AD-A009 519

# FAIL-SAFE/SAFE-LIFE INTERFACE CRITERIA

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Technology, Incorporated

Prepared for:

Army Air Mobility Research and Development Laboratory

April 1975

## EUSTIS DIRECTORATE POSITION STATEMENT

This report is considered to provide a very good review and analysis of fail-safe/safe-life concepts. Results of the program indicate that additional work in the area of structural, fail-safe design of helicopter landing gears and the preparation of clear definitions of fail-safety would be beneficial.

This report has been reviewed by this directorate and is considered to be technically sound.

The technical monitor for this contract was Mr. A. J. Gustafson of the Structures Technical Area, Technology Applications Division.

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Preparatory to developing fail-safe/safe-life design criteria for future helicopters, extensive literature and Government/industry surveys were conducted to define and evaluate the related state of the art. After selected fail-safe/safe-life design methodologies were defined, they were applied to the main and tail rotor systems (specifically the blade, blade retention device, hub, and rotating controls), the fuselage, and the landing

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gear. Next, in cost and weight trade-off studies, various combinations of design criteria for redundant structures, controlled-fracture structures, and monolithic structures with slow crack growth were evaluated with respect to their technical feasibility. Then after the most desirable fail-safe/safe-life design criteria for specific applications were assembled, they were compared with current design practices and the military specifications in MIL-S-8698 and AR-56 to identify specification areas needing definition, expansion, and/or refinement and to make recommendations accordingly for the improvement of these specifications.

## PREFACE

Technology Incorporated, Dayton, Ohio, prepared this report in compliance with the requirements of Contract DAAJ02-74-C-0004. The program was sponsored by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia. The project monitor for the Army was Mr. Arthur J. Gustafson, Jr.

The program was conducted under the direction of Mr. R. B. Johnson, Jr., Manager, Systems Analysis Department. The project engineer was Mr. S. W. Russell. Mr. R. R. Yeager conducted the literature search and prepared the industry survey questionnaires. Mr. G. V. Feldt assembled, reviewed, and interpreted the published reports and industry responses gathered through the visits and questionnaires. Mr. Feldt also conducted the trade-off studies, summarized the criteria, and developed the recommended fail-safe/safe-life design criteria for helicopters.

The authors are grateful to the many Government, private, and industrial organizations and personnel who participated in this effort without cost to the Government.

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# 1. INTRODUCTION

## 1.1 BACKGROUND

Recent U.S. Army helicopter development programs have produced numerous fail-safe designs to meet the requirements of such specifications as those in AR-56, "Structural Design Requirements (Helicopters)." Since safe-life design concepts may be preferable in some applications, the optimum design may well be a rational mix of fail-safe and safe-life designs to minimize the cost and weight of the helicopter while maximizing the safety of the system.

In the development of new helicopter systems, such as the HLH, the innovative opportunities of designers have introduced design methodologies where the fail-safe and safe-life design concepts are interpreted rather broadly and sometimes ambiguously so that there is often not a clear-cut distinction between the two concepts.

Therefore, the purpose of this program was to analyze the existing fail-safe and safe-life design criteria and methodologies and organize them so that they would afford the design engineer various fail-safe and safe-life design options.

## 1.2 PROGRAM OBJECTIVES AND SCOPE

The primary objective of this program was the development of rational criteria for the fail-safe/safe-life design of helicopter component structures. To this end, current fail-safe and safe-life design criteria and methodologies were extensively reviewed and analyzed. Consequently, military structural design specifications with fail-safe design areas requiring definition, expansion, and/or refinement were identified.

The enormity of the subject area required imposing the following restrictions:

(1) The potential fail-safe and safe-life design criteria will be defined and design criteria will be recommended for only the following helicopter components: main and tail rotor systems, fuselage, and landing gear.

(2) The recommended fail-safe and safe-life design criteria will not specify which structural design technique and/or materials should be employed in satisfying the criteria. However, cursory technical, cost, and weight trade-offs will be conducted to identify promising structural design schemes and possible inspection requirements.

(3) The recommended design criteria will be evaluated with respect to MIL-S-8698 and AR-56, and specification areas in these documents requiring definition, expansion, and/or refinement will be identified.

The two design philosophies discussed in this report are not conflicting or contradictory; rather, they represent two different approaches whose common objective is to provide a design rationale which, when implemented, will provide a safe, airworthy structure.

## 1.3 REPORT ORGANIZATION

The report consists of the following: (1) Section 2 summarizes the survey of current fail-safe and safe-life design criteria and methodologies; (2) Section 3 describes the fail-safe and safe-life design philosophies; (3) Section 4 formulates and presents the potential design criteria for each of the specified components; (4) Section 5 conducts cursory technical, cost, and weight trade-off studies to identify those current design practices that are most promising in the light of the recommended criteria; (5) Section 6 evaluates MIL-S-8698 and AR-56 with respect to the recommended criteria; (6) Section 7 presents the study summary and conclusions; and (7) Section 8 offers recommendations.

In addition, Appendix A includes technical merit, cost, and weight rating tables for fuselage and dynamic components. Appendix B includes abstracts of the reports considered to be most pertinent to the criteria development and a table summarizing these reports according to subject areas. Appendix C summarizes the results of questionnaires given to selected Governmental and industrial organizations and interviews with Government and industry representatives.

## 2. FAIL-SAFE AND SAFE-LIFE DESIGN CRITERIA AND METHODOLOGY SURVEY

The survey to gather information on fail-safe and safe-life design criteria and methodologies consisted of: (1) searching the literature at appropriate sources; (2) developing an information retrieval system for the reviewed documents; (3) acquiring pertinent documents; (4) on the basis of the document findings, preparing two questionnaires, both to be mailed to selected organizations - one to be returned immediately, and the second to be retained for guidance in subsequent interviews; and (5) visits to these organizations to elicit additional information according to the information inquiries in the second questionnaire.

### 2.1 LITERATURE SEARCH AND DOCUMENT ACQUISITION

In the literature search, abstracts or digests of pertinent reports documented in the last 10 years were reviewed. The information sources included the National Technical Information Service (NTIS), the Defense Documentation Center (DDC), and the American Helicopter Society.

After 356 abstracts or digests were reviewed, 240 complete reports were acquired in either microfiche or hard-copy form.

### 2.2 INFORMATION RETRIEVAL SYSTEM AND DATA COMPILATION

Each of the 356 abstracts or digests was assigned a 3-digit library number in a uniform-indexing information retrieval system. Then the following information for each document was keypunched on a computer card: the author, the issuing agency documentation number, and the assigned library number. If a document had two or more authors, a card was prepared for each author. Two copies of each card were prepared, and three separate decks of all cards were maintained. Each deck was updated weekly in three computer listings: one to alphabetically list the authors' last names, the second to alphanumerically order the issuing agency documentation numbers, and the third to numerically list the assigned library numbers.

To facilitate the compilation of data for each subject area in the survey, a methodology survey form (MSF) was established. As an investigator reviewed a particular document and extracted the relevant information, he checked off the corresponding subject areas covered by the extracted information. If the review revealed a subject area not included on the form but judged to be desirable, the area was added to the MSF and the corresponding relevant information was extracted.

Although most of the acquired 240 reports contained some pertinent information, 80 of these reports were considered of primary value to the study. Consequently, again for information retrieval and control purposes, these 80 reports were each assigned new 2-digit library numbers. To summarize the contribution and relevancy of these primary reports, Appendix B presents the abstract for each and Table B-1 which lists the subject areas and the headings for each of the new 2-digit library numbers. The solid circles in this table denote the subject areas associated with the respective library numbers whose corresponding documents are identified in the abstract listing.

## 2.3 GOVERNMENT/INDUSTRY SURVEY

The following organizations were selected for the questionnaire mailing and were subsequently visited for eliciting additional information on the basis of the questionnaires: Battelle Memorial Institute; Bell Helicopter; Boeing Vertol; Federal Aviation Administration; Hughes Aircraft; Kaman Aerospace Corporation; NASA Headquarters, Washington, D.C.; NASA, Langley Research Center; Naval Air Development Center, Air Vehicle Technology Dept.; Sikorsky Aircraft; and U. S. Army Aviation Systems Command.

Appendix C summarizes the results of both the questionnaires and the meetings.

## 3. FAIL-SAFE AND SAFE-LIFE DESIGN PHILOSOPHIES

To understand the aspects and interrelationship of the fail-safe and safe-life design philosophies before describing a rationale for interfacing their methodologies, the following paragraphs summarize the philosophy objectives and describe the technologies for their implementation.

Although the two philosophies have the common objective of designing for a reasonable assurance of flight safety, they involve different aspects of the fatigue phenomenon. As generally depicted in Figure 1, the fatigue process consists of the following: When areas of high stress concentration are subjected to cyclic loading, a structure undergoes a period of crack nucleation at both the atomic and microscopic levels. Under continued cyclic loading, a fatigue crack will initiate and subsequently propagate to critical dimensions where fracture occurs. While the safe-life philosophy is based on predicting the time when a crack will begin, the fail-safe philosophy is based on: (1) making critical structures capable of retaining sufficient strength for flight safety during the inception and initial propagation of a crack, and (2) providing an inspection procedure to detect the crack before it reaches its critical length so that the probability of catastrophic failure before crack detection is extremely remote.

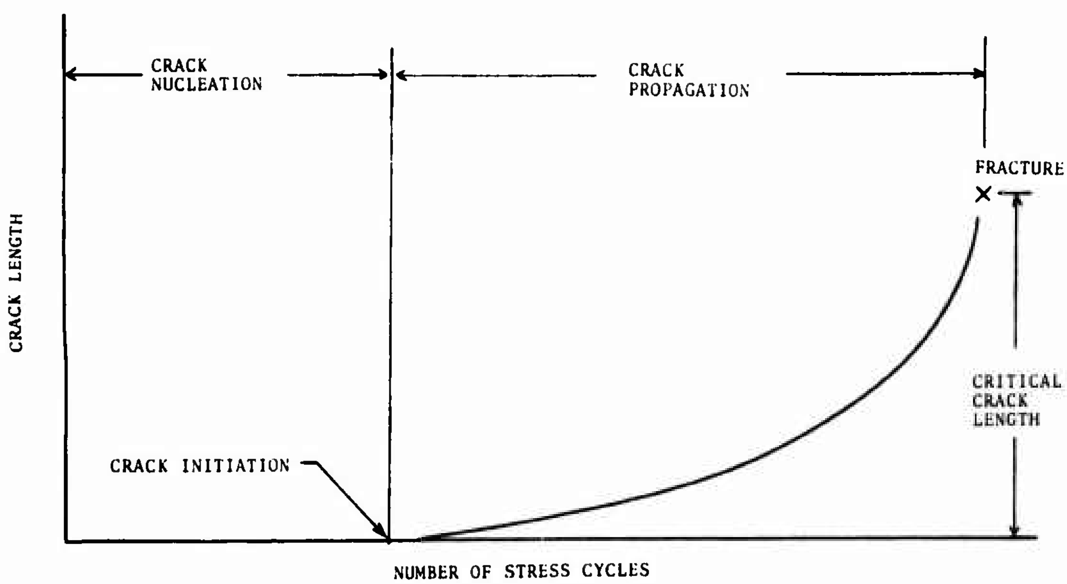


Figure 1. Characterization of (Generally Accepted) Fatigue Process.

In this section, the fail-safe and safe-life design philosophies are described separately and independently to emphasize the distinction between the two. However, in practice no aircraft structure is designed to either rationale exclusively.

## 3.1 BACKGROUND

In past and current helicopter designs, the objective of the safe-life design criteria has been a specified safe service life with a remote probability of fatigue failure. However, the need to apply effective fail-safe design principles for flight safety over the service life became apparent when the number of in-service failures began mounting because of cracks generally not initiated by fatigue crack nucleation (decohesion of atomic bonds resulting in striations) but by such causes as undetected production flaws and maintenance errors.

## 3.2 SAFE-LIFE DESIGN PHILOSOPHY

Because of the objective of the safe-life design criteria, the designer is concerned primarily with the so-called fatigue damage accumulation part of the curve in Figure 1.

In the development of a safe-life estimate for an aircraft structure, those areas considered to be most fatigue critical are normally analyzed according to three types of complementary data analyses: (1) a loading history representative of the aircraft's intended usage, (2) a derived S-N curve representative of loading and structure, and (3) a cumulative damage summation. However, since the analytical interpretation and resulting treatment of these data differ in the various safe-life design methodologies, this section will describe the more prominent methodologies and relate them to the needs for current helicopter structures.

The loads analyst derives in a statistical format the flight loads expected for a new helicopter on the basis of data recorded on similar helicopter types. Much of the resultant flight loads spectrum is also based on his judgment and experience.

The flight loads on helicopter airframe structures are primarily due to maneuver and gust conditions. These loads are normally in the form of low-frequency, high-amplitude (wide dispersion from low to high amplitude) cycles. The flight loads on helicopter dynamic component structures are primarily due to the steady-state harmonic loading inherent in rotating structures. These loads are normally in the form of high-frequency, low-amplitude (narrow dispersion from low to high amplitude) cycles.

Based on all flight load spectra, the fatigue load spectrum for a particular helicopter usage normally represents the most severe loads expected. Several methods are used to derive a spectrum: top-of-scatter of flight load spectra, 90% probability of occurrence, mean flight loads, mean flight loads plus three sigma, etc.

In establishing the fatigue load spectrum according to the safe (fatigue) analysis procedure, the degree of the severity of flight loads must be considered relative to the S-N curve representation (fatigue strength) and the service life calculation.

Fatigue strengths of materials and structural components are also statistical expressions. As for the development of the flight load spectrum, various techniques are utilized to establish a fatigue strength curve (S-N) which has sufficient conservatism to describe a minimum fatigue strength for all structures in the fleet. As apparent in both the Government/industry and the literature surveys, constant-amplitude (S-N curve) testing is most applicable for predicting the fatigue life of helicopter components, and load spectrum testing, e.g., using S-N curves derived from material coupon tests and applying sequential flight-by-flight loading, is most applicable for predicting the fatigue life of most airframe members.

Fatigue strengths are normally expressed by reducing the mean fatigue strength by varying multiples of the standard deviation of the test data, e.g., mean strength minus three sigma (the result is termed the statistically adjusted fatigue strength). Miner's theory of linear damage accumulation is still widely used in calculating fatigue lives.

Finally, the safe service life is obtained by multiplying the calculated fatigue life by a factor K:

$$K \cdot \text{calculated life} = \text{service life}$$

This factor may range from 0.25 (indicating a service life of four times the expected fatigue life) to 1.00 (indicating a service life equal to the calculated fatigue life). The magnitude of the factor depends largely on the technique used in determining the fatigue strength and the fatigue load spectrum. Table 2 in Section 4.1.1 presents many of the safe (fatigue) life determination techniques.

The traditionally accepted practice for the safe usage of the helicopter critical structure (principally the dynamic components) has been to retire each component at the end of its projected service life. Unfortunately, the service record of helicopter structural failures indicates that many components have failed before their safe-life retirement time because of causes other than statistically predictable fatigue phenomena, as illustrated in Figure 2. The obvious need for additional design criteria to account for these nonpredictable failures has led to the development and application of damage-tolerant design concepts, fracture mechanics analysis, and the fail-safe design philosophy.

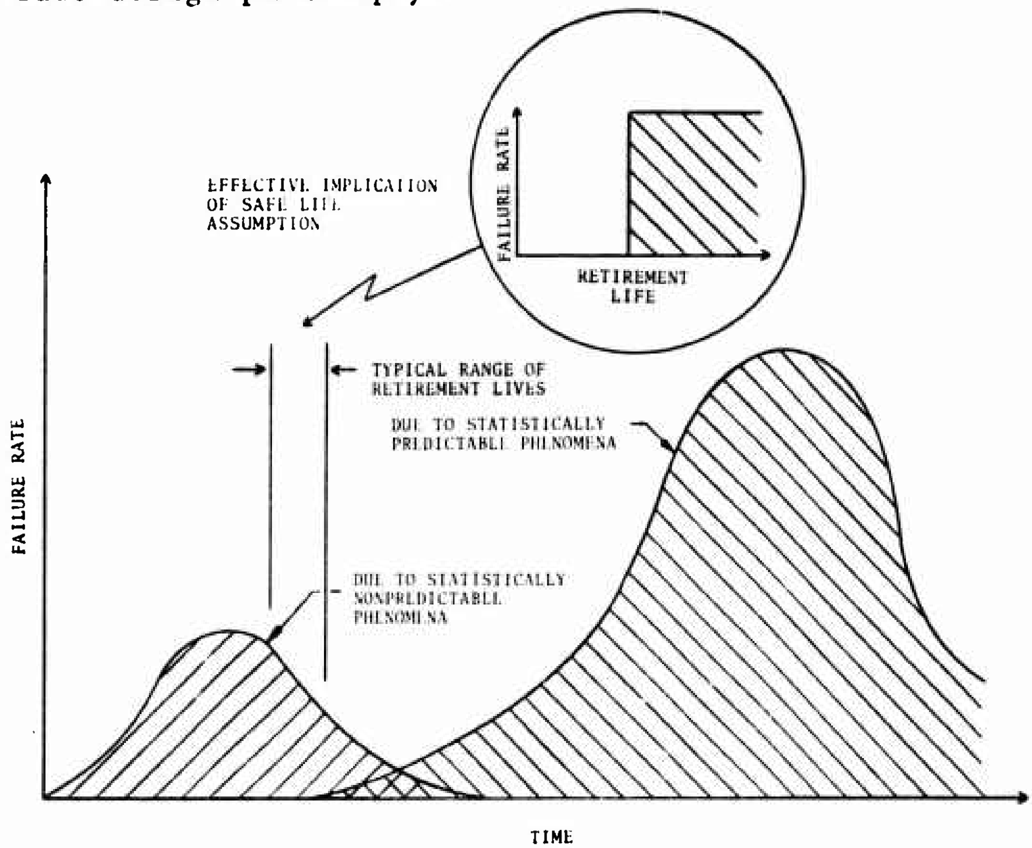


Figure 2. Graphical Description of Safe-Life Design Philosophy Shortcoming.

