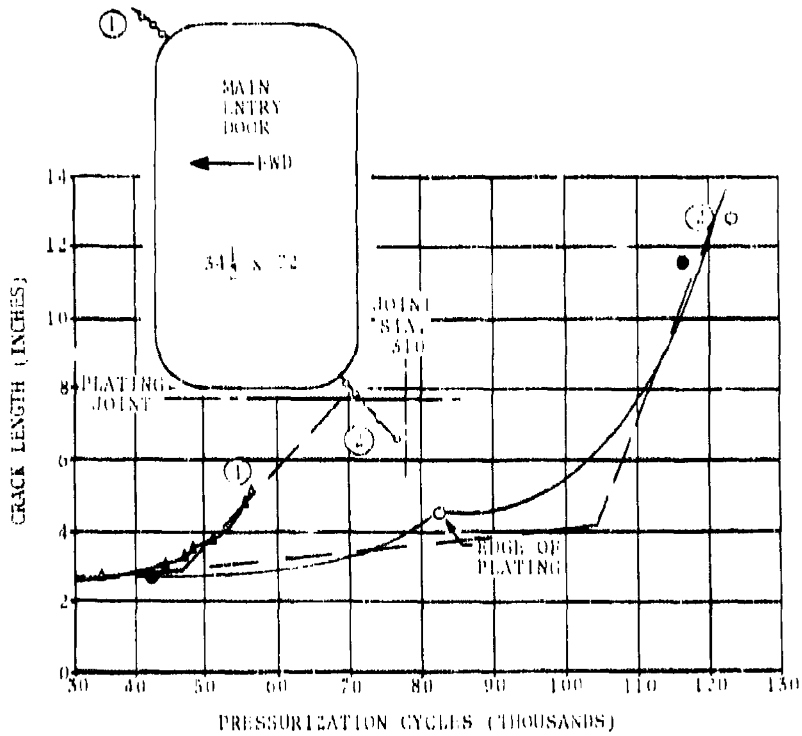


Figure 36. Crack Propagation Rate for Typical Main Entry Door

4.3 Actual Element Crack Growth Rate

Crack growth rate, like fatigue life, is a probabilistic variable and must be treated statistically. The scatter present in the growth rate distribution reflects both the basic characteristics of the fatigue process, the corrosion environment variation, and the load environment variation of typical aircraft structures. On the basis of the results from full-scale testing, Eggwertz (Reference 11) has determined that the standard deviation for the crack growth rate is approximately one-half that of the fatigue life. Taken from Reference 12, Figure 37 characterizes the variability of crack initiation and growth in distributional forms. In the simulation both the slow and fast crack growth rates are assumed to be normally distributed with means equal to the slow and fast growth rates determined in Section 4.2. The standard deviation is set equal to one half that used in the time to crack initiation distribution. Both of these parameters, the mean slow and mean fast growth rates, are required as SAIFE input.

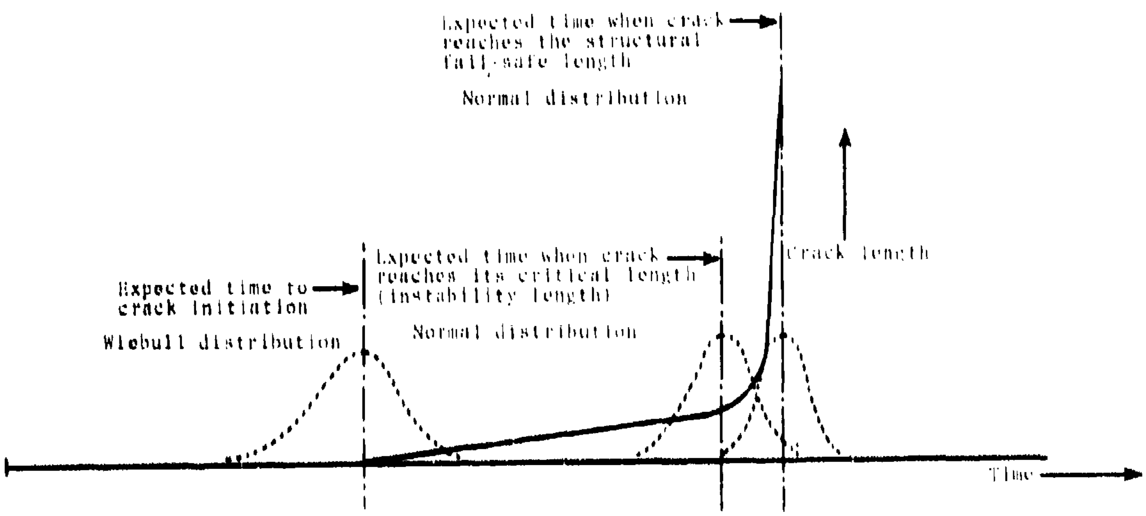


Figure 37. Typical Crack Initiation and Growth Rate Distributions

The slow and fast crack growth rates for each individual element on each aircraft in the fleet are selected by a Monte Carlo method. The selection is controlled so that the fatigue life, slow growth rate, and fast growth rate of an individual element all reflect a consistant material scatter and load environment.

4.4 Corrosion Effects on Strength

References 8, 13, and 14 present test results which show that the reduction in static strength because of corrosion is negligible until the loss of cross-sectional area becomes an extremely significant portion of the element cross section. A paper presented at the 1972 Tri-Service Conference on Corrosion (Reference 15) presented test data on the effect of corrosion on static strength. The data scatter was so large that it must be concluded that the effect of corrosion on static strength cannot be measured accurately by present standards.

An examination of the growth rates defined in Block 4.1 and the detection probabilities defined in Section 5.6 indicates that corrosion will certainly be detected before any detectable reduction in static strength has taken place. Reference 9 supports this finding with regard to detection.

Reference 16 examines the effects of corrosive environments on fatigue life of aluminum alloys under maneuver spectrum loading. The test results showed that in general all the aluminum alloy plate materials tested experienced significant and progressive reductions of mean fatigue lives for increasingly severe corrosive environments. The average crack propagation rate was approximately tripled by the static and cyclic corrosion environments. However, the effective crack length as measured after specimen failure appeared to be unaffected by environment. This suggests that the effects on residual cracked strength of the various corrosive environments is negligible. Therefore, although it may be intuitively felt that corrosion must have an effect on static strength, the data presently available does not support that opinion. It is also apparent that state-of-the-art material selection and preventive coating applications have eliminated corrosion as a major factor in catastrophic accidents.

While corrosion does not affect the relationship between crack length and residual strength, it does accelerate strength degradation by its effect on crack propagation rate. When corrosion is present, the crack propagation rate is increased by the same factors used to decrease fatigue life as defined in Sections 3.6 and 3.7.

4.5 Strength Reduction Because of Crack Growth

For the simulated structural elements, it is assumed that the original ultimate strength, S_u, is constant until the time of crack initiation, t = 0, and that after crack initiation the subsequent residual strength, S, can be expressed as a function of time (flight hours). As discussed in Section 4.2, the two-part approximation to the growth rate of a single fatigue crack has the general configuration shown in Figure 38. It will also be assumed that the relationship between crack length and residual strength, S, is as shown in Figure 39, with S going to 1.0 at the crack length corresponding to level flight failure. If the curves in Figure 38 and Figure 39 are combined, the resultant composite curve illustrates the functional relationship between time (flight hours) and S(t), where S(t) is the residual strength of the element at time t after crack initiation. Such a curve for a typical element is shown in Figure 40. Approximately 10 percent of the structural elements under consideration have no change in fatigue crack growth rate at the critical crack length. For these elements the functional relationship between flight hours and residual strength, S(t), is illustrated by the two-part curve shown in Figure 41. Approximately 15 percent of the structural elements under consideration have critical crack lengths and average slow crack growth rates such that neither the fail-safe length nor the critical crack length can be reached in the aircraft's service life. For these elements the functional relationship between flight hours and residual strength, S(t), is the linear curve shown in Figure 42.

