A Computer Simulation of the Life of the Structure of a Fleet of Aircraft

(A life cycle risk and reliability model for aircraft structures)

Arnold E. Anderjaska (1)
(1) Technical Data Analysis, Inc.
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Abstract

The author was instrumental in the Federal Aviation Agency 1978 development of the "Structural Area Inspection Frequency Evaluation" (SAIFE) program used to assist in defining structural criteria and inspections on wide body transports and aging airplanes. In the context of today's environment of keeping aircraft in service beyond original design goals, the application and extension of the SAIFE program is becoming increasingly desirable since the objectives of this program are very similar to the current objectives of ensuring safety of aging aircraft fleet through inspection and risk mitigation. The author provides his perspectives on the development of this program including a large body of service data collected to support this program.

The paper discusses the SAIFE program development details for simulating the structural life of an airplane fleet as it goes through its life cycle from design through test, production, service, inspection and repair / modification to retirement or failure. The paper describes how the entire primary structure was analyzed from a bottom-up approach using information from element or sub-element members such as stringers, spar cap, frame, and adjacent skin. The simulation logic details are then presented to describe the prediction of the time of occurrence of design errors, test failures, production defects, corrosion, service damage, fatigue cracks and subsequent crack growth and time to failure for each element of each airplane based on statistical distributions. The probability of element failure is calculated for each crack due to a load exceeding the residual strength during crack growth. Thus element, element type and aircraft failure rates are generated. Initial inspections are first scheduled for each element of each airplane in the fleet. The detection of each defect is predicted based on the size of the defect and the level and type of inspection. If detected, repair is assumed and inspections may be tightened or if not detected or no crack, inspections may be relaxed, all based on the current structural history. Then, either both inspections and repair are continued or modifications are designed and scheduled for installation, depending on relative costs of the two options. If not detected, either failure (residual strength degraded to 1g (delta g = 0)) or retirement occurs depending on which is predicted to occur first. This provides for generation of a structural history and an estimate of the fleet failure rate.

1. Background

In the early 1970's the airlines were, as a part of the FAA certification process, establishing the initial structural inspection programs for the new wide-body airplanes. In recognition of the manufacturer's increased emphasis on fatigue analysis, fatigue tests and damage tolerance, the airlines were changing their process for establishing inspection programs from the subjective MSG-2 to the more objective MSG-3 process which attempts to recognize that the manufacture has estimated the fatigue crack initiation and growth in critical areas. The typical airline inspection program was very complex involving four levels of increasingly rigorous inspections (some times called A, B, C and D inspections) and sampling internal inspections of high time airplanes. The inspection program also involved exploratory, non-exploratory and special service bulletin and Airworthiness Directive inspections. Further, the method of inspection varied from visual to x-ray, ultrasound and eddy current surface or hole inspections. It is difficult to estimate probability of detection under such a program. Reference 1 developed a conceptual proposal that addressed this problem and instigated the development of SAIFE.

There was also a continuing need to make decisions on actions, such as the following, to prevent progressive type failures from fatigue and corrosion and to minimize burden of corrosion and fatigue cracks:

(1) Establishment of inspection policies and programs;
(2) Design and substantiation criteria changes and
(3) Correction of service problems.

These decisions are primarily based on two factors;

(1) The probability of structural defects (cracks, corrosion, damage etc.) and catastrophic failures and
(2) The burdens caused or alleviated by the proposed action.

Good decisions require the best possible estimate of these two factors. It is impossible to account for all of the factors and the variabilities involved in predicting these two factors. Real life is very complex. However, decisions have to be made and are made every day with only an implied prediction of these two factors without making a best estimate. Typically, these decisions are based on available analysis, tests and engineering judgment. Burdens caused or alleviated are usually only intuitively considered or at least not quantified in an organized manner. Currently there is more use of quantitative risk analysis to assist in making decisions. They are a valuable assist, but should not be the sole basis for these decisions. While quite effective, it is believed that the current risk analyses do not account for all of the following significant factors and variabilities:

Variation in loading environment and failing load exceedances; crack and corrosion initiation and growth between individual airplanes; the occurrence of corrosion, production defects, service damage, multiple cracks; the probability of inspection detection based on in-service performance, the maintenance policies, the multiple levels of inspection and feed-back from service experience.

The FAA funded the development of a computer simulation that would attempt quantify the engineering judgment and the burdens involved by use of all available information and resources to account for all significant factors in predicting the above two factors. It was intended for use in evaluation of:

(1) Possible actions on old age aircraft.

(2) The detailed criteria for the new fatigue rule.
(3) Proposed MSG-3 inspection programs.

The development was completed in 1978. It was used to a limited extent in evaluating changes in various fatigue criteria and maintenance policies on aircraft failure rates. It was also used in one case to help evaluate a question in the Boeing 747 certification. Since that era, Boeing has been basing their Probability of Inspection Detection (PODD) values those demonstrated in-service as was the case in SAIFE.

2. Introduction

Budget restraints, increased development costs and the ability of current military aircraft to continue with some modification to perform their mission, has resulted in increased use of aircraft far beyond their planned life. The Navy P-3A was first produced in 1961 and is still in use in the U.S as a civil fire fighter and by the military in other countries. A later version, P-3C, first produced in 1969, is still a Navy mainstay and will remain so for a considerable number of years. This is approximately three times its' planned life. The Air Force B-52's longevity is legendary in that if it were an Air Force officer, it would have been retired many years ago. While economic factors and regulations such as noise and aging aircraft tend to limit the longevity of the use civil aircraft in U.S., many continued past their planned life especially in third world countries.

As a result, there is an increased use of risk analyses and an increased need of good risk analyses. SAIFE has unique capabilities and possibilities of improvement. For these reasons we have revisited the legacy SAIFE program and converted for PC use. In this paper we will describe the program, its' basis and the results of its' demonstration and limited use.

3. General Program Description

The SAIFE program is a large, complex math model designed to simulate the structural performance of an aircraft fleet and the effectiveness of its design, test, production and maintenance program. The model covers all phases of a structures life from design through full scale fatigue test, production, service, inspection, repair and modification to retirement or failure. It recognizes that this is a dynamic problem with feedback and response.

To be realistic, the model attempted to account for all significant factors and variations by accounting for the following:

(1) Design and criteria errors;
(2) Test schedule, criteria, errors and results;
(3) Production schedule and defects;
(4) Service usage schedule and service damage;
(5) Corrosion and its' growth;
(6) Fatigue cracks and their growth;
(7) Variation in load environment between individual aircraft:
(8) Variation in defect initiation and growth between structural elements identical in structure and location;;
(9) The PODD of all levels of inspection
(10) The timing of various events of all aircraft in the active fleet and their feedback.

The FAA Mechanical Reliability Reports (MRR) and Service Defect Reports (SDR) were a major basis for the input data. U.S. civil transport operators and maintenance facilities were required by law to report any structural defect found. These data represents the largest amount of data on structural deficiencies detected in transport aircraft and is kept in a computerized data base. Reports on large narrow-body jet transports in 1964 through 1974 were reviewed and analyzed. These involved approximately 4800 pertinent reports. The data sample was assumed to represent 1406 aircraft and 45,791,114 flight hours on the basis of the data in the annual volumes of the FAA Statistical Handbook.

These data have been criticized as not being usable statistically. It is known that not all defects are reported, they probably cover the early life but not cover the entire life of the aircraft involved, a significant number are incomplete and there are some errors. However, it is a very large sample, and essentially the only readily available operational source for estimating the location, size, type, flight hours and cause of service defects and for estimating the effectiveness of inspections under real life conditions.

Currently only defects found in the first 14 years (approximately) are reported as SDRs, the latter defects are reported under the Aging Aircraft Program and only 10 years of data are kept in the data base. If all of this data is kept and available, it could be possible in the future to analyze the data for the complete program life cycle of a given structural version of a given model.

The complexity and magnitude of the project required using the most efficient technique and computer language available. The computer simulation language SIMSCRIPT II.5 (Trademark, Consolidated Analysis Centers, Inc., Los Angeles, California) was ideally suited to this project since it is designed to handle simulations where hundreds of events are happening concurrently and in a chronological sequence such as in the SAIFE application. SIMSCRIPT is also a desirable computer language from the user's viewpoint since its free-form English format makes it easy to interpret the source program and it reduces the coding and debugging effort. In addition, SIMSCRIPT provides system functions to generate the random numbers required in SAIFE.

Decisions and outcomes (i.e., fatigue crack initiation, crack growth rates) throughout the simulation are generated either probabilistically by random “Monte Carlo” picks from probability distributions or deterministically from criteria all based on information in the input data or embedded in the code. The input data was developed by an extensive review and analysis of test, and service data and the literature. Figure 1 illustrates the data sources and analytical functions that are integrated into the SAIFE logic

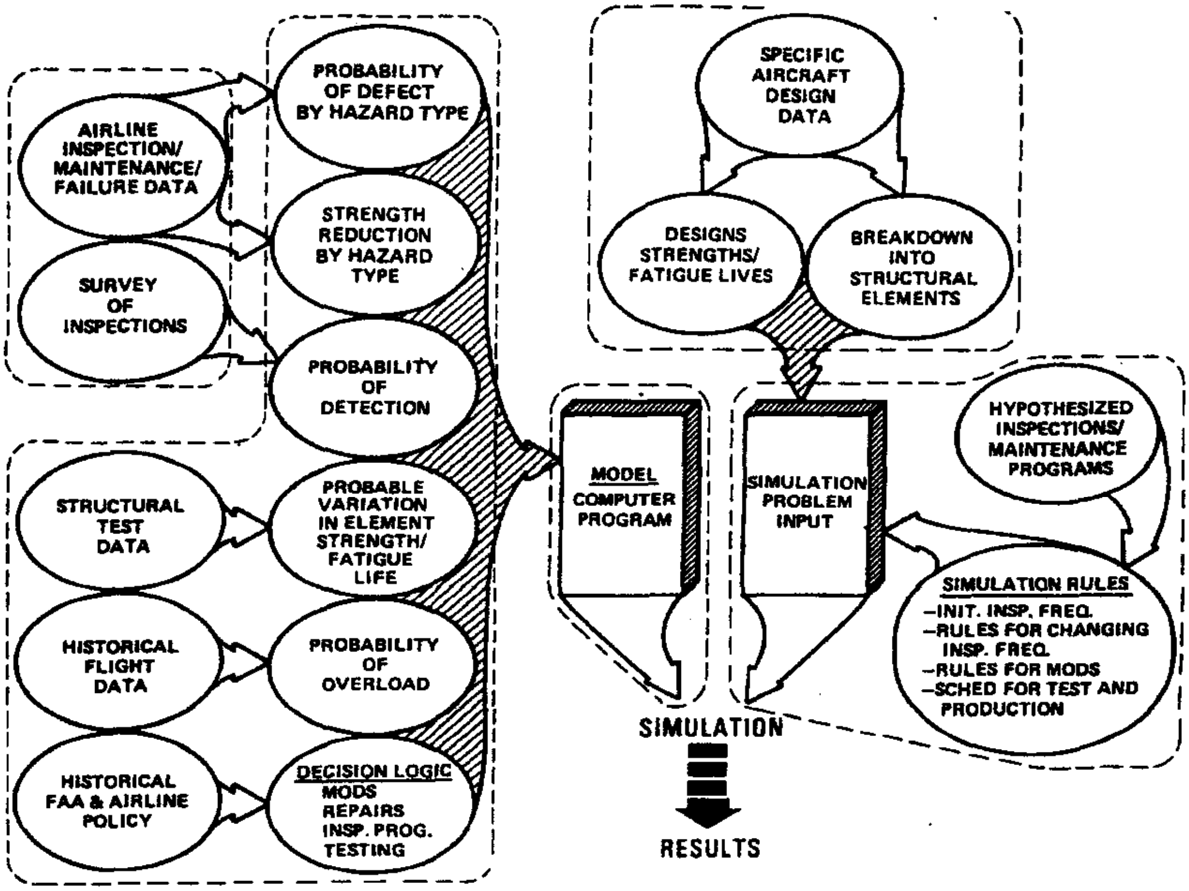
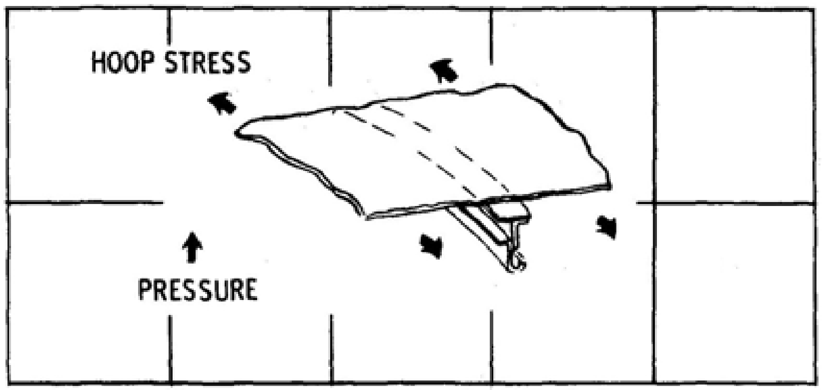
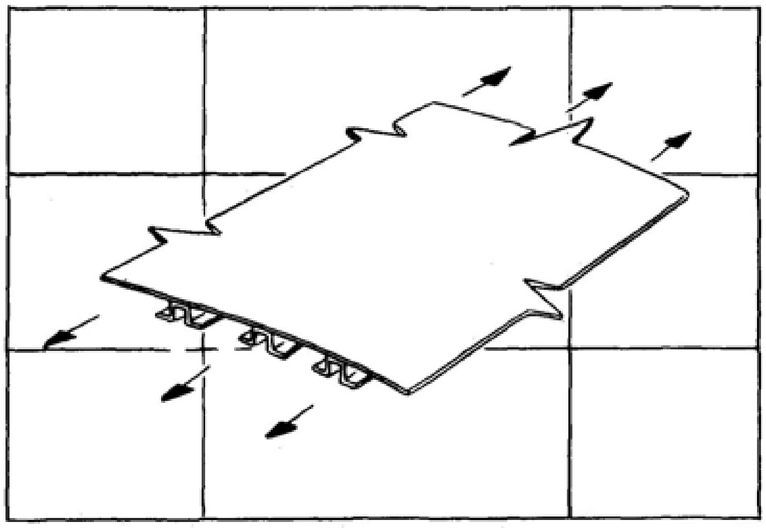


Figure 1. Approach to SAIFE Simulation Problem

The primary aircraft structure was divided in to small structure elements which are evaluated independently. The size of each element was determined by naturally occurring design points such as stringer or frame spacing. The elements were defined to include the attached skin and the attachment points to adjacent ribs or frames. Figure 2 shows typical elements.



Fuselage Frame Element



Wing Stringer Element

Figure 2. Typical Aircraft Elements Used in the SAIFE Simulation

4. Model Logic and Basis

The model logic and the basis for the input data and model criteria are described in this section. The essential approach is as follows:

(1) Read the fleet input data

(2) For each element;

(a) Read element input data:

(b) Predict the average fatigue life from fatigue analysis prediction;

(c) Predict full scale fatigue result and adjust inspections, implement modifications;

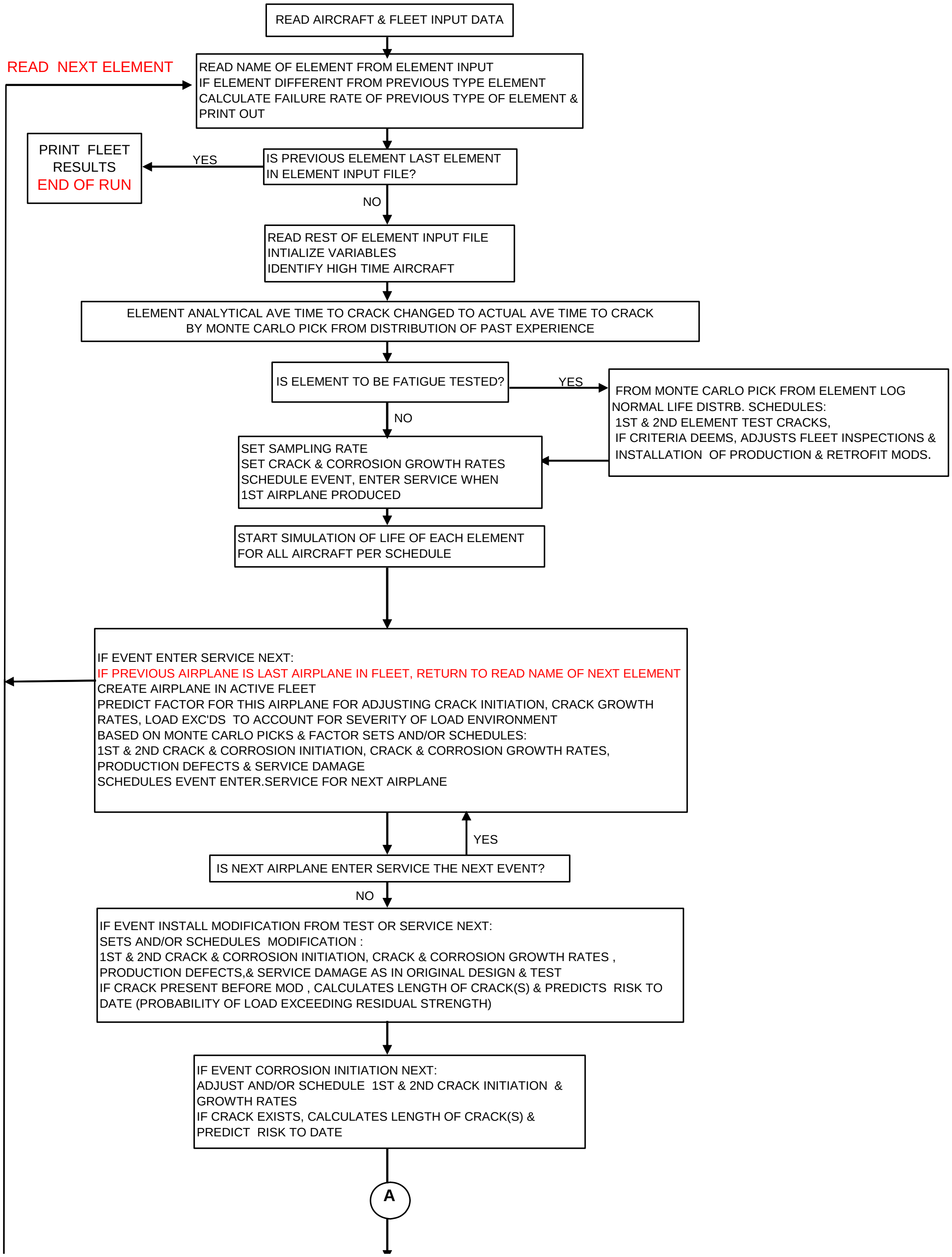
(d) As each airplane enters service predict production defects, service damage, fatigue crack and corrosion initiation and set crack and corrosion growth rates;

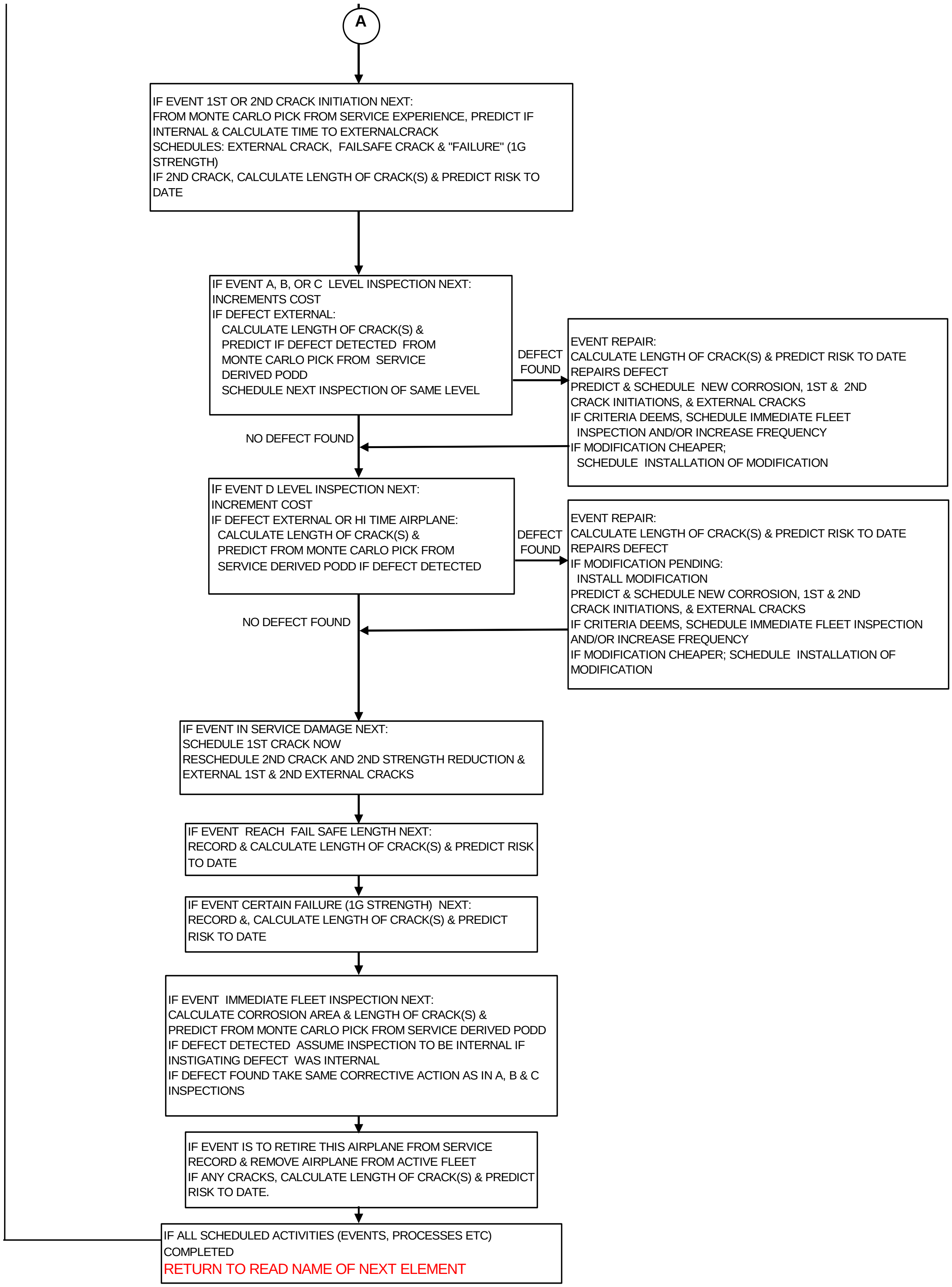
(e) Start simulation with defects occurring and growing with inspections, inspection change, modifications developed and installed, predicting risk to failure or retirement;

(f) Upon completing simulation on all airplanes, print out results and go to next element at (2) above.