## 3.3 FRACTURE CONTROL DESIGN PHILOSOPHY

As stated in an Air Force report[1]:

"...the majority of cracks found in aircraft structures were initiated from tool marks, manufacturing defects, and the like. When not detected, these cracks experience the driving forces of environment and service loading and may grow to serious proportions resulting in reduction of service life or complete loss of the aircraft..."

[1] Wood, H.A., FRACTURE CONTROL PROCEDURES FOR AIRCRAFT STRUCTURAL INTEGRITY, Air Force Flight Dynamics Laboratory, U.S. Air Force, NASA SP-309.

Consequently, fracture control (fail-safe) design technology has been applied to aircraft structures in both the fixed-wing and the rotary-wing aircraft.

The Government/industry review revealed that while the basic concept of fail safety, namely, critical load-carrying members so designed that the probability of catastrophic failure is extremely remote, was well understood, the concept of the type of structural configuration and design procedure for a fail-safe structure varied among the users. The lack of definitiveness and conciseness in the comprehension of this latter concept is understandable because of the broad terminology for the fail-safe design application. This broadness was particularly evidenced in the literature review where the various design methodologies had to be interpreted according to the time frame of their documentation.

Likely because of some of the inaccurate connotations associated with the term "fail-safe," the Air Force has preferred to use the term "fracture control." However, these terms are considered as synonymous throughout this report.

A fail-safe structure is defined as a structure that can so sustain a physical abnormality, such as a fatigue crack, damage, or deterioration, that the probability of catastrophic failure before the detection of the abnormality is extremely remote[2]. To design such a structure requires that its construction be highly tolerant of damage (a crack or flaw) and that its inspection be based on knowledge of the fatigue and fracture phenomena.

### 3.3.1 Structural Damage-Tolerant Design

Throughout this report, the term "damage tolerance" signifies that a flawed structure can retain a high percentage of its unflawed static strength and that the flaw will propagate slowly under cyclic loading. The damage tolerance is a function of the design configuration as well as the material resistance to fracture and flaw growth.

The selection of materials requires a trade-off of many concurrent requirements. In addition to such material qualities as high strength but low weight to meet static strength requirements, resistance to fracture and flaw growth under cyclic loading must now be considered. Reference 1 states that, as long as materials are similar in geometry, the

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[2] Jensen, H.T., THE EVOLUTION OF FAIL-SAFE CONCEPTS ROTORCRAFT, Sikorsky Aircraft, Division of United Aircraft Corporation, June 1965.

plane strain fracture toughness index, $K_{IC}$, is sufficient to select superior materials with the desired fracture resistance. However, trade-offs are difficult since in some material groups the toughness decreases as the yield strength increases, as illustrated in Figure 3.

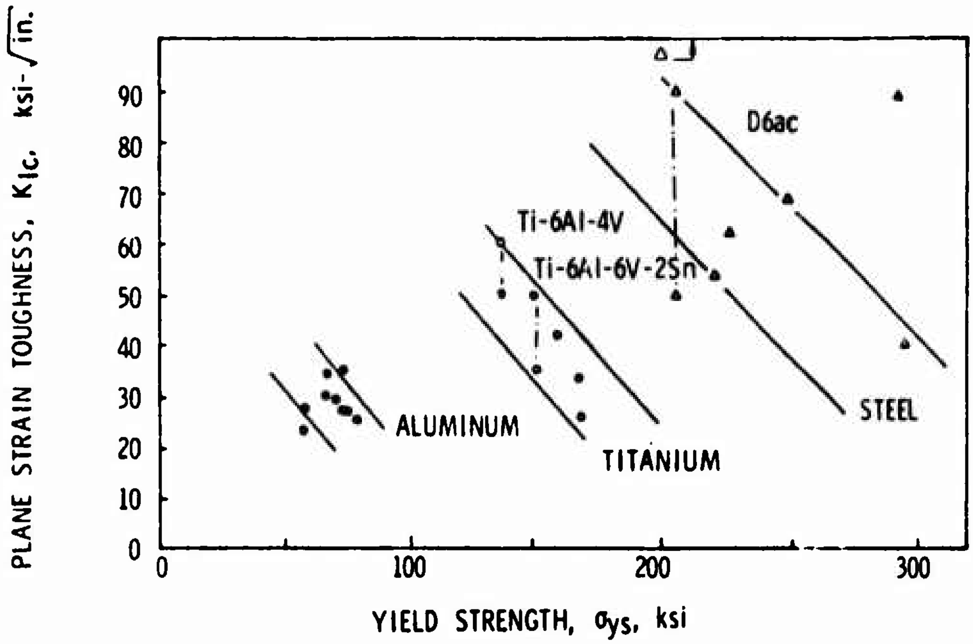


Figure 3. Trends in Toughness Variations.

The material selection is further complicated by the cyclic flaw growth behavior of candidate materials. This behavior is not as clearly defined as the fracture toughness and yield strength relationship. Materials within a group or class generally fall within a narrow scatterband with little dependence on toughness. Relative crack growth rate relationships are shown in Figure 4. The central part of the growth rate curve in this figure may be approximated by the following equation (Reference 1):

$$\frac{da}{dN} = 8\left(\frac{\Delta K}{E}\right)^2$$

Reference 1 states that crack growth under a spectrum of varying-amplitude loading causes crack-growth retardation to vary because of the load interactions. Consequently, materials must be assessed individually to determine their flaw-resistance qualities. However, in general, a material with a superior fracture toughness ($K_{IC}$) will normally have a superior crack-retardation resistance.

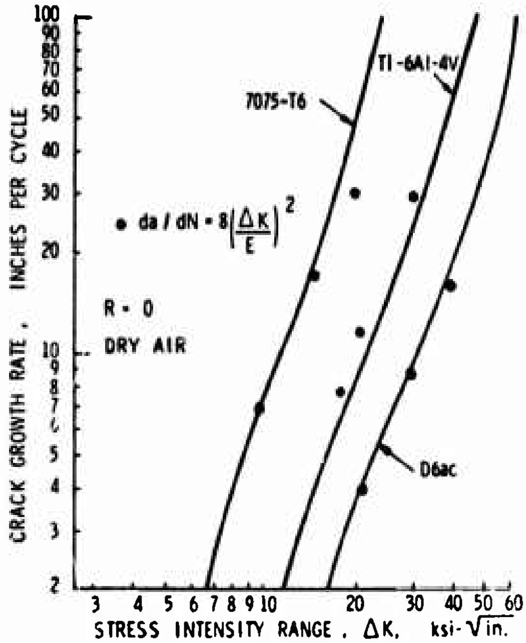


Figure 4. Fatigue-Crack-Growth Data for Typical Aircraft Structural Materials.

Although the residual strength and crack propagation characteristics will not be detailed further because of their complexity, the related residual strength and residual life phenomena will be treated as an integral part of the fracture mechanics considered in the fail-safe design. The pertinent design considerations are shown schematically in Figure 5. The 100% residual strength (shown in Figure 5(a)) is the same as the unflawed, ultimate strength of the structure (a positive margin of safety is assumed). Upon nucleation of a crack and the subsequent propagation under repeated loading, the residual strength of the structure decreases. The residual strength curve represents the degradation of static strength as the crack size, or the cracked portion of the total structural cross-sectional area (percentage of damage), increases. Note that at 100% damage the residual strength is equal to zero.

Figure 5(b) illustrates the crack growth phenomenon (crack size versus time of crack growth). The crack begins at $a_i$ (initial crack size) and subsequently propagates to fracture at 100% damage.

The design configuration chosen to carry the principal loads can have a marked effect on the curves of Figure 5. Fundamentally, all structures may be represented by one or a combination of several of the configurations shown in Figure 6.

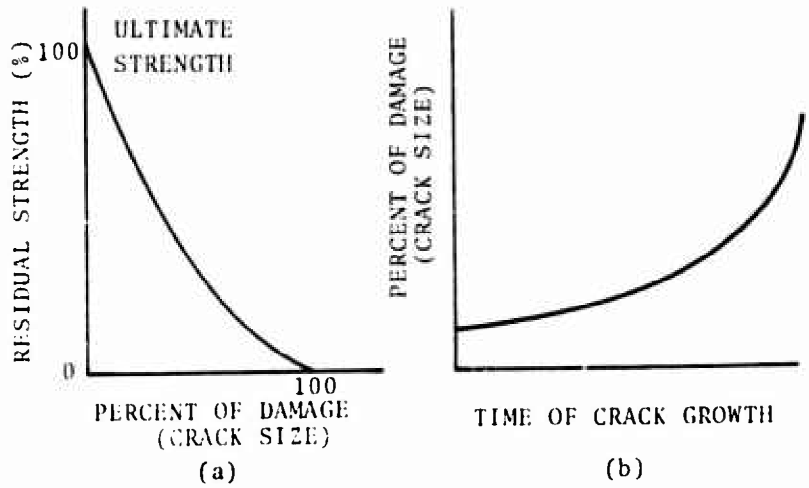


Figure 5. Schematic Representation of Structural Residual Strength and Crack Growth Tendencies.

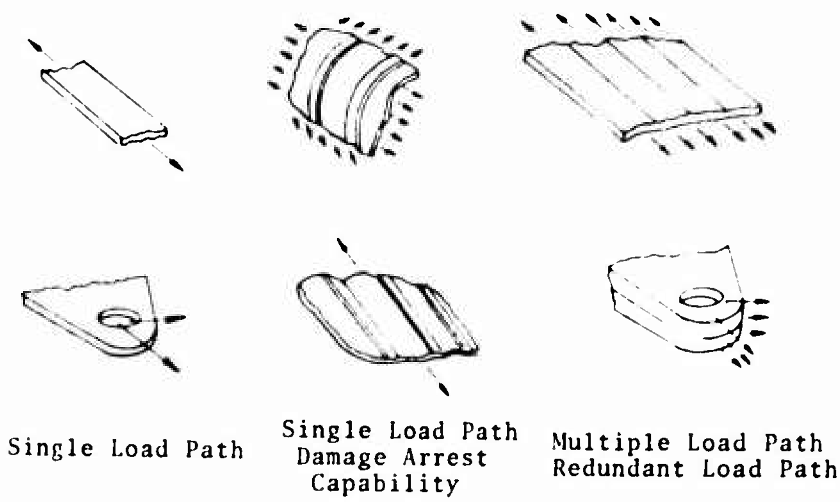


Figure 6. Structural Arrangements.

The residual strength and crack propagation of single load path configurations are illustrated in Figure 5. However, the crack propagation rates for the single load path configuration with a crack-arrest capability and those for the multiple and redundant load path configurations are somewhat different (see Figure 7).

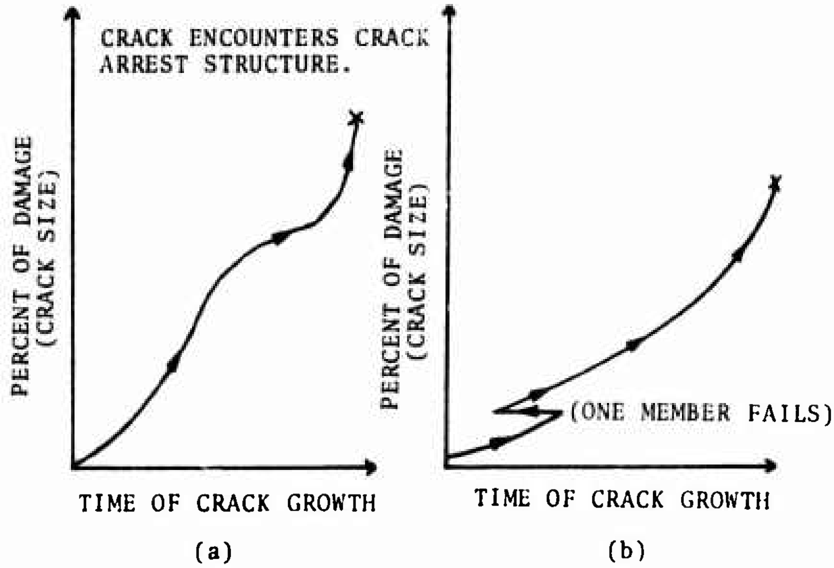


Figure 7. Schematic Representation of Crack Growth Tendencies of Structures With a Crack-Arrest Capability and a Multiload Path Structure.

Figure 7(a) shows the generalized crack propagation characteristics of the single load path configuration in a positive crack arrest structure. The potential advantages of this configuration are apparent when its extended time of crack growth is considered. There are two basic types of structure which may be characterized as positive crack arrest structures: the integral stiffener/crack arrest structure and the riveted or bonded stiffener/crack arrest structure. Work conducted by C.C. Poe (as noted by Hardrath[3]) serves to illustrate the relative crack growth tendencies of these two structures (see Figure 8). The crack growth rate for each structure approximates that of an unstiffened panel when the crack is small and is located between stiffeners. As the crack approaches a stiffener, its rate of growth slows appreciably; however, the two structures react differently from this point on. The riveted or bonded stiffener will allow the crack to pass under it, so that the stiffener remains intact. After the crack passes under the stiffener, its propagation rate will approximate the rate before it encountered the stiffener. This crack performance is not generally so with an integral stiffener since the stiffener must fail before the crack can propagate further in the panel. Therefore, once the stiffener has failed, the crack growth rate is accelerated rather than constrained. Since the examples shown in Figure 8 are not compared on a

[3] Hardrath, H.F., STRUCTURAL INTEGRITY IN AIRCRAFT, January 1973.

"by weight" basis, they should not be used to compare the relative virtues of the two structural types, but rather to illustrate the tendencies in the crack growth phenomenon of each.

Figure 7(b) depicts the crack propagation characteristics of the redundant load path configuration. Reference 1 and others point out the importance of considering the possibility that both or all members of a redundant load path configuration are flawed. It cites the case where "...if stress corrosion is responsible for the existence of subsurface cracks in one member, there is no assurance that each adjoining member does not contain cracks of a similar character."

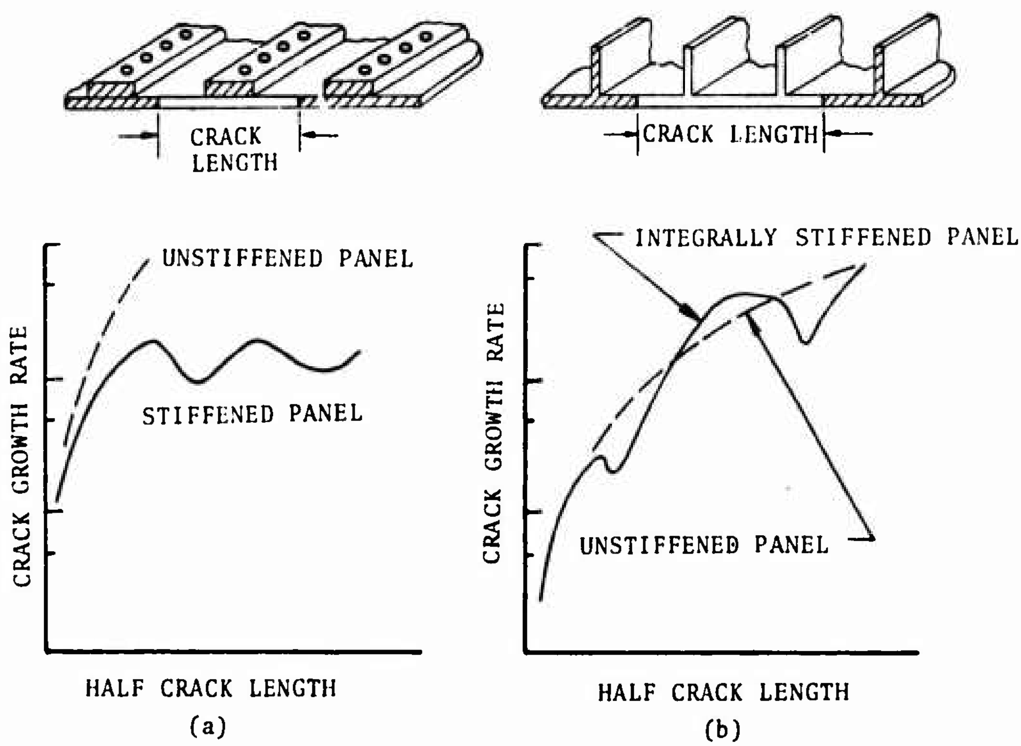


Figure 8. Crack Growth Rate Comparison of Stiffened Panels.

As discussed in Section 3.3.2, the fatigue and fracture phenomena may be used to design a fail-safe structure. However, unless the structure is tolerant of cracks, that is, it can sustain large and slow-growing cracks, the cost and weight penalties generally associated with fail safety are generally prohibitive. Frequent member failures, costly unscheduled maintenance, and aircraft downtime are unacceptable despite the fail-safe capability.