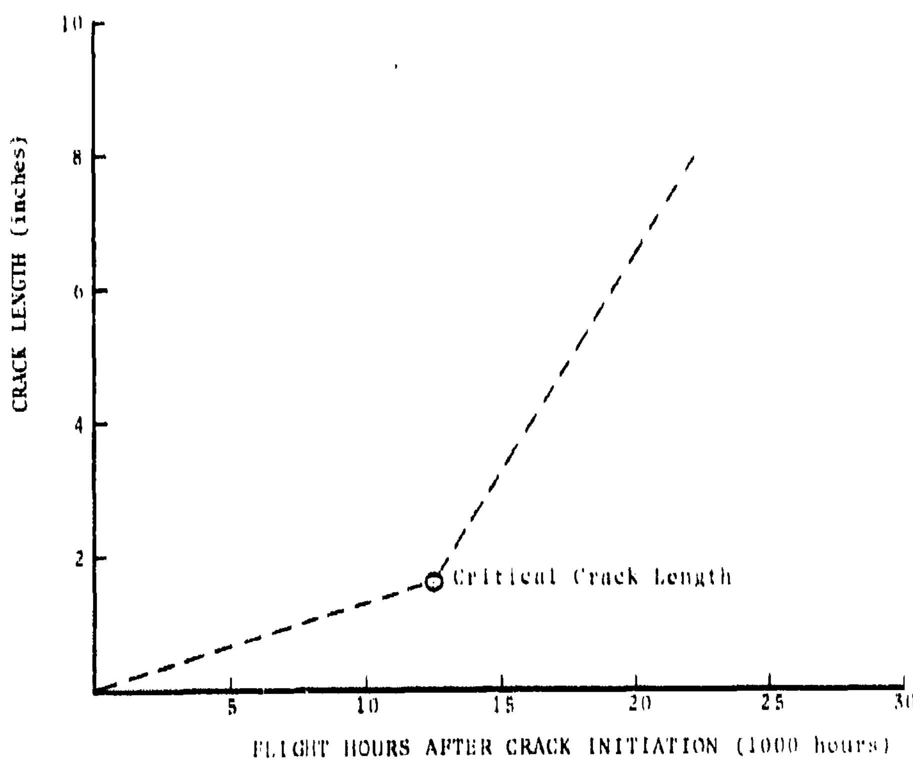


Figure 38. Linearized Crack Length vs. Time Relation Used in SAIFE

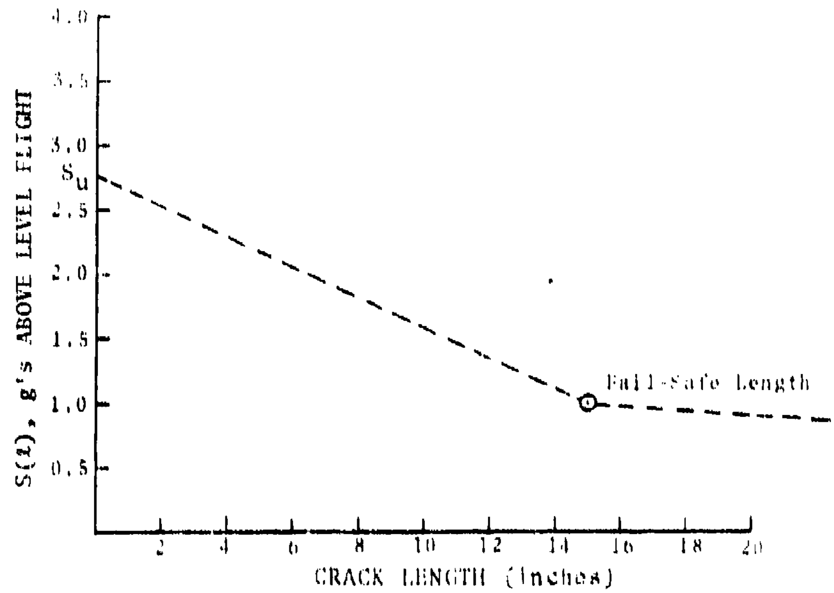


Figure 39. Linearized Strength vs. Crack Length Relation Used in SAIFE

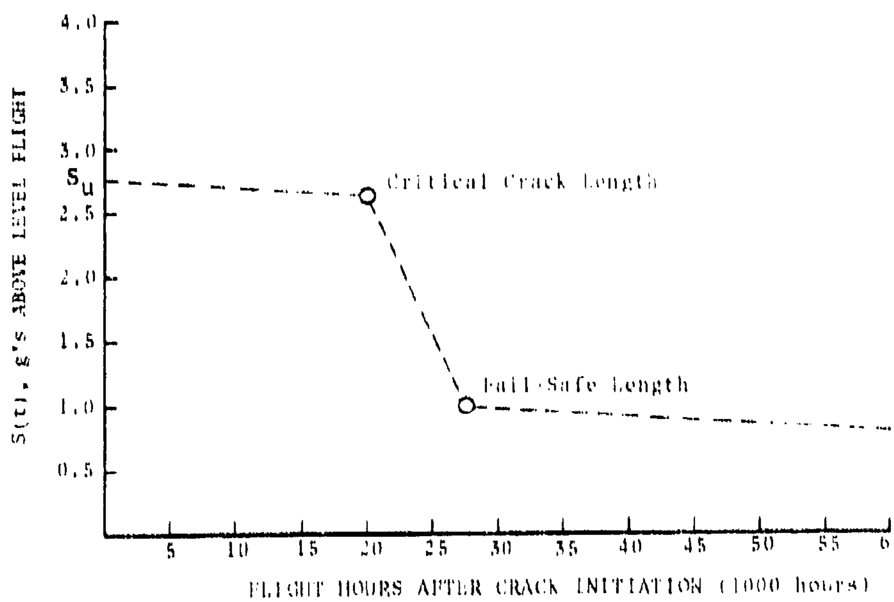


Figure 40. Resulting Linearized Strength vs. Time Relation

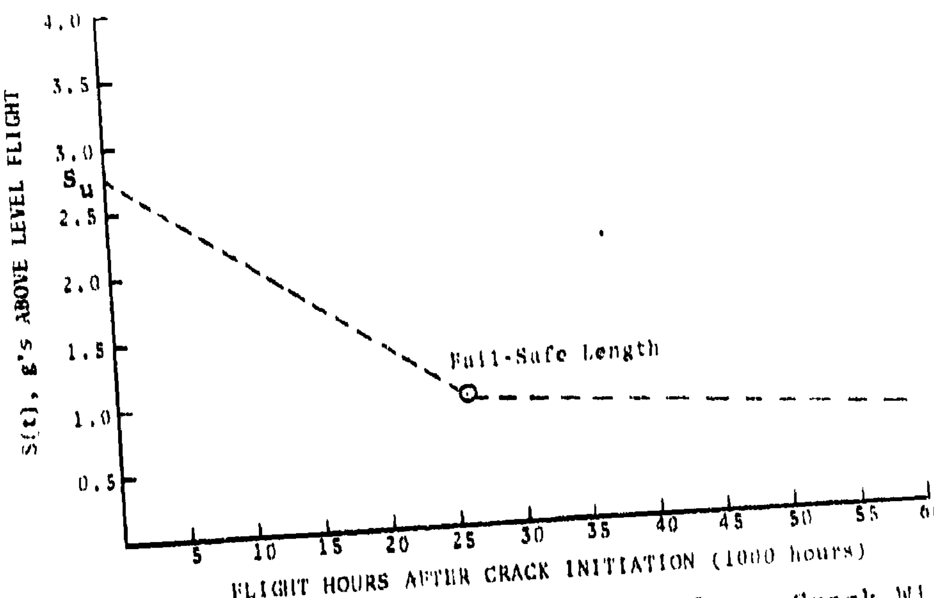


Figure 41. Strength vs. Time Relation for a Crack With Unchanging Growth Rate

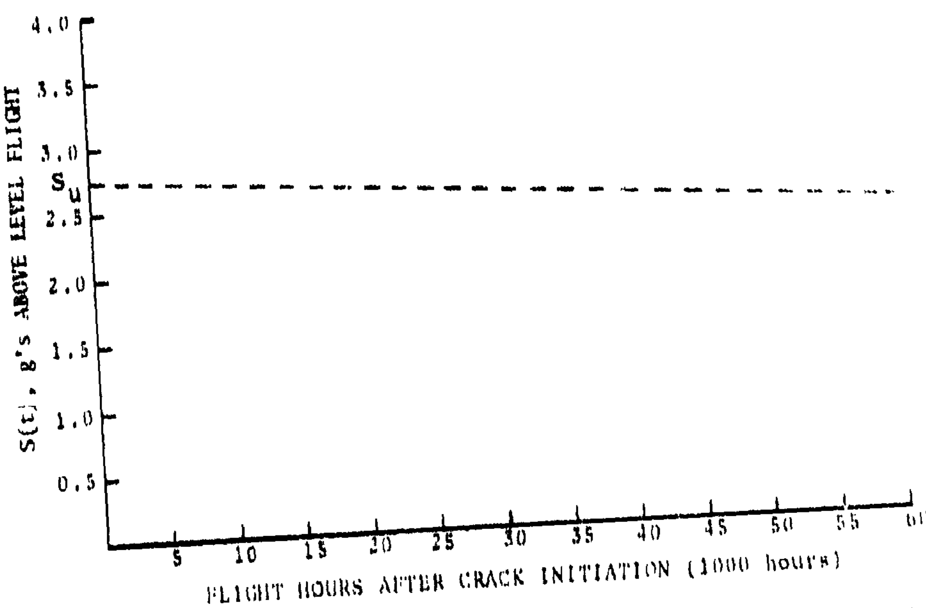


Figure 42. Strength vs. Time Relation for a Crack Failing to Reach Critical Length Before Aircraft Retirement

If there is a second crack initiation, a two-part approximation to the growth rate of the sum of the two crack lengths is calculated. The slow growth portion of the summed crack approximation is defined by two points: The first point is the length of the first crack when the second crack initiates, and the second point is the sum of the crack lengths when the first crack reaches critical crack length. The fast growth portion of the approximation is determined by a least-squares fit of the sum of the crack lengths when the second crack initiates, when the second crack reaches critical length, and when the aircraft is retired. A new S(t) curve is now determined as before, except that t=0 now represents the time of the second crack initiation. S(t=0) is now equal to S_u minus the loss of residual strength because of the first crack growth before the second crack initiation. If a third crack initiates, the same procedure is repeated.

4.6 Load Exceeds Reduced Strength

This function involves a probabilistic determination of the maximum flight load experienced by an airplane and the comparison of this load with the strength of the elements to project time to failure after a crack initiation. The data presented in Reference 17 was used to define the distribution of positive and negative normal accelerations experienced by two aircraft equipped with NASA VGH recorders, which provide continuous time-history records of indicated airspeed, normal acceleration, and pressure altitude. The data were collected over a 2-year period on two identical four-engine turbojet transport airplanes during routine commercial operations of a single airline. The data covered flights mostly over the eastern half of the Continental United States and a few to the West Coast and to northern South America. The data consisted of 3766 flight hours of operational maneuver and gust accelerations and 219.7 flight hours of check-flight maneuver accelerations. These data compared closely with those for another type of four-engine turbojet transport. The operational maneuver, operational gust, and check-flight maneuver accelerations, both positive and negative, were combined, and the exceedances per flight hour were calculated for each deviation from level flight in 0.1g increments. A least-squares curve-fit computer program was then used to fit an exponential curve to the exceedances. The equation for the exponential curve is as follows:

$$P(S_a) = A \exp[bS_a] \qquad (25)$$

where $P(S_a)$ is the number of flight loads per hour which exceed the load level S_a, and A and b are input parameters. A plot of the observed exceedances is presented in Figure 43. The least-squares curve fit is adjusted to give the closest fit at the high "g" load portion since this is where the element failures are most likely to occur.

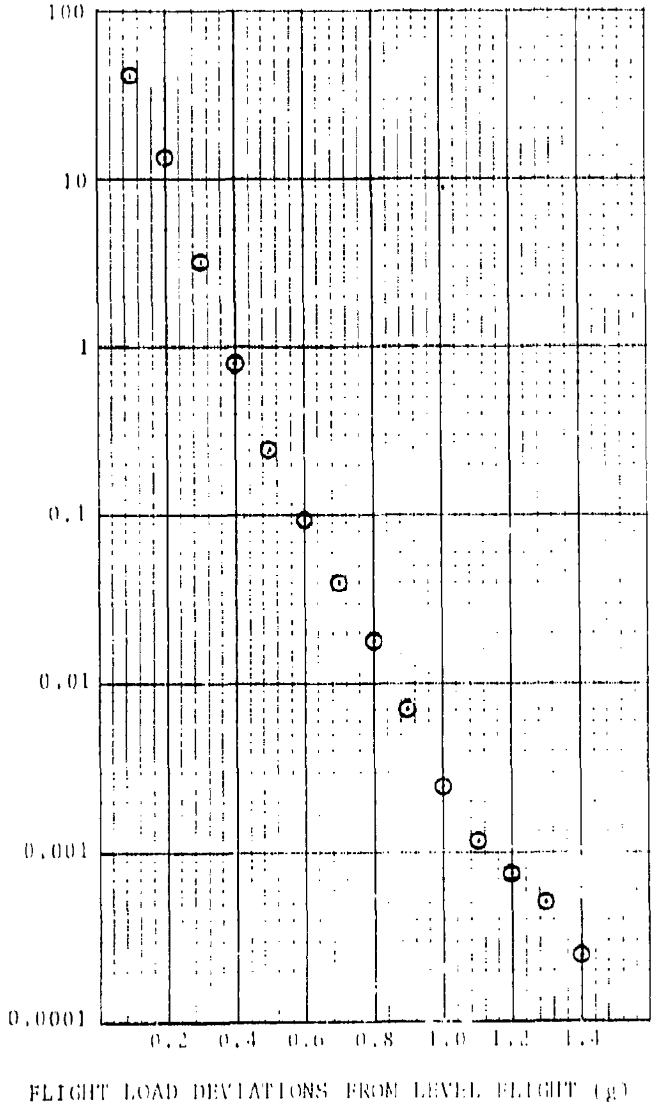


Figure 43. Flight Load Exceedances

Fatigue failure of the structure occurs when the flight load exceeds the residual strength. The equations defining residual strength, S(t), as illustrated in Figure 40, are

$$
\left.
\begin{aligned}
S(t) &= S_u - R_1 t, & t &\le t_1 \\
S(t) &= S_1 - R_2(t - t_1), & t_1 &< t \le t_2 \\
S(t) &= S_f - R_3(t - t_2), & t &> t_2
\end{aligned}
\right\} \qquad (26)
$$

where t_1 = time at which crack reaches critical length

t_2 = time at which crack reaches fail-safe length

R_1 = strength degradation rate for $t \le t_1$

R_2 = strength degradation rate for $t_1 < t \le t_2$

R_3 = strength degradation rate for $t > t_2$

S_u = ultimate strength

$S_1 = S_u - R_1 t_1$

S_f = fail-safe strength

With S(t) as expressed above, the number of flight loads per hour which exceed the residual strength at time t is

$$
\left.
\begin{aligned}
P\,[S(t)] &= \Lambda \exp\left[b(S_u - R_1 t)\right], & t &\le t_1 \\
P\,[S(t)] &= \Lambda \exp\left[b\, S_1 - R_2(t - t_1)\right], & t_1 &< t \le t_2 \\
P\,[S(t)] &= \Lambda \exp\left[b\, S_f - R_3(t - t_2)\right], & t &> t_2
\end{aligned}
\right\} \qquad (27)
$$

Adopting the same assumptions as Lundberg and Eggwertz (Reference 18), the above expressions for the residual strength exceedance rate can be substituted for the risk function, $\lambda(t)$, in the reliability formula. Recall from Volume I the reliability formula

$$
F(t) = 1 - \exp\left[- \int_0^t \lambda(t)\, dt\right] \qquad (28)
$$

Making the above substitutions yields

$$
F(t) = 1 - \exp\left\{ - \int_0^t \Lambda \exp\left[bS_u - bR_1 t\right] dt\right\}, \quad t \le t_1 \qquad (29)
$$

$$F(t) = 1 - \exp\left\{ - \int_0^{t_1} A \exp\left[bS_u - bR_1 t \right] dt \right.$$

$$\left. - \int_{t_1}^{t} A \exp\left[bS_1 - bR_2(t - t_1) \right] dt \right\}, \qquad (30)$$

$$t_1 < t \leq t_2$$

$$F(t) = 1 - \exp\left\{ - \int_0^{t_1} A \exp\left[bS_u - bR_1 t \right] dt \right.$$

$$- \int_{t_1}^{t_2} A \exp\left[bS_1 - bR_2(t - t_1) \right] dt \qquad (31)$$

$$\left. - \int_{t_2}^{t} A \exp\left[bS_f - bR_3(t - t_2) \right] dt \right\}, \quad t > t_2$$

where $F(t)$ is the probability of failure before time t of an aircraft structure which had a crack initiation at time $t=0$.