(3) Upon completing simulation of all elements, print out fleet results and end simulation.

This logic is shown in the Figure 3 flow chart below.


READ AIRCRAFT & FLEET INPUT DATA
READ NEXT ELEMENT
READ NAME OF ELEMENT FROM ELEMENT INPUT
IF ELEMENT DIFFERENT FROM PREVIOUS TYPE ELEMENT
CALCULATE FAILURE RATE OF PREVIOUS TYPE OF ELEMENT &
PRINT OUT
PRINT FLEET RESULTS
END OF RUN
YES
IS PREVIOUS ELEMENT LAST ELEMENT
IN ELEMENT INPUT FILE?
NO
READ REST OF ELEMENT INPUT FILE
INTIALIZE VARIABLES
IDENTIFY HIGH TIME AIRCRAFT
ELEMENT ANALYTICAL AVE TIME TO CRACK CHANGED TO ACTUAL AVE TIME TO CRACK
BY MONTE CARLO PICK FROM DISTRIBUTION OF PAST EXPERIENCE
IS ELEMENT TO BE FATIGUE TESTED?
YES
FROM MONTE CARLO PICK FROM ELEMENT LOG
NORMAL LIFE DISTRB. SCHEDULES:
1ST & 2ND ELEMENT TEST CRACKS,
IF CRITERIA DEEMS, ADJUSTS FLEET INSPECTIONS &
INSTALLATION OF PRODUCTION & RETROFIT MODS.
NO
SET SAMPLING RATE
SET CRACK & CORROSION GROWTH RATES
SCHEDULE EVENT, ENTER SERVICE WHEN
1ST AIRPLANE PRODUCED
START SIMULATION OF LIFE OF EACH ELEMENT
FOR ALL AIRCRAFT PER SCHEDULE
IF EVENT ENTER SERVICE NEXT:
IF PREVIOUS AIRPLANE IS LAST AIRPLANE IN FLEET, RETURN TO READ NAME OF NEXT ELEMENT
CREATE AIRPLANE IN ACTIVE FLEET
PREDICT FACTOR FOR THIS AIRPLANE FOR ADJUSTING CRACK INITIATION, CRACK GROWTH
RATES, LOAD EXC'DS TO ACCOUNT FOR SEVERITY OF LOAD ENVIRONMENT
BASED ON MONTE CARLO PICKS & FACTOR SETS AND/OR SCHEDULES:
1ST & 2ND CRACK & CORROSION INITIATION, CRACK & CORROSION GROWTH RATES,
PRODUCTION DEFECTS & SERVICE DAMAGE
SCHEDULES EVENT ENTER.SERVICE FOR NEXT AIRPLANE
YES
IS NEXT AIRPLANE ENTER SERVICE THE NEXT EVENT?
NO
IF EVENT INSTALL MODIFICATION FROM TEST OR SERVICE NEXT:
SETS AND/OR SCHEDULES MODIFICATION :
1ST & 2ND CRACK & CORROSION INITIATION, CRACK & CORROSION GROWTH RATES ,
PRODUCTION DEFECTS,& SERVICE DAMAGE AS IN ORIGINAL DESIGN & TEST
IF CRACK PRESENT BEFORE MOD , CALCULATES LENGTH OF CRACK(S) & PREDICTS RISK TO
DATE (PROBABILITY OF LOAD EXCEEDING RESIDUAL STRENGTH)
IF EVENT CORROSION INITIATION NEXT:
ADJUST AND/OR SCHEDULE 1ST & 2ND CRACK INITIATION &
GROWTH RATES
IF CRACK EXISTS, CALCULATES LENGTH OF CRACK(S) &
PREDICT RISK TO DATE
A


A
IF EVENT 1ST OR 2ND CRACK INITIATION NEXT:
FROM MONTE CARLO PICK FROM SERVICE EXPERIENCE, PREDICT IF INTERNAL & CALCULATE TIME TO EXTERNALCRACK
SCHEDULES: EXTERNAL CRACK, FAILSAFE CRACK & "FAILURE" (1G STRENGTH)
IF 2ND CRACK, CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE
IF EVENT A, B, OR C LEVEL INSPECTION NEXT:
INCREMENTS COST
IF DEFECT EXTERNAL:
CALCULATE LENGTH OF CRACK(S) &
PREDICT IF DEFECT DETECTED FROM
MONTE CARLO PICK FROM SERVICE
DERIVED PODD
SCHEDULE NEXT INSPECTION OF SAME LEVEL
DEFECT FOUND
EVENT REPAIR:
CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE
REPAIRS DEFECT
PREDICT & SCHEDULE NEW CORROSION, 1ST & 2ND CRACK INITIATIONS, & EXTERNAL CRACKS
IF CRITERIA DEEMS, SCHEDULE IMMEDIATE FLEET INSPECTION AND/OR INCREASE FREQUENCY
IF MODIFICATION CHEAPER;
SCHEDULE INSTALLATION OF MODIFICATION
NO DEFECT FOUND
IF EVENT D LEVEL INSPECTION NEXT:
INCREMENT COST
IF DEFECT EXTERNAL OR HI TIME AIRPLANE:
CALCULATE LENGTH OF CRACK(S) &
PREDICT FROM MONTE CARLO PICK FROM
SERVICE DERIVED PODD IF DEFECT DETECTED
DEFECT FOUND
EVENT REPAIR:
CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE
REPAIRS DEFECT
IF MODIFICATION PENDING:
INSTALL MODIFICATION
PREDICT & SCHEDULE NEW CORROSION, 1ST & 2ND CRACK INITIATIONS, & EXTERNAL CRACKS
IF CRITERIA DEEMS, SCHEDULE IMMEDIATE FLEET INSPECTION AND/OR INCREASE FREQUENCY
IF MODIFICATION CHEAPER; SCHEDULE INSTALLATION OF MODIFICATION
NO DEFECT FOUND
IF EVENT IN SERVICE DAMAGE NEXT:
SCHEDULE 1ST CRACK NOW
RESCHEDULE 2ND CRACK AND 2ND STRENGTH REDUCTION & EXTERNAL 1ST & 2ND EXTERNAL CRACKS
IF EVENT REACH FAIL SAFE LENGTH NEXT:
RECORD & CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE
IF EVENT CERTAIN FAILURE (1G STRENGTH) NEXT:
RECORD &, CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE
IF EVENT IMMEDIATE FLEET INSPECTION NEXT:
CALCULATE CORROSION AREA & LENGTH OF CRACK(S) &
PREDICT FROM MONTE CARLO PICK FROM SERVICE DERIVED PODD
IF DEFECT DETECTED ASSUME INSPECTION TO BE INTERNAL IF INSTIGATING DEFECT WAS INTERNAL
IF DEFECT FOUND TAKE SAME CORRECTIVE ACTION AS IN A, B & C INSPECTIONS
IF EVENT IS TO RETIRE THIS AIRPLANE FROM SERVICE
RECORD & REMOVE AIRPLANE FROM ACTIVE FLEET
IF ANY CRACKS, CALCULATE LENGTH OF CRACK(S) & PREDICT RISK TO DATE.
IF ALL SCHEDULED ACTIVITIES (EVENTS, PROCESSES ETC) COMPLETED
RETURN TO READ NAME OF NEXT ELEMENT

4.1 Input Fleet Data

The program input consists of three parts. The first part consists of input variables which pertain to the aircraft type under consideration. These variables are input only once per simulation run and are constant from element to element. If the user desires to input random number seeds, the ten seeds are input after the aircraft input. The second part of the input consists of long list data if standard output is not used. The third part of the input consists of input variables whose values are unique to each element. These variables must be input in their entirety for each element being simulated. The time line was based on the assumption that 3000 flight hours = 1 calendar year and that all aircraft accumulated 3000 hours per year.

MODEL - This identifies the aircraft type under consideration. A theoretical hybrid based on the 747 wing and the DC-10 fuselage was used in the demonstration.

SIZE.OF.FLEET - This integer variable is the number of aircraft in the fleet being simulated. A fleet of 500 aircraft was used in the demonstration.

USAGE.LIFE - This real variable is the service life in flight hours of the aircraft being simulated. All aircraft in the fleet must have the same service life. A service life of 60,000 hours was used.

BEGIN.PRODUCTION - This real variable is the time in flight hours relative to the start of the simulation when the first aircraft enters service. This variable in conjunction with the input variable START.TEST enables the user to start the fatigue test of the element before, after, or at the same time the first aircraft enters service. A start time of 150 hours was used.

PRODUCTION.TIME - This real variable defines the initial aircraft production rate. It is the time in flight hours between aircraft entering service. 50 hours was used.

2nd.PRODUCTION.TIME - This real variable defines the second aircraft production rate. It is the time in flight hours between aircraft entering service. 100 hours was used.

PRODUCTION CHANGE - This real variable is the simulation time when the second aircraft production rate takes effect. Note that this time is measured from the time that the first aircraft enters service and not from the start of the simulation. 5000 hours was used.

START.TEST - This real variable is the time in flight hours relative to the start of the simulation when the fatigue test of an element is begun. If no fatigue test is to be conducted, this variable is set to the machine upper limit. 0 hours was used.

TEST ACCELERATION FACTOR - This real variable is the fatigue test acceleration factor, that is, the quotient of the equivalent flight hours divided by the fatigue test hours. A factor or 100 was used.

C.GROWTH.RATE - This real variable is the corrosion area growth rate in square inches per hour for the aircraft being considered. The growth rate for each element in the aircraft is modified by its associated CRR (corrosion resistance rating).0.002 square inches per hour was used.

C7 - If a modification is developed because of a fatigue test failure, this real variable is the percentage (expressed as a decimal fraction) of the test life when the inspection frequency is increased.

C28 - This real variable is the percentage (expressed as a decimal fraction) reduction in the remaining fatigue life of an element when corrosion occurs in a stress concentration. 40% was used.

C29 - This real variable is the percentage (expressed as a decimal fraction) reduction in the remaining fatigue life of an element when corrosion occurs outside a stress concentration. 20% was used.

MU.R - This real variable is the mean of the log-normal distribution of the ratio of the actual average fatigue life to the predicted average fatigue life.

SIG.R - This real variable is the standard deviation of the log-normal distribution of the ratio of the actual average fatigue life to the predicted average fatigue life.

DLL - This real variable is the design limit load in g's above the 1-g level.

1ABCD (1) - This real variable is the inspection interval in flight hours of the A-level inspection. It remains constant throughout the simulation.

1ABCD (2) - This real variable is the inspection interval in flight hours of the B-level inspection. It remains constant throughout the simulation.

CABCD (*) - This one-dimensional real array of size four contains the inspection cost at each level of inspection. CABCD (l) corresponds to the A-level cost; CABCD (2) corresponds to the B-level cost; CABCD (3) corresponds to the C-level cost; and, CABCD (4) corresponds to the D-level cost.

S.OPT - This alpha variable is "YES" if the random number seeds are to be input; it is "NO" if seeds are not input.

LONG-LIST - This alpha variable is "NO" if standard output and "YES" for detailed output on particular aircraft. (LONG LIST DATA).

FAIL.OPT - This integer variable is "1" for output of probability of failure that is based on averaging individual element failure rates to obtain element type failure rates. This integer variable is "2" for output of probability of failure that is based on using a log-normal crack distribution and a curve fit of probability of failure versus crack length to obtain element type failure rates. This integer variable is "3" if both options are desired.

FAT.TEST.FACTOR - This real variable is the probability of a fatigue test being done on the structural element. This real variable is compared with a random number to determine if the fatigue test is done. If the fatigue test is not done, the fatigue test life is set to 9999999.

ACTUAL.AVG.FAT.LIFE - This real variable is the actual average fatigue life in flight hours determined by fatigue test. If this value is not known, input zero and SAIFE will determine it statistically.

LEAD.TIME - This real variable is the time in flight hours between when a decision is made to develop a structural modification and the time the modification is available for installation.

T.FREQ CHG - This real variable is the percentage expressed as a decimal fraction that the D-level inspection interval is reduced due to a fatigue test failure.

S.FREQ.CHG - This real variable is the percentage expressed as a decimal fraction that the D-level inspection interval is reduced due to service experience.

FREQ.DECREASE - This real variable is the percentage expressed as a decimal fraction that the C and D-level inspection intervals are increased due to favorable service experience.

A.REPAIR.COST - This real variable is the repair cost at the A-level inspection.

B.REPAIR.COST - This real variable is the repair cost at the B-level inspection.

C.REPAIR.COST - This real variable is the repair cost at the C-level inspection.

D.REPAIR.COST - This real variable is the repair cost at the D-lever inspection.

1ST.TOOLING - This real variable is the tooling cost of the first structural modification.

AD.TOOLING - This real is the tooling cost in the development of any additional structural modifications.

1ST.MD.COST - This real variable is the installation cost of the first structural modification.

AD.MD.COST - This real variable is the installation cost of any additional structural modification,

S.REPAIR.COST - This real variable is the repair cost of a defect detected during a special inspection.

SU - This real variable is the element ultimate strength in g s above the 1-g level.

SF - This real variable is the element fail safe strength in g s above the 1-g level.

S1 – S10 - These integer variables are the ten random number seeds, and are input only if S.OPT = "YES". Any integer value may be used as input.

LONG LIST DATA. Occasionally in the standard output, elements will appear with unusually long fatigue cracks or early element failures. It is desirable to have a more complete service history of aircraft with these early element-failures than that offered by the standard output. This service history is available through what is called the long list option. This output option is accessed by reading in alpha characters "YES" for the aircraft input variable LONG.LIST. After this input, the element description and identification numbers of the aircraft to be tracked are read in. The input variables for the long list option are listed and described below in the order in which they are read in by SAIFE.

NOE - This integer variable is the number of elements to be processed under the long list option.

ELID4*,*) - This two-dimensional alpha array of size four by NOE identifies each element to be processed. This identification must appear in the first sixteen columns of the data card and must be identical to the description read into the variable ELEMENT (*) described in ELEMENT DATA.

NOAC * - This one-dimensional integer array of size NOE is the number of aircraft to be tracked for each corresponding element.

TLID *,*) - This two-dimensional integer array of size NOE by NOAC *) contains the identification numbers of the aircraft to be tracked for a particular element.

4.2 Input Element Data.

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 up to 4000 elements. The basic element types are listed in. below. Whether the loading is primarily pressure or flight loads depends on the name of the element.

PRIMARY LOCATION-	FUS	Fuselage
	WNG	Wing
	WSC	Wing Center Section
AIRCRAFT ELEMNT-	ACC	Access
	DOR	Door
	FRM	Frame
	FRF	Frame- Flight Loaded
	MFR	Main Frame
	MFF	Mam Frame- Flight Loaded
	SPR	Spar
	SPS	Spar Splice
	STR	Stringer
	SWB	Spanwise Beam
	SWS	Spanwise Beam Splice
	WIN	Window
RELATIVE LOCATION-	AFT`	Aft
	CEN	Center
	FWD	Forward
	BOT	Bottom
	SID	Side
	TOP	Top
	LSA	Lower Surface Aft
	LSC	Lower Surface Center
	LSF	Lower Surface Forward
	USA	Upper Surface Aft
	USC	Upper Surface Center
	USF	Upper Surface Forward

The input variables which are unique to each element and must be read in for each element are listed and described below in the order in which they are read in by SAIFE.

ELEMENT (*) - This one-dimensional alpha array of size four identifies the element being simulated. The total length of this identification cannot exceed sixteen characters. The SAIFE program distinguishes between elements that are pressure loaded and flight loaded. Elements that are pressure loaded must start with characters FUS- and have MFR- as characters five through eight or FRM- as characters nine through twelve. Any characters can be used for the flight loaded elements.

PREDICTED.LIFE - This real variable is the average element fatigue life in flight hours predicted by analysis. If the actual average fatigue life is known, this variable can be entered as zero. Much of this data was extracted from References 2 and 3.

M1.MEAN - This real variable is the average first external crack growth rate in inches per flight hour. Much of this data for variables M1, M2, M3 and M4 were extracted from References 3 and 4.

M2.MEAN - This real variable is the average second external crack growth rate in inches per flight hour.

LGHT.TO.FAILURE - This real variable is the length in inches at which the crack reaches failure under a 1-g load.

CONE - This real variable is the crack length in inches at which the first external crack growth rate changes to the second external crack growth rate. (First external critical crack length).

FSAF.LGT - This real variable is the length in inches at which the crack reaches the fail-safe length.

BIRTH.DEFECT.PROBABILITY - This real variable is the probability of a production defect.

CRR - This integer variable is the corrosion resistance rating.

SDM.000URRENCE.RATE - This real variable is the occurrence rate of service damage per element per aircraft per flight hour.

l.PROB - This real variable is the probability of cracks originating internally.

C.PROB - This real variable is the probability of corrosion originating internally.

INT.LVL.INSP - This alpha variable is the letter identifying the lowest internal level inspection.

EXT.LVL.INSP - This alpha variable is the letter identifying the lowest external level inspection.

MOD.TEST - This alpha variable is input as "YES" if a structural modification is to be fatigue tested. Otherwise it is input as "NO".

LOCATED.IN.STRESS.CON - This real variable is the probability that there is corrosion in a stress concentration.

1.CDM.000URRENCE.RATE - This real variable is the initial corrosion occurrence rate in occurrences per element per aircraft per flight hour.

2.CDM.000URRENCE.RATE - This real variable is the second corrosion occurrence rate in occurrences per element per aircraft per flight hour.

CDM.RATE.CHANGE - This real variable is the aircraft service time in flight hours when the second corrosion occurrence rate takes effect.

L.EXT - This real variable is the length in inches at which a crack originating internally becomes external.

M3.MEAN - This real variable is the average third external crack growth rate in inches per flight hour.

.M4.MEAN - This real variable is the average fourth external crack growth rate in inches per flight hour.

CTWO -'This real variable is the crack length in inches at which the second external crack growth rate changes to the third external crack growth rate. (at the second external critical crack length).

CTHREE - This real variable is the length in inches at which the third external crack growth rate changes to the fourth external crack growth rate. (at the third external critical crack length).

INT.CONE - This real variable is the length in inches at which t e first internal crack growth rate changes to the second internal crack growth rate. (at the first internal critical crack length).

INT.CTWO - This real variable is the length in inches at which the second internal crack growth rate changes to the third internal crack growth rate. (at the second internal critical crack length).

IN.CTHREE - This real variable is the length in inches at which the third internal crack growth rate changes to the fourth internal crack growth rate. (at the third internal critical crack length).

IABCD(3) - This real variable is the initial inspection interval in flight hours of the C-level inspection.

1ABCD(4) - This real variable is the initial inspection interval in flight hours of the D-level inspection.

POP-SIZE - This integer variable is the number of elements of the same type on the aircraft. It is not necessary to input all elements of the same type. A sampling may be used and SAIFE will extrapolate the probability of failure calculation to the total number of elements actually in the aircraft.

MEAN -.This real variable is the result of fitting an exponential curve to flight or pressure load exceedance data. AMEAN*exp(BL) is the number of loads per hour which exceed the load level (L).

B - This real variable is the result of fitting an exponential curve to flight or pressure load exceedance data. AMEAN*exp BL) is the number of loads per hour which exceed the load level (L).

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 2. 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. 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.

4.3 Account for Variability in Fatigue Analysis Results.

The average fatigue life analytically predicted for each element design is the predicted average fatigue life and is a program input.

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 fatigue analysis.

(2) Actual average fatigue life – the fleet average fatigue life for an element determined by a Monte Carlo pick from a distribution based on previous studies.

(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 5. 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.

Since fatigue phenomena are not completely defined, the fatigue life prediction analysis should be performed statistically. In practice, 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 by analysis was studied by the Royal Aircraft Establishment in Reference 6. 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 4, the results of this comparison indicate that the analytical predicted average life frequently overestimated the actual average fatigue life which in any case usually varies considerably from that predicted by the analysis. To determine the: distribution shown in Figure 4, a computer program was used to fit the following statistical (log-normal) distribution to the data:

$$R = \text{Actual Life} / \text{Predicted Life}$$

$$\mu_R = 0.841$$

$$\sigma_R = 0.695$$

The parameters μR (distribution mean) and σR (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 4, where $\mu R = 1.000$ and $\sigma R = 0.695$.

With the application of Monte Carlo computer techniques, a log-normally distributed correction factor for the predicted average fatigue life Figure 4 is generated for each element design. This correction factor (R) yields the actual average fatigue life and is calculated for each element, but can be adjusted by a full scale fatigue test. The same statistical model is used for all the elements making up an aircraft model.