### 3.3.2 Technique of Fail-Safe Design

This section describes a generalized technique for formulating a fail-safe design. The goal of this technique is to determine the time it takes a crack to grow from some initial size to a magnitude where the structural strength has degraded to the extent that the structure can no longer meet the operational loading requirements. This delta time will be termed the effective crack life of the structural part.

The left-hand portion of Figure 9 illustrates the relationship between the residual strength of a structure (expressed as a percentage of its unflawed ultimate strength) and the crack size (expressed in units of percentage of damage where a 100 percent damage represents complete failure). As shown in this figure, the residual strength associated with the limit load stress (termed limit stress) defines the critical crack length, $a_{cr}$. The right-hand portion of Figure 9 shows the relationship between the crack size (again expressed in units of percentage of damage) and the time of crack growth; this relationship is termed the crack propagation curve. In this figure, the crack length parameter ($a_{cr}$) was transferred to the crack propagation curve to define a time, $T_{cr}$, beyond which continued operational loading could lead to catastrophic failure. Similarly, a crack length, $a_i$, was appropriately located on the ordinate of the crack propagation curve to represent the minimum detectable crack size and thus define the time, $T_i$, beyond which the crack is detectable. This latter crack size is largely a function of the inspection technique; Figure 10 compares the relative sensitivities of various nondestructive inspection techniques (Reference 1). The delta time between $T_i$ and $T_{cr}$ represents the effective crack life.

The inspection intervals should be based on the crack life. For example, if the inspection interval is defined as one-third of the crack life, a detectable crack will be inspected at least twice during the crack life, and the structure will have sufficient strength during this period to sustain the limit loading condition if it should occur.

In summary, through an artful manipulation of residual strength and crack size relationships in conjunction with graphical crack propagation tendencies representative of both structural strength and loading, an inspection scheme may be devised so that the probability of detecting a crack is reasonably high before the associated strength degradation reaches dangerous levels.

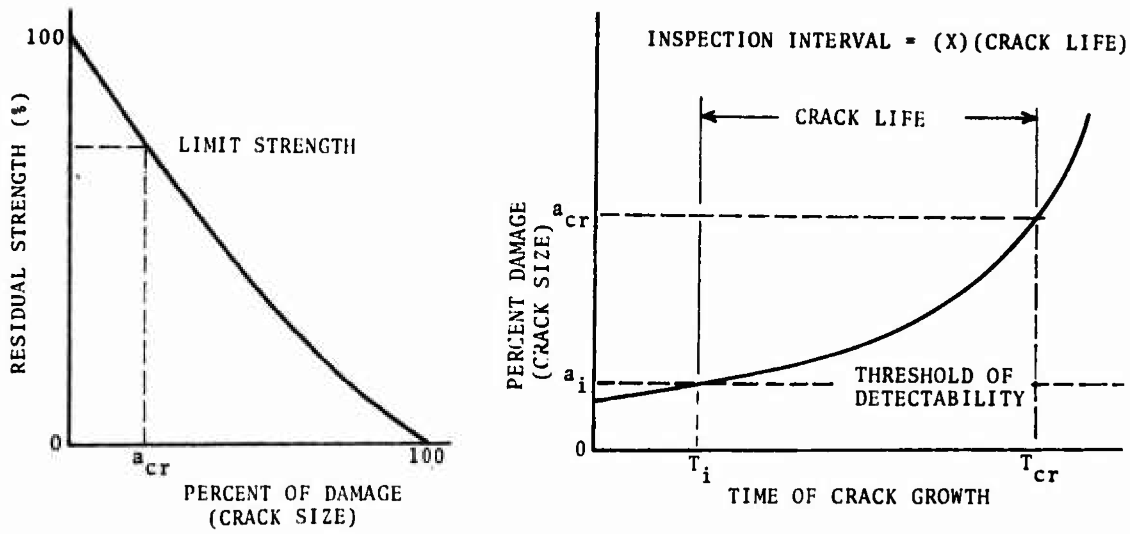


Figure 9. Schematic Representation of Residual Strength/ Residual Life Concept.

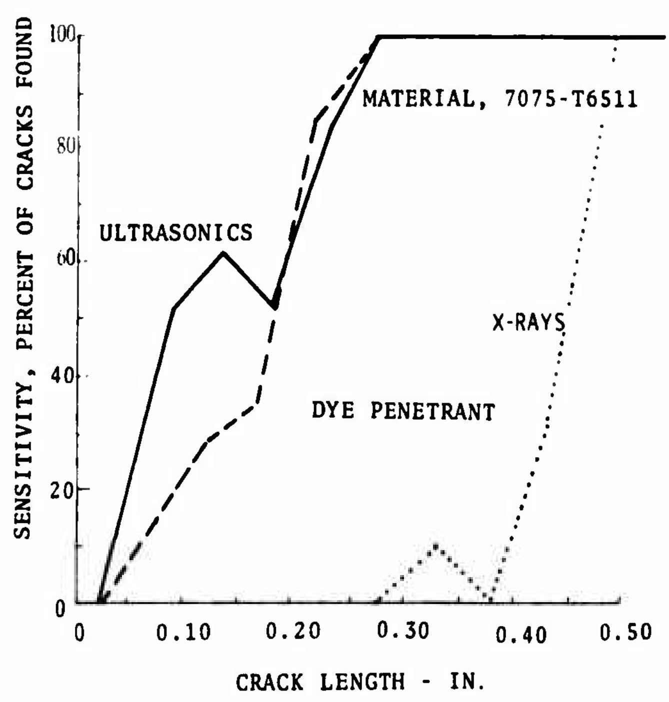


Figure 10. Demonstration of Flaw Detection Capability.

## 3.4 INTERFACE OF FAIL-SAFE AND SAFE-LIFE DESIGN PHILOSOPHIES

Reference 1 and others state that the resistance to crack initiation (crack nucleation in the conventional concept of fatigue) and the resistance to fracture (based on residual strength and crack growth rate characteristics) complement each other.

Section 3.5 summarizes the state of the art of fail-safe/ safe-life design criteria. Based on the discussion in the previous section, these criteria are presented in the form of case examples for the following reasons:

(1) Most importantly, if the design criteria can be proven to provide a low probability of catastrophic failure when a structure is damaged, they may be judged fail-safe design criteria.

(2) Fail-safe designs have various levels of efficiency; that is, some design criteria require more fail-safe provisions than others.

(3) The criteria for the case examples provide several candidates for nearly all critical helicopter structures.

(4) Case examples may reveal how criteria for a particular structure may be varied to suit the structure better while still maintaining the fail-safe design philosophy.

(5) Since the optimum interface between fail-safe and safe-life design philosophies cannot be determined for all helicopter structures collectively, it must be determined for each critical component individually.

## 3.5 PRESENTATION OF FAIL-SAFE/SAFE-LIFE DESIGN CRITERIA

The literature survey, described in Section 2.1, revealed two particularly descriptive reports on current fail-safe design criteria. Each report contains sets of fail-safe design criteria for various structural configurations and levels of inspectability. These criteria were selected as the fail-safe/ safe-life design criteria best representing the state of the art. The design criteria are presented and described separately in Cases 1 through 10.

The design criteria described in Cases 1 through 5 were obtained from a report by Reddick et al.[4], and the safe (fatigue) life criteria were obtained primarily from three other reports[5,6,7]. These criteria represent the initial design criteria from which the final fail-safe design criteria for the HLH rotor hub were chosen. Specifically, the criteria described in Cases 2 and 3 have been incorporated as part of the HLH rotor hub design criteria. Since the design service life requirements (for failed and unfailed configurations) and the requirements for fatigue strength reduction in these cases are representative, they are included in the following presentation.

The design criteria described in Cases 6 through 10 were obtained from Reference 1 and the safe (fatigue) life criteria from MIL-STD-1530[8]. They represent the results of exhaustive studies by the U.S. Air Force to describe a comprehensive fracture and fatigue control plan.

For each of the 10 cases, the fail-safe concept is described schematically and the advantages and disadvantages of the design criteria are cited. In general, Cases 1 through 5 represent design criteria applicable to aircraft structures which experience service loading similar to that of helicopter dynamic components, while Cases 6 through 10 represent design criteria applicable to helicopter fuselage structures.

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<u>FAIL-SAFE/SAFE-LIFE DESIGN CRITERIA</u>

CASE 1: Single load path, on-board flaw-indication device with cockpit display, fatigue life and residual life analyses, instantaneous flaw warning.

CASE 2: Single load path, on-board flaw-indication device, fatigue life and residual life analyses, inspection intervals defined.

CASE 3: Redundant load path, fatigue life analysis for failed and unfailed configurations, inspection intervals defined.

CASE 4: Redundant load path, fatigue life analysis for failed and unfailed configurations, inspection intervals not defined, retirement at service life.

CASE 5: Single load path, fatigue life and residual life analyses, inspection intervals defined.

CASE 6: Single load path, residual life analysis, inspection intervals defined, inspectable structure.

CASE 7: Single load path, residual life analysis, inspection intervals not defined, retirement at service life, noninspectable structure.[1]

CASE 8: Single load path with crack-arrest features, leak-before-break flaw-indication capability,[2] residual life analysis, inspection intervals defined, inspectable structure.[3]

CASE 9: Redundant load path, residual life analysis for failed and unfailed configurations, inspection intervals defined, inspectable structure.[3]

CASE 10: Redundant load path, residual life analysis for failed and unfailed configurations, inspection intervals not defined, retirement at service life, noninspectable structure.[1]

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[1] A structure is noninspectable when its critical crack length is less than its "through-the-thickness" crack length.

[2] When this design configuration does not have a leak-before-break flaw-indication capability, it is treated as a noninspectable redundant structure as in Case 10.

[3] A structure is inspectable when its critical crack length is greater than its "through-the-thickness" crack length.

## CASE 1

### Fail-Safe System Description

(1) Single load path.
(2) On-board flaw-indication device with cockpit display.
(3) Fatigue life and residual life analyses.
(4) Instantaneous flaw warning.

This fail-safe system ensures a timely detection of potentially dangerous cracks through an on-board failure warning device with a cockpit display. Assuming a leak-before-break condition indicator is used, the cockpit display will immediately inform the flight crew of a through-the-thickness crack. The residual life analysis must project an accurate time for the propagation from a through-the-thickness crack to the critical crack length. Figure 11 illustrates the Case 1 design criteria.

### Structural Requirements

(1) Static Strength

A minimum ultimate factor of safety of 1.5 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of components established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by $3\sigma$.

(3) Static Residual Strength

(a) A 100% residual strength is defined as being equal to the ultimate strength.

(b) A critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(4) Residual Fatigue Strength

(a) A crack life is defined as the time for a through-the-thickness crack to propagate to the critical crack size.

(b) A crack life should have a duration of at least one flight.

## Fail-Safe System Advantages

(1) Instantaneous warning of through-the-thickness crack.

(2) Redundant structure (with related weight penalty) which is required to have a high fail-safe efficiency.

(3) Extremely long crack life not required.

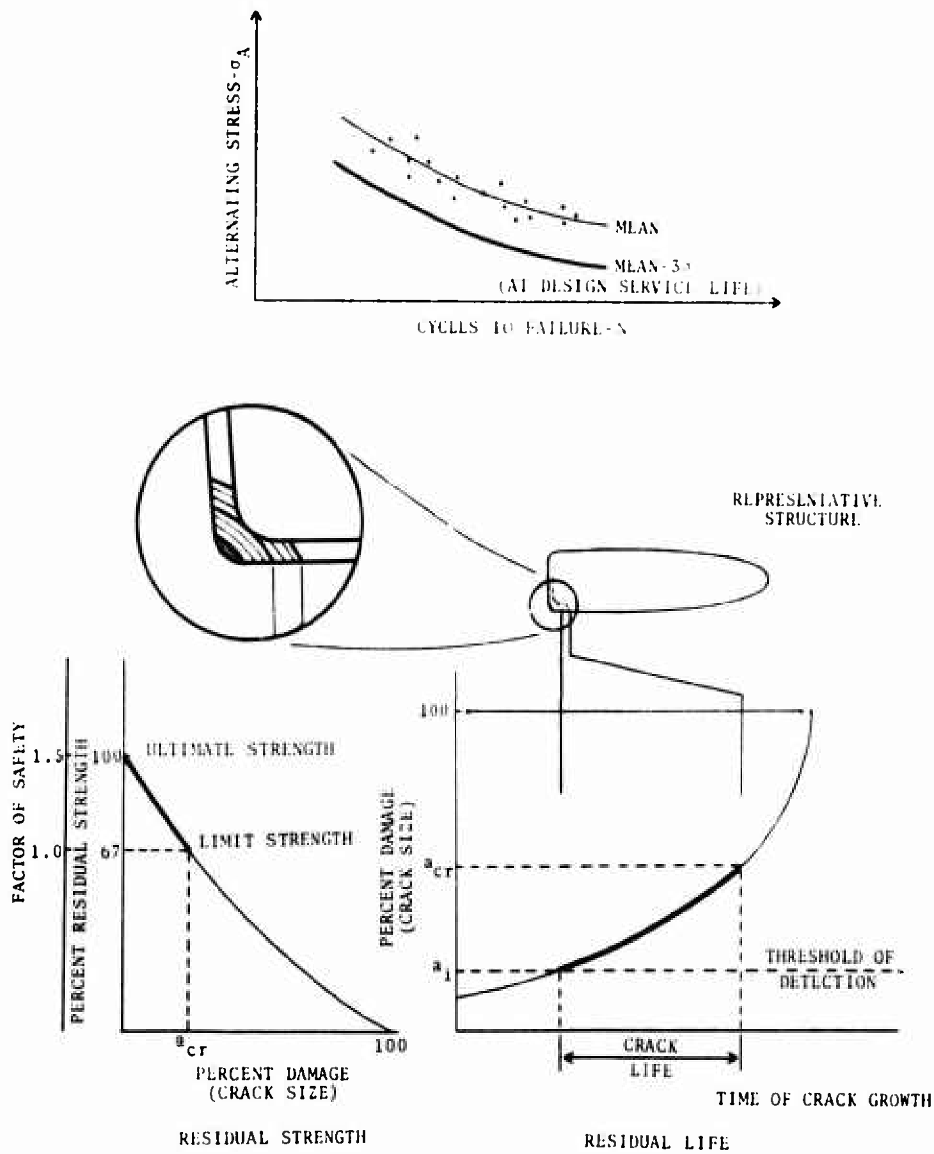


**Figure 11. Case 1 Fail-Safe/Safe-Life Design Criteria.**

### Fail-Safe System Disadvantages

The design, production, and maintenance complexity involved in incorporating an on-board flaw indicator with a cockpit display.

### Helicopter Structure Applicability

Case 1 is very applicable to critical primary load-carrying members of helicopter dynamic components (e.g., rotor blade spar and swashplate). It has significant cost and weight advantages in providing a truly fail-safe structure without incorporating a multiload path configuration.

In general, Case 1 may be applied to any flight critical structure; i.e., any structure whose loss could threaten the airworthiness of the vehicle.

## CASE 2

### Fail-Safe System Description

(1) Single load path.
(2) On-board flaw-indication device.
(3) Fatigue life and residual life analyses.
(4) Inspection intervals defined.

Case 2 ensures a timely detection of potentially dangerous cracks through an on-board failure warning device which must be periodically inspected on the ground. Assuming that a leak-before-break condition indicator is used, a through-the-thickness crack will be indicated when the aircraft flight system is shut down. The ground inspection interval is defined as one-third the crack life (time for a through-the-thickness crack to propagate to the critical crack size) as determined from the residual life analysis. Figure 12 illustrates the Case 2 design criteria.

### Structural Requirements

(1) Static Strength

A minimum ultimate factor of safety of 1.5 on the limit load stress.