To calculate the predicted times to failure from uniformly distributed random numbers, equate the random number RN to $F(t)$ and solve for t. For $F(t)$ as expressed above, and setting RN equal to $1 - F(t)$ for convenience, solving for t yields

$$t = -\frac{1}{bR_1} \ell n \left\{ \frac{bR_1 \ \ell n \ (RN)}{A \exp \ (bS_u)} + 1 \right\}, \qquad t \leq t_1 \qquad (32)$$

$$t = -\frac{1}{bR_2} \ell n \left\{ \frac{bR_2}{A \exp \ (bS_1 + bR_2 t_1)} \left[\ell n (RN) + \frac{A \exp \ (bS_1)}{bR_2} \right. \right.$$

$$\left. \left. - \frac{A \exp \ (bS_u)}{bR_1} \left(\exp \ (-bR_1 t_1) - 1 \right) \right] \right\}, \quad t_1 < t \leq t_2 \qquad (33)$$

$$t = -\frac{1}{bR_3} \ln \left| \frac{bR_3}{A \exp (bS_f + bR_3 t_2)} \left[\ln(RN) + \frac{A \exp (bS_f)}{bR_3} \right. \right.$$

$$- \frac{A \exp (bS_1)}{bR_2} \left(\exp (bR_2 t_1 - bR_2 t_2) - 1 \right)$$

$$\left. \left. - \frac{A \exp (bS_u)}{bR_1} \left(\exp (-bR_1 t_1) - 1 \right) \right] \right| , \quad t > t_2 \qquad (34)$$

When there is a second crack initiation, a new time to failure is calculated with t = 0 now corresponding to the time of second crack initiation and S_u now equaling the ultimate strength minus the loss of residual strength because of the first crack growth. The same procedure is repeated if there is a third crack initiation.

5. Detailed Description of Block 5.0, Periodic Inspection of Elements/Aircraft Deleted from Fleet (see Figure 44)

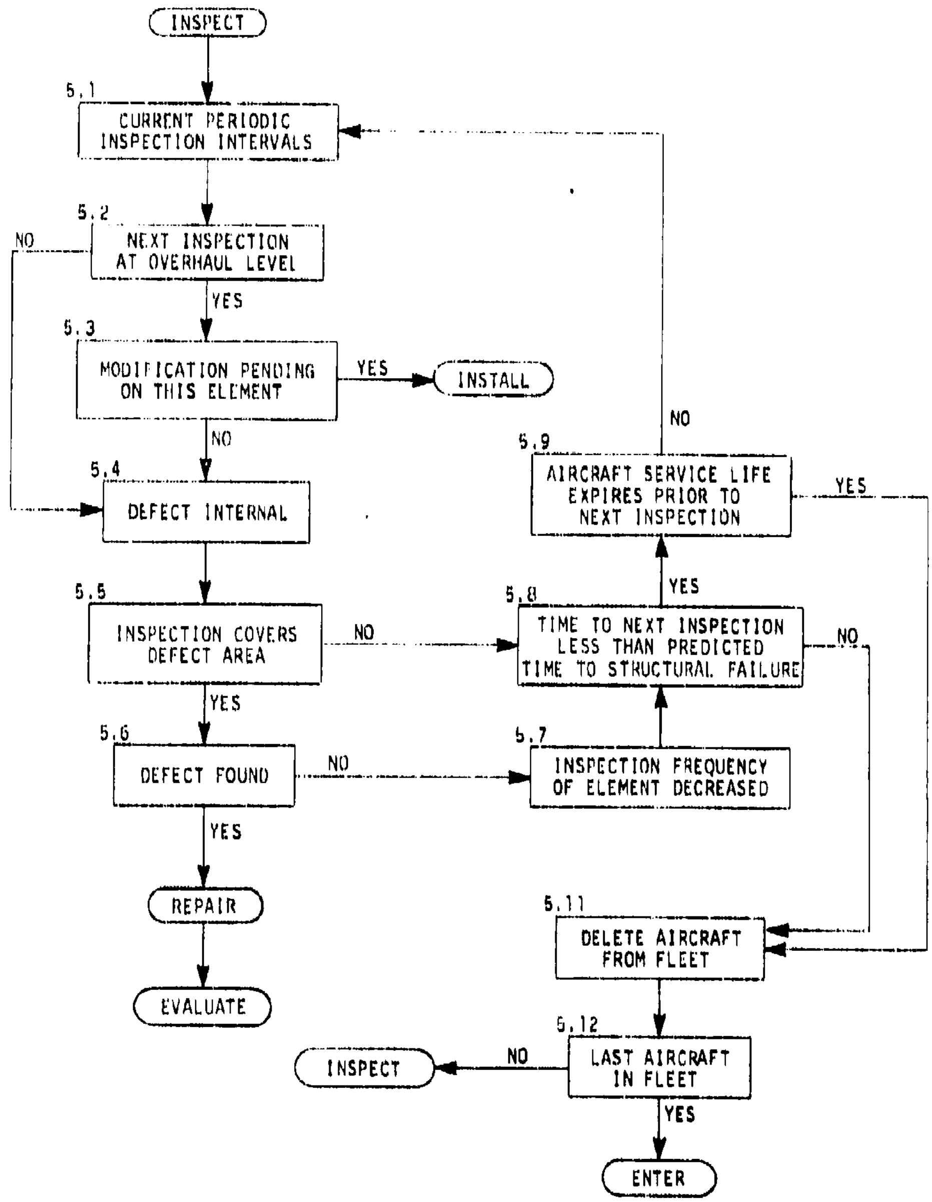


Figure 44. Detailed Flow Diagram - Periodic Inspection of Elements/Aircraft Deleted from Fleet

5.1 Current Periodic Inspection Intervals

The initial inspection intervals are input parameters. The inspection intervals can be those recommended by the Maintenance Review Board (MRB) or those submitted by an air carrier as part of the Standard Operations Specification. The simulation is designed to accept a standard ATA four-level inspection program. The four levels, A through D, are defined as follows:

A Preflight: Visual inspection conducted from ground level and primarily covering lower exterior wing and fuselage surfaces.

B Service: Close visual inspection of bottom of wing, lower fuselage, top of wing, and known problem areas. This also may include the front of the forward spar and rear of the aft spar in readily accessible areas.

C Phase: Close visual inspection of aircraft exterior and easily accessible interior areas, such as baggage compartments and door frames. NDI of selected areas of the aircraft.

D Overhaul: Detailed inspection of entire aircraft. This level may be conducted both on a sampling basis and during several separate inspections.

It is assumed that each higher-level inspection includes all the lower-level inspections down to the lowest level specified for a particular element type. Therefore, if a D-level inspection is being conducted and the lowest interval specified for that element was a B-level, then the current inspection would include the B-level, C-level, and D-level inspections.

Sampling inspections are accounted for in the D-level inspection logic by reducing the probability of defect detection. The amount of reduction is directly proportional to the size of the sample inspected. Each time an aircraft is inspected at the D-level, the probability of defect detection is multiplied by the fractional size of the sample. This decimal fraction is an input parameter.

The logic that follows in Block 7.1 automatically increases the frequency of inspection at certain levels depending upon the extent of the defects being found. The amount of increase is an input function. These changes in frequency are accounted for in this function, and the new intervals are used to schedule subsequent inspections.

5.2 Next Inspection at Overhaul Level

This function is a check of the inspection level being scheduled. It is present because the installation of modifications is normally scheduled only during overhaul inspections, and the modification installation takes precedence over the inspection process.

5.3 Modification Pending on This Element

Once the decision is made to develop a modification, it generally takes 3 to 6 months to design the modification, procure materials and/or parts, and set up the tooling required for installation. These lead times are input parameters that depend on the element being considered.

Because SAIFE processes time in flight hours, calendar days or months must be converted to the equivalent simulation time by Equation (35) or (36)

$$\text{Calendar Days} * 8.2 = \text{equivalent simulation time} \quad (35)$$

$$\text{Calendar Months} * 250 = \text{equivalent simulation time} \quad (36)$$

If the modification lead time requirements are satisfied, the logic progresses to the installation routine, and it is assumed that existing defects are repaired during the modification process. If the lead time requirements have not been satisfied, or if no decision has been made to modify the element, the logic continues through the inspection subroutine.

5.4 Defect Internal or External

The inspection level at which a defect will be detected depends partly on whether the defect is internal or external. Assuming that the occurrence of cracks or corrosion is a statistical trial and that each trial is independent of all other trials, then the probability of the corrosion being external in a single occurrence is simply the number of external occurrences divided by the total number of occurrences. Based on the MRR/SDR data, the probability of corrosion being external on each element type is summarized in Table 5.

To evaluate this function during the simulation, a random number from a uniform distribution is selected for each corrosion occurrence. If the random number is between zero and the probability established for the subject element, the corrosion is treated as being external; otherwise, it is treated as being internal.

TABLE 5. PROBABILITY OF A CORROSION DEFECT BEING EXTERNAL

	Total Occurrences	External Occurrences	Probability of Corrosion Being External
Fuselage			
Door frame	13	8	0.615
Floor beam	17	6	0.352
Keel beam	5	0	0.000
Main frame	22	2	0.091
Stringer	257	184	0.716
Pressure web	0	0	0.050*
Window frame	3	3	1.000
Wing			
Access frame	0	0	0.615*
Rib	3	2	0.667
Spar	57	10	0.175
Stringer	43	25	0.581
Wing Center Section			
Rib	0	0	0.175*
Spanwise beam	18	6	0.333
Stringer	92	32	0.348

* estimated

Crack defects are treated in the same manner as corrosion defects, with the probability of a crack being external, as summarized in Table 6.

TABLE 6. PROBABILITY OF A CRACK DEFECT BEING EXTERNAL

	Total Occurrences	External Occurrences	Probability of Crack Being External
Fuselage			
Door frame	82	19	0.232
Floor beam	90	22	0.244
Keel beam	40	2	0.050
Main frame	735	29	0.039
Pressure web	90	1	0.011
Window frame	50	28	0.560
Stringer	750	244	0.325
Wing			
Access frame	115	74	0.643
Rib	284	30	0.106
Spar	667	364	0.546
Stringer	516	397	0.769
Wing Center Section			
Rib	2	0	0.000
Stringer	202	132	0.653
Spanwise beam	118	63	0.534

As was the case for corrosion, this function is evaluated by selecting a random number from a uniform distribution for each crack occurrence. If the random number is between zero and the probability established for the subject element, the crack is treated as being external; otherwise, it is treated as being internal. However, in the case of cracks, SAIFE has the facility for cracks which initiate internally to become external. In general when a crack becomes external, there is a higher probability of it being detected since the lowest level external inspection is normally lower than the lowest level internal inspection. The point at which an internal crack becomes external is defined as a percentage of the critical crack length. This percentage is an input parameter.

5.5 Inspection Covers Defect Area

This function requires two program inputs: one input to specify the most frequent inspection interval that covers the internal portion of the subject element, and a second input to specify the most frequent inspection interval that covers the external portion of the subject element. It is assumed that once the most frequent interval is specified, the element is covered during all higher-level inspections. The simulation is set up to work with a four-level inspection program. The levels are those commonly referred to by the ATA as A, Preflight; B, Service; C, Phase; and D, Overhaul. The frequency of each level is also an input parameter, which can be varied to reflect the inspection program of different air carriers.

5.6 Defect Found

5.6.1 Crack Defect Found

The probability of detecting a crack depends on the defect size and the inspection level. The basic form of the probability equation is defined by Davidson in Reference 19. The equation form is an exponential with three constants: C_1, β_1, and ℓ_0, and one variable, ℓ. The probability $P(\ell)$ for each inspection level is defined as follows:

$$P(\ell) = \begin{cases} 0 & \text{for } \ell \leq \ell_0 \\ C_1\{1-\exp[-\beta_1(\ell-\ell_0)]\} & \text{for } \ell > \ell_0 \end{cases} \qquad (37)$$

where

C_1 = The maximum probability of detecting a crack at a given inspection level. If, for instance, C_1 = 0.9, then 1 out of 10 cracks would be missed regardless of its length. C_1 approaches 1.0 for overhaul and special inspections, but is significantly less than 1.0 for lower level inspections. This parameter is determined from MRR/SDR data by comparing the number of defects that were found at a particular inspection level with the number of defects that should have been found, but were missed. The number of defects that were missed was determined by the number of cracks that were found on higher-level inspections but could have been detected at the lower level because of their size and location. The specific defect size and location for each inspection level was determined from the Inspector's Survey and from the inspection level definition. The criteria for each inspection level are as follows:

Preflight - all cracks larger than 0.551 inch and located on the lower, exterior surfaces of the wing and fuselage could have been detected at this level.

Service - all cracks larger than 0.410 inch and located on the exterior of the upper and lower wing surfaces, the lower fuselage surface, and the fuselage near the wing root could have been detected at this level.

Phase - all cracks larger than 0.266 inch and located on the wing and fuselage exterior surfaces, cargo compartment, or accessible areas of the wing interior could have been detected at this level.

Overhaul - all cracks larger than 0.144 inch in all locations could have been detected at this level.