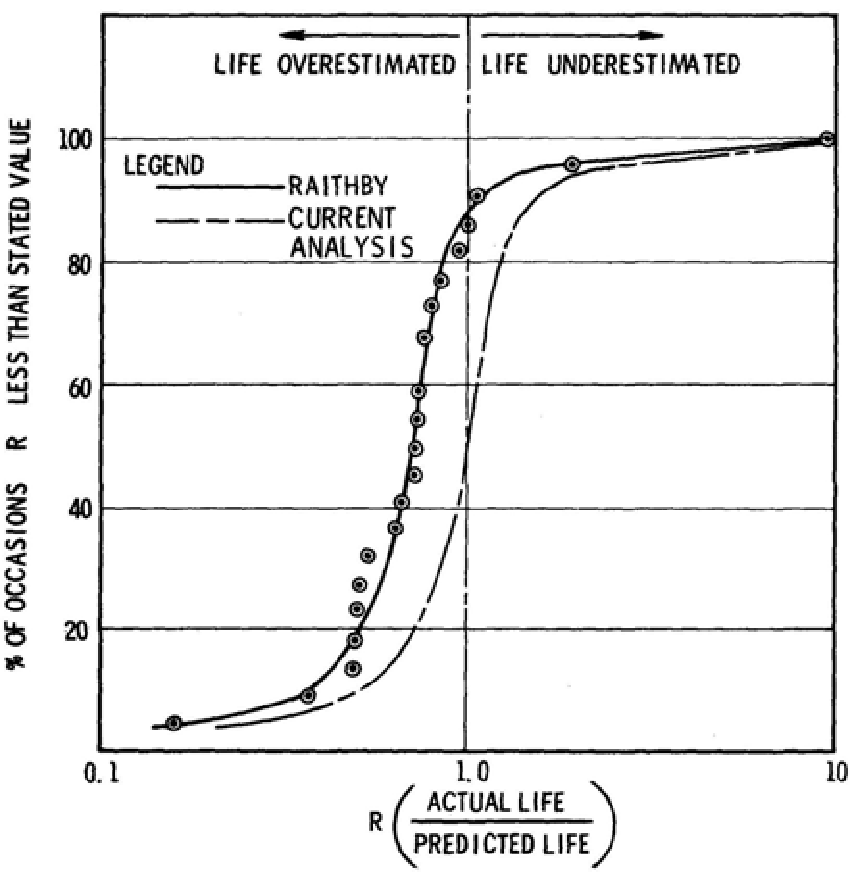


Figure 4. Comparison of Predicted and Actual Fatigue Life

4.4 Account for Full Scale Fatigue Test Results.

Usually in a development program a full scale fatigue test is performed starting after the 1st airplane is produced. The test accumulates equivalent flight much faster than service aircraft. The 1st and 2nd element fatigue test cracks are predicted statistically as described for "Fatigue Crack Initiation in 4.6.2 below. The average element fatigue life is not adjusted to agree with the fatigue test results but service inspections may be increased and/or modifications implemented if the test goals not met.

Based on the fatigue test results a modification may be developed and installed. The decision is based on the number of equivalent flight hours attained during fatigue testing. 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:

Test Hours * Fatigue Test Acceleration Factor = Flight Hours

The criteria for developing a modification are then as follows:

(1) If flight hours > or = 2 * service life a modification is not developed.
(2) If flight hours < 2 * service life 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.

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.

When a modification is required, it is assumed that the modification is designed to the element's original predicted average fatigue life. The input indicates whether a modification is to be fatigue tested before its incorporation into the fleet.

Modification development uses the logic established in 4.3 above. 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 4.3 above. The standard deviation is decreased by 15% and the mean is increased by 15%.

Monte Carlo techniques are again used along with the distribution established in Figure 4 to determine the actual average fatigue life of the modified element.

If a fatigue test the modified element was specified, 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 becomes the actual average fatigue life of the modified element.

4.5 Set Crack and Corrosion Growth Rates and Sampling Rate.

A typical, average 4 step linear crack growth curve as shown in Figure 5 below was used. These curves were obtained from References 3, 5 and 7 and provided in the element input. Essentially, the wing crack growth curves were based on the B-747 fatigue tests and the fuselage crack growth curves were based on the DC-10 fatigue tests. Figure 6 shows manufacturer's data of crack propagation in a typical service door.

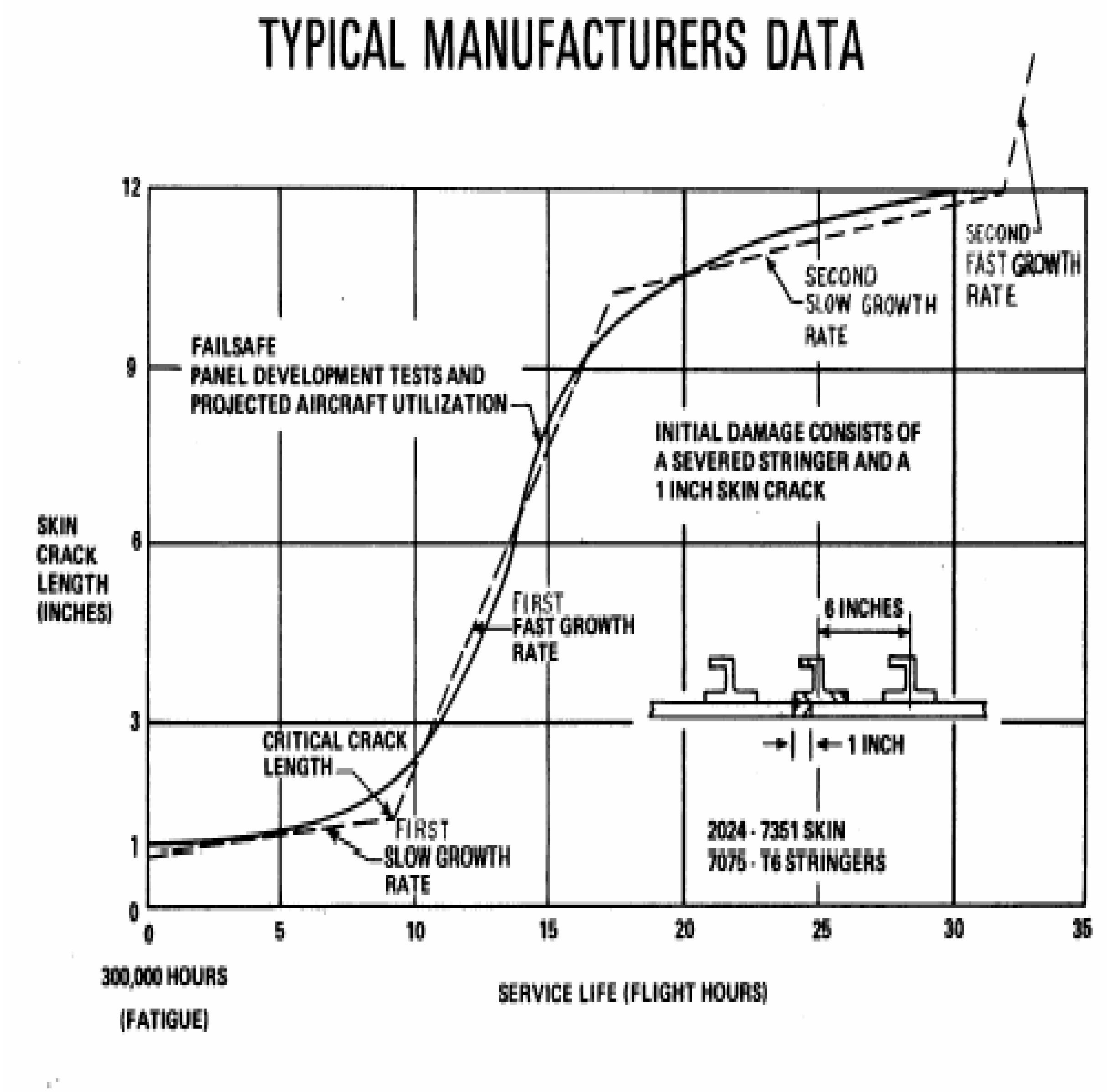


Figure 5. Four Step Linear Crack Growth Curve

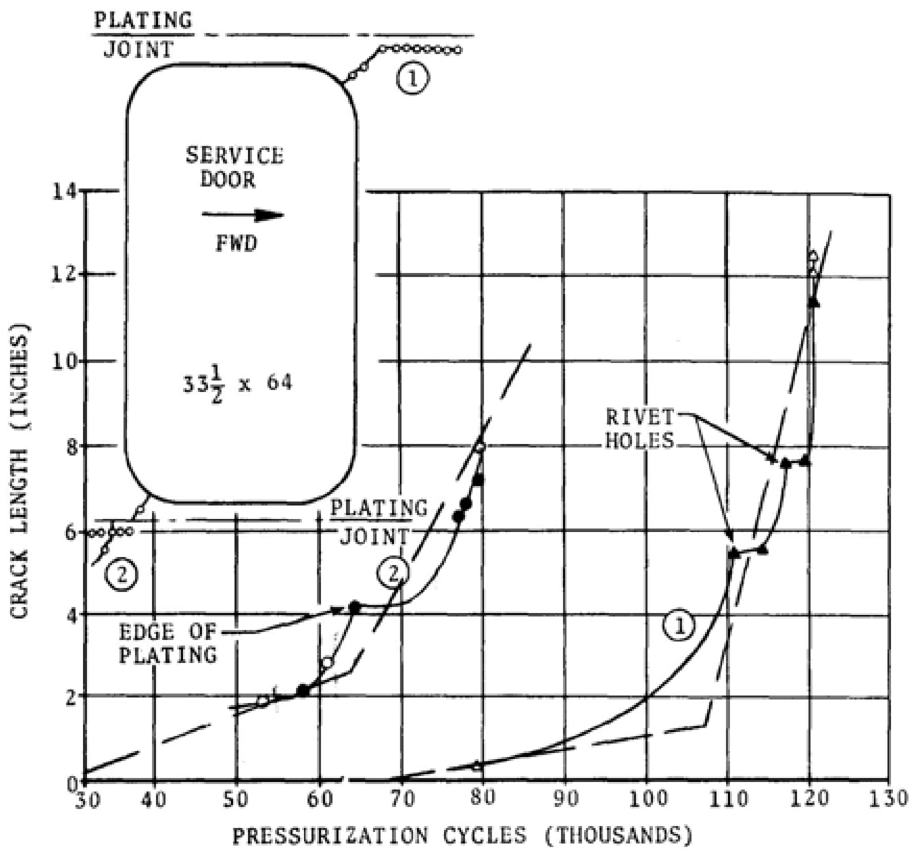


Figure 6. Crack Propagation Rate for Typical Service Door

Average corrosion growth rates were obtained from an analysis of the MRR/SDRs service defect reports with special attention paid to DC-9 wing center section stringers, which were extensively documented. These rates were provided in the element input.

It is airline practice to perform a detailed external and internal inspection on a sample of perhaps 10 high time airplanes. A typical initial sampling % (of X high time airplanes) and interval were provided in the element input.

4.6 Enter Service and Start Simulation.

As each airplane is produced its' entrance to service is scheduled and starting with the first element each of the initial variabilities are established for that element and airplane, The simulation of the life of the element is continued for all airplanes in the fleet until it fails or retired. The simulation then loops through all elements. Accounting for these variabilities is described below.

4.6.1 Account for Variability in Load Environment between Individual Airplanes.

A factor to cover the load spectrum variation between airplanes is generated by a Monte Carlo pick from a log normal distribution with a scatter equal to the structural scatter (difference between $\sigma = 0.14$ and $\sigma = 0.20$, Abelkis, Reference 10) in *Fatigue Crack Initiation and Growth* in 4.6.2 below. The result is divided by the mean to produce the actual factor for this airplane. The subsequent predicted Crack initiation and growth for elements are adjusted using this load environment factor.

4.6.2 Account for Variability in Production Defect and Service Damage Occurrences and Crack and Corrosion Initiation and Growth between Elements Identical in Structure and Location.

Production Defect: As one-time occurrences, production defects normally 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 (MRRs) and Service Difficulty Reports (SDRs) in conjunction with the service bulletins issued periodically by the airframe manufacturers and Air-worthiness. Directives issued by the FAA. 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. .

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 1. The equation for determining the probability of a production defect occurring is

$$P_p = \text{No. of Defect} / 1406 * \text{No. of Individual Elements}$$

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 1964 to 1974, per Reference 8, The FAA Statistical Handbook, 1973.

TABLE 1. PROBABILITY OF A PRODUCTION DEFECT OCCURRING

Element Type	No. of Production Defects Before 11040 Flt Hr	Ave. No. Elements Per A/C	Probability of Production Defect in Individ. Element
Fuselage			
Door frame	0	10	1.19 x 10 -5*
Window frame	0	10	1.19 x 10 -5 *
Main frame	17	180	6.72 x 10 -5
Floor beam	1	60	1.19 x 10 -5
Keel beam	2	60	2.37 x 10 -5
Pressure web	4	60	4.74 x 10 -5
Stringer	24	180	9.48 x 10 -5
Wing			
Access frame	0	50	1.19 x 10 -5*
Rib	2	100	1.42 x 10 -5
Spar	3	100	2.13 x 10 -5
Stringer	3	100	2.13 x 10 -5
Wing Center Section			
Rib	1	21	3.39 x 10 -5
Spanwise beam	0	21	1.19 x 10-5*
Stringer	1	21	3.39 x 10 -5

* estimated

A Monte Carlo pick from a uniform distribution determines whether the element has a production defect. Figure 7 is a histogram of times to crack initiation of the MRR/SDR cracks from production defects. A Monte Carlo pick from the Weibull distribution fitted to this data was used to determine the time of crack initiation when preceded by a production defect. The time from corrosion initiation to crack detection was determined by an analysis of the size of the defect and flight hours at the time of detection of each of the 59 fatigue cracks initiated by production defects. Several standard statistical models were fitted to the data in Figure 7, 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 Figure 7 Weibull distribution rather than from the original Weibull distribution discussed in Fatigue Crack Initiation and Growth below. Times to second and third crack initiations are still drawn from the original Weibull distribution.

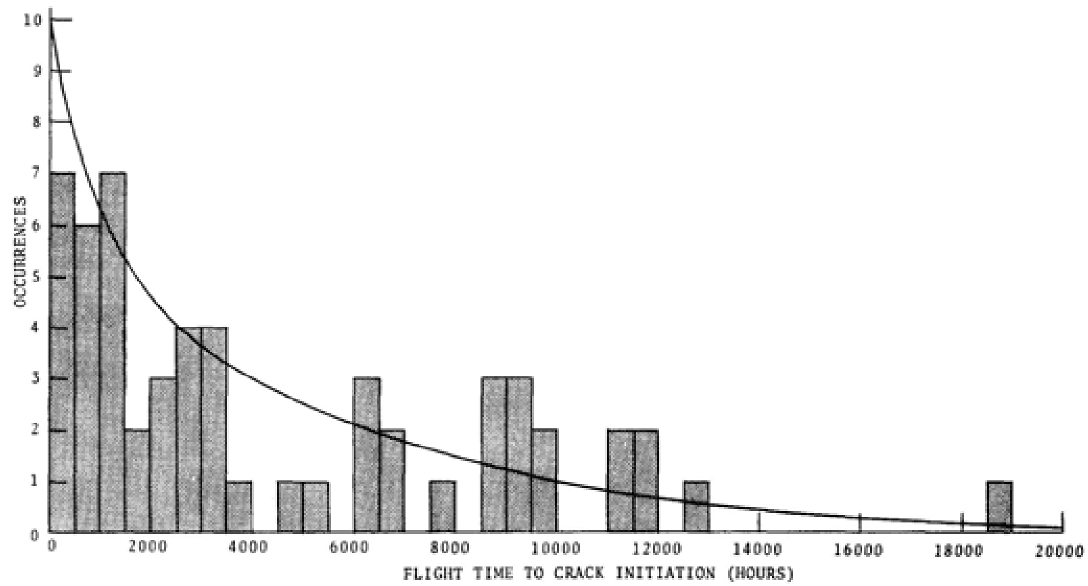


Figure 7. Histogram of Crack Occurrences on Production Damaged Elements

Service Damage- Service damage normally results in structural degradation because it initiates a fatigue crack. 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/ SDRs 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. It was assumed that service damage and crack initiation occur almost simultaneously. The time of service damage and the simultaneous fatigue crack was determined from an analysis of the size of the crack and the flight hours when detected. These times were used to construct a histogram of times to service damage. As indicated in Figure 8, 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/SDRs 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

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

where K_S = adjustment factor discussed below

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

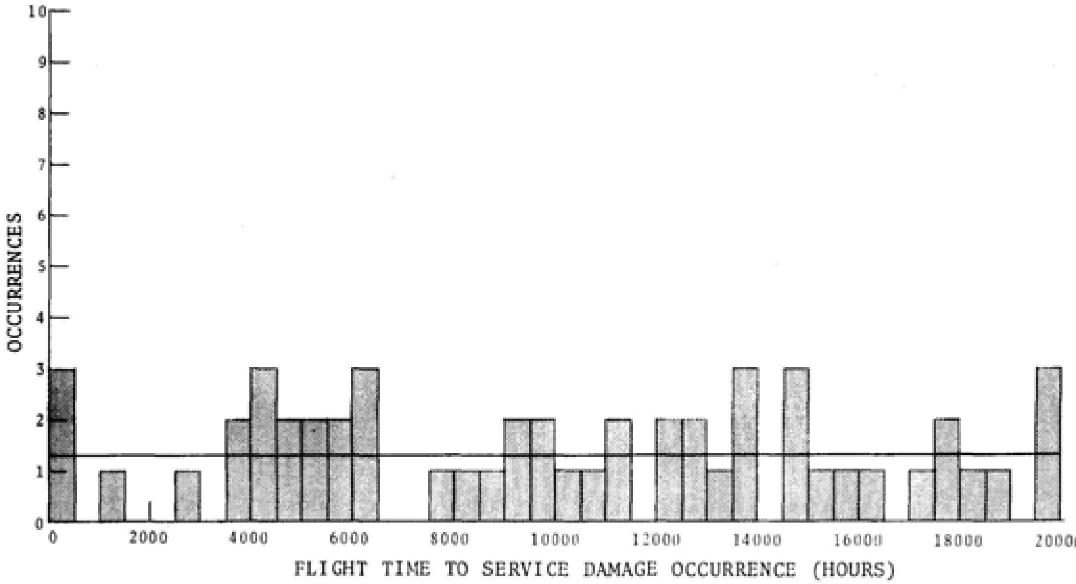


Figure 8. Histogram of Service Damage Occurrences

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 2. The service damage occurrence rate is adjusted by a factor of two because of the results of the Maintenance Inspectors Survey of 50 FAA inspectors, which indicated that service damage is twice as prevalent as actually reported.

A Monte Carlo pick from a uniform distribution produces RN which is used in the following equation to determine the time to crack initiation.

$$t = l/\lambda \ln(RN)$$

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 2. 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 (occu./flt. Hr.)
Fuselage			
Door frame	7	10	3.06 x 10 -8
Window frame	8	10	6.99 x 10 -9
Main frame	21	180	5.10 x 10 -9
Floor beam	1	60	2.43 x 10 -10
Keel beam	1	60	2.43 x 10 -10
Pressure web	0	60	2.43 x 10 -10*
Stringer	14	180	3.40 x 10 -9
Wing			
Access frame	0	50	1.31 x 10 -9*
Rib	1	100	4.36 x 10 -10
Spar	1	100	4.36 x 10 -10
Stringer	3	100	1.31 x 10 -9
Wing Center Section			
Rib	0	21	0
Spanwise beam	4	21	8.32 x 10-9
Stringer	0	21	0

* estimated

Fatigue Crack Initiation and Growth: 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. A method for the estimate of the expected time to first failure is outlined by Freudenthal (Reference 9). In his paper, Freudenthal shows that the cumulative distribution function has the form of the two-parameter Weibull:

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

where t = time to crack initiation

θ = characteristic value

b = shape parameter

By pooling the results from various sources (Figure 9), Freudenthal made an estimate of fatigue life scatter based on the representation of test data by the log-normal distribution. He concluded that a value of the standard deviation σ ($\log_{10}N$) = 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 10) in which he concludes that a value of σ = 0.14 would describe the basic fatigue life scatter and that a value of σ = 0.20 would account for basic scatter and the additional scatter introduced by load environment variation. Using the actual average fatigue life (t) previously generated and assuming a value for σ, the values of θ and b can be determined from the following relationship:

$$b = \pi / (2.303 * \sigma * (6)^{1/2})$$

$$\theta = t / \Gamma(1 + 1/b)$$

Where Γ = gamma function

A value of σ = 0.14 was assumed as scatter in the load environment was accounted for in Section 4.6.1 above.

Monte Carlo picks of a random number RN were then used in the following equation to generate the time to the 1st and 2nd element fatigue crack initiations. The resulting distributions are shown in Figure 10 for the first three cracks, but due to the complexity, the program is limited to the first two cracks.

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

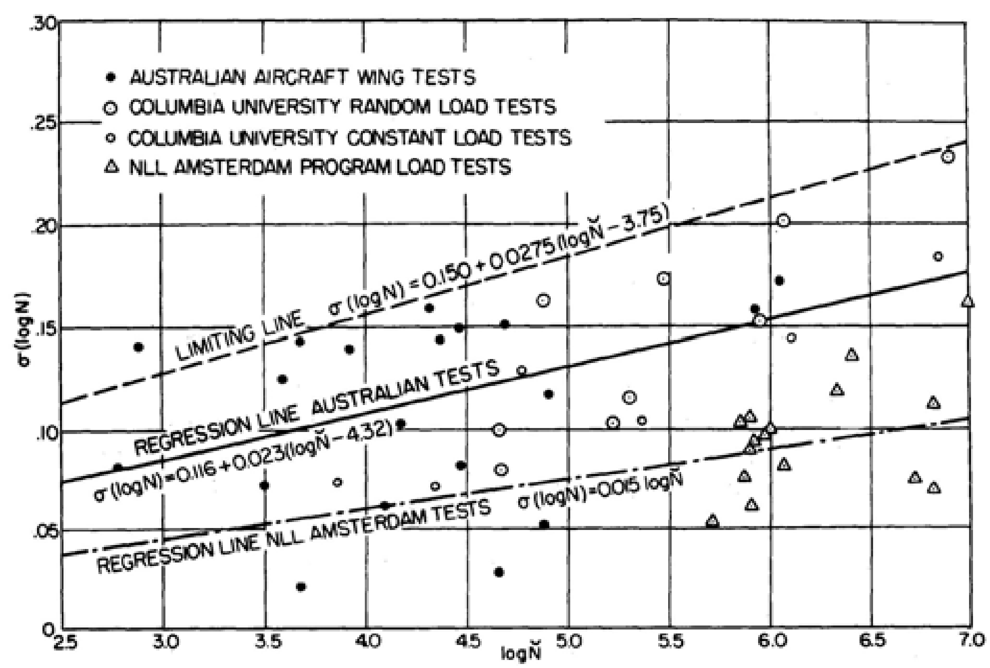


Figure 9. Relation Between σ($\log_{10}$ N) and log N ave.