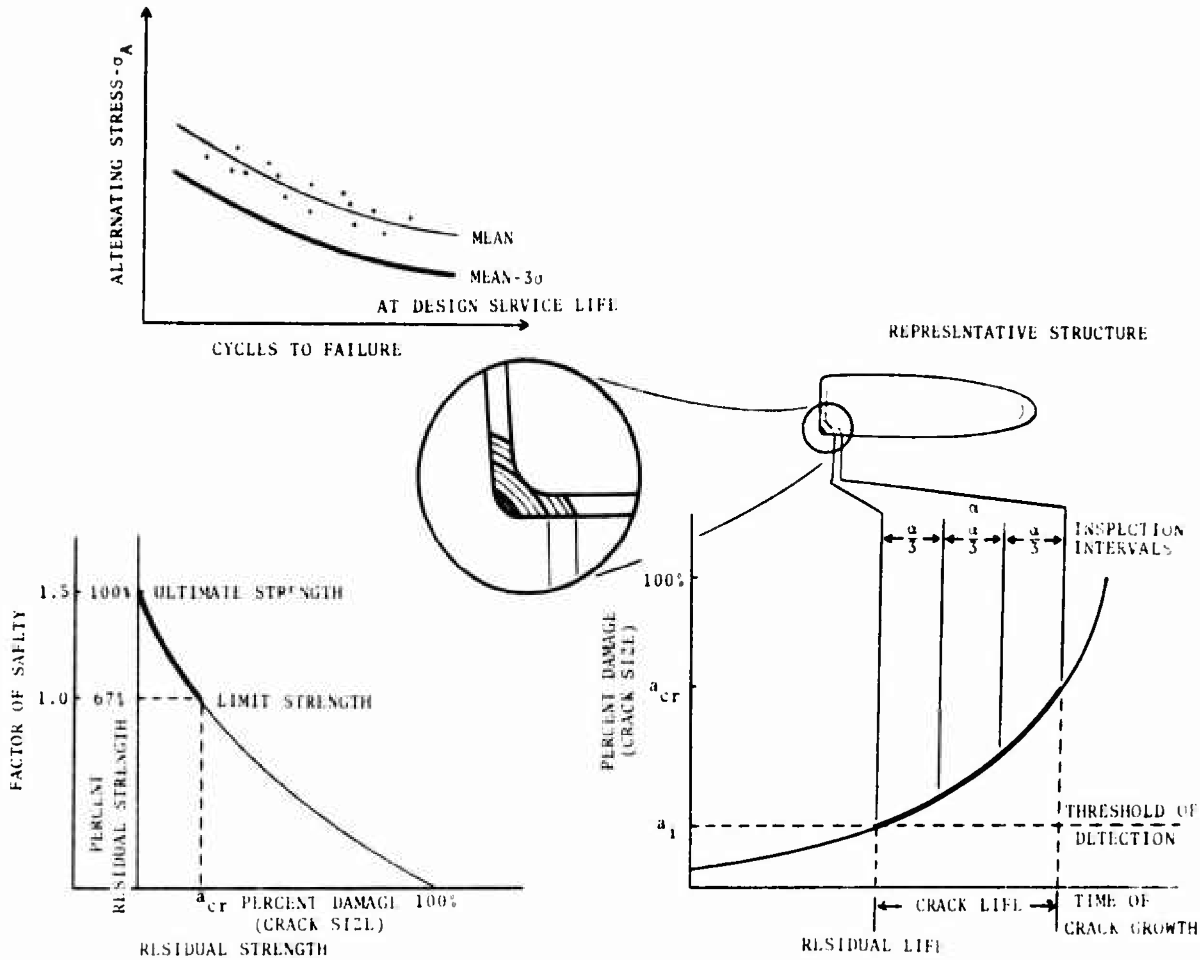


Figure 12. Case 2 Fail-Safe/Safe-Life Design Criteria.

(2) Fatigue Strength

(a) A mean fatigue strength of components established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduce by 3σ.

(3) Static Residual Strength

(a) A 100% residual strength is defined as being equal to the ultimate strength.

(b) A critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(4) Residual Fatigue Strength

(a) Crack life equals the time for a through-the-thickness crack to propagate to the critical crack size.

(b) Crack life is contractually agreed upon; it should last for the duration of at least three flights.

### Fail-Safe System Advantages

(1) A through-the-thickness crack is indicated when the aircraft flight system is shut down.

(2) Redundant structure (with related weight penalty) which is not required to have a high fail-safe efficiency.

(3) Extremely long crack life not required.

### Fail-Safe System Disadvantages

(1) The design, production, and maintenance complexity involved in incorporating an on-board flaw indicator.

(2) Ground inspection requirement.

### Helicopter Structure Applicability

Case 2 is very applicable to critical primary load-carrying members of helicopter dynamic components (e.g., rotor blade spar and swashplate). It has significant cost and weight advantages in providing a truly fail-safe structure without incorporating a multiload path configuration.

In general, Case 2 may be applied to any flight critical structure, i.e., any structure whose loss could threaten the airworthiness of the vehicle.

## CASE 3

### Fail-Safe System Description

(1) Redundant load path.

(2) Fatigue life analysis for failed and unfailed configurations.

(3) Inspection intervals defined.

Case 3 offers an extremely remote probability of catastrophic structural failure by providing a redundant load path and an inspection procedure. The criteria require that the redundant structure satisfy a contractually agreed upon service life (based on a conventional safe-life analysis) and that if one member fails, the remaining structure satisfy a slightly reduced service life. The inspection interval is defined as being equal to one-tenth of the service life of the failed configuration. Figure 13 illustrates the Case 3 design criteria.

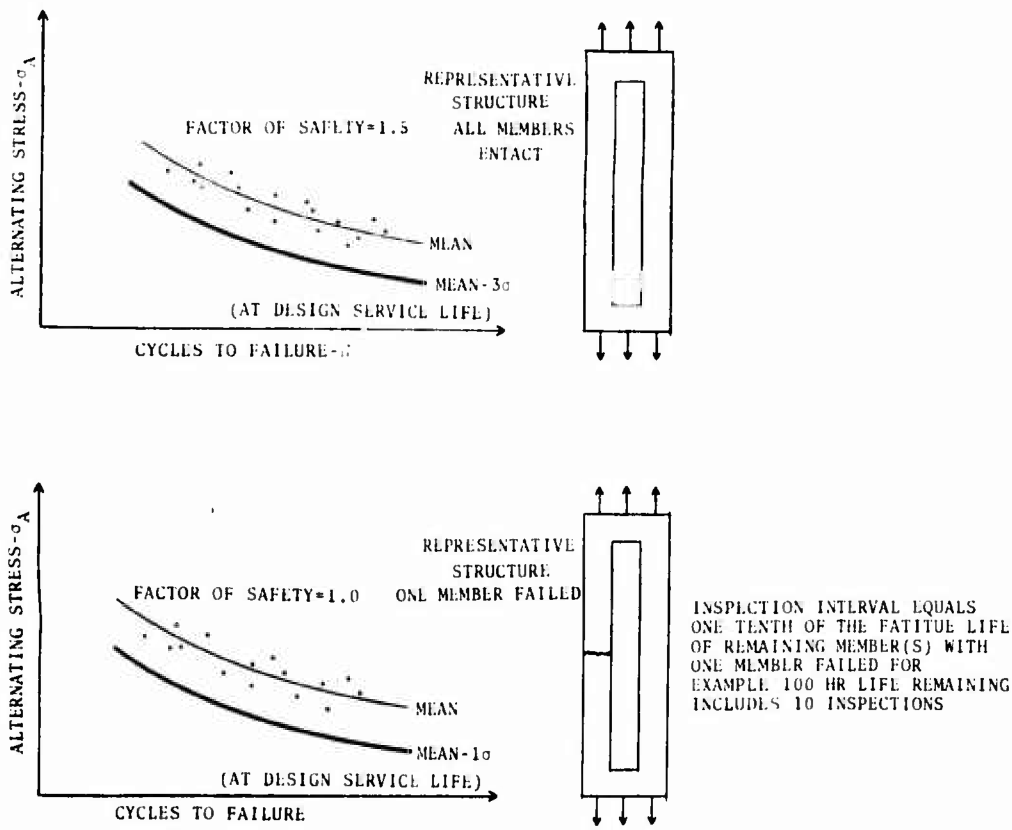


Figure 13. Case 3 Fail-Safe/Safe-Life Design Criteria.

## Structural Requirements

### Before Failure of One Member

#### (1) Static Strength

A minimum ultimate factor of safety of 1.5 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of components established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by $3\sigma$.

After Failure of One Member

(1) Static Strength

A minimum factor of safety of 1.0 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of failed component established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by $1\sigma$.

(c) The inspection interval equals one-tenth the service life.

## Fail-Safe System Advantages

(1) A high probability of detecting a member failure through inspection scheme.

(2) A reasonable and adequate reserve strength requirement consistent with inspection scheme.

(3) A redundant structure which decreases the vulnerability to catastrophic ballistic damage.

## Fail-Safe System Disadvantages

(1) The criteria ignore the possibility of one or more members being flawed when a single member fails.

(2) The criteria ignore residual life (fracture mechanics) factors whose recognition could increase the inspection intervals.

(3) Weight penalties inherent in a redundant structure.

## Helicopter Structure Applicability

Case 3 has its greatest applicability to structures where inherent redundancy is easily achieved and where it is very difficult to utilize damage-tolerant materials. This case mav be applied to dynamic or fuselage structural components.

Because Case 3 fails to recognize the static and the fatigue residual strength phenomena, it should be applied to those structures which have a moderate criticality for the vehicle's airworthiness

# CASE 4

## Fail-Safe System Description

(1) Redundant load path.

(2) Fatigue life analysis for failed and unfailed configurations.

(3) Inspection intervals not defined; retirement at service life.

Case 4 offers an extremely remote probability of catastrophic structural failure by providing a redundant load path. The criteria require that the redundant structure satisfy a contractually agreed upon service life (based on a conventional safe-life analysis) and that if one member fails, the remaining structure satisfy the same service life requirement at a slightly reduced fatigue strength (based on a conventional safe-life analysis). No inspection intervals are defined since the structure should maintain adequate static and fatigue strength for the specified service life even with one member completely failed. Figure 14 illustrates the Case 4 design criteria.

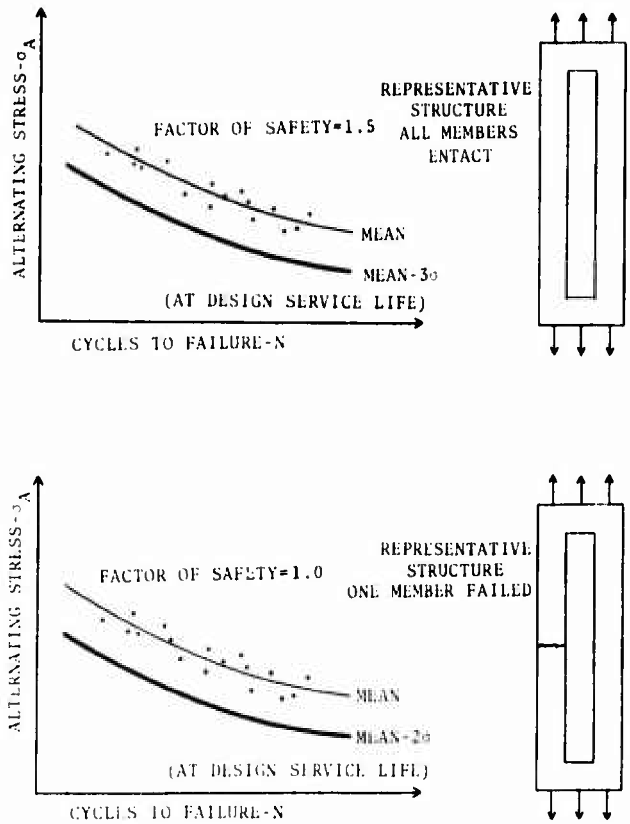


Figure 14. Case 4 Fail-Safe/Safe-Life Design Criteria.

## Structural Requirements

### Before Failure of One Member

(1) Static Strength

A minimum ultimate factor of safety of 1.5 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of components established by applying different constant-amplitude vibratory lcads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by 3σ.

After Failure of One Member

(1) Static Strength

A minimum factor of safety of 1.0 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of failed component established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by $2\sigma$.

## Fail-Safe System Advantages

(1) No costly inspection required to ensure structural reliability.

(2) A redundant structure which decreases the vulnerability to catastrophic ballistic damage.

## Fail-Safe System Disadvantages

(1) The criteria ignore the possibility of one or more members being flawed when a single member fails.

(2) The criteria ignore residual life (fracture mechanics) factors.

(3) Weight penalties inherent in a redundant structure.

## Helicopter Structural Applicability

Case 4 has its greatest applicability to structures where inherent redundancy is easily achieved and where it is very difficult to utilize damage-tolerant materials. Case 4 may be applied to dynamic or fuselage structural components.

Because Case 4 fails to recognize the static and the fatigue residual strength phenomena and does not have inspection requirements, it should be applied to those structures which have a moderate criticality for the vehicle's airworthiness.

CASE 5

## Fail-Safe System Description

(1) Single load path.
(2) Fatigue life and residual life analyses.
(3) Inspection intervals defined.

Case 5 offers a generalized method for detecting potentially dangerous cracks through the recognition of the residual life phenomenon. The ground inspection interval is defined as one-third the crack life (time for the crack to propagate from the threshold of the crack detectability, a function of the inspection method, to the critical crack size) as determined from the residual life analysis. Figure 15 illustrates the Case 5 design criteria.

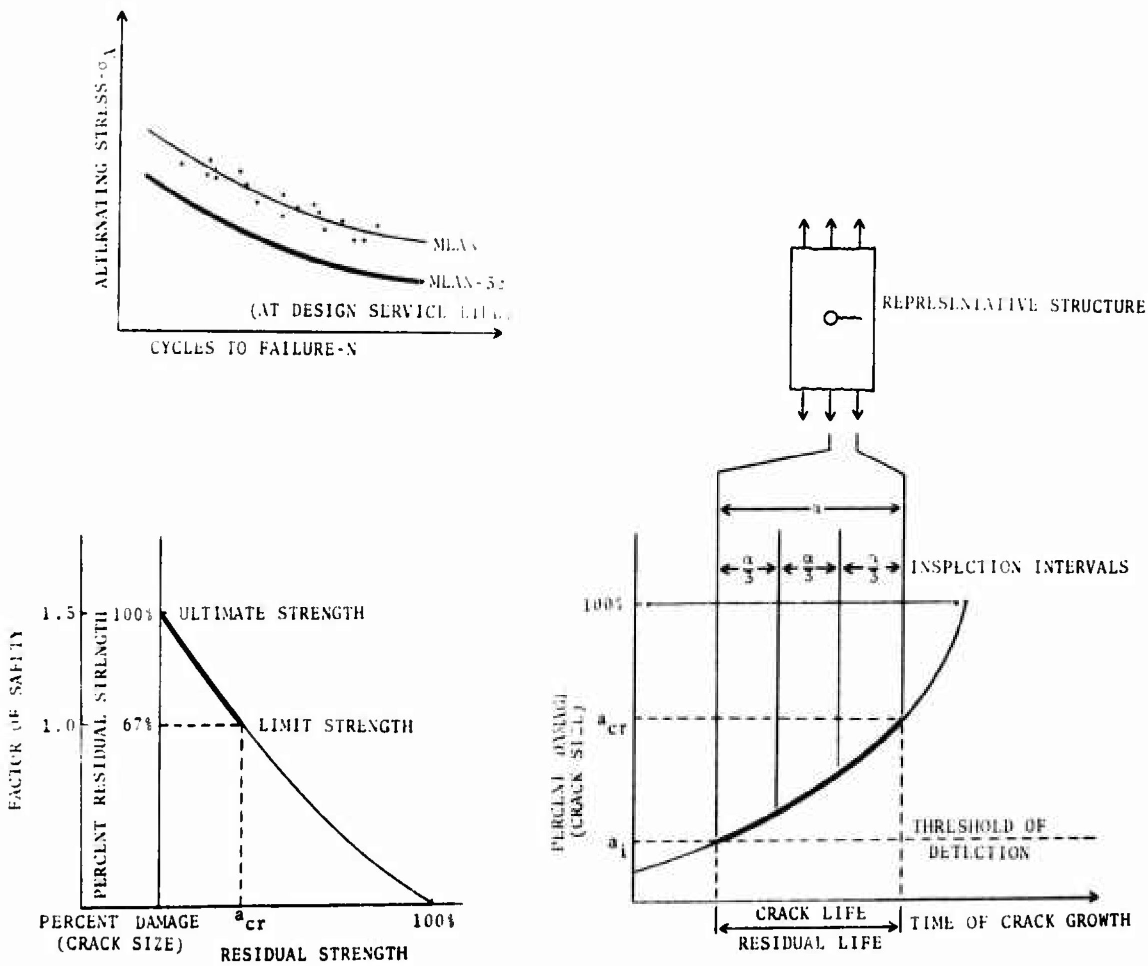


Figure 15. Case 5 Fail-Safe/Safe-Life Design Criteria.

## Structural Requirements

(1) Static Strength

A minimum ultimate factor of safety of 1.5 on the limit load stress.

(2) Fatigue Strength

(a) A mean fatigue strength of components established by applying different constant-amplitude vibratory loads. Fatigue loads should represent all involved loads simultaneously.

(b) A contractually agreed upon service life substantiated by a mean fatigue strength reduced by $3\sigma$.

(3) Static Residual Strength

(a) A 100% residual strength is defined as being equal to the ultimate strength.

(b) A critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(4) Residual Fatigue Strength

(a) The crack life equals the time for the initial crack, whose magnitude is defined by the relative sensitivity of the inspection technique, to propagate to the critical crack size.

(b) The crack life is contractually agreed upon.

## Fail-Safe System Advantages

(1) Redundant structure (with related weight penalty) which is not required to have a high fail-safe efficiency.

(2) The fail-safe criteria may be applied to a structure already in service.

## Fail-Safe System Disadvantages

(1) Detection of a large critical crack size will require high damage-tolerant materials, but visual inspection techniques may satisfy the criteria.

(2) Detection of small critical crack size will require non-destructive inspection techniques.

## Helicopter Structure Applicability

Case 5 offers general fail-safe design criteria applicable to nearly any aircraft structure, either a new design or an in-service design. To be competitive with other fail-safe criteria, Case 5 requires materials with slow crack growth and an inspection procedure which does not require complex equipment and/or extensive structural teardown or removal. Obviously, for an in-service structure not designed to fail-safe criteria, these factors may be economically prohibitive because of the retrofit costs.

## CASE 6

### Fail-Safe System Description

(1) Single load path.
(2) Residual life analysis.
(3) Inspection intervals defined.
(4) Inspectable structure.