Because of the very few MRR/SDR's that contained information on corrosion size, it was assumed that C_1 for corrosion was equal to C_1 for cracks at the same level of inspection.

β_1 = The shape parameter that determines how quickly the probability of detection approaches the maximum probability, C_1. β_1 is also determined from MRR/SDR data by plotting the cumulative distribution of defects detected at a given inspection level.

ℓ_o = The smallest defect that will be detected. This parameter was determined from the data in Volume III, the Inspector's Survey, by taking the logarithm of the responses at a given inspection level and computing the mean value. The mean of the logarithms was used because a plot of the responses indicated that they closely followed a log-normal distribution.

The result of this development is four equations: one for each periodic inspection level. The equations are as follows:

Preflight: A Level

$$P(\ell) = 0.273\{1.0 - \exp[-0.312(\ell - 0.551)]\} \qquad (38)$$

Service: B Level

$$P(\ell) = 0.462\{1.0 - \exp[-0.513(\ell - 0.410)]\} \qquad (39)$$

Phase: C Level

$$P(\ell) = 0.643\{1.0 - \exp[-0.686(\ell - 0.266)]\} \qquad (40)$$

Overhaul: D Level

$$P(\ell) = 0.990\{1.0 - \exp[-0.725(\ell - 0.144)]\} \qquad (41)$$

These equations are illustrated in Figure 45.

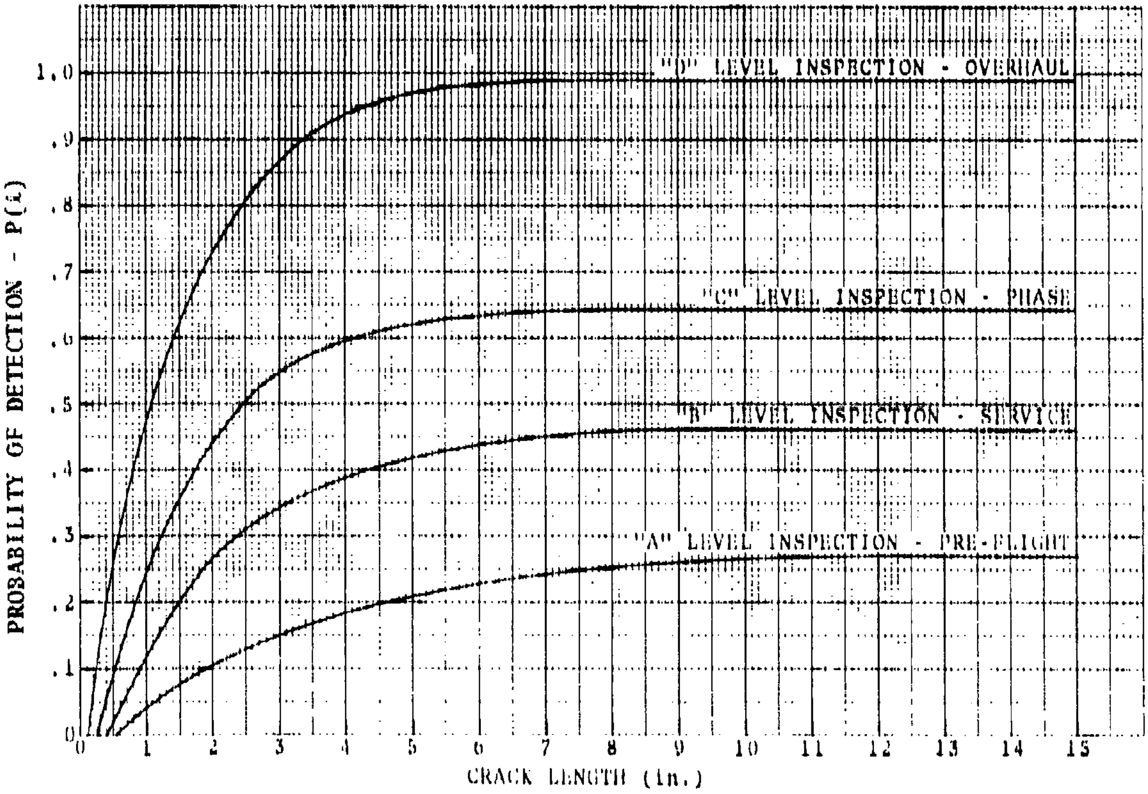


Figure 45. Probability of Crack Detection During a Periodic Inspection

Having determined the probability of detection for a given defect size, the determination of whether or not a defect was found is based on Monte Carlo methods. A random number is generated from a uniform distribution and if that number is less than or equal to the probability of detection, the defect is considered to be detected; otherwise, it is considered to be undetected.

If the defect is detected, the element is repaired or modified depending on previous simulation decisions. If the defect remains undetected, the simulation continues to conduct inspections until the defect is detected, a structural failure occurs, or the aircraft is retired from the fleet.

5.6.2 Corrosion Defect Found

The probability of detecting a corrosion defect is determined similarly as the probability of detecting a crack defect. The basic equation is the same, except that the corrosion area, a, is substituted for the crack length, ℓ, and the minimum detectable area, a_0, is substituted for the minimum crack length, ℓ_0. The equations for the probability of detection of corrosion then become the following:

Preflight: A Level

$$P(a) = 0.273\{1.0 - \exp[-0.312(a - 0.996)]\} \quad (42)$$

Service: B Level

$$P(a) = 0.462\{1.0 - \exp[-0.513(a - 0.635)]\} \quad (43)$$

Phase: C Level

$$P(a) = 0.643\{1.0 - \exp[-0.686(a - 0.565)]\} \quad (44)$$

Overhaul: D Level

$$P(a) = 0.990\{1.0 - \exp[-0.725(a - 0.351)]\} \quad (45)$$

These equations are illustrated in Figure 46.

Again, a Monte Carlo method, using a uniformly distributed random number, is employed to determine whether a defect is actually found or not.

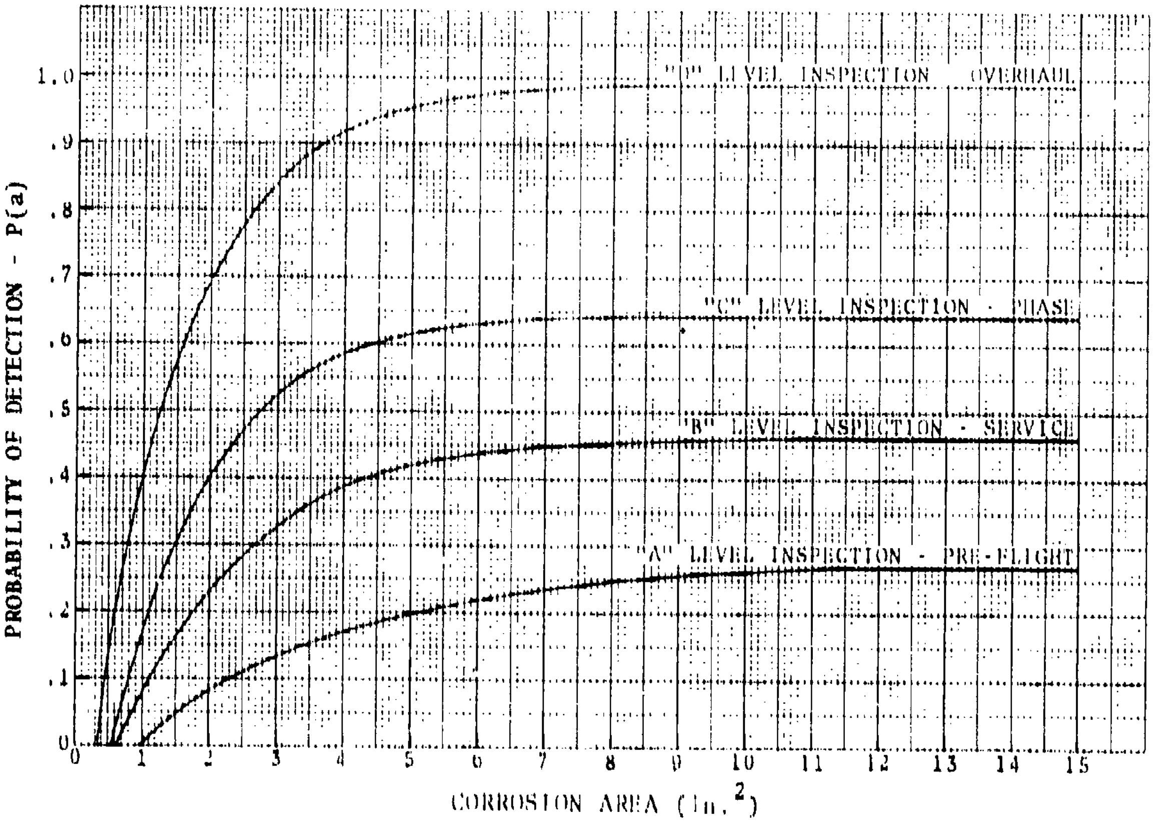


Figure 46. Probability of Corrosion Detection During a Periodic Inspection

5.7 Inspection Frequency of Element Decreased

Inspection intervals are normally extended in structural areas where few defects have been found. As the overhaul and phase inspections are conducted on each of the ten high-time aircraft in the fleet, the time of detection and the number of defects detected are recorded. If no defects are found on any of the ten high-time aircraft during one D-level interval, then the overhaul inspection interval is extended. The amount of decrease depends on the particular element and is, therefore, an input parameter.

Inspection interval extensions apply only to the phase and overhaul inspections.

5.8 Time to Next Inspection Less Than Time to Element Failure

This function compares the time to structural failure with the time of the next periodic inspection. If the time to the next inspection occurs first and the aircraft is not scheduled for retirement, then the inspection routine is repeated. If the time to failure occurs first, the logic continues to Block 5.10.

5.9 Aircraft Service Life Expires Prior to Next Inspection

This function simply compares the service life of the aircraft, as stated by the manufacturer or as determined by the user from service experience, with the number of flight hours that the aircraft has accumulated and the number of flight hours until the next periodic inspection is scheduled. If the accumulated flight hours plus the hours to the next inspection are equal to or greater than the aircraft service life, then the aircraft is retired from the fleet. If, however, the service life has not expired, the logic returns to Block 5.1 and the inspection routine is repeated until a defect is found, a structural failure occurs, or the aircraft is retired from service.

5.10 Delete Aircraft From Fleet

Aircraft are deleted from the fleet as their service life expires or as they experience structural failures. The aircraft service life is an input parameter that can be varied to determine the effect on safety of extending the service life with or without improving the design requirements.

5.11 Last Aircraft in Fleet

This function simply keeps track of the number of aircraft deleted from the fleet and compares that number with the number of aircraft originally produced. When the last aircraft has been deleted, SAIFE returns to Block 1.0 and repeats the simulation with the next element.

6. Detailed Description of Block 6.0, Repair Element to Original Strength (see Figure 47)

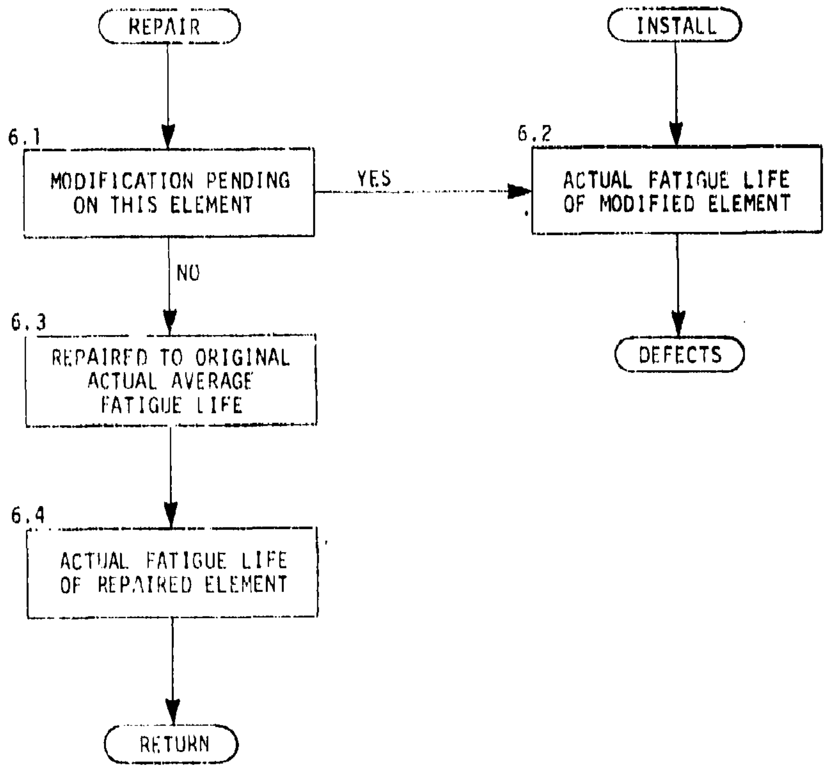


Figure 47. Detailed Flow Diagram - Repair Element to Original Strength

6.1 Modification Pending

As discussed in Section 5.3, if there is a structural modification pending on a particular aircraft, the SAIFE logic represents the installation of the modification as occurring during either an overhaul inspection or the repair of an in-service defect. If there are future in-service defects projected for the aircraft at the time of installation, these projections are cancelled and new ones are generated from their corresponding distributions as though the aircraft were just entering service. If the inspection intervals had been previously decreased, they are returned to their initial values as each aircraft is modified.