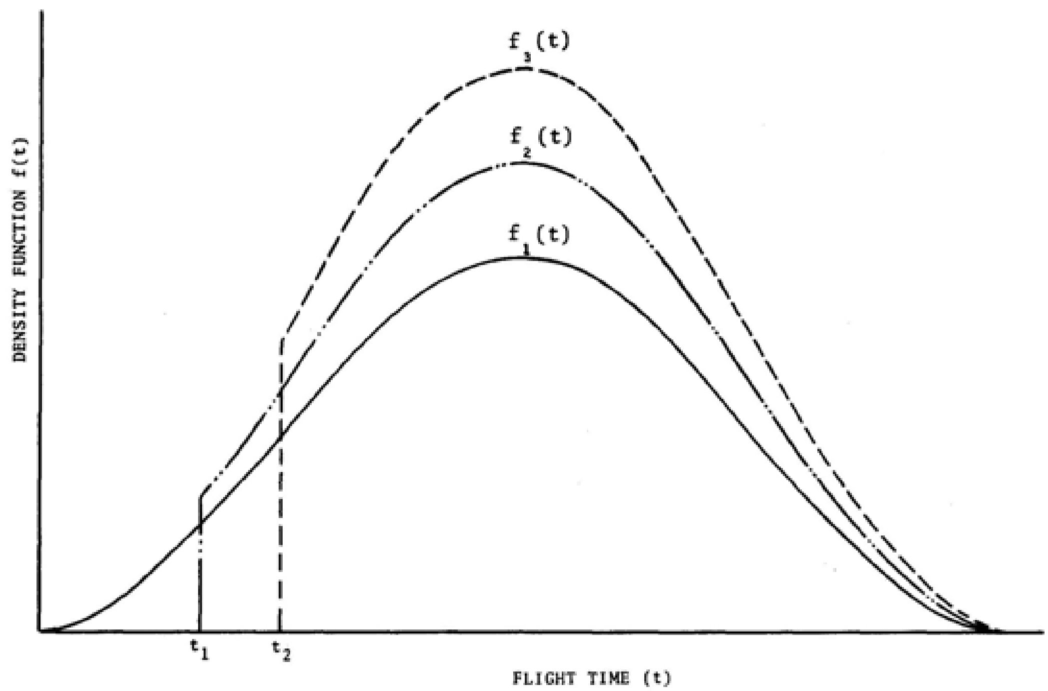


Figure 10. Truncated Fatigue Life Distribution for First, Second and Third Cracks in an Individual Element

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. Reference 12, characterizes the variability of crack initiation and growth in distributional forms. In the simulation all crack growth rates are assumed to be normally distributed with means as previously in shown 4.5 above. The standard deviation is set equal to one half that used in the time to crack initiation distribution. Both of these parameters, the mean crack growth rates, are provided in the element input.

The crack growth rates for each individual element on each aircraft in the fleet are selected by a Monte Carlo method then adjusted for the load environment factor of 4.6.1 above.

Whether a crack initiates internally or externally, is important from an inspection standpoint. The probability of a crack initiating externally was determined for each element type from MRR/SDR data and given in Table 3. A Monte Carlo pick from a uniform distribution determines whether the crack initiated externally.

TABLE 3. PROBABILITY OF A CRACK BEING EXTERNAL

Structural Element Type	No. of Total Occurrences	No of. 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

Corrosion Initiation and Growth: 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/SDRs. The first step in the formulation of the corrosion occurrence rate is a determination of the corrosion growth rate. For this determination, the size of the reported corrosion and the flight hours at detection were analyzed to estimate a nominal growth rate. The corrosion rate for each element type was multiplied by its' "corrosion resistance rating" (CRR) "adjustment Factor". The following CRRs were established by the MSG-2 conference convened for the certification of each aircraft type.

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

With the growth rate known, the MMR/SDR data was converted to the Corrosion Initiation Occurrences vs. Flight Hours for each element type as shown in Figure 11.

The occurrence rate for a fleet of 1406 aircraft is then derived from this data. The fleet size data were obtained from the FAA statistical handbook of aviation for 1970 and 1973 (References 8 and 13).

$$P_C = \frac{\Delta \text{Occurences}}{\Delta \text{ Flight Hours} * 1406 * \text{No. of Individual Elements}}$$

With the rate known (λ), a Monte Carlo pick from a uniform distribution produces RN which is used in the following equation to determine the time to corrosion initiation for each element.

$$t = l/\lambda \ln(RN)$$

As shown by the Figure 11curve below, typically corrosion has two rates, so two rates are provided in the input data.

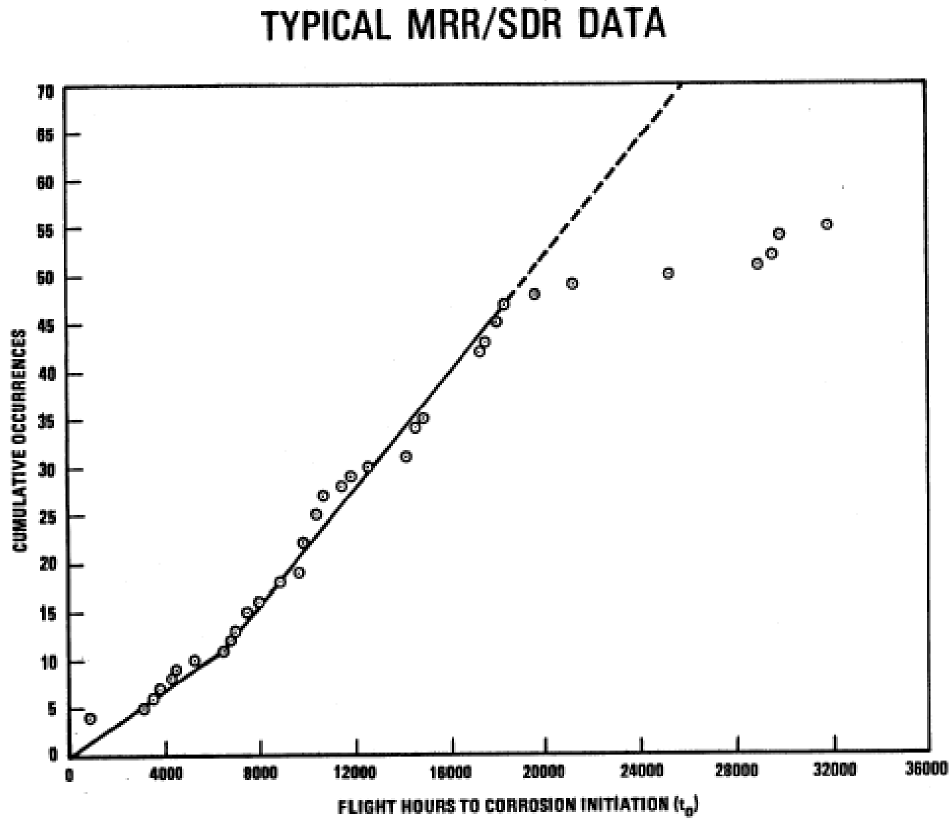


Figure 11. Corrosion Initiation Occurrences vs. Flight Hours

Whether corrosion initiates internally or externally, is important from an inspection standpoint. The probability of a corrosion initiating externally was determined for each element type from MRR/SDR data and given in Table 4 A Monte Carlo pick from a uniform distribution determines whether the corrosion initiated externally.

TABLE 4. PROBABILITY OF CORROSION BEING EXTERNAL

Structural Element Type	No. of Total Occurrences	No. of 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
Pressure web	257	184	0.716
Window frame	0	0	0.050*
Stringer	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*
Stringer	18	6	0.333
Spanwise beam	92	32	0.348

4.6.3 Account for Corrosion Effects

When corrosion initiates in the element, it shortens the time to fatigue crack initiation and increases fatigue crack growth rates. The amount of the change depends on whether the corrosion initiates in a stress concentration. So for the model to make the appropriate change, it must predict whether it initiates in a stress concentration. 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.

The corrosion occurrences documented in the MRR/SDRs 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).

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 5.

TABLE 5. 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 frame	13	5	0.385
Window frame	23	3	0.130
Main frame	3	2	0.385*
Floor beam	59	1	0.017
Keel beam	5	0	0.017*
Pressure web	1	0	0.136*
Stringer	257	55	0.136
Wing			
Access frame	0	0	0.385*
Rib	9	1	0.111
Spar	57	3	0.053
Stringer	3	3	0.056*
Wing Center Section			
Rib	0	0	0.056*
Spanwise beam	18	1	0.056
Stringer	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.

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 14 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. Reference 14 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.

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.

Corrosion also increases fatigue crack growth rates. The literature supports using the same factors for crack initiation for adjustment of fatigue crack growth rates.

References 15, 16, and 17 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 17) 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 previously defined and the detection probabilities defined indicates that corrosion will likely be detected before any detectable reduction in static strength has taken place. Reference 3 supports this finding with regard to detection.

Reference 18 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 are 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 minimized corrosion as a major factor in catastrophic accidents.

Consequently, when corrosion initiates, the only change in strength, is that the model changes fatigue crack initiation and crack growth rates. However, if a fatigue crack is present; it calculates the risk (see explanation in 4.6.4 below) incurred by the crack up to that time. Also, the subsequent corrosion growth affects its' probability of detection and possible affect on inspection changes and/or modification

4.6.4 Account for Fatigue Cracks Effect on Strength and Safety

When fatigue cracks initiate (at the scheduled time). A Monte Carlo pick from a uniform distribution determines whether it initiated internally. The probability of being external was determined from a review and analysis of the MRR/SDR data for each element type. The results were shown in Table 3 above. If it is internal, the time to become external is calculated and scheduled. Also the times to reach fail safe length and a length that corresponds to failure with a residual strength of 1g (delta g = 0) are calculated and scheduled. If it is a second crack in the element, the probability of the element failing prior to this time is calculated. (This calculation is also made when corrosion initiates, or a repair is made or a modification is installed, or the total crack length reaches fail safe length, or residual strength is 1g, or the airplane is retired)

Reduction in Strength: Strength is reduced because of crack growth. For the simulated structural elements, it is assumed that the original ultimate strength, Su, 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 4.5, the four-part approximation to the growth rate of a single fatigue crack has the general configuration shown in Section 4.5 and Figure 5 above. It will also be assumed that the relationship between crack length and residual strength, S, is a multipart linear curve as shown in Figure 12, with S going to 1.0 at the crack length corresponding to level flight failure. If it is assumed that residual strength is a linear relationship with crack length and the strength at fail safe length known, four part linear curve of flight hours vs. residual strength can be developed as shown in curve Figure 13.

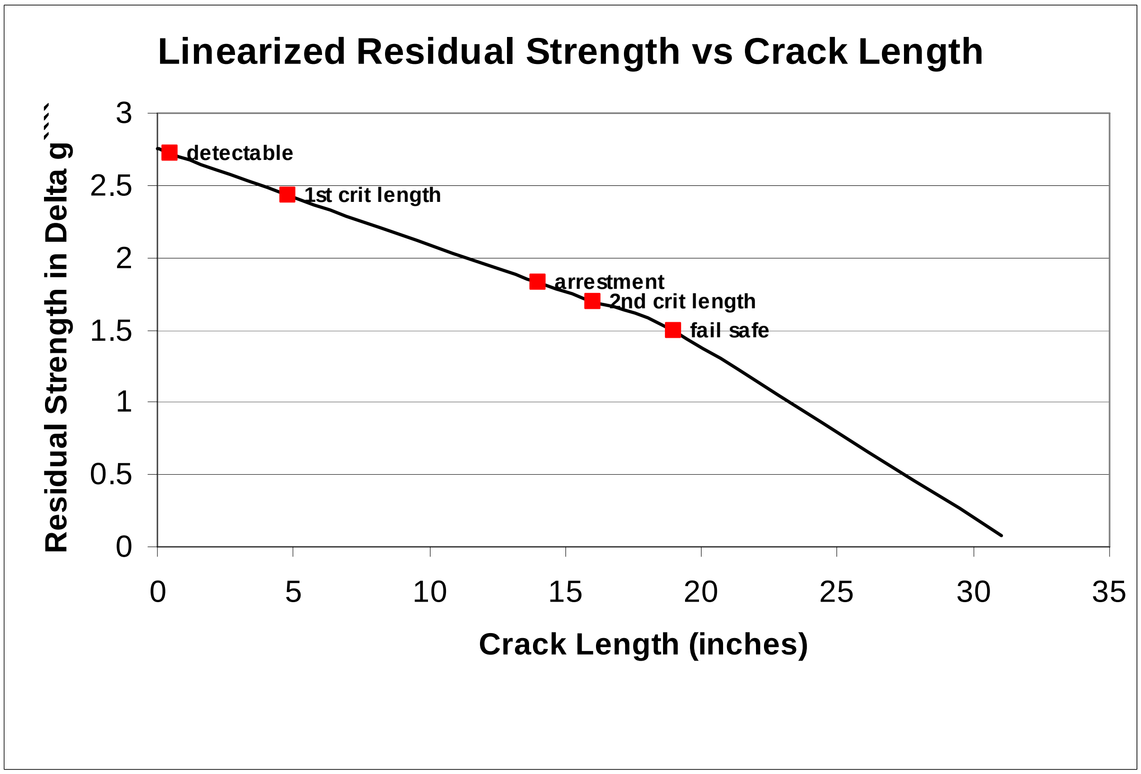


Figure 12. Residual Strength vs. Crack Length

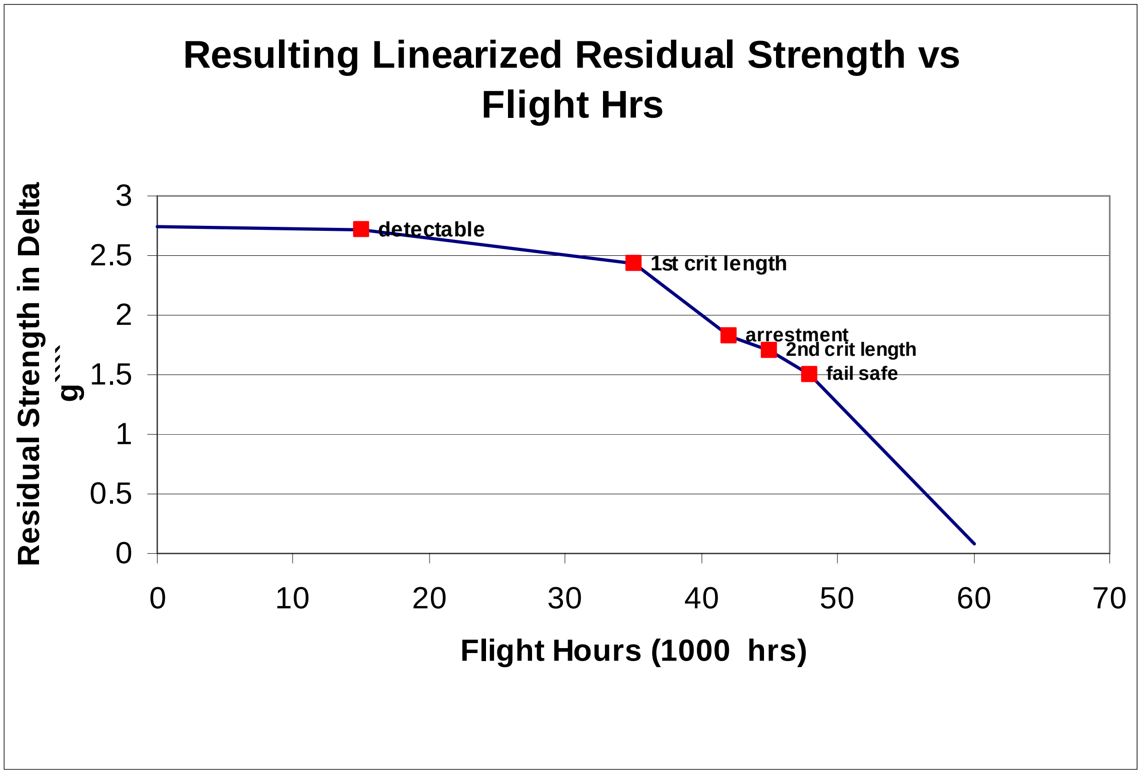


Figure 13. Residual Strength vs. Flight Hours

If there is a second crack initiation, the growth rate of the sum of the two crack growth rates is calculated. If a third crack initiates, the same procedure could be repeated.

Probability that Load Exceeds Reduced Strength: This 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 19 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 0.1 g 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:

$$\mathbf{P(S_a) = A\ exp[bS_a]}$$

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 14. 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. The resulting average exceedance rates are adjusted to account for the variation in spectrum severity between individual airplanes.

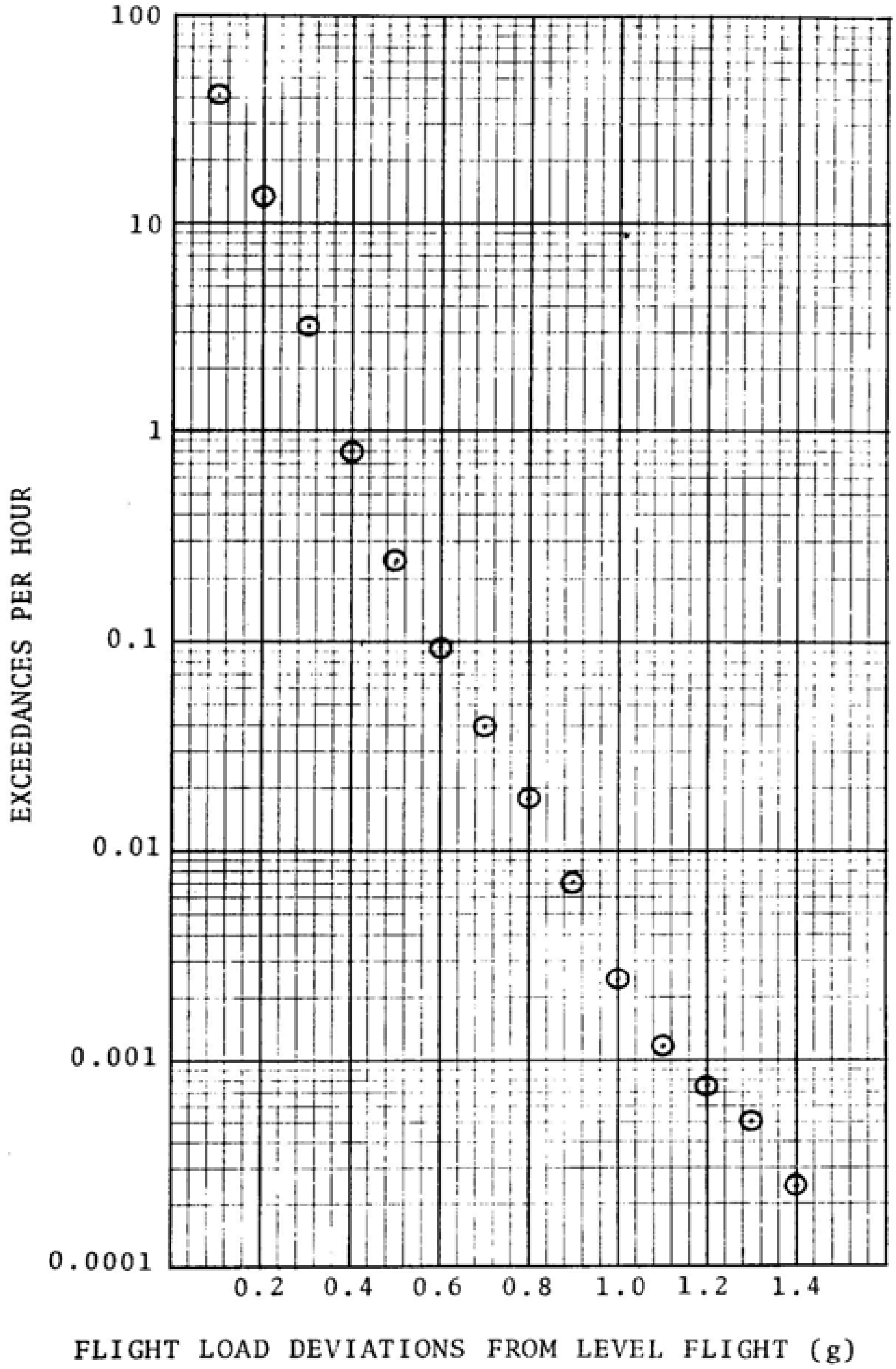


Figure 14. Flight Load Exceedances

The following equation defines the residual strength for each segment up to the time of the calculation as illustrated in Figure 13 above, where the initial strength is S_u the ultimate strength:

$$\mathbf{S(t_i) = S_{i-1} - R_i\,(t_{i-1} - t_i)}$$

Where **t_i = time at end of segment i**
R_i = strength degradation rate of segment i
$S(t_i)$ = strength at the end of segment i

With **$S(t_i)$** as expressed above, the number of flight loads per hour that exceed the residual strength at time **t_i** is

$$\mathbf{P\,[S(t_i)] = A \exp\,[bS_i - R_i\,(t_{i-1} - t_i)]}$$

Adopting the same assumptions as Lundberg and Eggwertz (Reference 20), the above expressions for the load exceedance of residual strength can be substituted for the risk functions λ (t) in the reliability formula:

$$\mathbf{F(t) = 1 - \exp\,[-\int_0^t - \lambda\,(t)\,dt\,]}$$

Making the above substitutions yields

$$\mathbf{F(t_{i-1} - t_i) = 1 - \exp\,\{-\int_{ti-1}^{ti} A \exp\,[bS_i - R_i\,(t_{i-1} - t_i)]\,dt\}}$$

$F(t_{i-1} - t_i)$ represents the reliability degradation over the period of segment **$t_{i-1} - t_i$.**
These degradations are added up over all the segments to the time of the calculation to give the probability of element failure F (t) due to the crack up to the time t of the calculation.

When there is a second crack initiation, a new probability of failure is calculated with (t – 0) corresponding to the time of the second crack initiation with the initial strength equal to the residual strength at that time due to the first crack. The same procedure could be repeated for a third crack initiation.

4.6.5 Account for Inspection, Repair, Inspection Change and/or Modification

Inspection: Inspections are made at four different levels as scheduled and as rescheduled as test and service experience deems. Detection is base on the appropriate probability of defect detection (PODD). If defects are found, a decision to whether to implement a modification is made based on cost of the alternative to repair and inspection. In any case the defect is repaired or modified. A decision is also made on whether the inspection program is to be changed, based on the severity of this defect and the severity and number of those found on other aircraft in the fleet.

4.6.5 Inspection, Repair, Risk Incurred, Inspection Change and/or Modification

Inspection: 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 with typical initial intervals, were defined as follows with the help of a survey of approximately 50 FAA inspectors:

A- Check, 25 hours: Visual inspection conducted from ground level and primarily covering lower exterior wing and fuselage surfaces.

B- Service, 300 hours: 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 1000 hours: 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, 3000 hours: 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-leve1 inspections.