Case 6 ensures a timely detection of potentially dangerous cracks in single load path structures which are inspectable. To qualify as inspectable, a structure must have a critical crack length greater than the through-the-thickness crack length, which in turn suggests that the structure be fabricated of damage-tolerant materials. The criteria require that an initial crack of finite magnitude be assumed prior to the first flight. Inspection intervals are defined such that the probability of detecting the crack before it reaches critical dimensions is highly probable. Figure 16 illustrates the Case 6 design criteria.

### Structural Requirements

(1) Static Strength

The structure shall sustain 100% of the design ultimate load.

(2) Fatigue Strength

The structure shall sustain the spectrum of the flight vehicle usage load, in a flight-by-flight sequence, for the service life times a factor of 4 without fatigue cracking.

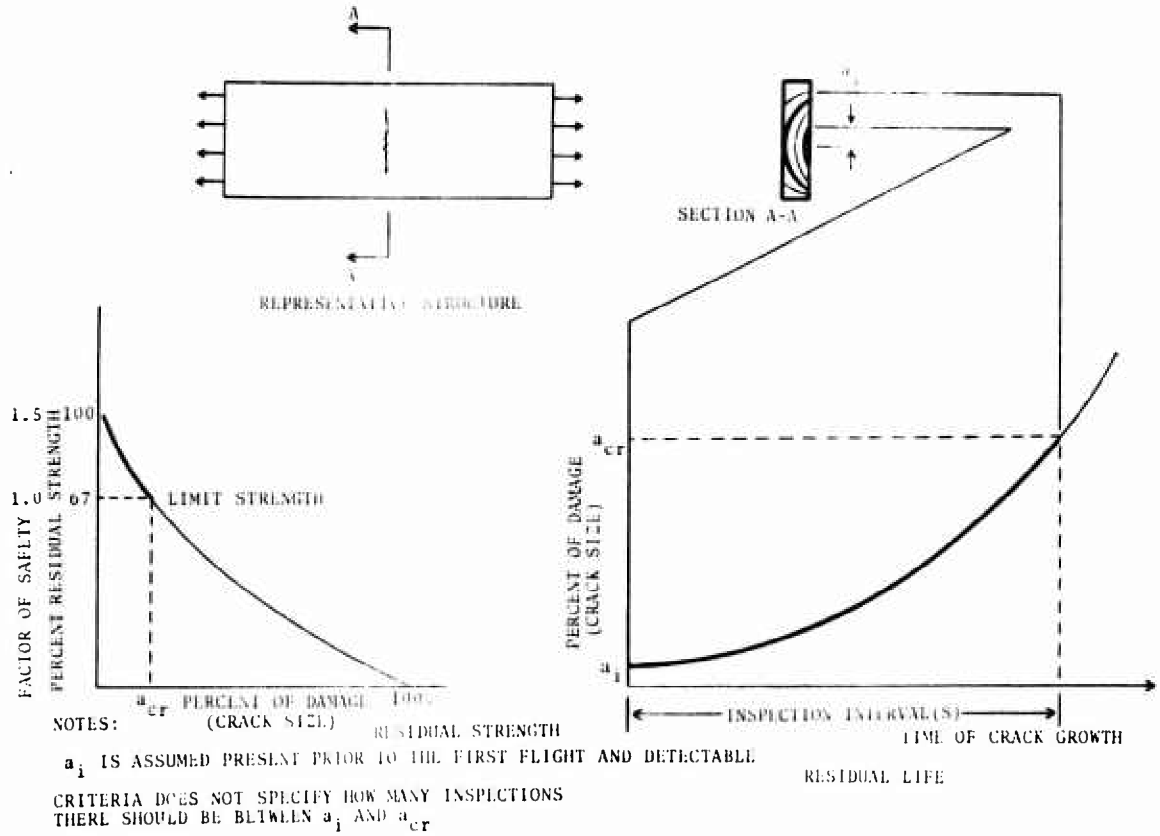


**Figure 16. Case 6 Fail-Safe/Safe-Life Design Criteria.**

(3) Static Residual Strength

(a) The critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(b) The structure shall carry the limit load at the end of one inspection interval.

(4) Residual Fatigue Strength

(a) Crack life equals the time for a through-the-thickness crack to propagate to the critical crack size.

(b) The crack life must have a duration equal to at least one inspection interval.

### Fail-Safe System Advantages

(1) A redundant structure (with related weight penalty) which is not required to have a high fail-safe efficiency.

(2) Extremely long crack life not required.

(3) A leak-before-break condition indicator which would tend to reduce the required crack life and increase the probability of detection may be readily included in the criteria.

### Fail-Safe System Disadvantages

A ground inspection procedure requiring difficult non-destructive and/or visual inspection.

### Helicopter Structure Applicability

Case 6 is particularly applicable to the critical primary fuselage load-carrying members since (1) the fatigue requirements specify a flight load spectrum type of testing which, in general, is more easily made representative of fuselage loading where static loading conditions are critical, and (2) the inherent damage-tolerance of most semimonocoque fuselage structures allows the assumption of an initial finite crack size prior to the aircraft's first flight without imposing economically prohibitive inspection intervals over its design service life.

Case 6 should be applied to helicopter dynamic components; however, a leak-before-break condition indicator would probably be necessary to make the design economically feasible while still satisfying the criteria.

## CASE 7

### Fail-Safe System Description

(1) Single load path.

(2) Residual life analysis.

(3) Inspection intervals not defined; retirement at service life.

(4) Noninspectable structure.

Case 7 ensures a safe residual life for single-load-path structures which are noninspectable. To be noninspectable, a structure must have a critical crack length smaller than its through-the-thickness crack length, which in turn suggests that the structure be fabricated of damage-tolerant materials. The criteria require that an initial crack of finite magnitude be assumed prior to the first flight. With this initial damage assumption, the probability of the crack propagating to critical dimensions within the contractually agreed upon service life must be extremely remote. Figure 17 illustrates the Case 7 design criteria.

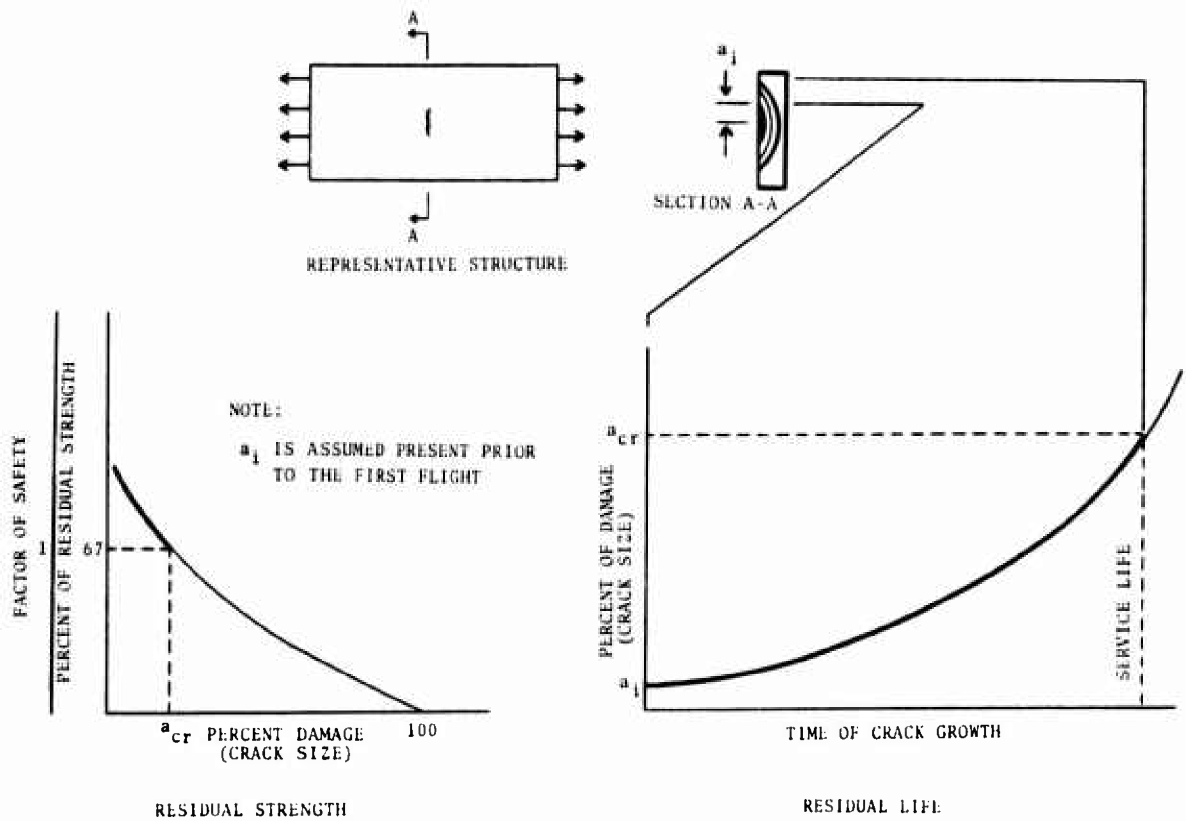


**Figure 17. Case 7 Fail-Safe/Safe-Life Design Criteria.**

## Structural Requirements

### (1) Static Strength

The structure shall sustain 100% of the design ultimate load.

### (2) Fatigue Strength

The structure shall sustain the spectrum of flight vehicle usage loading, in a flight-by-flight sequence, for the design service life times a factor of 4 without fatigue cracking.

(3) Static Residual Strength

(a) The critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(b) The structure shall carry the limit load at the end of one design service lifetime.

(4) Residual Fatigue Strength

(a) The crack life equals the time for a crack to propagate from an assumed initial size to the critical crack size.

(b) Crack life must have a duration of at least one design service lifetime.

## Fail-Safe System Advantages

(1) A redundant structure (with related weight penalty) which is not required to have a high fail-safe efficiency.

(2) Costly, frequent inspection not required.

(3) The criteria provide for a high probability that a crack will not propagate to critical dimensions within the service life.

## Fail-Safe System Disadvantages

Since the criteria do not require inspections during the service life, the effects of minor ballistic damage, maintenance errors, corrosion, etc., may tend to accelerate the crack growth beyond that projected by the residual life analysis.

## Helicopter Structure Applicability

Case 7 may be most applicable to those primary fuselage load-carrying members whose inspection would normally require major structural disassembly and/or removal. Obviously, leak-before-break flaw indication devices cannot be made compatible with the Case 7 criteria. Case 7 should not be applied to helicopter dynamic component structures.

CASE 8

Fail-Safe System Description

(1) Single load path with crack-arrest features.
(2) Leak-before-break flaw indication capability.
(3) Residual life analysis.
(4) Inspection intervals defined.
(5) Inspectable structure.

Case 8 ensures a timely detection of potentially dangerous cracks for single load path structures with crack-arrest features which have a leak-before-break flaw detection capability (this capability requires that the critical crack length be greater than the through-the-thickness crack length). The criteria require that an initial crack of finite magnitude be assumed prior to the first flight. Inspection intervals are defined such that the probability of detecting the crack before it reaches critical dimensions is highly probable. The leak-before-break flaw detection capability requires very frequent inspection intervals, an inspection after each flight. Figure 18 illustrates the Case 8 design criteria.

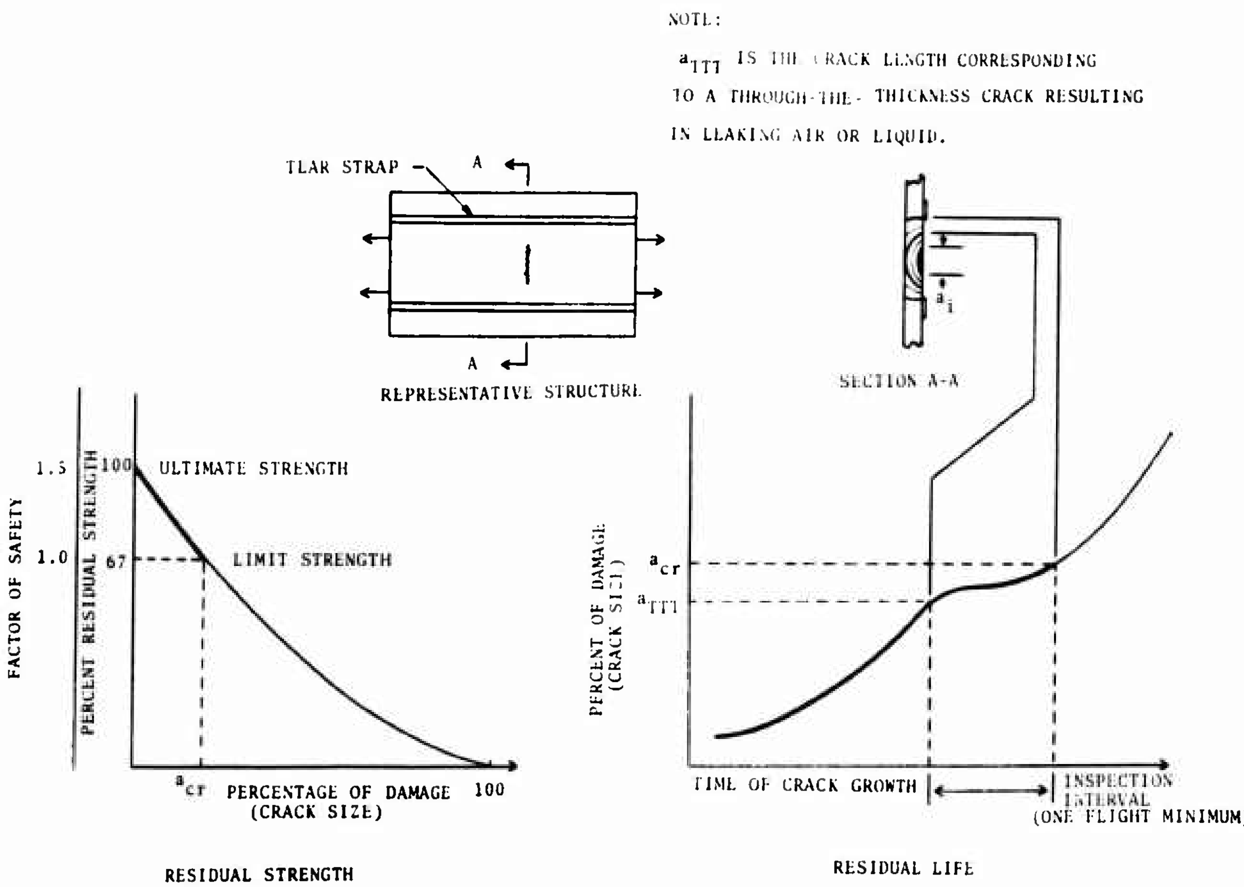


Figure 18. Case 8 Fail-Safe/Safe-Life Design Criteria.

## Structural Requirements

(1) Static Strength

The structure shall sustain 100% of the design ultimate load.

(2) Fatigue Strength

The structure shall sustain the spectrum of flight vehicle usage loading, in a flight-by-flight sequence, for the design service life times a factor of 4 without fatigue cracking.

(3) Static Residual Strength

(a) A critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(b) The structure shall carry the limit load at the end of one inspection interval.

(4) Residual Fatigue Strength

(a) The crack life equals the time for a through-the-thickness crack size to propagate to the critical crack size.

(b) The crack life should have a duration of at least one flight.

## Fail-Safe System Advantages

(1) A through-the-thickness crack is indicated when the aircraft flight system is shut down.

(2) A redundant structure (with related weight penalty) which is not required to have a high fail-safe efficiency.

(3) Extremely long crack life not required.

(4) A positive crack-arrest structure which greatly increases the resistance to crack growth while minimizing weight.

## Fail-Safe System Disadvantages

This system has virtually no disadvantages aside from the requirement for ground inspection between flights.

## Helicopter Structure Applicability

Case 8 is especially applicable to the primary fuselage structure that affects the high criticality for the vehicle's airworthiness. The criteria for Case 8 are nearly the same as those for Case 2 except for the safe (fatigue) life aspects.

## CASE 9

### Fail-Safe System Description

(1) Redundant load path.

(2) Residual life analysis for failed and unfailed configurations.

(3) Inspection intervals defined.

(4) Inspectable structure.

Case 9 ensures the timely detection of potentially dangerous cracks through redundant load paths and inspection intervals based on residual life analyses of both failed and unfailed configurations. The criteria require that an initial crack of finite magnitude be assumed prior to the first flight. An inspection interval is defined as the time it takes a crack to propagate from an initial value to the critical crack length for the condition when one member is completely failed (the critical crack size is greater than the through-the-thickness crack size). Figure 19 illustrates the Case 9 design criteria.

### Structural Requirements

(1) Static Strength

The structure shall sustain 100% of the design ultimate load.

(2) Fatigue Strength

The structure shall sustain the spectrum of flight vehicle usage loading, in a flight-by-flight sequence, for the design service life times a factor of 4 without fatigue cracking.