6.2 Actual Fatigue Life of Modified Element

When a structural modification is installed on an aircraft, the actual fatigue life of the modified element is determined in the same manner as when the aircraft entered service. The actual life is drawn from a statistical distribution of fatigue lives about the actual average fatigue life of the element type. If the modification has not been fatigue tested, the actual average fatigue life of the element type is statistically determined as it was at the start of the simulation in Block 1.4. If the modification has been fatigue tested, the actual average fatigue life of the element type is assumed to be the same as that predicted by analysis.

6.3 Repaired to Original Actual Average Fatigue Life

When it has been determined that a repair is required and if there is no modification pending on the element, the element is repaired before any more flight hours are allowed to accumulate on the aircraft. All defects present at the time of repair, whether detected or not during the scheduled inspection, are assumed to be repaired at this time. It is also assumed that the element strength is restored to its original static strength. However, in-service defects (fatigue cracks, service damage, and corrosion) predicted to occur after the repair is accomplished are not affected by the repair process and they are allowed to occur at their originally determined times.

6.4 Actual Fatigue Life of Repair Element

As discussed in Section 6.3, previously projected fatigue cracks that have not been initiated at the time of repair are unaffected and retain their original initiation times. Those cracks that are repaired have new times to crack initiation determined in the same manner as when the aircraft entered service; that is, from a fatigue life distribution reflecting basic fatigue scatter and load environment variation, times are randomly drawn about the element average fatigue life.

7. <u>Detailed Description of Block 7.0, Special Inspection and Increased Inspection Frequency (see Figure 48)</u>

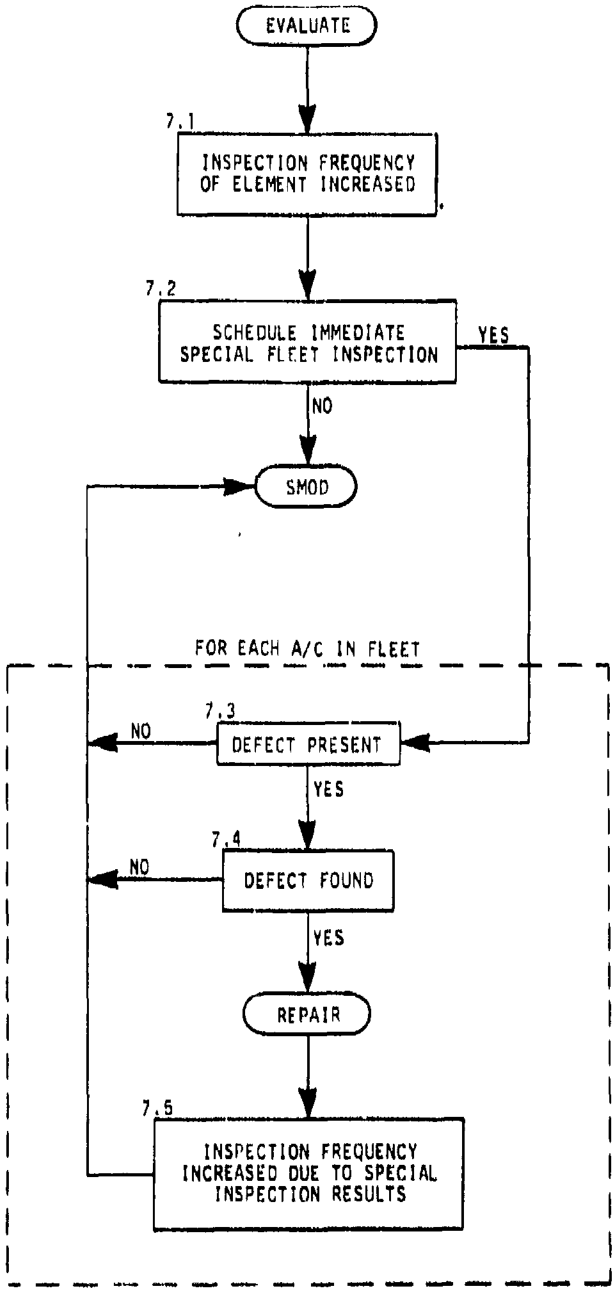


Figure 48. Detailed Flow Diagram - Special Inspection and Increase Inspection Frequency

7.1 Inspection Frequency of Element Design Increased

The frequency at which certain inspections are conducted is increased when it has been determined that the present frequency is not adequate. The percentage increase in frequency is an input parameter that depends on the safety criticality of the element. Three criteria are used to determine whether a frequency increase is necessary. If any one of the three are satisfied, the frequency is increased. The three criteria are as follows:

(1) A crack greater than fail-safe length is detected.

(2) Large cracks are detected in an individual element, such that the sum of the lengths of the cracks present plus the projected growth of the largest crack through the next inspection interval results in a one-half reduction in the fail-safe strength.

(3) Small cracks are detected in the same element on numerous aircraft*, such that the total strength reduction resulting from all of the cracks divided by the number of aircraft in the fleet equals 20 percent of the original fail-safe strength of an individual element.

The first two criteria deal with defects that very seldom occur, but when they do there is a high probability of an aircraft accident or of extensive unscheduled maintenance.

The third criterion deals with the potential safety hazard resulting from the very small but finite possibility of the occurrence of a flight load that exceeds the design strength along with the greatly increased probability that the strength of the element has been slightly reduced because of a small crack.

7.2 Schedule Immediate Special Fleet Inspection

The criteria defined in Section 7.1 are also used to determine whether a special fleet-wide inspection is required. When a special inspection is called for, the subject element is carefully inspected on every aircraft in the fleet. If a special inspection is not called for, the logic continues to Block 8.0 where it is determined whether or not a modification to the element is required.

7.3 Defect Present

This function is a simple yes-or-no indicator that determines whether a crack defect has initiated on the element being covered by the special inspection. If no cracks are present, the logic continues immediately to Block 8.0.

7.4 Defect Found

During a special inspection, the probability of finding a defect is significantly improved over the probability of finding a defect during a regularly scheduled inspection. This is due to the fact that the location and nature of the defect are reasonably well specified before the inspection is conducted. The probability of detecting a crack during a special inspection is determined from the equation

$$P(\ell) = 0.999\{1.0 - \exp[-0.971(\ell - 0.102)]\} \tag{46}$$

And the probability of detecting corrosion is

$$P(a) = 0.999\{1.0 - \exp[-0.971(a - 0.180)]\} \tag{47}$$

where

ℓ = length of the crack present
a = area of corrosion present

As was the case for the probability of detection curves in Section 5.6, these equations were determined from the defects reported in the MRR/SDR data and the Survey of Inspectors; these equations are illustrated in Figures 49 and 50. The MRR/SDR data are presented in Volume III of this report. The defect is considered to be found when a random number generated by the simulation is equal to or less than the probability generated from the above equations for cracks or corrosion.

7.5 Inspection Frequency Increased Because of Special Inspection Results

After a special fleet inspection is completed, the magnitude of the defects found is compared with the first and second criteria defined in Section 7.1, and a second reduction in inspection frequency may be instituted as a result of the special inspection. For the inspection interval required by the second criterion in Section 7.1, the interval most recently set by Block 7.1 is used.

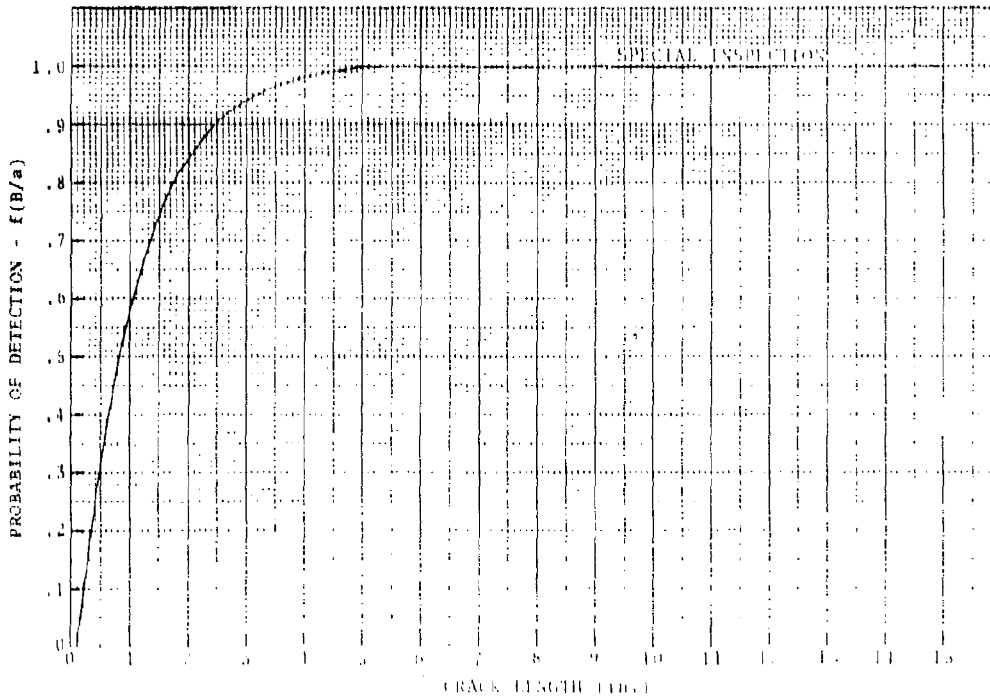


Figure 49. Probability of Crack Detection During a Special Inspection

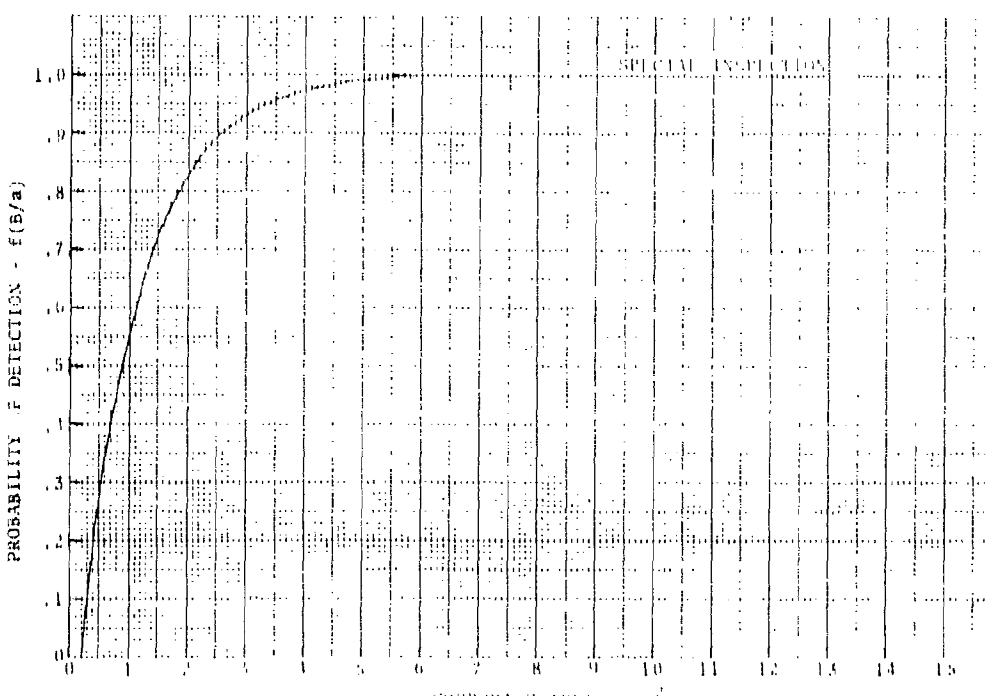


Figure 50. Probability of Corrosion Detection During a Special Inspection

8. <u>Detailed Description of Block 8.0, Develop Modifications Because of Service Experience (see Figure 51)</u>