Sampling inspections are accounted for in the D-level inspection logic in that a specified number of high time airplanes receive an internal inspection. The logic 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.

The size of the defect(s) is calculated and if initiated internally, it has previously been determined whether it has become external. If the inspection is only an external inspection, it can only detect any external portion of the defect and only the external size in considered. The probability of detection depends on the size of the defect and the level of inspection.

The PODD curves in Figures 15 and 16 were developed by an extensive review and analysis the MRR/SDRs for large civil transports for a ten year period from 1964 to 1974. It was primarily based on reports from B747, B 727, B720, B707, DC8 and DC9 aircraft. It also included the larger defects from accidents and incidents. It considered size of the defect when detected, the inspection schedule at the time of detection and the estimate defect growth rate. The PODD is the ratio of the number defects found of at a given size (at a given inspection level) to the number of defects found and missed of the same size at that inspection level.

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.

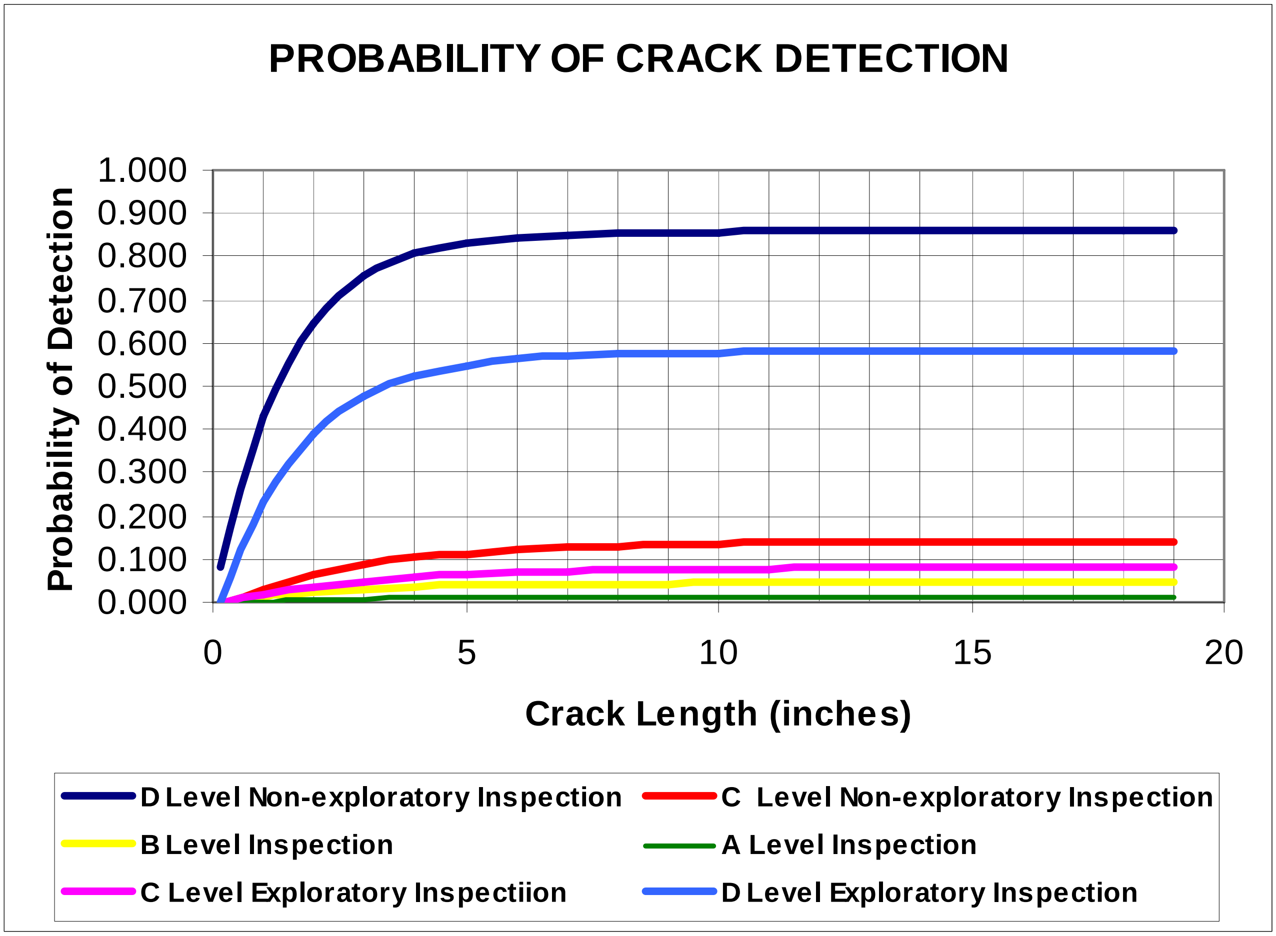


Figure 15. Probability of Crack Detection

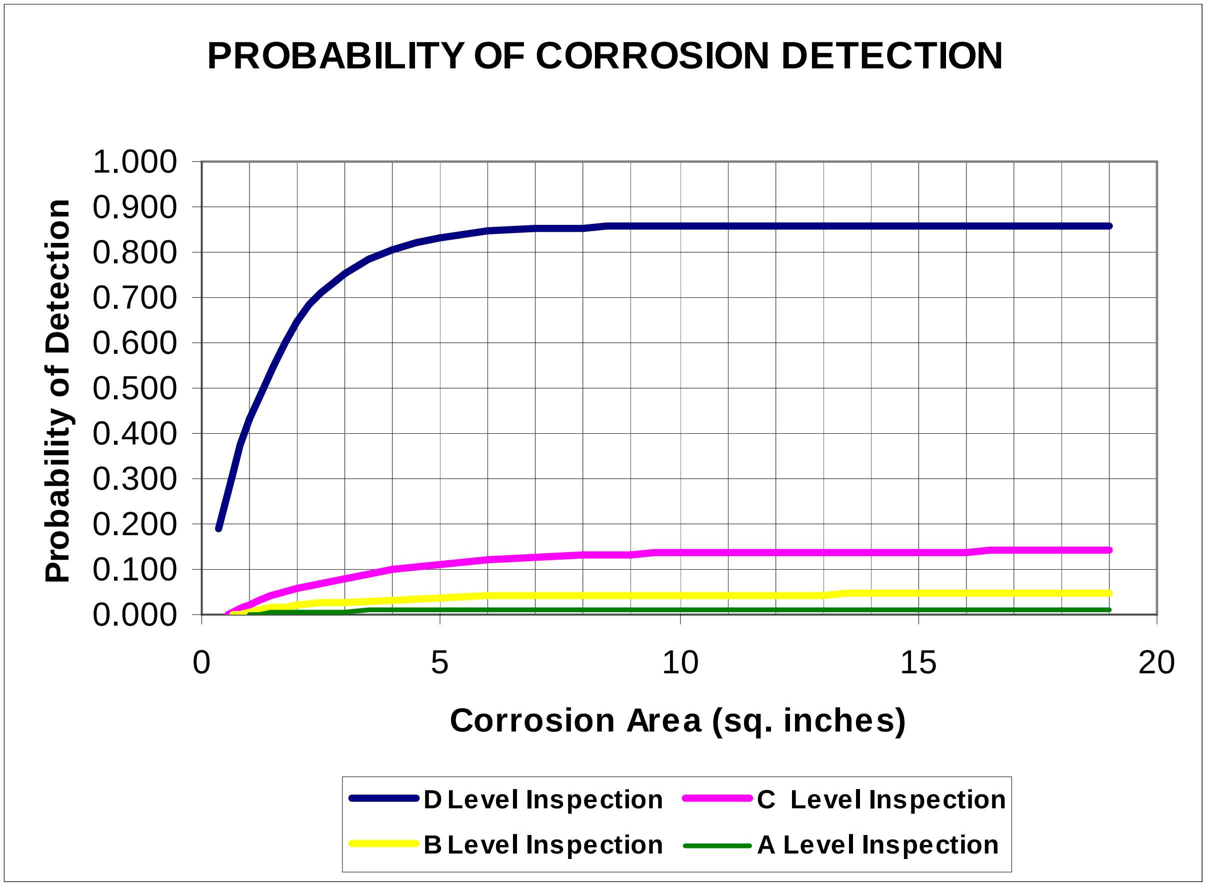


Figure 16. Probability of Corrosion Detection

Repair: When a defect is found 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. If corrosion existed, its' acceleration of crack initiation and growth is removed. However, in-service defects (fatigue cracks, service damage, and corrosion) predicted to initiate after the repair is accomplished are not affected by the repair process and they are allowed to initiate at their originally determined times.

As discussed above, 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.

Risk Incurred: As discussed under *Probability that Load Exceeds Residual Strength* in 4.6.4 above, an estimate is made of risk incurred by the repaired crack(s) prior to repair. This is expressed as the probability of airframe failure due to crack(s) that initiated in the element.

Inspection Change: 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 or on any inspection on any aircraft, 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.

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 is 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.

The criteria defined above 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 determine whether or not a modification to the element is required.

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 approximately equal to that of a non-exploratory D level inspection and is internal if the defects generating the special inspection were found in an internal inspection.

After a special fleet inspection is completed, the magnitude of the defects found is compared with the first and second criteria defined above, and a second reduction in inspection frequency may be instituted as a result of the special inspection.

Modification: When each aircraft enters service or when a defect is found in service, it is determined 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 determine whether the modification is available for installation. It is determined whether or not a prescribed structural modification has been installed in an aircraft before it enters service. The modification has been installed if

$X4 * C2 > X8 + C11 + C12$ (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).

Calendar days * 8.2 = equivalent simulation time

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.

If a defect is found in service and no modification is pending, the program examines service experience to date and makes a decision on whether to implement a modification. The SAIFE logic bases the decision solely on economic considerations. It assumes that each approach provides adequate safety. 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 6. The values shown in Table 6 were determined from a report (Reference 21) 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.

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

$$C_{MOD} < C_{REPAIR} + C_{INSP}$$

where
C_{MOD} = modification cost per flight hour

C_{REPAIR} = repair cost per flight hour

C_{INSP} = increased inspection cost per flight hour

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

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 4.3. 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).

If element input data specifies that any modification is to be fatigue tested, 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.

TABLE 6. ECONOMIC PARAMETERS REQUIRED FOR DEVELOPING A SERVICE MODIFICATION

	Inspection Level			
	A	B	C	D
Man- Hrs/Insp.	24*	472*	861*	1250
NDI Cost/Insp.	-	-	$618*	$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

4.6.6 Failure or Retirement and Risk Incurred.

The simulation of the life an element continues through the life of each airplane until failure or retirement. Failure at this time is defined as the residual strength has been reduced to 1 g (delta g = 0). At failure, the airplane is removed from the active fleet. At this time, the simulation estimates the risk incurred by cracks individual elements as in 4.6.4; displays the element results After all active airplanes are accounted for, and moves to the next element. When all elements of type are completed, it estimates the risk incurred by the element type, expressed as a failure rate and displays the element type results. When all elements are completed, it estimates the risk incurred by the fleet, expressed as a failure rate and displays the fleet results.

4.7 Display Results.

At completion of the simulation, the results are displayed as shown for a sample run of 4 elements in Table 7 (sheets 1 through 6). Sheets 1 through 4 give the results after simulation of each element is completed. Sheet 5 gives the summary results after simulation of all elements of a given type is completed and sheet 6 gives the fleet results after simulation of all elements at the end of the simulation.

TABLE 7 Sheet 1

AIRCRAFT TYPE: 747DC10

NUMBER OF AIRCRAFT IN FLEET: 500 AIRCRAFT SERVICE LIFE: 60000 HOURS

STRUCTURAL ELEMENT: FUS-MFR-SID-0540

PREDICTED AVERAGE FATIGUE LIFE: 422910 HOURS ACTUAL AVERAGE FATIGUE LIFE: 318594 HOURS
FATIGUE TEST LIFE:9999999 HOURS

NUMBER AND TIME TO INITIATION OF AIRCRAFT DEFECTS

	FIRST CRACK	CORROSION	SERVICE DAMAGE	PRODUCTION DEFECTS
OCCURRENCES	1	0	1	0
MIN(HRS)	53423	0	53423	-----
MAX(HRS)	53423	0	53423	-----
AVG(HRS)	53423	0	53423	-----

NUMBER AND LENGTH OF CRACKS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	0	0
MIN(IN)	0.	0.	0.	0.	0.
MAX(IN)	0.	0.	0.	0.	0.
AVG(IN)	0.	0.	0.	0.	0.

NUMBER AND AREA OF CORROSION DEFECTS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	0	0
MIN(SQ.IN)	0.	0.	0.	0.	0.
MAX(SQ.IN)	0.	0.	0.	0.	0.
AVG(SQ.IN)	0.	0.	0.	0.	0.

INSPECTION INTERVALS(HRS)					MOD NO	SAMPLING	TIME
INITIAL	25	315	1000	1600	0	30	
2	25	315	1125	2400	0	21	2200
3	25	315	1266	3600	0	15	4600
4	25	315	1424	5400	0	11	8200
5	25	315	1602	8100	0	8	13600
6	25	315	1802	12150	0	6	21700
7	25	315	2253	15188	0	7	33850
8	25	315	2816	18984	0	8	49038
9	25	315	3520	23730	0	9	71819
10	25	315	4399	29663	0	10	86865

CRACK LENGTHS AND CORRESPONDING CUMULATIVE PROBABILITY OF FAILURE

AIRCRAFT NO.	FLT. HOURS	CRK.LGT.	PROB. OF FAILURE
462	60000	.33	+2.E-014

SERVICE DAMAGE AIRCRAFT NO. 462

NUMBER OF SPECIAL INSPECTIONS CONDUCTED: 0
NUMBER OF STRUCTURAL MODIFICATIONS: 0
FINAL ACTUAL AVERAGE MODIFIED FATIGUE LIFE: 318594 HOURS
NUMBER OF AIRCRAFT MODIFIED IN SERVICE: 0
ESTIMATED ELEMENT FAILURE RATE:+6.44E-022/HR.

STRUCTURAL FAILURES

AIRCRAFT NO.	FLT. HOURS

RESIDUAL STRENGTH EQUALS FAIL-SAFE STRENGTH

AIRCRAFT NO.	FLT. HOURS

TABLE 7 Sheet 2

STRUCTURAL ELEMENT: FUS-MFR-SID-0560

PREDICTED AVERAGE FATIGUE LIFE: 434010 HOURS   ACTUAL AVERAGE FATIGUE LIFE: 163258 HOURS
FATIGUE TEST LIFE: 231536 HOURS

NUMBER AND TIME TO INITIATION OF AIRCRAFT DEFECTS

	FIRST CRACK	CORROSION	SERVICE DAMAGE	PRODUCTION DEFECTS
OCCURRENCES	6	0	0	0
MIN(HRS)	39538	0	0	-----
MAX(HRS)	58817	0	0	-----
AVG(HRS)	49843	0	0	-----

NUMBER AND LENGTH OF CRACKS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	0	0
MIN(IN)	0.	0.	0.	0.	0.
MAX(IN)	0.	0.	0.	0.	0.
AVG(IN)	0.	0.	0.	0.	0.

NUMBER AND AREA OF CORROSION DEFECTS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	0	0
MIN(SQ.IN)	0.	0.	0.	0.	0.
MAX(SQ.IN)	0.	0.	0.	0.	0.
AVG(SQ.IN)	0.	0.	0.	0.	0.

INSPECTION INTERVALS(HRS)					MOD NO	SAMPLING	TIME
INITIAL	25	315	1000	1600	0	30	
2	25	315	1125	2400	0	21	2200
3	25	315	1266	3600	0	15	4600
4	25	315	1424	5400	0	11	8200
5	25	315	1602	8100	0	8	13600
6	25	315	1802	12150	0	6	21700
7	25	315	2253	15188	0	7	33850
8	25	315	2816	18984	0	8	49038
9	25	315	3520	23730	0	9	71819
10	25	315	4399	29663	0	10	86865

CRACK LENGTHS AND CORRESPONDING CUMULATIVE PROBABILITY OF FAILURE

AIRCRAFT NO.	FLT. HOURS	CRK.LGT.	PROB. OF FAILURE
117	60000	.18	+8.E-015
209	60000	.72	+3.E-014
282	60000	.73	+5.E-014
401	60000	.34	+2.E-014
416	60000	.05	+3.E-015
437	60000	.78	+6.E-014

NUMBER OF SPECIAL INSPECTIONS CONDUCTED: 0
NUMBER OF STRUCTURAL MODIFICATIONS: 0
FINAL ACTUAL AVERAGE MODIFIED FATIGUE LIFE: 163258 HOURS
NUMBER OF AIRCRAFT MODIFIED IN SERVICE: 0
ESTIMATED ELEMENT FAILURE RATE:+5.50E-021/HR.

STRUCTURAL FAILURES

AIRCRAFT NO.	FLT. HOURS

RESIDUAL STRENGTH EQUALS FAIL-SAFE STRENGTH

AIRCRAFT NO.	FLT. HOURS

TABLE 7 Sheet 3

STRUCTURAL ELEMENT: FUS-MFR-SID-1720

PREDICTED AVERAGE FATIGUE LIFE: 158130 HOURS   ACTUAL AVERAGE FATIGUE LIFE: 102233 HOURS

FATIGUE TEST LIFE:9999999 HOURS

NUMBER AND TIME TO INITIATION OF AIRCRAFT DEFECTS

	FIRST CRACK	CORROSION	SERVICE DAMAGE	PRODUCTION DEFECTS
OCCURRENCES	14	1	0	0
MIN(HRS)	15005	2829	0	-----
MAX(HRS)	54643	2829	0	-----
AVG(HRS)	38023	2829	0	-----

NUMBER AND LENGTH OF CRACKS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	1	1	8
MIN(IN)	0.	0.	6.67	.90	1.39
MAX(IN)	0.	0.	6.67	.90	5.29
AVG(IN)	0.	0.	6.67	.90	2.91

NUMBER AND AREA OF CORROSION DEFECTS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	1	0
MIN(SQ.IN)	0.	0.	0.	34.84	0.
MAX(SQ.IN)	0.	0.	0.	34.84	0.
AVG(SQ.IN)	0.	0.	0.	34.84	0.

INSPECTION INTERVALS(HRS)					MOD NO	SAMPLING	TIME
INITIAL	25	315	1000	3200	0	15	
2	25	315	1125	4800	0	11	3800
3	25	315	1266	7200	0	8	8600
4	25	315	1424	10800	0	6	15800
5	25	315	1602	16200	0	5	26600
6	25	315	2002	20250	0	6	42800
7	25	315	2002	7088	0	17	57918
8	25	315	2002	2481	0	49	57918
9	25	315	2002	20250	1	6	59334
10	25	315	2503	25313	1	7	80648
11	25	315	3129	31641	1	8	85711

CRACK LENGTHS AND CORRESPONDING CUMULATIVE PROBABILITY OF FAILURE

AIRCRAFT NO.	FLT. HOURS	CRK.LGT.	PROB. OF FAILURE
64	54618	6.67	+3.E-012
15	57068	2.94	+2.E-013
55	55068	10.49	+9.E-012
303	32568	1.39	+8.E-014
364	26468	2.09	+2.E-013
73	54168	2.24	+2.E-013
305	32368	2.76	+2.E-013
19	59348	.80	+5.E-014
98	55398	.90	+6.E-014
106	54748	.80	+5.E-014
158	49548	.19	+8.E-015
166	48748	.75	+4.E-014
211	44248	1.42	+1.E-013
231	42248	.38	+4.E-014

NON-EXPLORATORY DETECTION LEVEL AT 57918 MODIFICATION 0
RELATIVE CONST OF CONTINUING INSPECTION & REPAIR VS MODIFICATION:
ICPH = .002 MCPH = .065 RCPH = .093 TIME = 57918
CORROSION AIRCRAFT NO. 285

NUMBER OF SPECIAL INSPECTIONS CONDUCTED: 2
NUMBER OF STRUCTURAL MODIFICATIONS: 1
FINAL ACTUAL AVERAGE MODIFIED FATIGUE LIFE: 166313 HOURS
NUMBER OF AIRCRAFT MODIFIED IN SERVICE: 495
ESTIMATED ELEMENT FAILURE RATE:+4.47E-019/HR.

STRUCTURAL FAILURES

AIRCRAFT NO.	FLT. HOURS
------------	----------

RESIDUAL STRENGTH EQUALS FAIL-SAFE STRENGTH

AIRCRAFT NO.	FLT. HOURS
------------	----------

TABLE 7 Sheet 4

STRUCTURAL ELEMENT: FUS-MFR-SID-1740

PREDICTED AVERAGE FATIGUE LIFE: 157620 HOURS   ACTUAL AVERAGE FATIGUE LIFE: 343268 HOURS
FATIGUE TEST LIFE:9999999 HOURS

NUMBER AND TIME TO INITIATION OF AIRCRAFT DEFECTS

	FIRST CRACK	CORROSION	SERVICE DAMAGE	PRODUCTION DEFECTS
OCCURRENCES	3	0	2	0
MIN(HRS)	2615	0	2615	-----
MAX(HRS)	36817	0	30910	-----
AVG(HRS)	23447	0	16762	-----

NUMBER AND LENGTH OF CRACKS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	2	0	1
MIN(IN)	0.	0.	3.97	0.	2.49
MAX(IN)	0.	0.	7.62	0.	2.49
AVG(IN)	0.	0.	5.80	0.	2.49

NUMBER AND AREA OF CORROSION DEFECTS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	0	0
MIN(SQ.IN)	0.	0.	0.	0.	0.
MAX(SQ.IN)	0.	0.	0.	0.	0.
AVG(SQ.IN)	0.	0.	0.	0.	0.