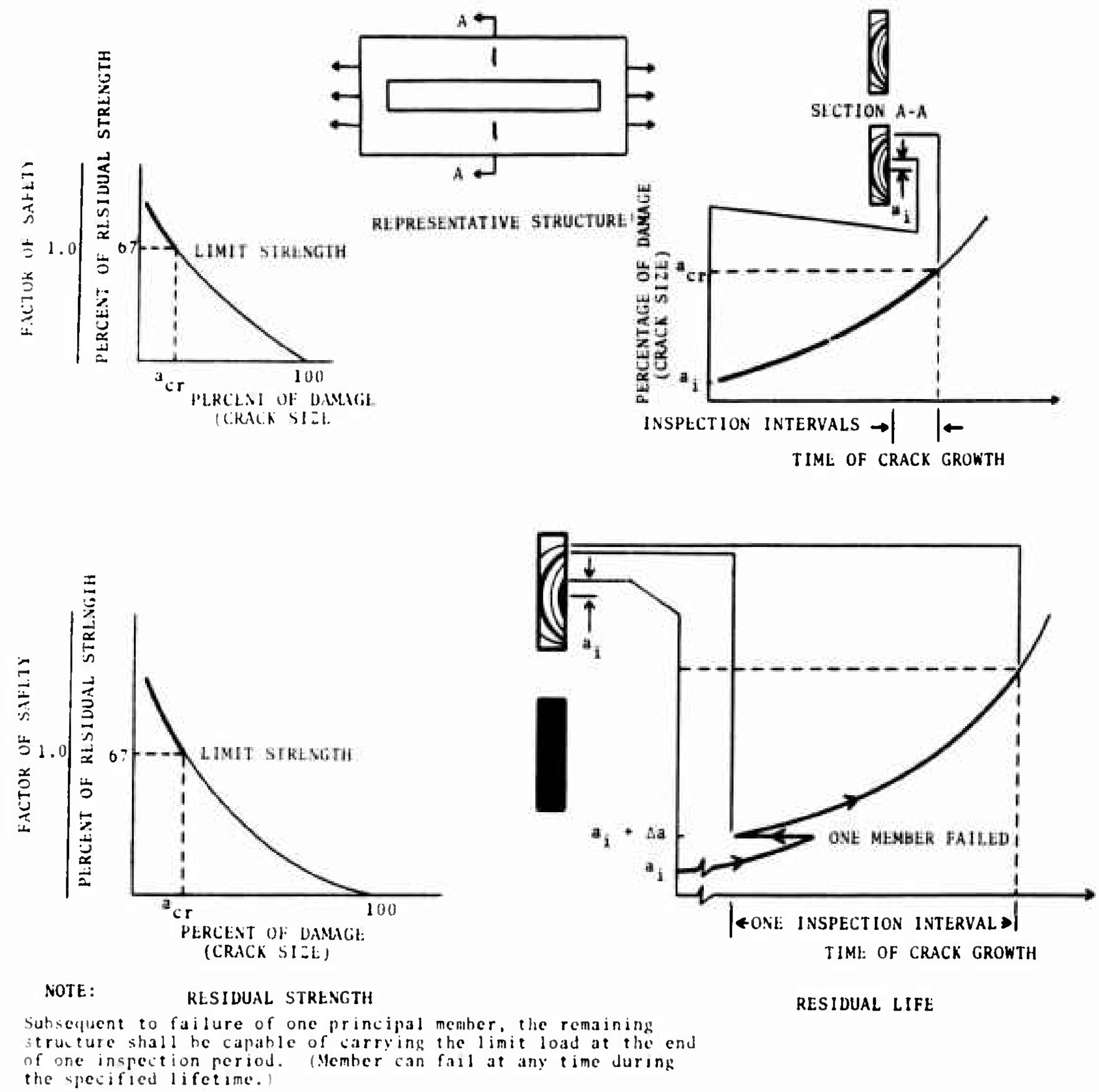


Figure 19. Case 9 Fail-Safe/Safe-Life Design Criteria.

(3) Static Residual Strength

(a) After the failure of one principal member, the remaining structure shall be capable of carrying the limit load at the end of one inspection interval.

(b) The critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(4) Residual Fatigue Strength

(a) The crack life for the failed configuration (one member failed) equals the time for the crack to propagate from the assumed initial crack length to the critical crack length.

(b) The required crack life is contractually agreed upon.

### Fail-Safe System Advantages

(1) A high probability of detecting a member failure through the inspection scheme.

(2) A reasonable and adequate reserve strength requirement consistent with the inspection scheme.

(3) A redundant structure which decreases the vulnerability to catastrophic ballistic damage.

### Fail-Safe System Disadvantages

Weight penalties inherent in a redundant load path configuration.

### Helicopter Structure Applicability

Case 9 is especially applicable to the primary fuselage structure that affects the high criticality for the vehicle's airworthiness. Aside from the fatigue life analysis, the criteria for Case 9 are similar to those for Case 3 except for two important differences: (1) a crack is assumed prior to the first flight; and (2) inspection intervals are based on safe crack-growth predictions resulting from a residual life analysis.

The fail-safe residual life inspection requirement for Case 9, rather than those for Case 3, should be applied to the dynamic components.

## CASE 10

### Fail-Safe System Description

(1) Redundant load path.

(2) Residual life analysis for failed and unfailed configurations.

(3) Inspection intervals not defined; retirement at service life.

(4) Noninspectable structure.

Case 10 ensures a safe residual life for a redundant load path structure which is noninspectable (critical crack length less than the through-the-thickness crack length). The criteria require that an initial crack of finite magnitude be assumed prior to the first flight. The criteria require that one service lifetime equals at least the time it takes a crack to propagate to critical dimensions for the condition when one member is completely failed. Figure 20 illustrates the Case 10 design criteria.

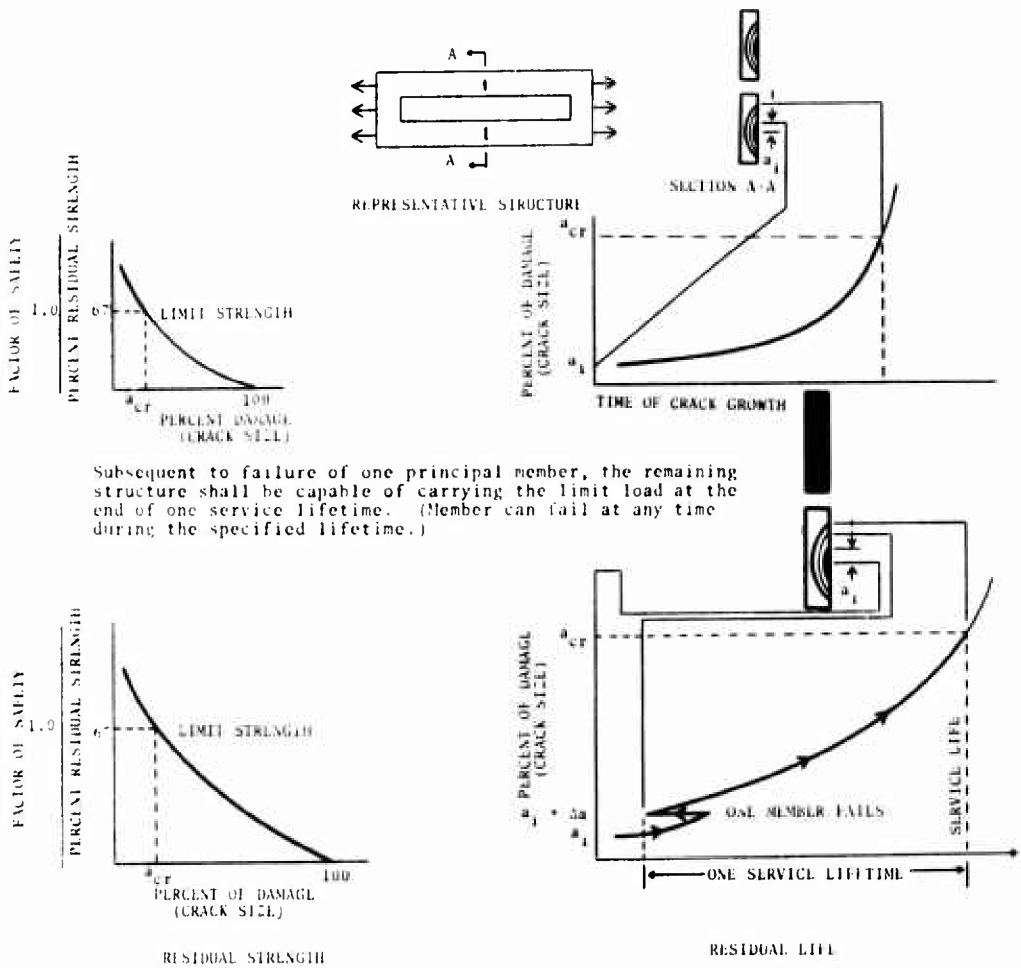


Figure 20. Case 10 Fail-Safe/Safe-Life Design Criteria.

## Structural Requirements

(1) Static Strength

The structure shall sustain 100% of the design ultimate load.

(2) Fatigue Strength

The structure shall sustain the spectrum of flight vehicle usage loading, in a flight-by-flight sequence, for the design service life times a factor of 4 without fatigue cracking.

(3) Static Residual Strength

(a) After the failure of one principal member, the remaining structure shall be capable of carrying the limit load at the end of one service life.

(b) The critical crack size is defined by that residual strength magnitude which is equal to the limit load stress.

(4) Residual Fatigue Strength

(a) The crack life for the failed configuration (one member failed) equals the time for the crack to propagate from the assumed initial crack length to the critical crack length.

(b) The crack life must have a duration of at least one design service life.

## Fail-Safe System Advantages

(1) No costly inspections to ensure structural reliability.

(2) A redundant structure which decreases the vulnerability to catastrophic ballistic damage.

(3) The low probability of catastrophic failure in service because of the requirement that the crack life (with one member failed) be at least equal to the design service life.

## Fail-Safe System Disadvantages

(1) Extreme weight penalty for redundant load path configuration, particularly with the residual life requirement of the design criteria.

(2) No inspections which may be considered a serious deficiency in the fail-safe scheme.

## Helicopter Structure Applicability

Case 10 is strongly recommended for the primary fuselage structure that has a high criticality for the vehicle's airworthiness, and requires major disassembly and/or removal for inspection.

In comparison with the criteria for the other cases, the criteria for Case 10 are not generally economically feasible for dynamic components.

## Comparison of Case Design Criteria

Although the design criteria for the 10 cases have marked similarities, each case represents a unique set of design criteria.

Case 1 represents the ultimate in fail-safe design criteria since it provides for instantaneous warning of a through-the-thickness crack. It does not matter if the structure is configured with single or redundant load paths since the crack will be detected in time for the helicopter to return safely. The literature survey revealed no in-service helicopter with this instantaneous warning system incorporated.

Cases 2 and 6 bear a strong similarity but have two important differences: First, the safe (fatigue) analyses (constant-amplitude S-N vs. load spectrum) are different, primarily because the design criteria for Case 2 were intended for dynamic components whereas those for Case 6 were not. Second, Case 6 requires the assumption that a crack of magnitude $a_i$ be present prior to the aircraft's first flight. This has a subtle effect on the required crack life, the effect for Case 6 being more severe. However, the two criteria are much the same in that they offer excellent fail-safe schemes.

Although Cases 3 and 9 provide fail safety for the same type of structure (inspectable, redundant load path structure), they achieve their goal through much different means. Case 3 does not consider the residual life phenomenon at all; instead, it establishes a safe service life for the structure with one principal member failed and devises a conservative inspection schedule based on the reduced service life. Case 6 uses the results of the residual life analysis to establish appropriate inspection intervals. Since it recognizes that all members of the redundant structure may be damaged, all members are assumed to have an initial crack of magnitude $a_i$ prior to the first helicopter flight. The inspection interval is determined by the situation where one member has completely failed and the other member(s) have sustained damage of magnitude ($a_i + \Delta a$). The time for the crack to propagate from this magnitude to the critical crack size is defined as the inspection interval. This requirement ensures that at least one inspection will be conducted when a crack of detectable magnitude is present. Cases 3 and 9 also treat the safe (fatigue) life analyses differently. A similar comparison can be made between Cases 4 and 10. Cases 9 and 10 offer superior fail-safe assurances because of their recognition and utilization of the crack propagation/residual strength phenomenon.

Aside from the differences in the treatment of the safe (fatigue) life analysis, Cases 5 and 6 are virtually the same. Both characterize and utilize the basic crack propagation (residual strength phenomenon) to establish the economical inspection intervals while providing fail safety. Also based on these principles, Case 8 has two additional requirements: a leak-before-break condition-indicating device and a positive damage-containment structure such as one containing tear straps. Such provisions ensure a fail-safe design scheme which may be applied to structures which may not conveniently incorporate fracture tough materials, slow crack-growth materials, and slow crack-growth structural detail design.

The criteria for Case 7 are based on the recognition and utilization of the residual strength/crack propagation phenomenon to provide a structure such that a crack, assumed to be of finite magnitude a, and present prior to the first flight, will not propagate to its critical length by the end of one service life.

The following comparison of the 10 cases is based on the generally accepted definition of a fail-safe structure:

> "Any structure whose characteristics are such that in the presence of abnormalities, such as fatigue cracking and/or physical damage or deterioration, the probability of a catastrophic failure prior to detection of the abnormality is extremely remote." (Reference 2)

In the light of this definition, the validity of Cases 4, 7, and 10 as fail-safe structural design criteria becomes questionable since they do not explicitly require inspections.

Case 4, in particular, represents no more than a safe (fatigue) analysis for a failed and unfailed configuration in a redundant load path structure.

Although the criteria described in Cases 7 and 10 do not require inspection, they should be considered valid fail-safe criteria since they meet the requirement, "...the probability of a catastrophic failure..is extremely remote," because of the following two safe crack-growth provisions:

(1) The criteria in each of the two cases specify that a finite crack be assumed present prior to the first flight.

(2) The criteria in each of the two cases ensure that the structure will have adequate residual strength to carry the limit load stresses at the end of one design service life.

The criteria for Cases 7 and 10 are particularly applicable to those structures whose failure detection through visual inspection, nondestructive inspection, or flaw indication (leak-before-break) devices is not feasible.

Table 1 summarizes the remaining design criteria for Cases 1 through 3 and Cases 5 through 10, and cites those criteria judged most desirable. Based on an analysis of all known fatigue and fracture mechanisms, the resulting heirarchy of suggested design criteria is ordered primarily according to the highest probability of detecting an abnormality prior to catastrophic failure of the structure or at least an adequate assurance that the structure will function without catastrophic failure.

The family of fail-safe design criteria listed in Table 1 should be applied to all critical helicopter structures with minor to moderate revisions in the criteria (particularly in the safe fatigue life requirement).

## 3.6 SUMMARY

Although the safe-life and fail-safe design philosophies are both necessary for aircraft structural integrity, their interface is sometimes hard to define. Such interfacing ultimately depends on the desired degree of fail safety. Of the 10 cases of design criteria considered most representative of the state of the art, only the criteria for Case 4 were not considered fail-safe design criteria.

## TABLE 1. FAIL-SAFE DESIGN CRITERIA SUMMARY

| Case | Adequate Safe (Fatigue) Life Requirements | Specific Helicopter Structural Application | Criteria Assure Fail Safety (Method) | | Recommended Heirarchy | |
|---|---|---|---|---|---|---|
| | | | Through Inspections | Based On | Single Load Path | Redundant Load Path |
| 1 | Yes | Dynamic Component | Yes | Residual Strength/Crack Propagation Rate | 1 | - |
| 2 | Yes | Dynamic Component | Yes | Residual Strength/Crack Propagation Rate | 3 | - |
| 3 | Yes | Dynamic Component | Yes | Reduced Service Life of Failed Configuration | - | 3 |
| 5 | Yes | Dynamic Component | Yes | Residual Strength/Crack Propagation Rate | 4 | - |
| 6 | Yes | Airframe | Yes | Residual Strength/Crack Propagation Rate | 5 | - |
| 7 | Yes | Airframe | No | Safe Crack Growth/Retirement at Design Service Life | 6 | - |
| 8 | Yes | Airframe | Yes | Residual Strength/Crack Propagation Rate | 2 | - |
| 9 | Yes | Airframe | Yes | Residual Strength/Crack Propagation Rate of Failed Configuration | - | 1 |
| 10 | Yes | Airframe | No | Safe Crack Growth of Failed Configuration/Retirement at Design Service Life | - | 2 |

To design a fail-safe structure requires consideration of the following:

(1) Adequate fatigue analyses and tests.

(2) A safe-life analysis compatible with the structural loading.

(3) Knowledge of the structural residual strength and crack growth characteristics.

(4) A definition of the inspection procedures.

(5) Assurance of safe crack growth for a prescribed time period.

(6) A redundant load path and leak-before-break flaw-indication device considered as desirable but not necessary.

# 4. STATE-OF-THE-ART FAIL-SAFE/SAFE-LIFE DESIGN METHODOLOGY

## 4.1 INTRODUCTION TO HELICOPTER DYNAMIC COMPONENT DESIGN METHODOLOGY

The rotor blades, hub, and controls and the drive system represent the components which make the helicopter unique among aircraft. Since these components are loaded at frequencies which are multiples of rotor speed, they experience 7 to 70 million load cycles per 1000 operating hours. These loads have a much narrower dispersion from low to high amplitude than those of fixed-wing aircraft or even of other helicopter components such as the fuselage, as illustrated in Figure 21. Consequently, the study of structural fatigue due to high-cycle, low-amplitude loads has been the prime concern of helicopter designers and analysts over the past 20 years (Reference 6).