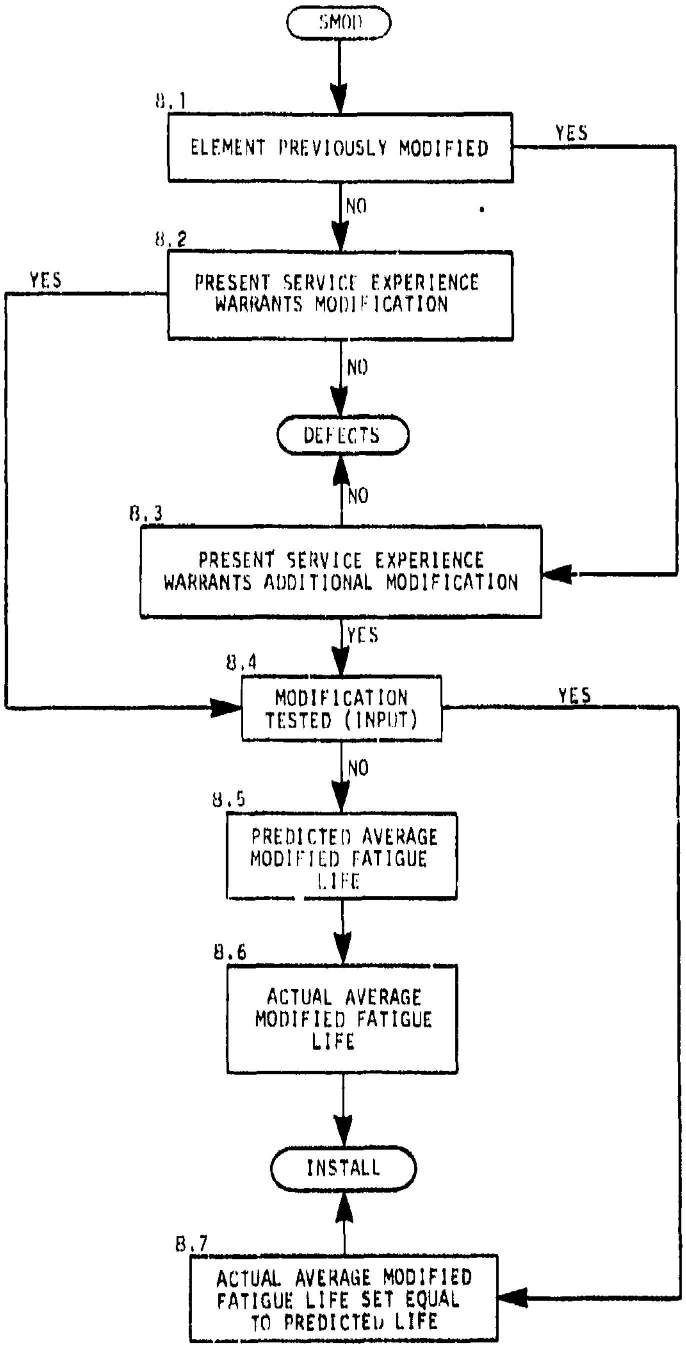


Figure 51. Detailed Flow Diagram - Develop Modification Because of Service Experience

8.1 Element Previously Modified

As a simple variable, this function indicates whether an element was previously modified. If the element was not modified, Block 8.2 examines present service experience and makes a decision on the development of a first modification. If the element was previously modified, Block 8.3 examines present service experience and makes a decision on the development of an additional modification.

8.2 Present Service Experience Warrants Development of a Modification

The SAIFE logic bases the decision on whether or not to develop a structural modification because of service experience solely on economic considerations. Since the economic parameters considered depend on element type and are subject to change with time, they are necessarily part of the input data. Values for most of these parameters are related to the inspection level at which the maintenance action is performed. This relationship is required because the complexity of the elements generally increases with higher inspection levels. The economic parameters identified and the values currently used are listed in Table 7. The values shown in Table 7 were determined from a report (Reference 20) given at the ATA Maintenance Conference held September 26 to 28, 1968. The estimated values are based on the relative complexity of each inspection level.

TABLE 7. ECONOMIC PARAMETERS REQUIRED FOR DEVELOPING A SERVICE MODIFICATION

	Inspection Level			
	A	B	C	D
Man-Hrs/Insp.	24*	472*	861*	1250
NDI Cost/Insp.	-	-	$648*	$941*
Man-Hrs/Repair Task	2*	4.8	8.8*	12.7
Material Cost/Repair Task	$3*	$6*	$12*	$18
Man-Hrs/Mod Task	-	-	-	683
Material Cost/Mod Task	-			$1339
Labor + Overhead Rate	$15/Hr at all levels			

* estimated values

The decision to develop a modification is made by comparing the cost per flight hour of the modification with the repair cost per flight plus the increased inspection cost per flight hour. The modification cost per flight hour is found by dividing the total fleet modification cost by the remaining service life of the fleet. The repair cost per flight hour is found by dividing the total fleet repair costs since the last modification by the fleet flight time since the last modification. The increased inspection cost per flight hour is found by dividing the projected increased inspection costs by the remaining service life of the fleet. A modification is justified when

$$C_{MOD} < C_{REPAIR} + C_{INSP} \tag{48}$$

where C_{MOD} = modification cost per flight hour

C_{REPAIR} = repair cost per flight hour

C_{INSP} = increased inspection cost per flight hour

8.3 Present Service Experience Warrants Development of Additional Modification

The decision to develop an additional modification because of service experience is based on the same economic parameters considered in Block 8.2. Currently the economic parameter values used are the same as those for the first modification decision; these values are listed in Table 7.

8.4 Modification Tested

This function indicates whether or not a modification has been tested before its incorporation into the fleet. It is an input parameter that is determined for each element.

8.5 Predicted Average Modified Fatigue Life

If a decision has been made to develop a modification, it is assumed that the modification is again designed to the predicted average fatigue life of the original design as defined in Block 1.3.

8.6 Actual Average Modified Fatigue Life

If the modification is not fatigue tested, it is subject to the same type of variation between actual and predicted fatigue life as was the original design. To determine the actual average fatigue life of the modification, a random draw is once more made from the log-normally distributed correction factor described in Section 1.5. Although the form of the correction factor distribution is the same as it was for the original design, it is assumed that the accuracy of the modification design is improved. This increased accuracy is accounted for by decreasing the standard deviation and increasing the mean in the above distribution. The standard deviation is reduced by 15% and the mean is increased by the quantity 0.15 (1.0-mean).

8.7 Actual Average Modified Fatigue Life Set Equal to Predicted Life

If the decision was made to fatigue test the modification, it is assumed that the actual average fatigue life of the modification will attain its predicted life or be redesigned and retested until it does.

III. SUMMARY

The implementation of the SAIFE logic, as described in Section II, required incorporating decision functions to realistically simulate the evaluation process which establishes or modifies the structural inspection intervals for commercial jet transport aircraft. Some of these decision functions were based on previous research as referenced in this volume, and others were developed during the current study. A partial list of the latter include the following:

(1) Probability of a production defect occurring.

(2) Service damage occurrence rates.

(3) Corrosion occurrence rates.

(4) Probability of corrosion occurring in a stress concentration.

(5) Probability of crack detection during a periodic or a special inspection.

(6) Probability of corrosion detection during a periodic or a special inspection.

(7) Pressurization load exceedances per flight hour.

The results of the demonstration computer runs indicate that the implemented SAIFE logic can successfully simulate and quantify the evaluation process for the establishment or revision of the structural inspection intervals for commercial jet transport aircraft.