INSPECTION INTERVALS(HRS)					MOD NO	SAMPLING	TIME
INITIAL	25	315	1000	3200	0	15	
2	25	315	1125	4800	0	11	3800
3	25	315	1266	7200	0	8	8600
4	25	315	1424	10800	0	6	15800
5	25	315	1602	16200	0	5	26600
6	25	315	2002	20250	0	6	42800
7	25	315	2503	25313	0	7	67150
8	25	315	2503	8859	0	20	87223

CRACK LENGTHS AND CORRESPONDING CUMULATIVE PROBABILITY OF FAILURE

AIRCRAFT NO.	FLT. HOURS	CRK.LGT.	PROB. OF FAILURE
194	58262	3.97	+6.E-013
489	43273	7.62	+4.E-012
474	44773	2.49	+2.E-013

SERVICE DAMAGE AIRCRAFT NO. 489
NON-EXPLORATORY DETECTION LEVEL AT 72712 MODIFICATION 0
RELATIVE CONST OF CONTINUING INSPECTION & REPAIR VS MODIFICATION:
ICPH = 0. MCPH = .092 RCPH = .001 TIME = 72712
SERVICE DAMAGE AIRCRAFT NO. 474
RELATIVE CONST OF CONTINUING INSPECTION & REPAIR VS MODIFICATION:
ICPH = .000 MCPH = .188 RCPH = .005 TIME = 87223

NUMBER OF SPECIAL INSPECTIONS CONDUCTED: 1
NUMBER OF STRUCTURAL MODIFICATIONS: 0
FINAL ACTUAL AVERAGE MODIFIED FATIGUE LIFE: 343268 HOURS
NUMBER OF AIRCRAFT MODIFIED IN SERVICE: 0
ESTIMATED ELEMENT FAILURE RATE:+1.72E-019/HR.

STRUCTURAL FAILURES

AIRCRAFT NO.	FLT. HOURS

RESIDUAL STRENGTH EQUALS FAIL-SAFE STRENGTH

AIRCRAFT NO.	FLT. HOURS

TABLE 7 Sheet 5

SUMMARY OF STRUCTURAL ELEMENT: FUS-MFR-SID

NUMBER AND TIME TO INITIATION OF AIRCRAFT DEFECTS

	FIRST CRACK	CORROSION	SERVICE DAMAGE	PRODUCTION DEFECTS
OCCURRENCES	24	1	3	0
MIN(HRS)	2615	2829	2615	-----
MAX(HRS)	58817	2829	53423	-----
AVG(HRS)	39797	2829	28982	-----

NUMBER AND LENGTH OF CRACKS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	3	1	9
MIN(IN)	0.	0.	3.97	.90	1.39
MAX(IN)	0.	0.	7.62	.90	5.29
AVG(IN)	0.	0.	6.09	.90	2.87

NUMBER AND AREA OF CORROSION DEFECTS DETECTED AT EACH LEVEL OF INSPECTION

	A-LEVEL	B-LEVEL	C-LEVEL	D-LEVEL	SPECIAL
OCCURRENCES	0	0	0	1	0
MIN(SQ.IN)	0.	0.	0.	34.84	0.
MAX(SQ.IN)	0.	0.	0.	34.84	0.
AVG(SQ.IN)	0.	0.	0.	34.84	0.
INSPECTION INTERVALS(HRS)					
INITIAL	25	315	1000	3200	
SHORTEST	25	315	1000	1600	
LONGEST	25	315	4399	31641	

NUMBER OF SPECIAL INSPECTIONS CONDUCTED: 3
NUMBER OF STRUCTURAL MODIFICATIONS: 1
NUMBER OF AIRCRAFT MODIFIED IN SERVICE: 495

```
ESTIMATED ELEMENT TYPE FAILURE RATE USING AVG:+2.72E-017/HR
ESTIMATED ELEMENT TYPE FAILURE RATE:+9.16E-017/HR.
SAMPLE CRK. LGT. MEAN(IN)   2.13        SAMPLE STD. DEV.   2.585
CRK. LGT. VS PROBABILITY CURVE FIT CONST: A = -13.425759315491   B =    .259096533060

             STRUCTURAL FAILURES                                  RESIDUAL STRENGTH EQUALS FAIL-SAFE STRENGTH
AIRCRAFT NO.       FLT. HOURS       STA. NO.                    AIRCRAFT NO.       FLT. HOURS       STA. NO.
------------       ----------       --------                    ------------       ----------       --------


AVERAGE FLIGHT CRACKS/FAIL-SAFE LENGTH FOR THIS ELEMENT TYPE        0.      0.      0.      0.      0.
AVERAGE PRESSURE CRACKS/FAIL-SAFE LENGTH FOR THIS ELEMENT TYPE        .227    .227    .227    .227    .227
5 LARGEST FLIGHT CRACKS/FAIL-SAFE LENGTH BASED ON LGT VS PROB CURVE   0.       0.      0.     0.      0.
5 LARGEST PRESSURE CRACKS/FAIL-SAFE LENGTH BASED ON LGT VS PROB CURVE   .393    .296    .283    .275    .250
```

TABLE 7 Sheet 6

SUMMARY OF COMPLETE AIRCRAFT STRUCTURE

EST.AIRCRAFT FAILURE RATE OF FLIGHT LOADED STRUCTURE USING AVG: 0.0 /HR.

EST.AIRCRAFT FAILURE RATE OF PRESSURE LOADED STRUCTURE USING AVG:+2.72E-017/HR.

EST. AIRCRAFT FAILURE RATE OF FLIGHT LOADED STRUCTURE: 0.0 /HR.

EST. AIRCRAFT FAILURE RATE OF PRESSURE LOADED STRUCTURE:+9.16E-017/HR.

5 LARGEST ELEMENT TYPE AVERAGE FLIGHT CRACKS/FAIL-SAFE LENGTH	0.	0.	0.	0.	0.
5 LARGEST ELEMENT TYPE AVERAGE PRESSURE CRACKS/FAIL-SAFE LENGTH	.227	.227	.227	.227	.227
5 LARGEST FLIGHT CRACKS/FAIL-SAFE LENGTH BASED ON LGT VS PROB CURVE	0.	0.	0.	0.	0.
5 LARGEST PRESSURE CRACKS/FAIL-SAFE LENGTH BASED ON LGT VS PROB CURVE	.393	.296	.283	.275	.250

AIRCRAFT FAILURES

NONE

5. Model Demonstration, Parametric Study and Results

The final version of the SAIFE model was demonstrated using a complete model of one wing and one side of the fuselage (2074 elements) and a 10% sample. The results were evaluated and compared with service history. Subsequently, a parametric study was made to determine the effect of major variables and to aid in development of the initial Supplemental Structural Inspection Documents (SSIDs) for aging transports. It was also used in one case during the certification of the B-747.

5.1 Model Demonstration and Results

The complete model and a 10% sample were run and compared with service history. The results are shown Tables 8, 9 and 10.

The ratio of total number of cracks detected in the simulation to the number of cracks reported in MRRs (Column (a) of Table 8 was 2.30 and 3.02 for demonstration is considered reasonable for the following reasons. When comparing the service history with the SAIFE demonstration output, four factors that affect the two data sets must be considered. The net result of these factors should be more defects in the SAIFE output than in the service history. Of these four factors, the first three increase the number of defects presented in the demonstration output, but the fourth decreases the number of defects. The four factors are as follows:

a) The MRR/SDR data represents generally the first half of the service life of aircraft because the data were collected from the U.S. air carrier fleet while the SAIFE output represents the entire service life of all the aircraft in a given fleet.

b) Not all aircraft defects are reported in the MRR/SDR documents.

c) The service history is based on narrow-body aircraft which have fewer elements than the hypothetical wide body aircraft used in the demonstration.

d) Improved analysis techniques, design criteria, and manufacturing methods should result in fewer defects on the wide-body aircraft represented in the demonstration than on the narrow-body aircraft reported in the MRR/SDRs.

Also the higher ratio is also some-what supported by an USAF study in which it was determined by a complete structural teardown and NDI inspection on two KC-135 full scale wing fatigue tests, that only one-fifth of the cracks present were found in the normal test inspection program.

Columns (c) and (d) of Table 8 list the larger cracks experienced in the full demonstration and the sample. The sample results are based on the extrapolation method of predicting largest cracks in the complete fleet based on the distribution of the frequency and length of cracks in the sample. The agreement between the full demonstration and the sample is good, indicating that 70 to 90% of fail safe crack length would be equaled or exceeded 5 times in the life of the fleet and that cracks as large as 161% (full demo, 31 inches in side stringer) and 135% of fail-safe length (sample, 26 inches also in side stringer) would occur. These large cracks did not generate any “Certain Failures" reports, which were defined as a residual strength which has degraded to 1g (delta g = 0) in these particular simulation runs.

With regard to these large cracks, it should be noted that: The program does not consider wing fuel leaks or fuselage pressure loss from large cracks in the detection of cracks. This is in part because large cracks have existed in the past and have not been detected by these means. In the case of the wing, leakage may not occur or may be small because sealant, multiple layers and compression while on the ground. In the case of the fuselage pressure shell, leakage may be small, especially for a circumferential crack in stringer and skin from flight loads, until shortly before the final failure or skin “flapping” In both cases, typically additional sealing or replacement of seals is the usual response to the initial indication of leakage.

Columns (e) and (f) of Table 8 give the estimated failure rates based on two different estimation methods for both the full demonstration and sample. The method of column (e) merely divides the failure rates of the sample by the decimal percent of the sample. This method ignores the possibility that the larger exposure in a simulation of the complete airplane would result in longer cracks with a much higher risk of failure. The method of column (f) is based on extrapolation of the sample crack frequency, length and probability of failure to cover the complete airplane. This method is considered more realistic. The sample and full demonstration failure rate estimates are in reasonable agreement although, the sample failure rate estimates are generally lower.

Great credence should not be placed on the absolute value of the estimated failure rates because the input and relationships in the simulation are only approximate and because of the statistical nature of the simulation, the results may vary considerably from run to run. However, it is of interest to note that the simulation, which is evaluating a typical wide-body design operating under typical inspection programs and practices, predicts failure rates (5.83×10^{-10} for full demo, 2.84×10^{-10} for sample) which meet the widely accepted criteria of less than one failure in 10^{9} airplane hours. At these failure rates, no failure would be expected in the simulation covering 3×10^{7} airplane flight hours. **<u>The current service record of U. S. wide-bodies provides support for these predictions.</u>** Sample estimates of failure rates and the percent of fail-safe crack length equaled or exceeded by the 5 largest cracks will be used in the forthcoming parametric trend studies to gauge the effect on safety by varying design parameters, inspection programs and operating practices. Crack length was added as an indicator because failure rate estimates are quite volatile.

Table 9 simulation results show good agreement with MRR data in the percent of cracks detected at each inspection level. Approximately 67% of the cracks were detected in the simulation in the close or detailed inspection (overhaul and special) compared to 78% reported in the MRRs. Only 20 to 30% of the cracks were detected in area or cursory type inspections. Here again, there is good agreement between the full demonstration and sample.

Table 10 shows good correlation between the average length crack detected in the sample (1.718 inches) the full demonstration (1.515 inches) and the MRRs (1.567 inches). This correlation supports the lower inspection reliability curves, based on MRR studies, and used in the program demonstration.

The fuselage side stringers and wing lower surface center stringers dominate the failure rate prediction for flight structure and the fuselage window frames dominate the failure rate for pressure structure in the particular simulation run made for the full demonstration.

From the short list and input for fuselage side stringer at station 1100 it can be deduced that the dominating crack was initiated by service damage at 21447 flight hours on aircraft number 408, was external and grew without detection to 19.44 inches at 58255 flight hours and to 31.19 inches at retirement at 60,000 flight hours without experiencing a load in excess of 1g residual strength. A and B level inspections were not considered effective in this area and non exploratory C and D inspections were being made at 3520 to 4399 and 23,730 to 29663 hour intervals during this period with the knowledge that one crack had been found in this area. Actually with these intervals, this aircraft would have received no D inspections and only one C inspection after it reached fail safe length. The problem illustrated by this case does not lend itself to easy solution. The actual fatigue life was adequate. Possibly a more rigorous C inspection, with a maximum interval of one per year (i.e. 3000 hours) would help.

The short list computer printout for the risk dominating wing lower surface center stringer elements (stations 0543 & 0807) indicated that Station 0543 element had a marginal fatigue life (i.e., 66752 hours vs. 120,000 hours) but as indicated by a fatigue test life of 9,999,999 hours, did not have a valid fatigue test and service repair and inspection costs did rot justify a service modification; the D inspection interval was reduced as a result of service cracks but the dominate fatigue crack of 2.91 inches was never detected prior to retirement. This type of problem could be alleviated by a more complete or realistic fatigue test. Similar to fuselage stringer 1100, the dominate crack (4.23 inches) in station 0807 element was also initiated early by service damage and not detected prior to retirement under the long inspection intervals late in the program. However in this case the actual fatigue life was marginal but was not detected in a valid fatigue test and service cracks did not generate a service modification or an inspection interval reduction.

A short list computer printout for the risk dominating fuselage window frame element (station 0930) indicated that the initial element actual fatigue life was marginal. This was detected in the fatigue test thus generating a production modification but no retrofit on early service aircraft. The production modification was not fatigue tested and had a higher but still marginal fatigue life. Service cracks detected on early unmodified elements generated a double reduction in the external D inspection interval to 2481 hours but no increase in sampling as no cracks were ever found in the internal sampling inspections. Apparently two internal cracks initiated simultaneously on opposite corners on aircraft 489 and grew at twice the rate of a single crack to a total length of 6.8 inches before becoming external. These cracks were subsequently missed in several external non-exploratory D and C inspections (A and B inspections were not considered effective for this area) and grew to a total length 8.08 inches when terminated by retirement. This type of problem could be alleviated by fatigue testing of modifications, more thorough evaluation and investigation of cracks detected in service and more frequent internal sampling.

TABLE 8. SUMMARY OF SAIFE DEMONSTRATION RESULTS

Element Type	(a) SAIFE Cracks Detected/ MRR-SDR Cracks		(b) First Cracks Occurring/ Cracks Detected		Length Equaled or Exceeded by 5 Longest Cracks		(d) Fail-Safe Crack Occurrences	
	Full	Sample	Full	Sample	Full	Sample	Full	Sample
Door Frame	3.02	14.90	1.49	1.36	26.60	–	0	
Window Frame	8.67	19.36	2.75·	2.28	38.95	–	0	
Fuselage								
- Main Frame, Bottom	5.78	11.05	1.76	1.53	15.45	–	0	
- Main Frame, Side	3.62	6.22	2.17	1.93	15.28	–	I0	
- Main Frame, Top	1.74	8.18	3.00	2.13	14.64	–	0	
- Stringer, Bottom								
- Stringer, Side	1.74	1.60	3.10	2.73	41.00	–	2	3
- Stringer, Top	1.39	2.12	3.07	3.00	23.30	--	0	
Wing								
- Access Frame	1.73	1.20	2.89	4.67	31.83	–	0	
- Spar, Aft	0.33	0.96	1.56	2.36	13.59	–	0	
- Spar, Center	83.80	19.33	2.13	8.06	22.45	–	0	
- Spar, Forward	0.00	0.00	--	0.00	2.19	–	0	
- Stringer, Aft	5.78	2.31	2.53	3.26	46.15	–	0	
- Stringer, Center	3.37	3.91	3.05	3 38	53.31	–	0	
- Stringer, Forward	0.30	0.45	9.07	8.44	26.00	---	0	
							0	
Wing Center Section								
- Stringer, AFt	4.00	0.00	5.56	--	18.15	--	0	
- Stringer, Center	0.00	0.00	---	--	7.15	--	0	
- Stringer, Forward	0.00	0.00	-----	--	1.15	--	0	
- Spanwise Bean, Aft	0.00	1.94	3.18	1.86	21.56	–	0	
- Spanwise Beam, Center	0.00	0.00	----	---	1.15	–	a	
- Spanwise Beam, Forward	1.88	0.00	--	--	1.25	-	0	
Pressure Loaded Total	4.57	4.12	1.92	1.81	49.70	41.40	0	3
Flight Loaded Total	1.72	2.07	3.29	3.13	67.25	86.25	2	
Total	2.30	3.02	2.74	2.51			2	3

TABLE 8 (Continued)

Element	(e) Estimated Failure Rate Using Average: Full	Sample	(f) Estimated Failure Rate: Full	Sample	(g) Actual Failure Rate: Full
Door Frame	2. 54E-15	3. 58E-15	6.70E-15	2.01E-13	*
Window Frame	5.02E-14	1. 78E-14	1. 16E-11	3.90E-14	
Fuselage					
- Main Frame, Bottom	4. 54E-18	6.47E-18	4. 54E-18	1.08E-15	
- Main Frame, Side	9.82E-18	9.49E-14	1. 18E-16	1.84E-14	
- Main Frame, Top	6.70E-18	2. 17E-17	6.70E-18	2.85E-16	
- Stringer, Bottom					
- Stringer, Side	1. 61E-11	2.55E-13	3.63E-10	2. 43E-I0	
- Stringer, Top	2.45E-16	1. 61E-17	2.45E-16	8.60E-17	
Wing					
- Access Frame	3.98E-12	2.90E-12	4. 34E-12	3. 82E-12	
- Spar, Aft	8. 55E-13	1.30E-12	1.09E-12	1. 44E-12	
- Spar, Center	1. 85E-11	6.97E-12	6. 19E-11	1. 12E-11	
- Spar, Forward	1. 95E-14	0.00E-00	1. 61E-14	0.00E-00	
- Stringer, Aft	3.14E-12	2.80E-12	8.35E-12	3.99E-12	
- Stringer, Center	4.64E-12	1.22E-11	1. 11E-10	1. 64E-11	
-Stringer, Forward	4. 63E-13	3.08E-12	2.04E-12	3. 44E-I'2	
Wing Center Section					
- Stringer, Aft	7.81E-13	3.08E-14	7. 57E-13	0.00E-00	
- Stringer, Center	2.90E-14	1. 49E-15	2.60E-14	0.00E-00	
- Stringer, Forward	5.07E-15	0.00E-00	4.37E-15	0.00E-00	
- Spanwise Beam, Aft	1. 18E-12	3.49E-13	5.86E-12	9.88E-13	
- Spanwise Beam, Center	1. 54E-13	1. 94E-13	1. 38E-13	0.00E-00	
- Spanwise Beam, Forward	7.39E-14	4. 69E-15	5.83E-14	0.00E-00	
Pressure Loaded Total	4.80E-14	1. 03E-14	6. 26E-13	4.23E-14	
Flight Loaded Total	6. 71E-11	3.02E-11	7. 51E-I0	2. 84E-I0	
Total	5.00E-11	3.02E-11	5.83E-I0	2.84E-I0	

* Note: No actual failures occurred in demonstration run.

TABLE 9.

COMPARISON OF CRACKS DETECTED AT EACH INSPECTION LEVEL PER MILLION FLIGHT HOURS

	FULL		SAMPLE		MMR/SDR	
	Cracks Detected	% of Total	Cracks Detected	% of Total	Cracks Detected	% of Total
A Check	24.87	9.56	25.34	7.82	2.87	4.3
Service	20.89	8.03	20.81	6.42	7.93	11.8
Phase	28.49	10.95	29.86	9.22	10.94	16.3
Overhaul	147.24	56.59	200.45	61.87	24.21	36.1
Special	38.69	14.87	47.51	14.66	21.14	31.5
Total	260.18	100.00	100.00	321.98	67.09	
100.00						

TABLE 10. COMPARISION OF SIZE OF CRACKS DETECTED

	FULL Average Length (inches)	SAMPLE Average Length (inches)	MMR/SDR Where Reported (inches)
A Check	1.573	1.943	----
Service	1.719	1.812	----
Phase	1.688	2.505	---
Overhaul	1.375	1.467	---
Special	1.771	1.812	---
Fuselage Total	1.741	1.815	1.99
Wing Total	1.118	1.470	2.16
Total	1.515	1.718	2.089 (1.567)*

* All reports assuming 5/8” length when not reported

5.2 Model Parametric Study, Use and Results

The complete model and a 10% sample were run in a parametric study to determine the effect of major variables and to aid in development of the initial Supplemental Structural Inspection Documents (SSIDs) for aging transports. It was also used in one case during the certification of the B-747.

Variation in Usage Life: The usage life was varied from the base case planned life (60,000 hrs) and the log of flight structure and pressure structure failure rates were plotted against the usage life in Figure 17. The base case (60,000 hrs in this case) failure rate value was the log mean of 3 runs. This procedure was used on throughout this section.

The failure rate seems to become asymptotic at high usage lives, possibly because of the rate of high loads. A review of the detailed results indicated that the crack growth time was a major factor. The implications were that the wide-body safety level was satisfactory for the planned life (60,000 hrs) However, the safety level with normal practices was inadequate for extended use beyond the planned life.

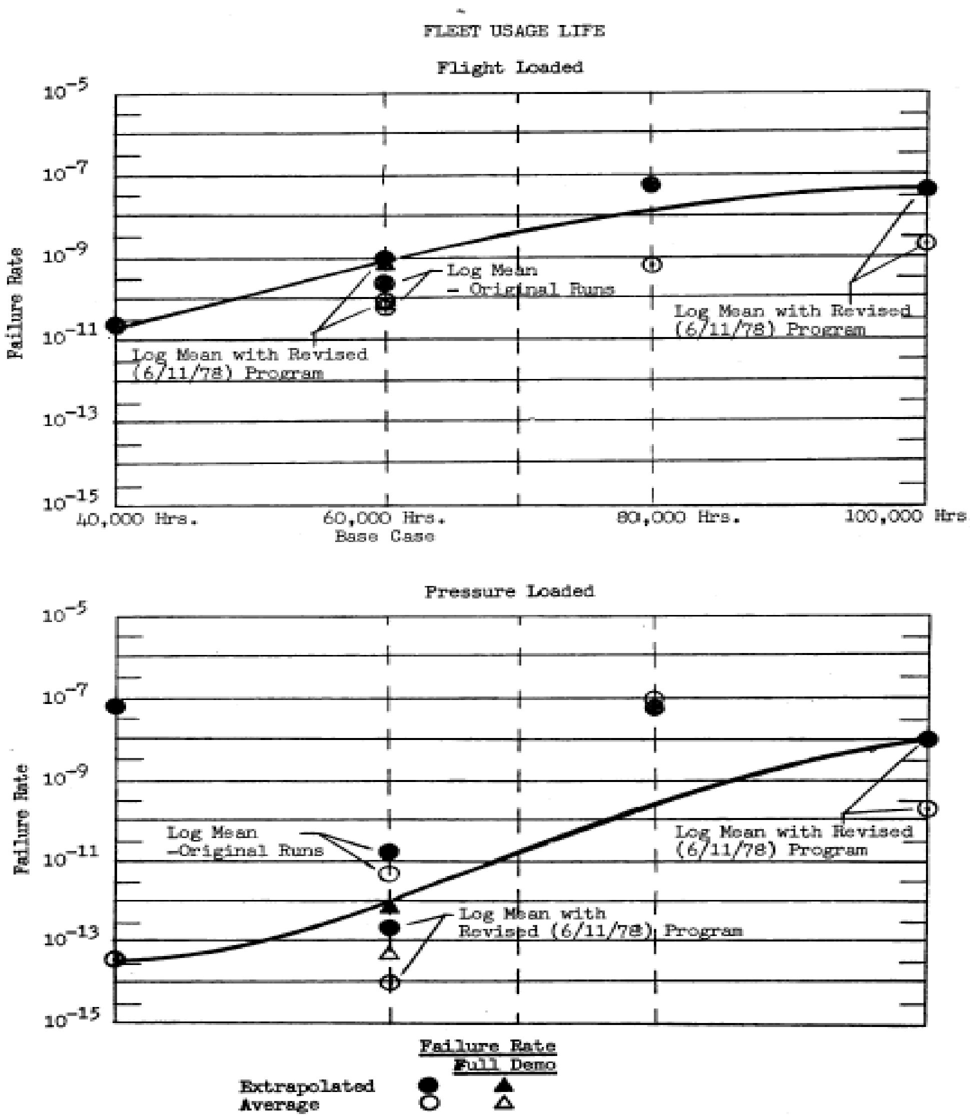


Figure 17. Variation in Usage Life

Action on Old Age Aircraft: The base case was a usage life of 100,000 hours with normal with a normal practices and inspections. Two possible courses of action to permit extension beyond the planned usage life (60,000) were evaluated. In the one a special complete internal and external inspection was performed at the end of the planned usage life (60,000 hrs). In the other, and audit was made at the planned usage life (60,000 hrs) and the D check was subsequently limited to 15,000 hours in all areas, with internal NDT inspections in areas of low fatigue life with poor deductibility. Thee runs were made in all three cases. The results are shown in Figure 18.