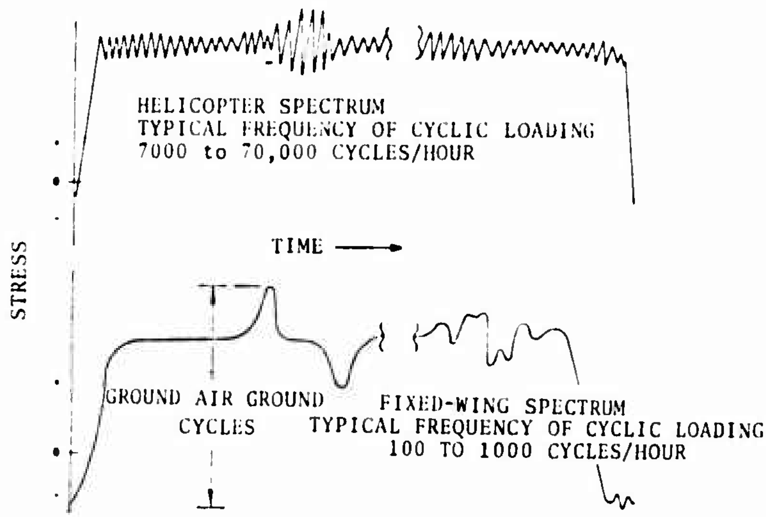


Figure 21. Typical Stress Spectra for a Helicopter Blade Root-End Component and a Transport Airplane Wing Root.

Innovative fail-safe design techniques for dynamic components have been prompted because of the inadequacy of relying wholly on the safe-life philosophy. As a result of such reliance, manufacturing defects due to deficient equipment and human error and service-induced defects due to operation in hostile environments and maintenance error have remained undetected. Consequently, with such defects and the extreme high-cycle loading on the dynamic components, rotor systems have failed with the consequent loss of aircraft[9].

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[9] Thompson, G.H., and Weiss, W.L., FAIL-SAFETY FOR THE H-46 ROTOR BLADE, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

The following sections describe the safe-life design philosophy as applied to the dynamic components and discuss, with particular emphasis on the fail-safe aspects, the design methodologies for each of the prime dynamic components, namely, rotor blade, blade retention, hub, and rotating controls.

### 4.1.1 Safe-Life Determination for Helicopter Dynamic Components

Since the safe-life design of a component is based on the component removal before the completion of its service life so that its fatigue failure is extremely remote, the fatigue characteristics of the component must be known so that its safe service life may be established.

The effective service life of a component depends primarily on the load spectrum to which the component is subjected and the resistance (commonly termed fatigue strength) of the component to such loads. Figure 22 illustrates a general method for determining a safe service life. Since, as stated in Reference 6, the parameter values on the left of the diagram are statistical, the resulting service life is also statistical.

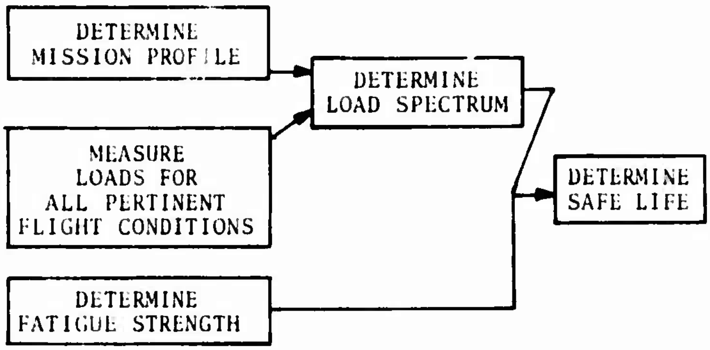


Figure 22. Method for Determining Safe Life.

To establish the safe operating life for dynamic components at the design outset, industry and Government representatives stated that the most widely used technique is the constant-amplitude S-N curve testing and cumulative damage calculation method and that the load spectrum testing technique is rarely used because the loads spectrum is initially not sufficiently definitive and changes with subsequent vehicle modifications and mission variations.

Usage of the constant-amplitude S-N curve technique to establish conservative mission profiles must rely on experience and judgment, and requires extensive aircraft instrumentation to measure loads for specific flight conditions and bench tests of full-scale production components to determine their fatigue strength. Then the theory of cumulative damage is applied to the combined resultant data to project the expected number of load cycles to failure. Table 2 summarizes the more prominent safe-life techniques used by helicopter manufacturers.

TABLE 2. COMPARISON OF FATIGUE LIFE DETERMINATION TECHNIQUES

| Technique | Endurance Limit Used | Flight Loads Used | K-Factor (K x Calculated Life = Service Life) |
|---|---|---|---|
| A | Bottom of scatter or 80% of mean | Top of scatter | 0.75 |
| B | Mean minus 3 Sigma | Top of scatter | 0.85 |
| C | Mean minus 3 Sigma | Top of scatter | 1.00 |
| D | Mean | 90% Probability of occurrence | Varies with number of test specimens and redundancy: 0.181 for 10 specimens* |
| E | Mean | Mean | 0.25 |
| F | Random selection + 3 Sigma | Probability curves fitted to data by least squares fit. ± 3 Sigma | 1.00** |
| G | Mean minus 3 Sigma* | Mean plus 3 Sigma* | 1.00 |
| H | Special testing technique employed. Result used directly to obtain life | Top of scatter | Varies with number of test specimens and redundancy: 0.25 for steel and 0.125 to 0.167 for aluminum; 6 specimens tested* |

* For safe-life structure only.
** Random Monte Carlo technique used.

The differences among the analysis techniques are due primarily to the mode and magnitude of the conservatism in their analytical methods. As stated in Reference 6, the techniques yield moderately large differences in retirement lives. Although these differences are appreciable, they have generally little practical bearing since most of the fatigue failures leading to catastrophic accidents have been due to causes not accounted for in the statistical results of these methods.

In summary, although the safe-life design philosophy provides a conservative safe fatigue life estimate, it does not account for most of the causes inducing catastrophic failure. Consequently, the need to apply fail-safe design principles to critical dynamic components is obvious.

The following sections deal with the fail-safe design methodology for helicopter dynamic components, specifically the rotor blade, blade retention mechanism, hub, and rotating controls. In these sections, the principles for the fail-safe design methodology are considered to complement those for the safe-life design methodology.

### 4.1.2 Fail-Safe/Safe-Life Design Methodology for Main and Tail Rotor Blades

The major subassemblies in the conventional main rotor blade are the spar, the root-end attachment, and the aft fairing assembly. Of these, the critical load-carrying members are the spar and the root-end attachment. Many of the innovative fail-safe design techniques have originated in the design of the main rotor blades.

As stated in Section 3, the three fundamental configurations available for designing damage-tolerant structures are (1) single load path; (2) single primary load path with auxiliary crack-arrest features; and (3) multiple and redundant load paths, each consisting of damage-tolerant materials. By necessity, only the single load path configuration for the rotor blade spar has gained wide acceptance; however, design concepts for the other two design configurations are being developed. Some of these design concepts are illustrated in Figure 23.

Figure 23(a) shows two conventional single load path rotor blade spars. Either the D-shaped spar or the C-shaped spar may incorporate internal crack-arrest structures as shown in Figure 23(b). In this design the increased spar wall thickness provides stiffening as well as structural areas resistant to crack propagation. The internal crack-arrest structure is also attractive from a ballistic impact/survivability point of view[10]. The "D" spar configuration has higher torsional stiffness per pound than the conventional "C" spar, while the "C" spar is simpler to fabricate.

The closed (monolithic) "D" spar lends itself well to differential-pressure condition-indicating systems. Using this basic spar configuration, Sikorsky developed the blade inspection method (BIM) which measures pressure differential by pressurizing the spar cavity, and Boeing-Vertol developed the integral spar inspection system (ISIS) which measures pressure differential by evacuating the sealed spar cavity. Both systems incorporate an indicator connected to the spar at

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[10] Rich, M.J., VULNERABILITY CONSIDERATIONS IN THE DESIGN OF ROTARY WING AIRCRAFT STRUCTURES, Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut, 1969.

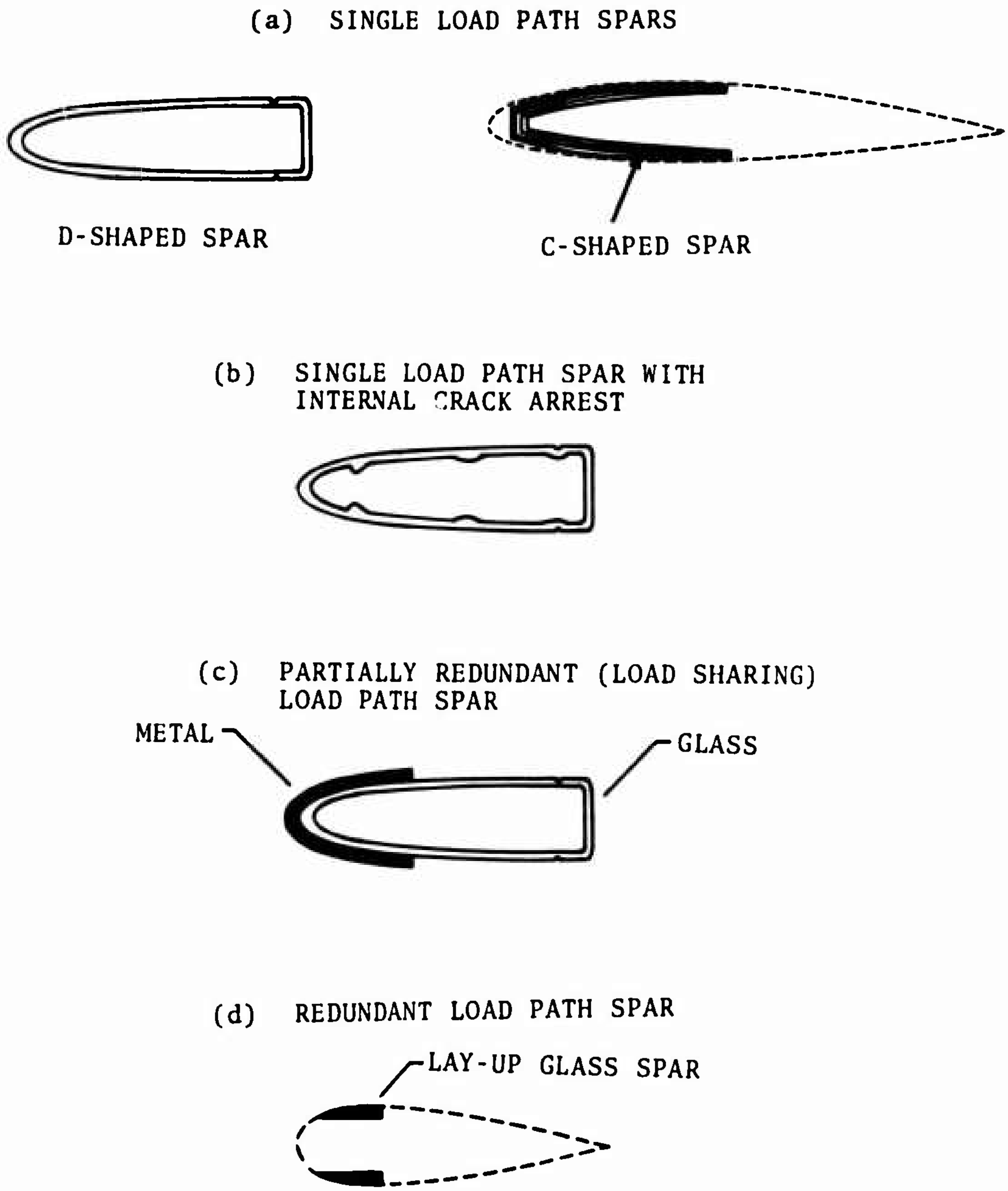


Figure 23. Rotor Spar Design Techniques.

the root of the blade. This indicator registers a warning whenever the pressure differential is lost because of a crack in the spar (Reference 9 and a report by Field et al.[11]). The pressure-differential condition-indicating system is lightweight and relatively inexpensive. Although such a condition-indicating system has a high confidence level, as indicated in the Government/industry survey, the period between the time when the system initially indicates the pressure-differential loss and the time when the crack reaches its critical length is relatively short, typically 20 to 50 hours. However, the detection time for the proposed HLH blade spar, which has a configuration similar to that in Figure 23(c), is claimed to be 200 hours[12].

Composite materials have become increasingly more important in the design of rotor blade spars. As indicated in Figure 23, some of the spar configurations may be constructed of composite materials, but those shown in Figures 23(c) and (d) must be so constructed to prevent excessive weight penalties.

Metals and composite materials generally have different failure modes. While a crack in metallic materials will normally propagate perpendicular to the principal tensile load, a crack in composite materials will propagate according to the type of composite and loading. Table 3 (taken from Reference 11) compares the "hard" failure mode of high modulus composites, such as boron-epoxy, with the "soft" failure mode of low modulus composites. The failure mode of the lower modulus composites is often characterized by more resin breakdown and less fiber breakage. The air permeability of composites changes with resin breakdown to the extent that it is technically feasible to incorporate pressure-differential condition indicators in unidirectional epoxy-fiber composite spars.

As shown in Figure 23(c), this combination of material characteristics is included in the metal-composite design configuration where the metal part of the spar will fail first with the glass composite remaining undamaged. In discussing tests on this configuration, Reference 11 states:

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11 Field, D.M., Finney, R.H., and Stratton, W.K., ACHIEVING FAIL SAFE DESIGN IN ROTORS, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

12 Scarpati, T., Sanford, R., and Powell, R., THE HEAVY LIFT HELICOPTER ROTOR BLADE, The Boeing Vertol Company, Philadelphia, Pennsylvania and U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, May 1973.

"...The tests of glass composites, in combination with steel and titanium, have included undamaged specimens, specimens with prior damage to the metal, specimens with prior damage to the glass, and specimens with simulated bullet damage to both the metal and glass. In all cases, the metal failed first; the damage did not propagate to the glass, and the effect on the glass composite was no greater than the net area lost by any initial damage to the glass..."

Additionally, the failure modes of the metals and composites were unchanged while the crack propagation rate of the metal was markedly reduced by a factor of more than three. According to the preceding phenomena, the proposed HLH rotor blade spar can satisfy the following severe operational design criteria:

(1) A greater than 3600-hour service life based on standard safe-life fatigue analysis.

(2) A fail-safe blade with 200 hours of safe operation after its first failure.

The design of the HLH rotor blade satisfies the above criteria while maintaining a reasonable structural weight.

TABLE 3. FATIGUE FAILURE THEORY UNIDIRECTIONAL COMPOSITES

| | BORON | GLASS |
|---|---|---|
| INITIATING | Random fiber breaks | Random fiber breaks<br>Micro cracks in resin |
| PROPAGATING | Accumulation of fiber<br>Possible fatigue of fibers<br>Limited disbonding and resin breaks | Propagation of resin cracks over broad area<br>Extensive disbonding<br>Gradual loss in stiffness |
| NEAR FAILURE | Small loss in stiffness<br>Rapid accumulation along "weak plane" | Large loss in stiffness<br>Extension longitudinal cracks |
| FAILURE | Sharp break | General breakdown over large area |

Finally, Figure 23(d) illustrates a redundant spar design concept where the spar would be constructed of composite materials. This concept is documented in a Sikorsky Aircraft report[13]. The design study was conducted to construct and test a twin-beam composite rotor blade. Although the damage-tolerant aspect of the blade was not investigated thoroughly, and the blade was not designed to any specific fail-safe design criteria, the reference makes the statement that:

> "...the composite blades exhibit twice the fatigue strain capability of comparable aluminum blades. In addition, crack propagation is considerably slower and critical crack size larger in the composite blade."

While the design technique appears feasible, totally redundant spars, whether they be composite, metal, or metal-composite combinations, are not currently practical because of the 20% to 40% weight penalty (Reference 11). However, this design has potential weight savings and low costs that may materialize through improved aerodynamic geometry, dynamic tuning, and damage tolerance (all made possible by constructing the proposed blade entirely of composite materials), as documented in Reference 13.

Also critically important to the reliable operation of the rotor blade is the root-end retention system, consisting of a root-end and a root-end grip attachment. There are numerous fail-safe design configurations for root-end retention. The more prominent configurations are shown in Figure 24.

The single pin wraparound concept shown in Figure 24(a) may be constructed of metallic or composite materials with the fittings constructed of metal characterized by slow crack growth. When composites are incorporated in such a construction, the resulting structure is extremely strong, and flaw-indicating devices are virtually unnecessary. However, it is feasible to include vacuum paths running through the interior of the composite wraparound structure. Then these paths may be interconnected in the spar pressure-differential condition-indicating system so that one indicator may monitor the entire spar (Reference 11). Additionally, with hingeless rotors, changes in stiffness of the wraparound structure caused by damage proliferation may be detected as changes in the droop and dynamic response (this is known as passive fault condition indicating).

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[13] Salkind, M.J., THE TWIN BEAM COMPOSITE ROTOR BLADE, Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut, May 1973.