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APPENDIX

SAIFE LOGIC FLOW DIAGRAMS

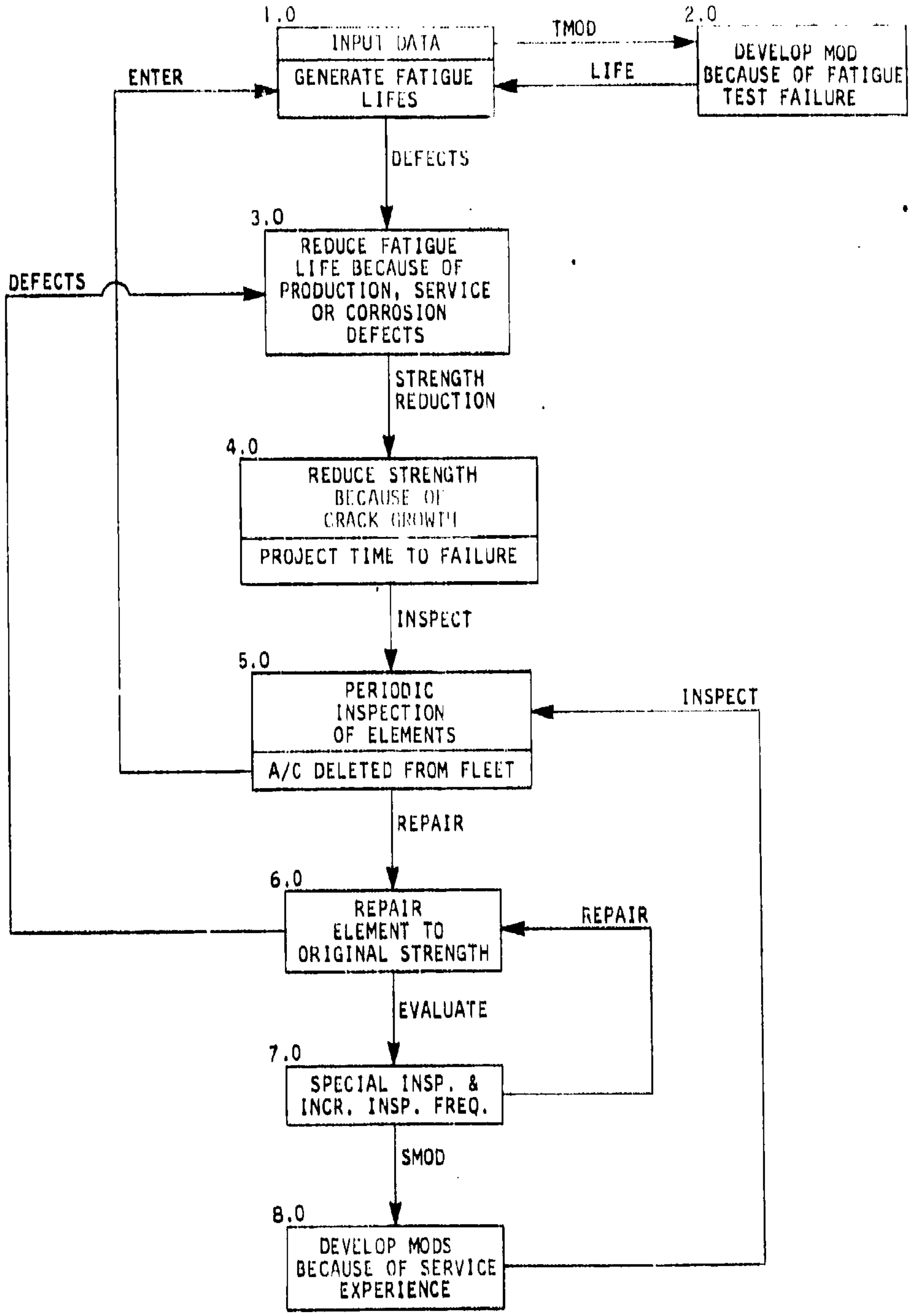

Figure 52. Flow Diagram Showing Major Aspects of SAIFE Logic

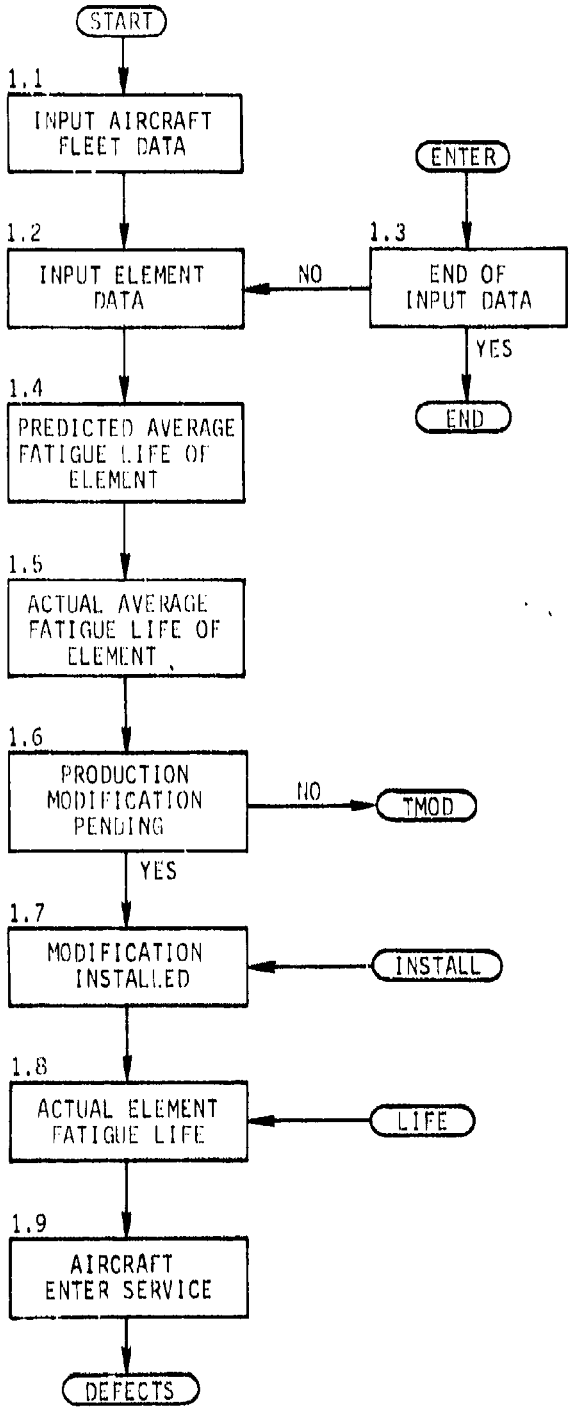

Figure 53. Detailed Flow Diagram - Input Data/Generate Fatigue Lives

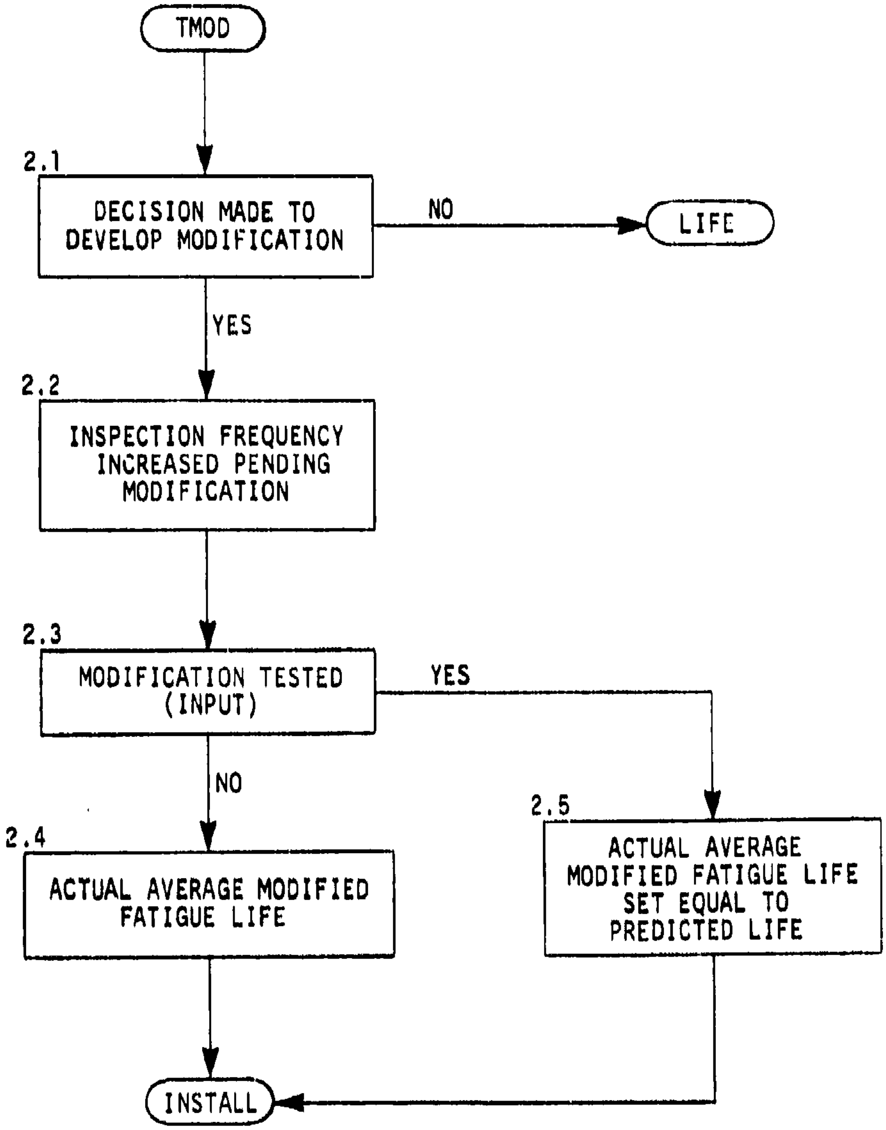

Figure 54. Detailed Flow Diagram - Develop Modification Because of Fatigue Test Failure

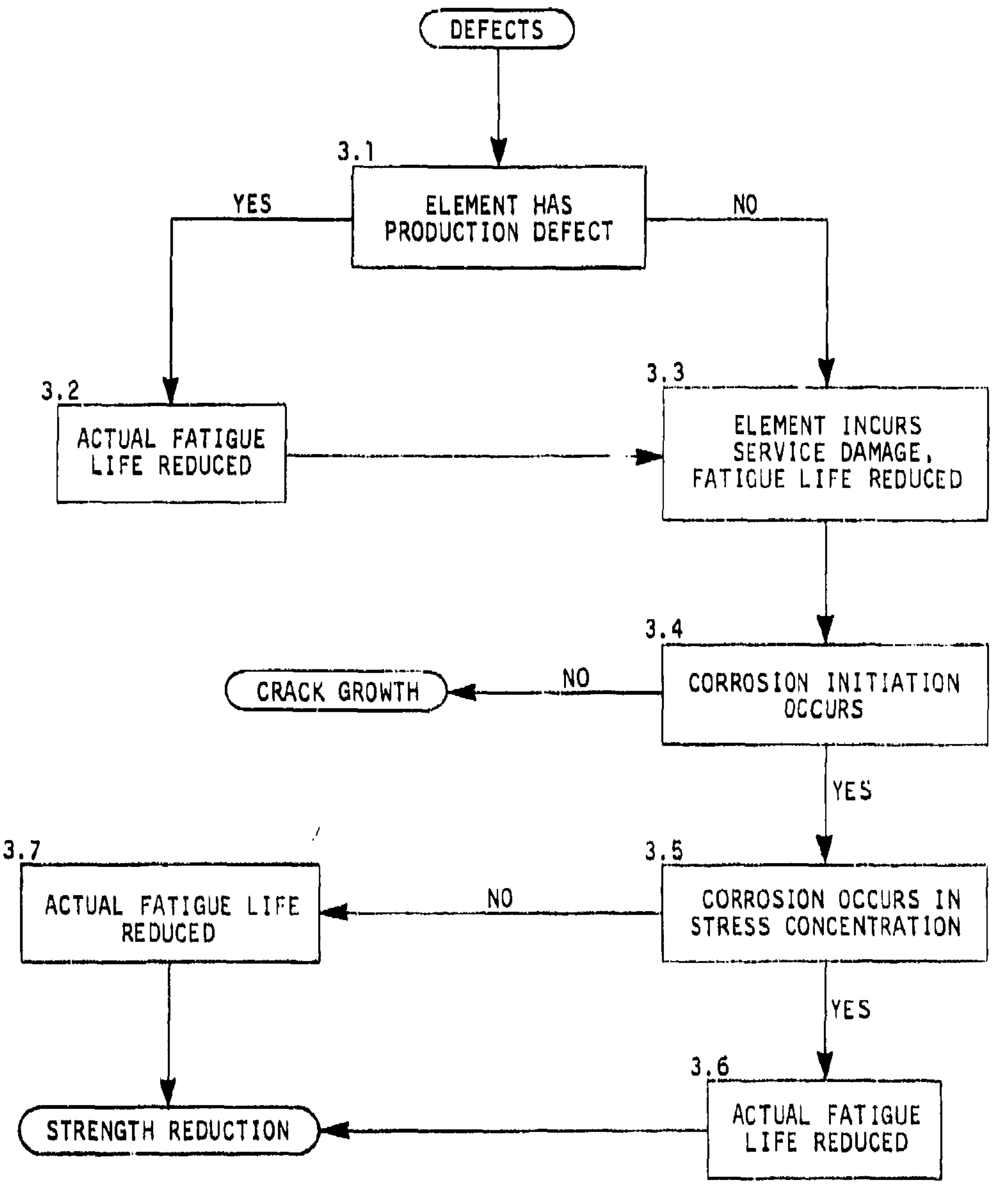

Figure 55. Detailed Flow Diagram - Reduce Fatigue Life Because of Production, Service, or Corrosion Defects

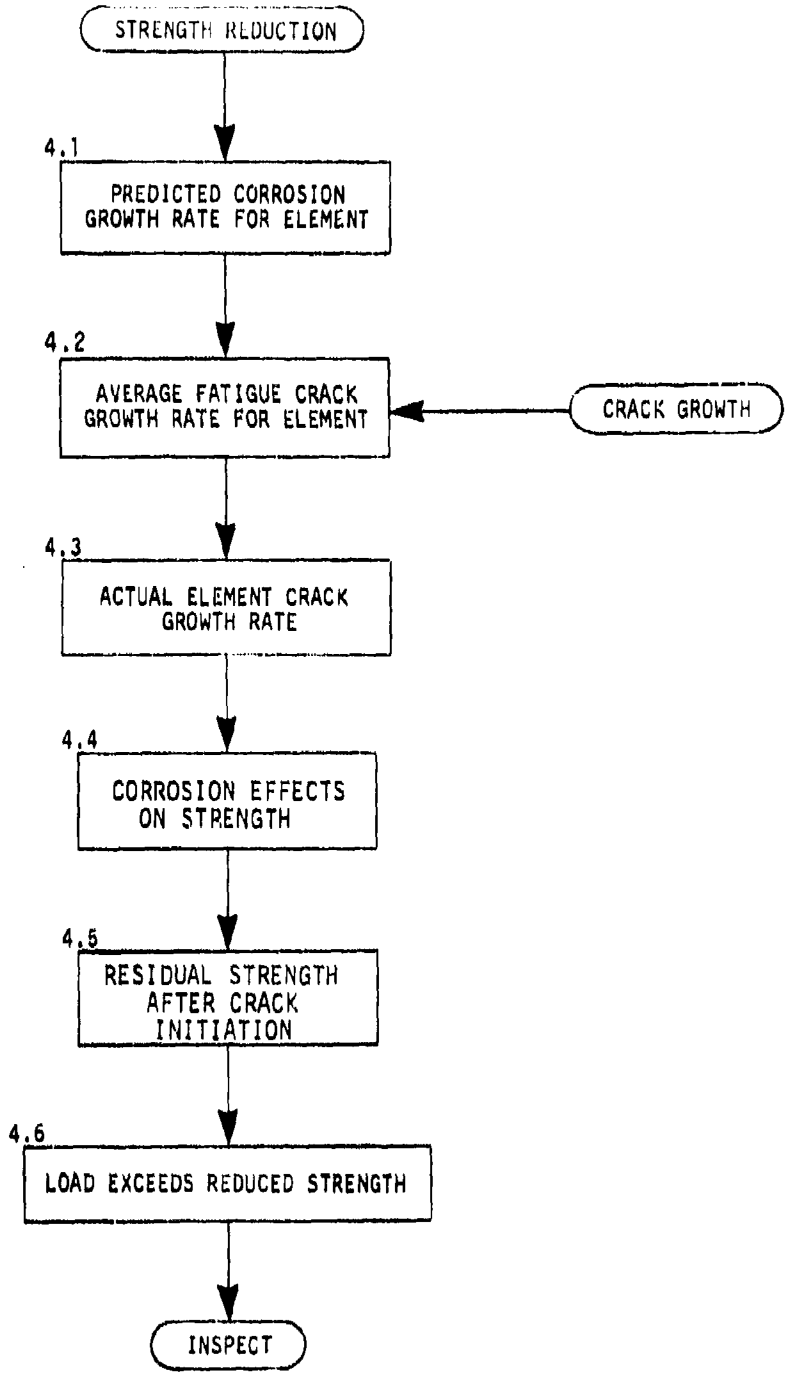

Figure 56. Detailed Flow Diagram - Reduce Strength Because of Crack/Corrosion Growth/Predict Time to Failure