A review of the results indicated that perhaps runs with corrective action underestimated their effectiveness. However, the audit approach seemed more effective and was in large part incorporated into the initial SSIDs. The implications were that the safety level with a normal program was inadequate for 100,000 hours and that the corrective actions evaluated would perhaps provide an adequate safety level. More runs would be use full.

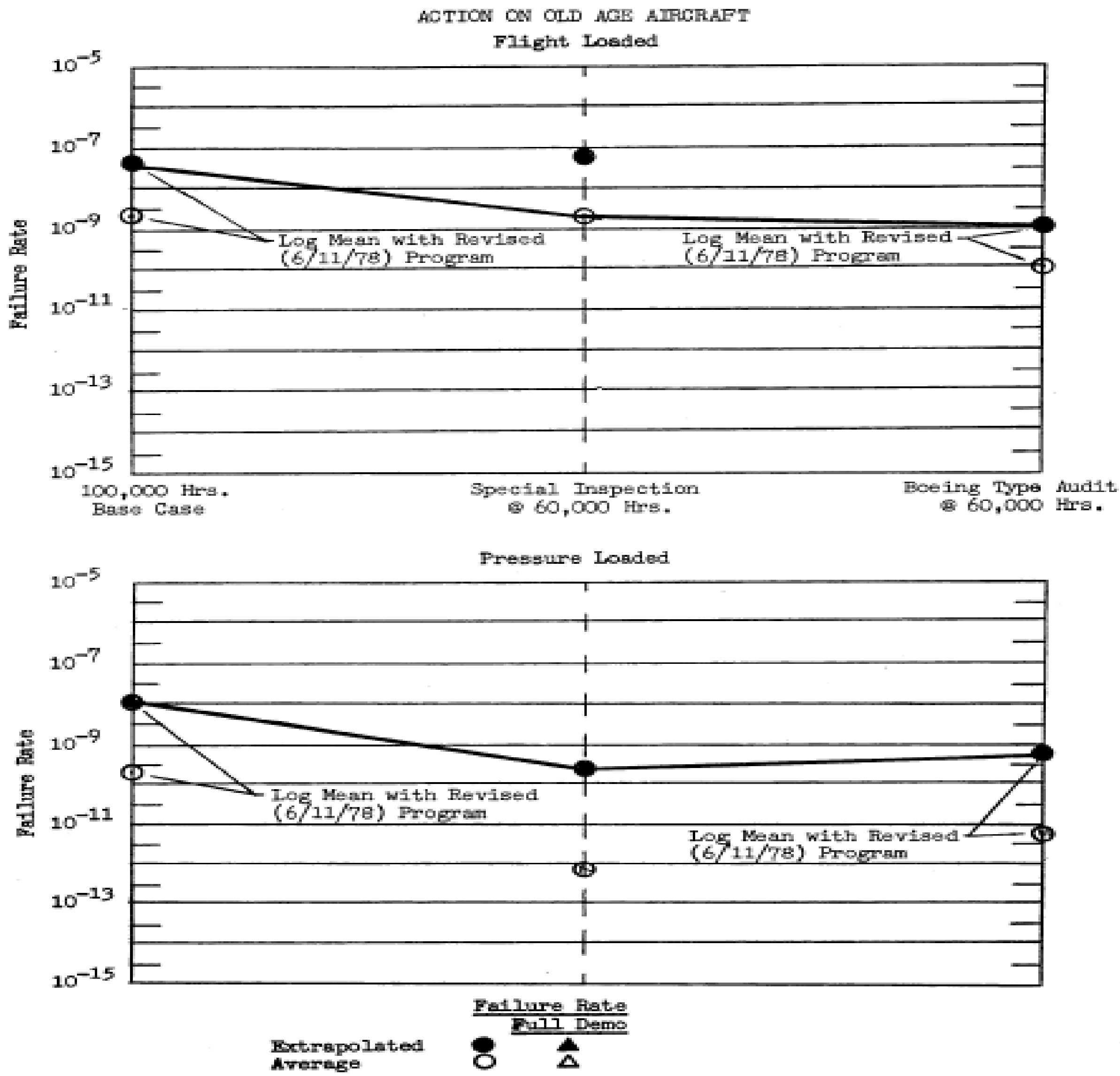


Figure 18. Action on Old Age Aircraft

Variation of Critical Crack Length: The critical crack length was defined as the first crack length at which the crack propagation rate sharply increases. The base case was the typical wide-body critical crack length (5 to 14 inches) that varied as a function of material and stress level. The results are shown in Figure 19.

A review of the results indicated that a shorter critical crack length reduces strength faster and drastically reduced the safety level. A detail review indicated that the runs were typical. The implications were that a reduction the typical critical crack length would drastically reduce the safety level by an increase in length would result in only a small increase in safety.

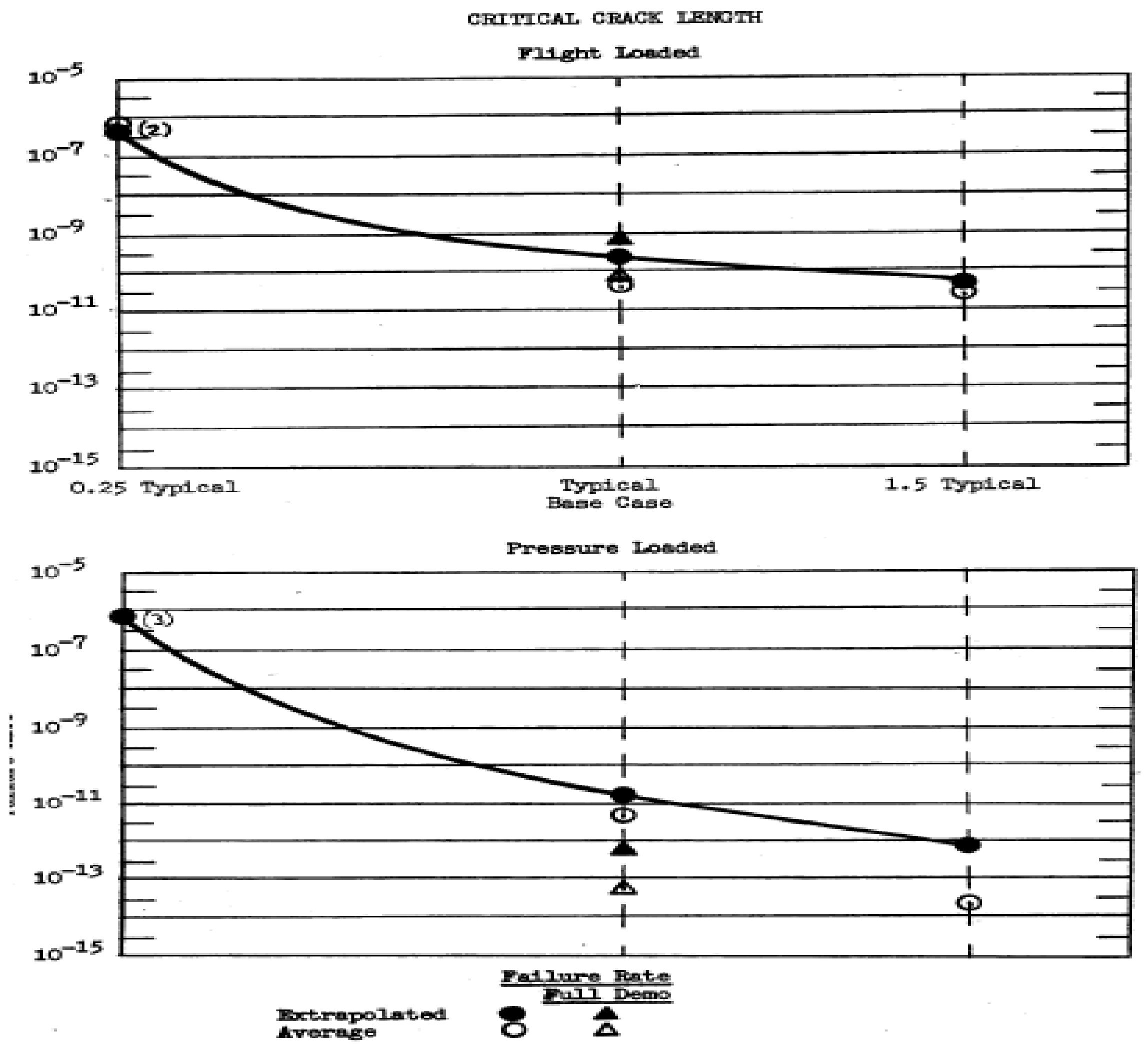


Figure 19. Variation of Critical Crack Length

Variation in Fatigue Life Goal: The analytical and test fatigue life goals was varied by multiplying the typical twice the planned usage life by factors of 0.5 and 2.0. The results are shown in Figure 20.

A review of the results indicated that the short fatigue life goal was compensated, perhaps un-realistically, by an increase in number of service modifications. The simulation probably does not fully account for corrosion, production and service damage. The implications are that low fatigue life is not extremely critical for fail-safe structure but exceptionally long fatigue live (> 4 * Usage life) significantly increases the safety level of fail-safe structure.

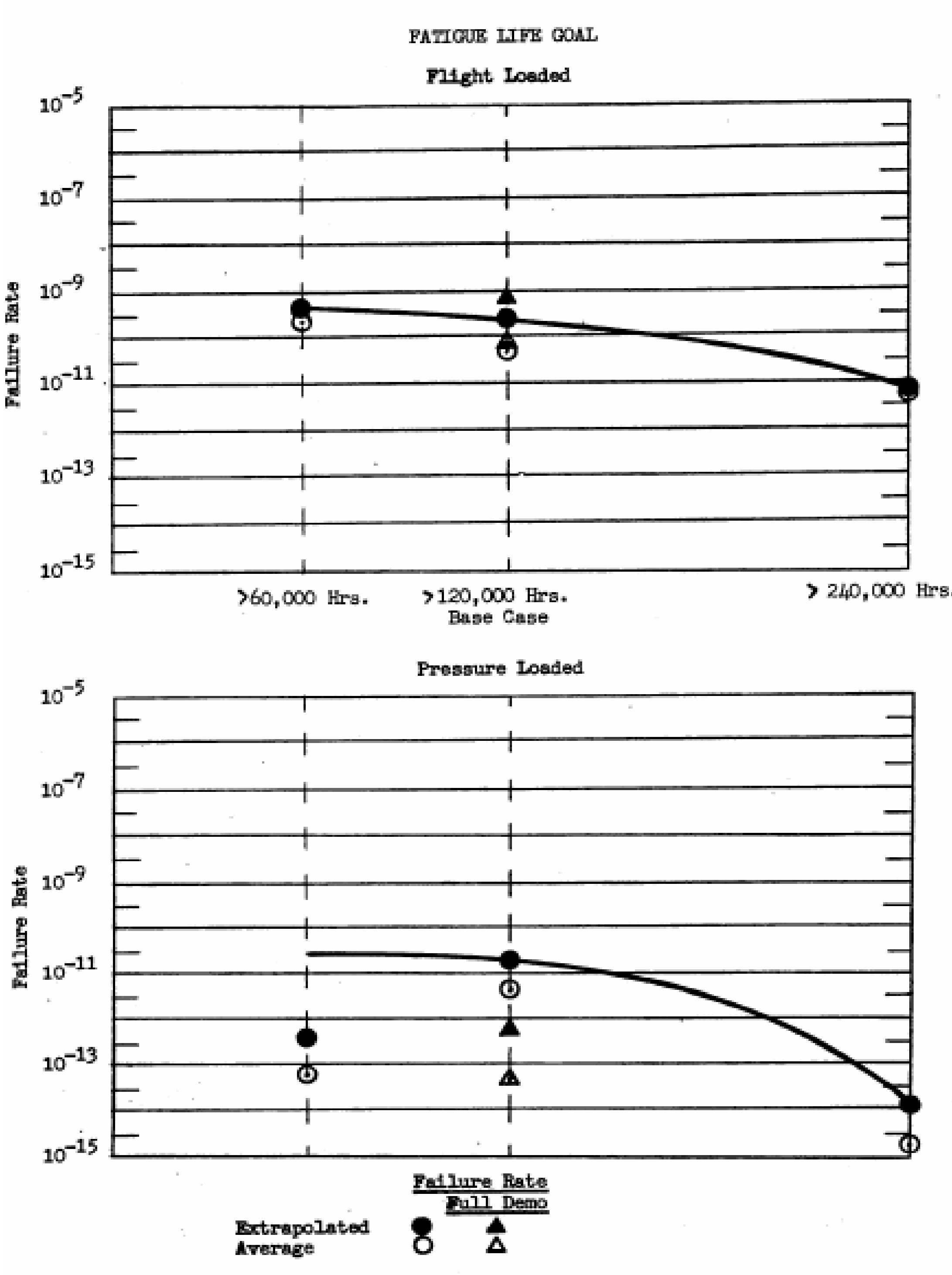


Figure 20. Variation in Fatigue Life Goal

Variation in Fleet Size: The fleet size was reduced and the production time limited to the base case of a fleet of 500 airplanes. The results are shown in Figure 21.
A review indicated that the failure rate increases with small fleet because the service feedback increases with fleet size.

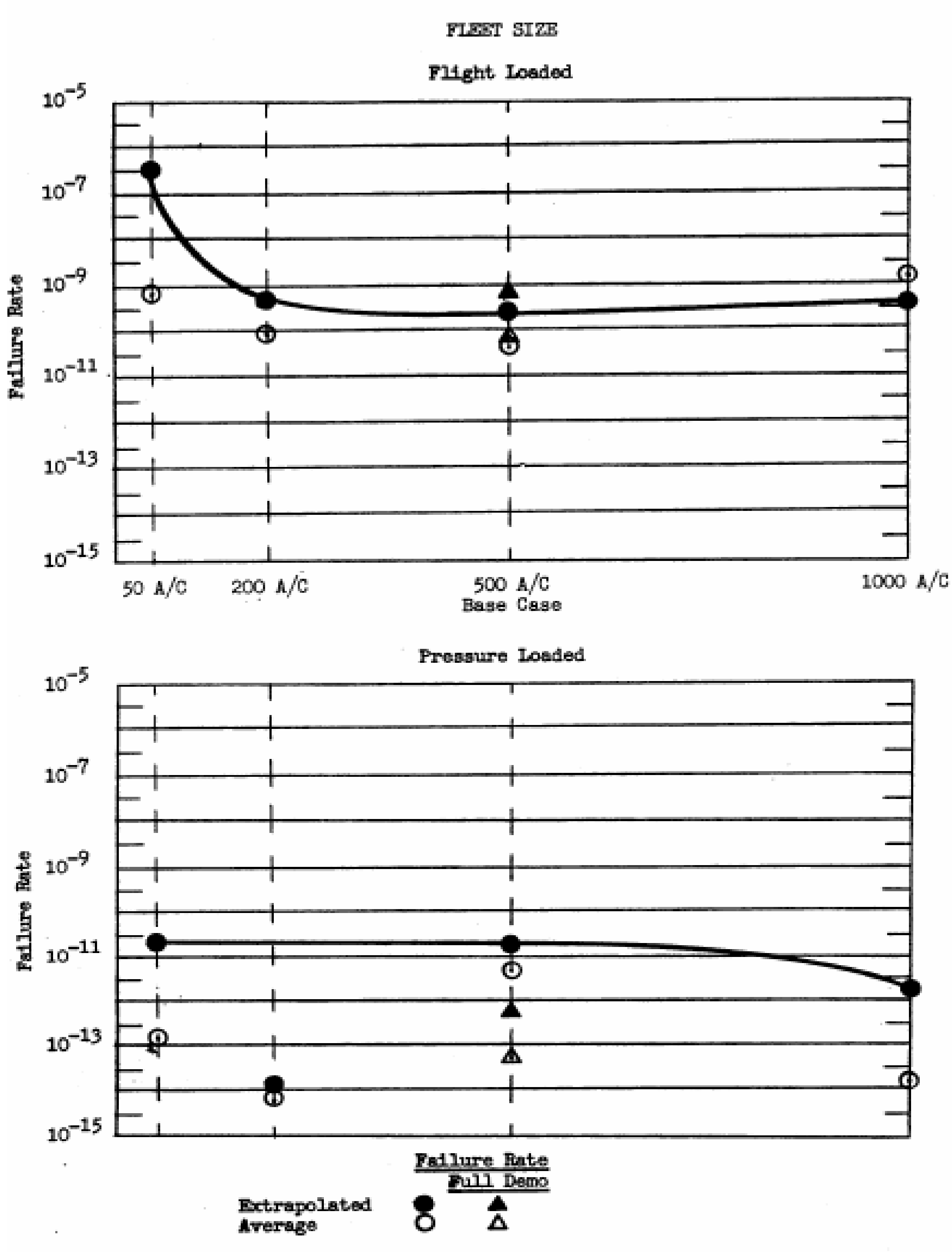


Figure 21. Variation in Fleet Size

Variation in Fail Safe Crack Length: The typical fail safe crack length for civil transports (23 to 46 inches) varied as function of crack stopper capability, stress level and material). The results are shown in Figure 22

A review of the results indicated that the effect of a reduction in the typical fail safe crack length was similar to the variation in the critical crack size, but more drastic. The implications are that a reduction in the typical fail safe length would drastically reduce the safety level, but an increase would provide only a small safety improvement except in pressure structure.

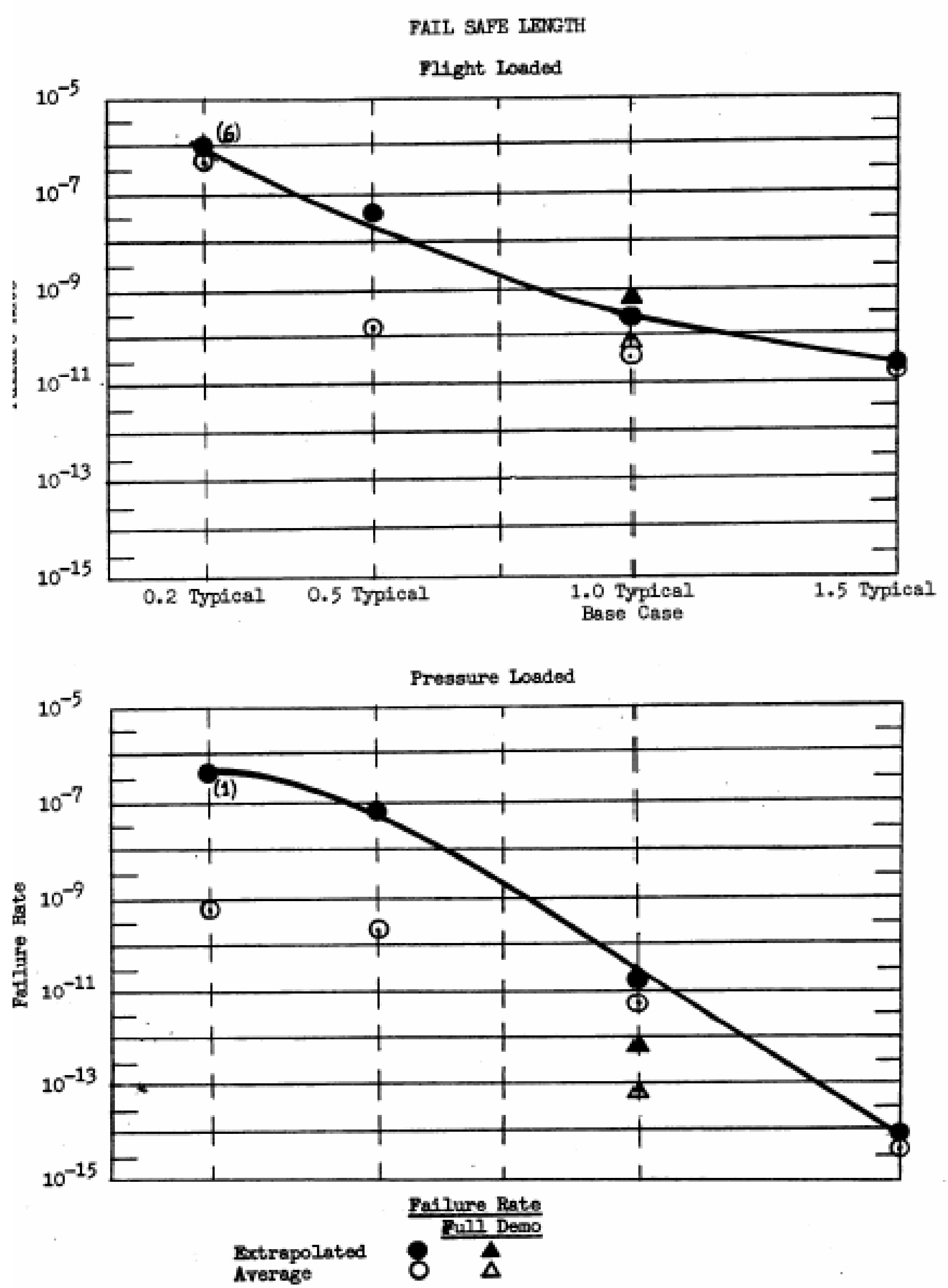


Figure 22. Variation in Fail Safe Crack Length

Variation in Fail Safe Load: The typical fail safe load for civil transports varied as function of crack stopper capability, stress level and material. The results are shown in Figure 23. The results and implications are similar to the fail safe crack length study..