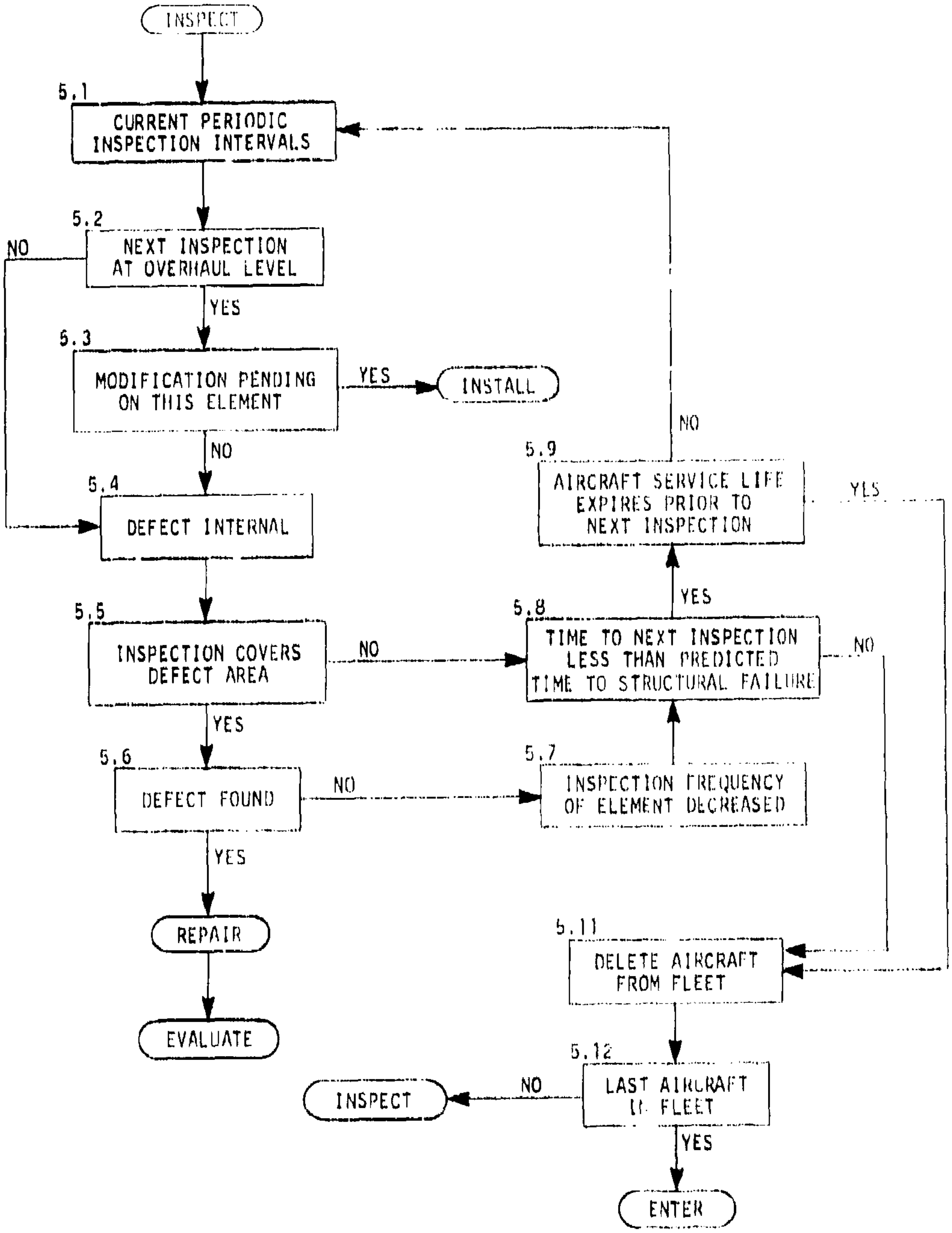

Figure 57. Detailed Flow Diagram - Periodic Inspection of Elements/Aircraft Deleted from Fleet

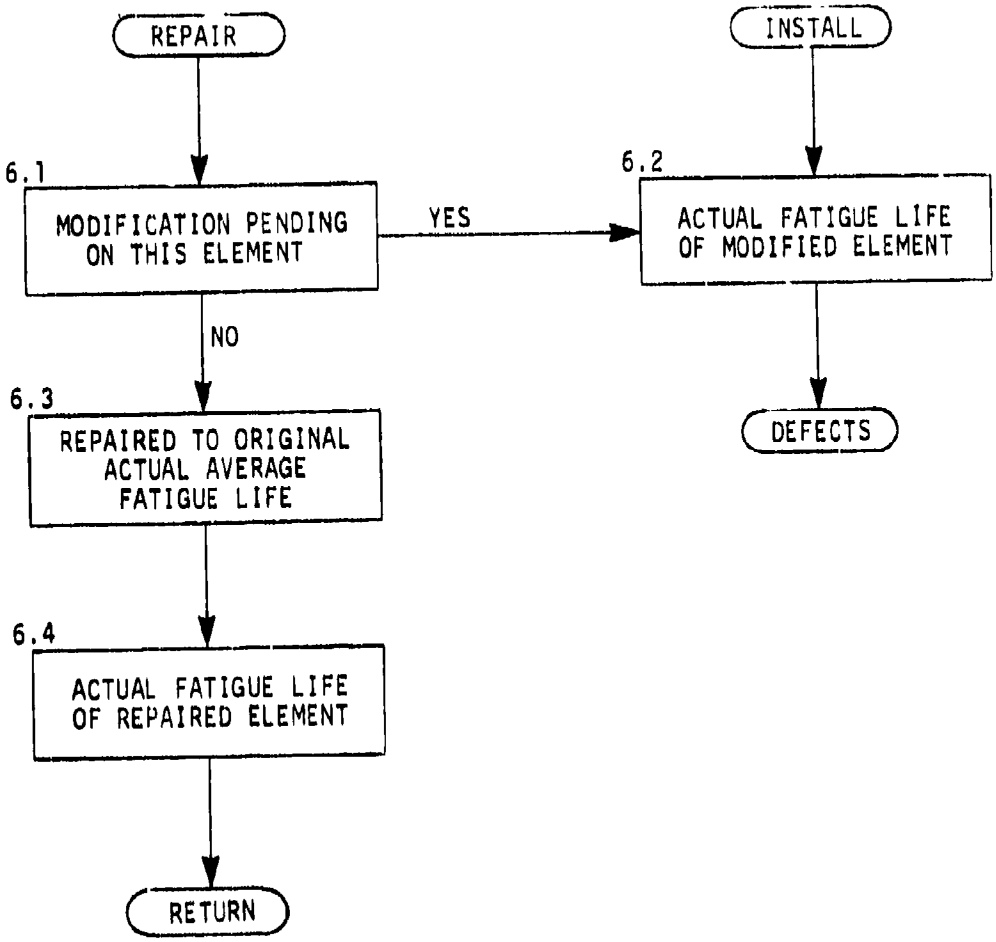

Figure 58. Detailed Flow Diagram - Repair Element to Original Strength

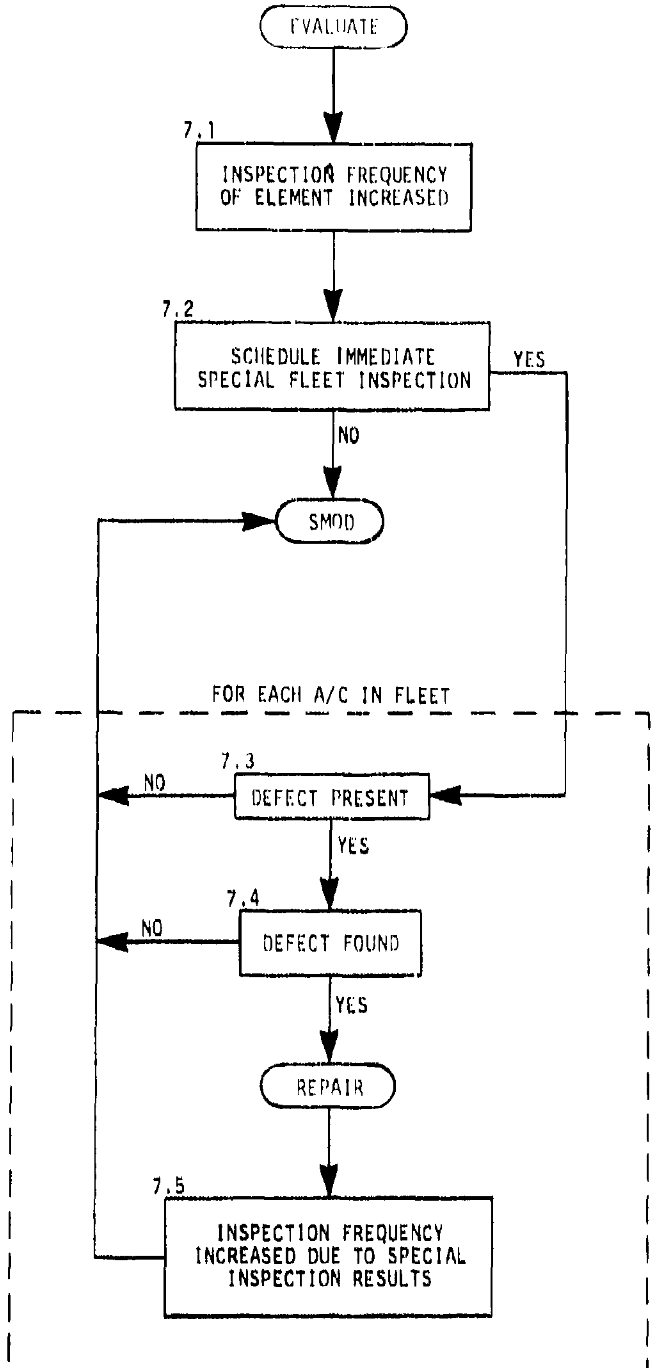

Figure 59. Detailed Flow Diagram - Special Inspection and Increase Inspection Frequency

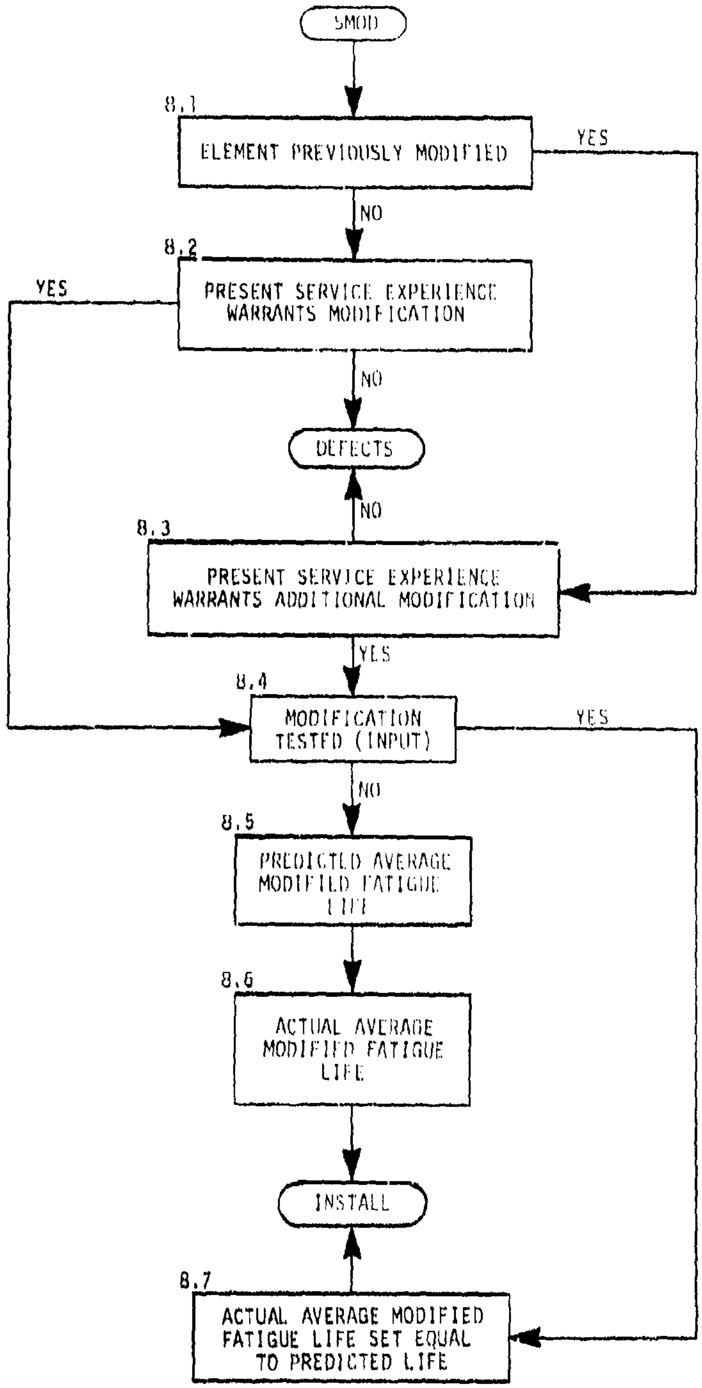

Figure 60. **Detailed Flow Diagram - Develop Modification Because of Service Experience**