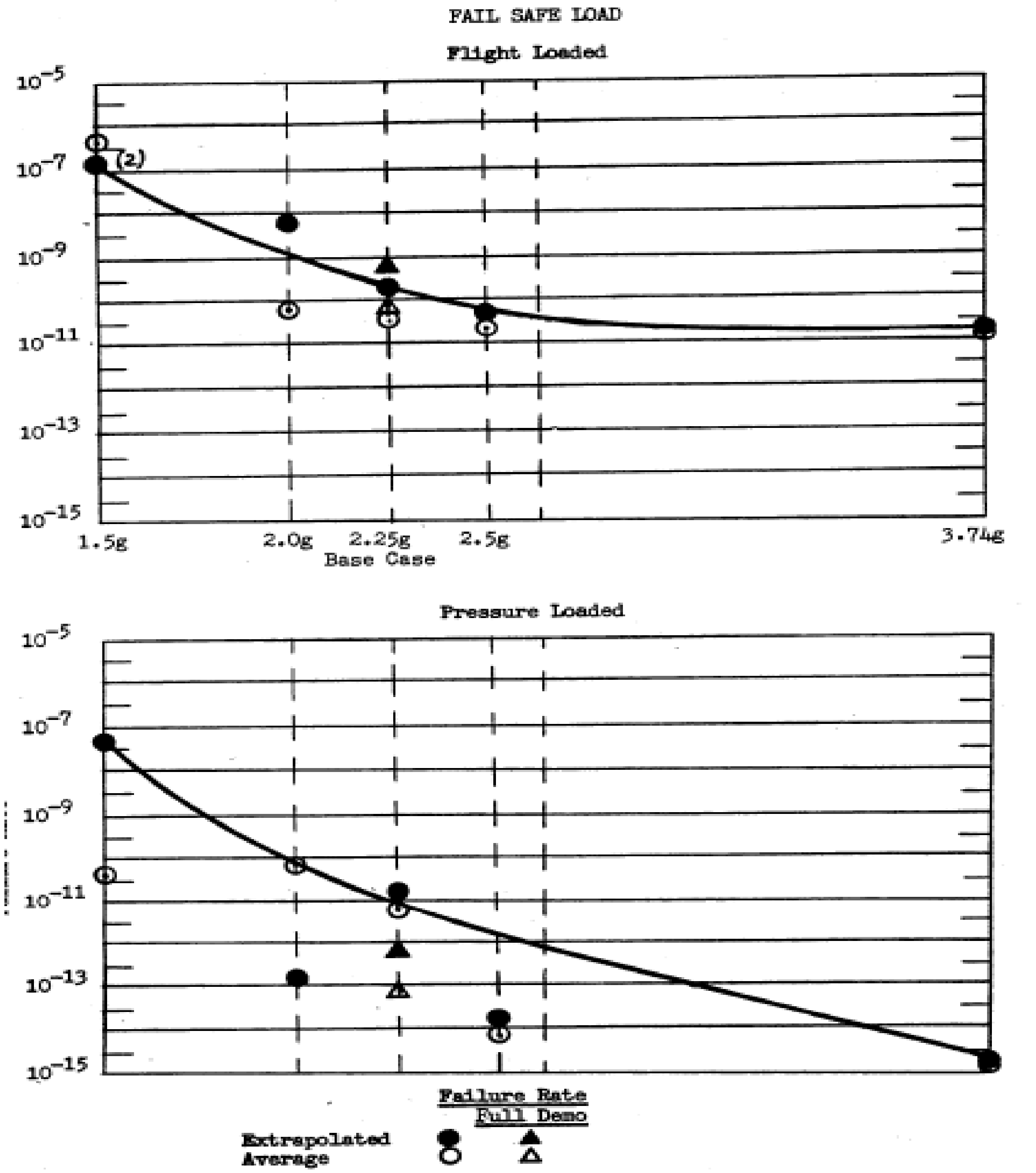


Figure 23. Variation in Fail Safe Load

Variation in Inspection Change Criteria: In the base case, the D check interval was reduced to 30%, if crack detected would grow to 0.5 the fail safe length (5 to 11.5 inches) prior to the next D check. In the first alternate evaluated, the 0.5 was changed to 0.25. In the second alternative, the external D check interval was limited to 3200 hours instead of 32,000 hours and the sample internal D check interval was limited to 4* 12,000 hours instead of 10* 32,000 hours. The results are shown in Figure 24. Neither of the alternatives improved the safety level..

A detailed review indicated that the runs on alternatives were unduly pessimistic; probably they should have been made with the full model instead of the 10% sample. The implications, probably erroneously, were that drastically tighter inspection change criteria would be required to achieve any improvement in the safety level.

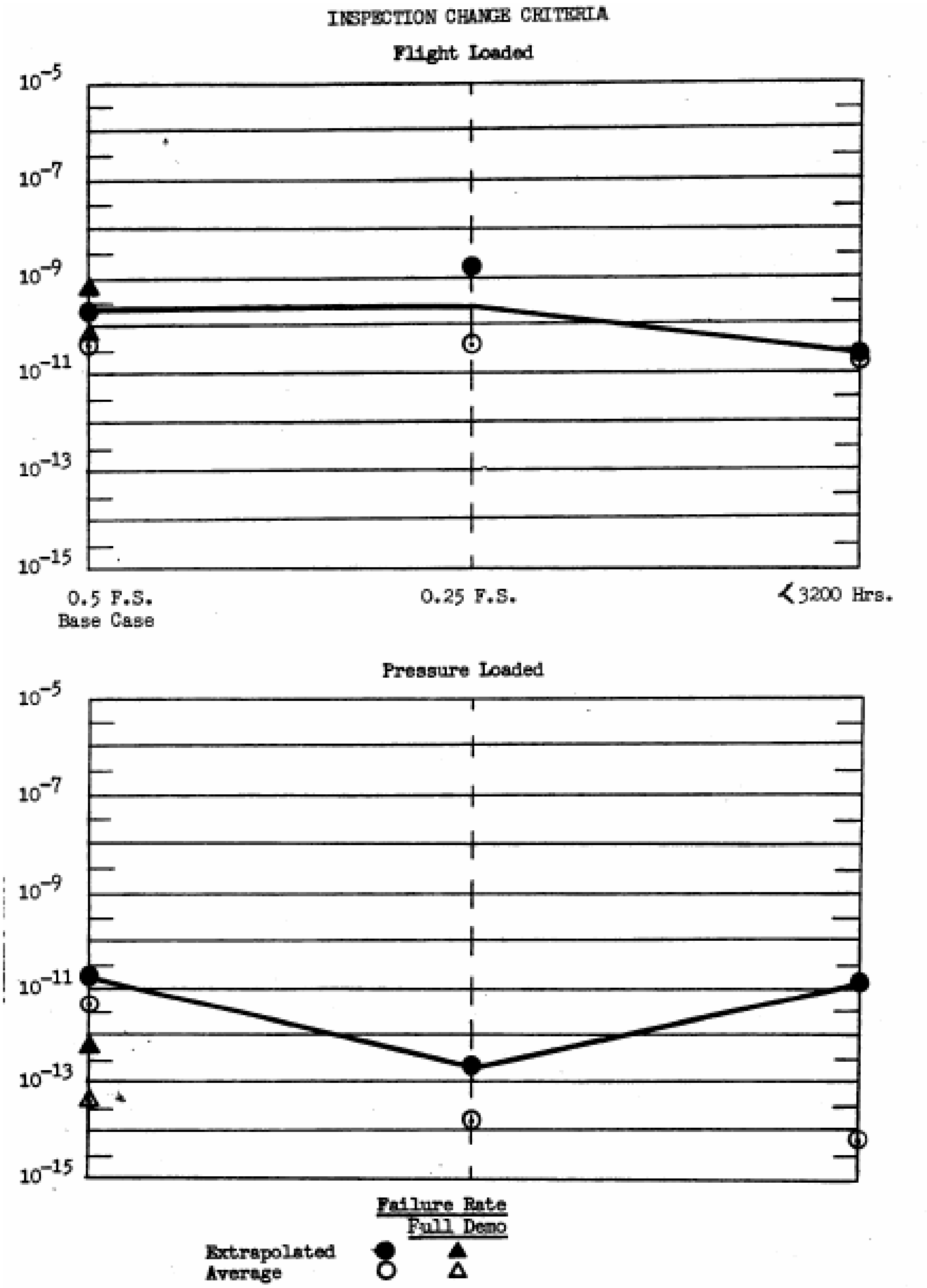


Figure 24. Variation in Inspection Change Criteria

Variation in Initial Internal Sample: The internal D check sample was varied 2.5% initial sample to 100% sample throughout the program for the 10 high time airplanes. On cracks found in internal inspection, program calls for an interval reduction based on further growth in the internal inspection interval. The results are shown in Figure 25. There were fewer reductions in the D check interval in the 100% sample run which is the opposite of what would be expected..

A detailed review indicated the 100% sampling run was pessimistic but all runs on the alternatives were 10% sample runs. They probably should have been runs with the full model. A detailed review indicated that the runs on alternatives were unduly pessimistic; probably they should have been made with the full model instead of the 10% sample. The implications are that the program may not be valid for this comparison.

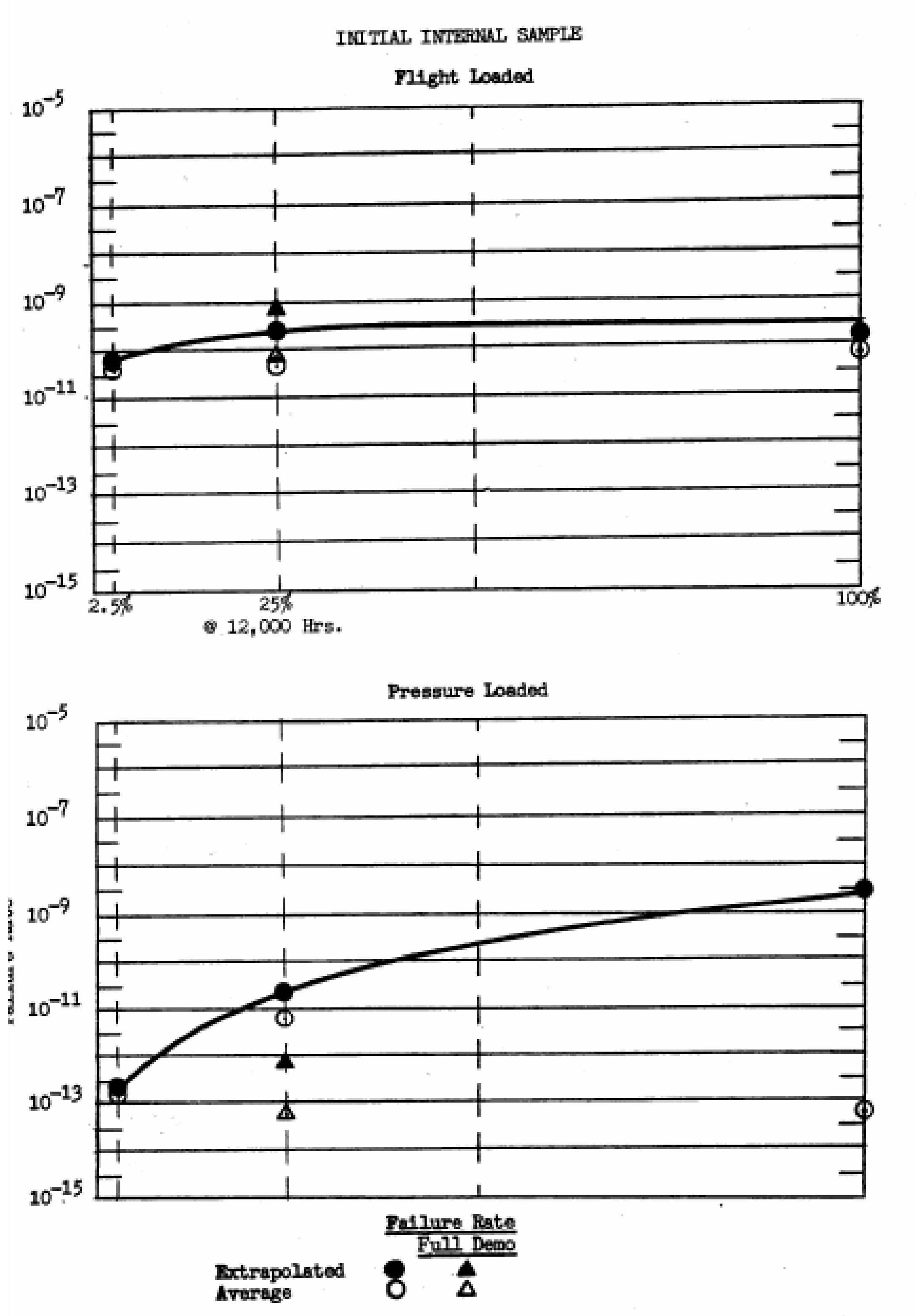


Figure 25. Variation in Initial Internal Sample

Variation in Service Modification Policy: The program calls for in service modifications when they are cheaper than continuing increased inspection and repair. The policy was varied by multiplying repair costs by a factor of 10. The results are shown in Figure 26. There were fewer reductions in the D check interval in the 100% sample run which is the opposite of what would be expected.

A detailed review indicated that the runs with increased repair costs were not typical, in that the number of modifications remained unchanged whereas they should have increased 30% and the safety level decreased. In the case, where there was one additional modification, the safety level increased. All runs on the alternatives were 10% sample runs. The detailed review indicated that the runs on alternatives were unduly pessimistic; probably they should have been made with the full model instead of the 10% sample. The implications were that any safety improvement for a more stringent policy was not sufficient to overcome the inherent run variability.

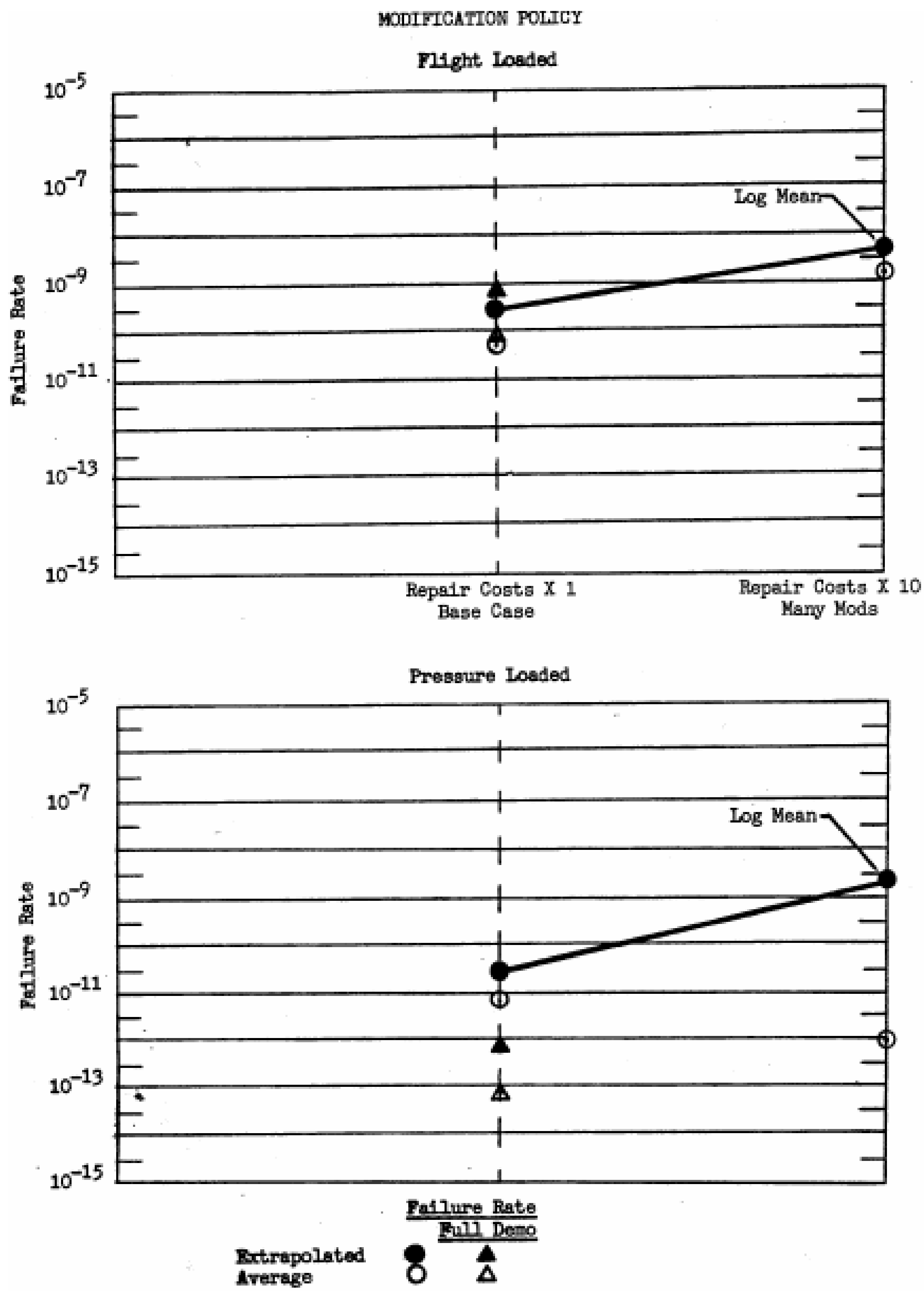


Figure 26. Variation in Service Modification Policy

6. Summary and Conclusions

6.1. Summary

The results of demonstration of the program agreed reasonably well with civil transport service experience as documented by the MRR/SDRs for the period of 1964 through 1974. As shown, the percent of structural cracks detected in the demonstration and their size, agreed well with those reported in the MRR/SDRs. Also the distribution of cracks between elements was in fair agreement with the MRRR/SDR results. The predicted wide body aircraft fatigue failure rates seem reasonable and to date seem to be supported by their subsequent service experience. It should be noted that this program can be used for individual elements, types of elements or a complete airplane, with the analysis covering the entire life cycle to a specified time period in service.

The subsequent parametric study and use have in almost all cases provided results that are rational and reasonable. The program has the capability of providing detailed results on each element on each airplane. These detailed results have provided valuable insights on how large cracks can occur. The study also provided valuable insights on the importance and effect of various design parameters and test and maintenance practices.

Credibility has always been a problem with conventional risk analyses. The recent need and dependence for such analyses, has resulted in their improvement and more acceptance of their credibility. Their credibility has been improved by:

- (a) Limiting the assumptions in number and to those generally accepted or as being conservative and no proof of their invalidity was available.
- (b) Cautioning that validity is limited to the case defined by the assumptions.
- (c) Improving the realism of the analysis by including more variables such a prescribed specific periodic inspection with a high probability of detection (POD).

However, such analyses have not apparently, considered the variability of flight loads spectrum between individual airplanes in predicting whether a load will exceed the degraded strength. Also the assumed POD is usually based on the results of controlled inspection experiments, not of actual in service performance and is likely un-conservative. In spite of such shortcomings, the existing conventional risk analyses are useable, needed and valuable and are constantly being improved.

Conversely, the SAIFE risk analysis attempts to cover all significant variables, makes many assumptions and consequently has a greater credibility problem. This is a daunting and difficult task. However, considerable effort was made to make it realistic with some credible. During development, a deliberate effort was made to achieve results comparable to service experience. The changes made were not arbitrary changes to achieve the desired results, but rather were based on detailed analysis of the results, with changes to make it more realistic. One such change, was to account for the typical civil transport crack stoppers which are not generally considered in conventional risk analyses. The SAIFE program could also be improved in areas such as the following:

a) The POD is not just a function of the level and type of inspection and the defect size as is the case in SAIFE. Several additional factors, some of which may exist for considerable time and affect many inspections. The POD of the individual inspector, which in the case of specialized NDT inspection wherein one inspector may make all of these inspections for an entire airline of maintenance facility. The inspection instructions may be inadequate and affect many inspections. The prescribed inspection may be impossible to accomplish, especially with the time and resources available. This may be the case, when thousands of holes are involved or as was the case where the Navy visual inspection of one of their transport aircraft for wing stringer cracks through inspection access holes. A subsequent teardown showed that many stringers were cracked and missed in the inspection.
b) Feed back could be provided from structural elements that similar in structure and stress spectrum instead of only providing feed back from elements identical in structure and location.
c) While element failures, which are also considered aircraft failures, can be estimated from the failure rates generated, a direct Monte Carlo determination of whether a specific element failed due to a load exceeding the residual strength would be desirable.

There is a continuing need to make decision on actions such as the following to prevent progressive type structural failures from fatigue and corrosion:

a) Design and substantiation criteria changes;
b) Correction of service problems;
c) Establishment of inspection and maintenance policies and programs.

These decisions are essentially based on two factors:

a) The probability of structural defects occurring and of catastrophic failure.
b) The burdens caused and alleviated by the proposed action.

Good decisions require the best possible estimates of these two factors. It is and impossible task to consider all of the factors and variabilities involved in predicting these two factors. However, decisions have to be made and are made every day with only an implied prediction of these two factors without making a “best estimate”. These decisions are based on the available analysis, tests, data and engineering judgment and some times a conventional risk analysis that only considers some of the factors.

The SAIFE program attempts to account for all significant factors and variabilities and does provide a “best estimate” of these factors with some credibility. Because of its’ unique capability, Technical Data Analysis Inc. has re-visited the program and converted it to the PC environment.

6.2 Conclusions.

1. The SAIFE program can be used on individual element, types of elements or a complete airplane.
2. Its' use in evaluating design, tests and maintenance criteria and practices, could provide valuable insights and relative merit and effectiveness of alternatives.
3. As a supplement to conventional risk analyses, it could provide insights and a comparison of results that could be valuable
4. SAIFE should be improved and further use explored.

7. Acknowledgements

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