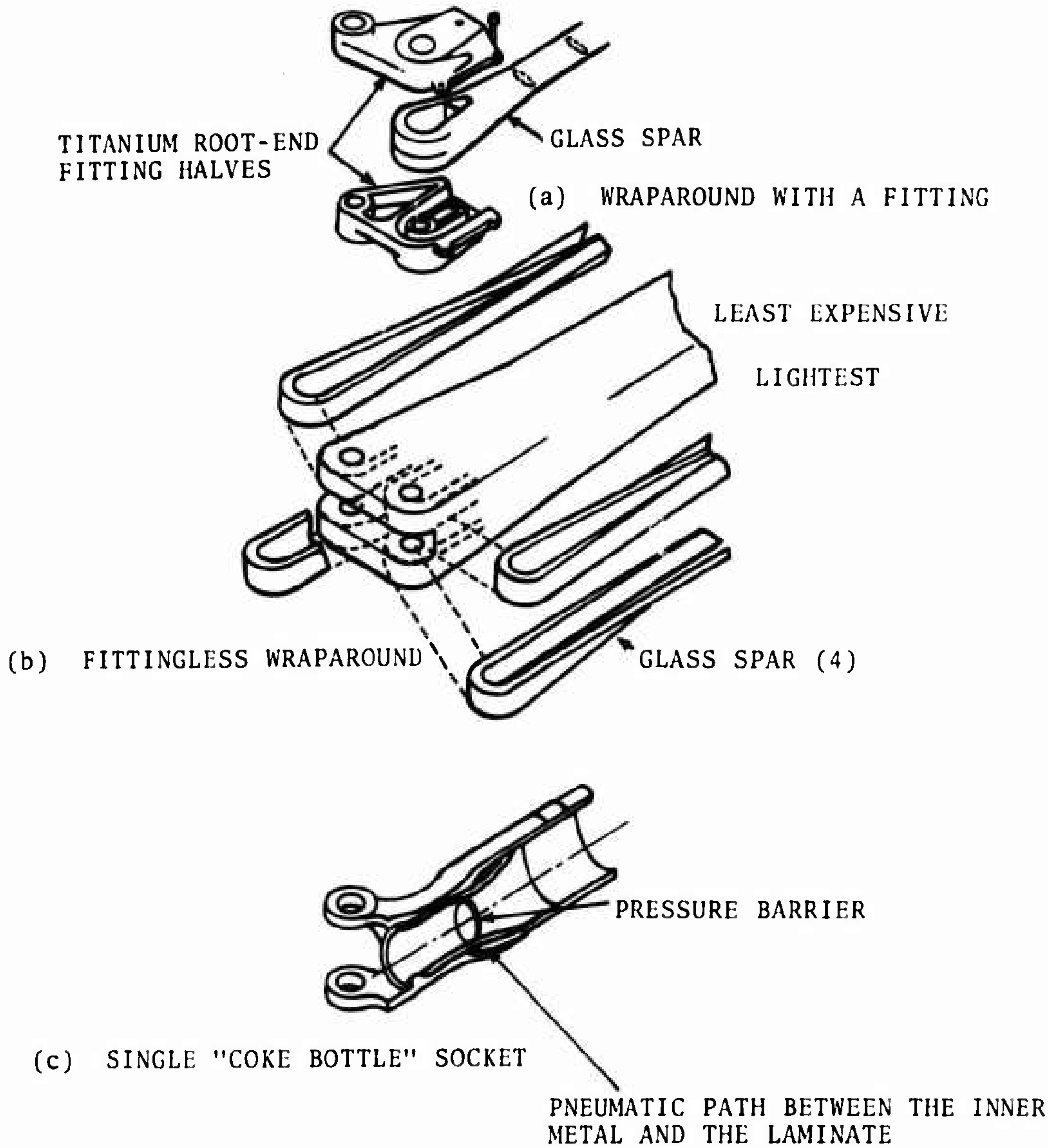


**Figure 24. Rotor Blade Root-End Design Configurations.**

The multipin design shown in Figure 24(b) incorporates most of the advantages of the wraparound concept (Figure 24(a)) and provides redundancy in the attachment area. This design represents a low-cost, lightweight, root-end configuration.

Figure 24(c) shows the single "coke bottle" socket configuration. The socket has a level of inherent fail safety by virtue of the entrapment of the spar in the coke bottle structure. In addition, the multipin construction may provide redundancy in the attachment structure. However, coke bottle designs (single or double) usually have cost and/or weight penalties because of their fabrication complexity.

Other root-end retention designs, most notably the flange-type construction, require satisfying fail-safe criteria that make their practical usage difficult.

Of the numerous variations of the basic configurations shown in Figure 24 (particularly in the geometrical orientation of the multipin attachment), the foregoing root-end retention systems may be used with "C" or "D" spars and constructed of either metallic or composite materials.

In summary, many design configurations are feasible, that is, capable of meeting fail-safe design requirements. Reference 11 suggests that although the multipin fittingless wraparound design (Figure 24(b)) may not represent the optimal root-end configuration for all rotor systems, it will generally provide the lightest and least expensive fail-safe root-end retention system configuration for new designs. (This design was chosen for the HLH root-end retention system (Reference 12).)

The aft fairing subassembly is generally not a critical structure. However, if it is damaged, it may cause increased loading on the spar. Consequently, this subassembly should be highly damage tolerant and have a low cost and weight profile. The following paragraphs briefly discuss the prominant aft fairing construction techniques.

Essentially, the aft fairing subassembly is composed of a core material covered by a skin material. In early helicopter designs, rotor blades were constructed similarly as their fixed-wing counterparts, that is, by mechanically fastening sheet metal skins to ribs to form a box-type construction. Such constructions proved to be unsatisfactory because of excessive fatigue failures in the joints induced by stress concentrations and fretting corrosion (Reference 6).

Currently, nearly all in-service rotor blade designs incorporate a honeycomb core, normally either aluminum or Nomex honeycomb, in the aft fairing assembly. Of the two core constituents, Nomex honeycomb offers the advantages of freedom from corrosion and fabrication flexibility. Although Nomex honeycomb is more expensive than aluminum honeycomb, many sources (such as Reference 12) contend that Nomex will prove to be more cost effective because of its superior handling capability and damage tolerance. In any event, the honeycomb core makes the aft fairing subassembly more crack resistant.

Similarly, composite materials (particularly fiberglass) are preferred for the aft fairing skin because of their durability, fatigue resistance, and light weight (their durability has been proven by their years of service on CH-47B/C helicopters (Reference 12)).

In summary, by a judicious selection, usage, and integration of composite and metallic materials, conventional safe-life and fracture mechanics principles, and pressure-differential condition-indicating systems, fail-safe rotor blades, like those in the HLH system, may be constructed so that they will satisfy the fail-safe/safe-life design criteria outlined in Section 3 without incurring excessive cost and weight penalties. Further, since the structural requirements for the tail rotor blade are not as severe as those for the main rotor blades, the fail-safe design techniques for the main rotor blade may also be applied to the tail rotor blade.

### 4.1.3 Fail-Safe/Safe-Life Design Methodology for Main and Tail Rotor Head and Rotating Controls

The main rotor consists of a hub assembly and a blade retention-control assembly. The subassemblies incorporated in a given rotor head configuration depend to a large extent on the type of rotor head control system. Each configuration is a yoke-type structure which transmits the aerodynamic loads in the form of (1) centrifugal force, (2) beam shear and bending, and (3) chordwise shear and bending as torque to the main rotor mast.

Normally installed between the rotor blade and the yoke is the rotor blade grip whose function is to provide various levels of rotor blade freedom (pitch, lead-lag, and flapping) as well as to serve as the vital link in the blade retention system. The blade-grip attachment, grip-yoke attachment, grip, and yoke are all critical structures subjected to dynamic loading.

The blade-to-grip attachment has been described in some detail in the previous section. The grip may be attached to the yoke by a pinned design configuration, by a needle bearing-tension strap arrangement, or more recently by using elastomeric bearings. The various attachment techniques are detailed in Reference 4 and three other sources[14,15,16].

The primary rotating control structure for the main rotor system consists of a swash plate, pitch links, and a pitch horn (the pitch horn is either mechanically or integrally attached to the grip). In-flight cyclic and collective controls are transmitted from the stationary structure to the rotating structure through the swash plate. The control commands for the grip and blade pitching action are transmitted from the swash plate to the pitch horn through pitch links. The pitch links are normally secured at either end in a pinned structural configuration. In summary, the critical structures for the main rotor rotating controls are the swash plate, swash plate/pitch link attachment, pitch link, pitch link/pitch horn attachment, and the pitch horn.

The tail rotor head has virtually the same construction as the main rotor head. The tail rotor blade is secured to the tail rotor grip by a pinned attachment, which is in turn attached to the tail rotor yoke. Collective pitch control is effected by a push-pull rod in the center of the driveshaft which actuates the pitch arm assemblies. The critical tail rotor head and rotating control structures are the pitch arm/control shaft attachment, pitch arm, pitch arm/grip attachment, grip/blade attachment, grip, grip/yoke attachment, and the yoke.

The design criteria techniques outlined in Section 3 may be applied to the critical main and tail rotor head and

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[14] Personnel of the Directorate for Product Assurance Systems Performance Assessment Division, MANAGEMENT SUMMARY REPORT, OH-58A, U.S. Army Aviation Systems Command, Directorate for Product Assurance, USAAVSCOM Technical Report 73-2, U.S. Army Aviation Systems Command, St. Louis, Missouri, January 1973, AD-756-415.

[15] Personnel for the Directorate for Produce Assurance Systems Performance Assessment Division, MANAGEMENT SUMMARY REPORT, AH-1G, U.S. Army Aviation Systems Command, Directorate for Product Assurance, 72-28, U.S. Army Aviation Systems Command, St. Louis, Missouri, AD-756-377, July 1972.

[16] Cook, T.N., Young, R.L., and Starses, F.E., MAINTAINABILITY ANALYSIS OF MAJOR HELICOPTER COMPONENTS, Kaman Aerospace Corporation, USAAMRDL-TR-73-43, Eustis Directorate, U.S. Army Air Mobility Research and Development Lab, Fort Eustis, Virginia, August 1973, AD-769-941.

rotating controls. Since the available information was not specific about fail-safe design criteria applied to each of the foregoing critical subassemblies, the techniques to accomplish the goals of the various fail-safe design criteria will not be evaluated as were those for the other helicopter structures. Rather, a few illustrative examples will be cited.

After the design criteria for the first five case examples listed in Section 3 were judged acceptable for the design of the proposed HLH hub, the criteria for Cases 2 and 3 were chosen.

The HLH hub is fully articulated and incorporates elastomeric bearings which react the blade centrifugal loads while accommodating the coincident pitch, flap, and lead-lag blade action (Reference 4). The crossbeam, the loop, and the hub plates are all designed redundantly as per the criterion requirements of Case 3 (this redundancy, along with a reduced service life requirement for the failed configuration and an inspection procedure to detect the failure within the reduced service life, ensures a fail-safe structure). The pitch housing, or what might be called the grip, was designed according to the design criteria of Cases 2 and 3. Although designed with a single load path, the pitch housing barrel has a fail-safe status because of its incorporation of slow crack-growth materials and a pressure-differential crack-indication system (after the pitch housing was equipped with the indicator, it was sealed and evacuated of air to create a partial vacuum inside the structure). In addition, both ends of the pitch housing, that is, the outboard blade and pitch link attachment lugs, were designed redundantly.

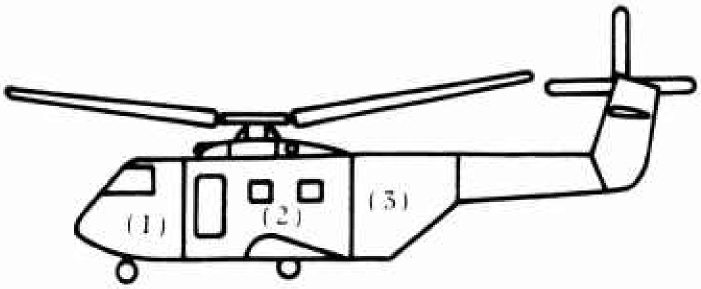
Similarly, several of the subassemblies in the proposed UTTAS rotor head and rotating controls were designed fundamentally according to the design criteria for Case 2. The swash plate and the blade-retention pin were equipped with pressure-differential-loss detection systems similar to those in the ISIS. The main rotor hub and pitch shaft have a lubricating oil passage within the critical structural areas to facilitate crack detection. These leak-before-break crack-detection provisions combined with a damage-tolerant structure (as substantiated by analysis and test) ensure fail-safe structures.

On the basis of the above, the state of the art permits designing fail-safe rotor head and rotating control components.

## 4.2 INTRODUCTION TO HELICOPTER FUSELAGE DESIGN METHODOLOGY

Since the structural loadings on the helicopter fuselage are due primarily to the steady-state and transient loads resulting from flight maneuvers and landings and secondarily to sonic or acoustic loading in the engine exhaust area and to resonance of the secondary structure, the various fuselage components are designed (sized) for static loading conditions (Reference 7 and a Sikorsky Aircraft report[17]). The critical loading condition, which may occur during flight or landing, involves crippling, buckling, and ultimate strength.

The primary fuselage structural components are shown in Figure 25. Because of the wide variety of fuselage construction configurations, only the outer shell construction will be considered relative to various design criteria.




(1) FORWARD FUSELAGE
(2) CENTER FUSELAGE
(3) AFT FUSELAGE

**Figure 25. Primary Fuselage Structure.**

Basically, three types of airframe constructions have been used for the fuselage outer shell: thin skin/stringer, sandwich, and sculptured plate.

As the preferred construction, because of its durability, the semimonocoque fuselage structure (skin-stringer-frame construction) is characterized by multiple stringers formed over structural framing to which a metal sheeting is attached. Most of its maintenance problems have involved loose rivets or hardware and broken or cracked parts.

The sandwich construction has gained wide acceptance for specific structural requirements. Basically, this construction

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[17] Rich, M.J., VULNERABILITY AND CRASHWORTHINESS IN THE DESIGN OF ROTARY WING VEHICLE STRUCTURES, Sikorsky Aircraft Division, United Aircraft Corporation, 1968.

requires bonding face sheets to an inside core. Honeycomb material is normally used in the sandwich construction.

The sculptured plate structure is used primarily for highly loaded panels or in shell-like applications. This structure is fabricated from a single piece of material by selectively removing material to form thin walls strengthened by integral stiffening structures and attachment sections.

As indicated in the literature review, research and development for helicopter fuselage structures has dealt mainly with ballistic tolerance, vulnerability and crashworthiness, composite material applications, and damage tolerance. Except for crashworthiness, these structural aspects have been deeply affected by the fracture control philosophy and techniques.

The three constructions for the fuselage outer shell will be considered relative to the safe-life and fail-safe design philosophies in the following sections. Section 4.2.1 will discuss the application of fail-safe design criteria to the helicopter fuselage structure, while Section 4.2.2 will present current fuselage design methodologies and objectives. Section 4.2.3 gives a detailed summary of the recommended fail-safe/safe-life design criteria.

### 4.2.1 Current Helicopter Fuselage Design Methodology

The design methodology for constructing helicopter fuselages has been characterized by the classical method of designing to react ultimate loads and checking for fatigue in selected areas (Reference 7). Such designing has been considered appropriate since the fuselage structure does not undergo the cyclic loading of dynamic components; rather, its critical loading condition occurs generally under extreme maneuver, takeoff, or landing conditions where it must have adequate static strength to react these steady-state loads.

The fuselage structure is also subjected to transient loads during flight, ground-to-air and air-to-ground transitions, and ground operations; however, the capability of withstanding these loads is secondary to the static strength required for the steady-state loads.

The static strength design practices are straightforward, well-established, and successful. First, the maximum expected loads for the various fuselage components are determined through a blend of experience gained in previous flight loads recording programs, established analysis procedures, and engineering judgment. This determination generally establishes the design condition for the limit load which the fuselage structure must sustain. An additional static strength design criterion is the factor of safety, namely, 1.5 times the limit load.

The directives to conduct a fatigue analysis of the fuselage structures and then to establish its safe-life design are as follows:

(1) Determine the fatigue sensitive areas of the fuselage structure, i.e., those areas where stress concentrations are coupled with relatively high working stresses.

(2) After establishing flight loads spectra, derive specific component loading spectra.

(3) Establish component fatigue loading spectra.

(4) Establish the fatigue strength of the component, usually in terms of the classical S-N curve.

(5) Establish a life calculation technique.

(6) Establish the means for accounting for statistical variations in component/material fatigue strengths and flight test data, and for manufacturing, maintenance, and operational anomalies.

Section 3 describes some of the current fatigue life determination techniques. Reference 6 cites that although these various techniques have significant differences in calculating the retirement lives, the results may be considered similar because of their conservatism in providing a confident safe-life estimate and in ensuring that the probability of fatigue failure is extremely remote. In practice, these techniques generally do not account for most of the helicopter structural failures; this is the basic fault of relying wholly on the safe-life design methodology to ensure flight safety. On the other hand, the fail-safe design methodology includes such intangibles as engineering errors in interpretation of data, manufacturing errors not found by inspection, maintenance errors, undetected battle damage, and pilot operating errors.

While all safe-life techniques may be applied to the various helicopter constructions, their preference rating depends on their damage-tolerant capabilities, as discussed in Section 4.2.2, in addition to their cost and weight design trade-offs, as discussed in Section 5.

As stated previously, the criticality of the fuselage structure has depended primarily on static loading conditions and to a lesser degree on fatigue loading. However, as revealed in the Government/industry survey, fatigue loading will become increasingly more important since future designs may have transmission and rotor-induced loads carried through a fuselage superstructure. Such loads would severely increase the structural fatigue requirement so that the fuselage would be more resistant to crack initiation and propagation.

Although the design techniques discussed in the previous section have generally sufficed because of their being based on a blend of inherent redundancy and safe (fatigue) life analyses, the inclusion of fail-safe principles in these techniques will become necessary with increasing structural demands.

The following section discusses the application of the fail-safe design principles to the fuselage structure.

### 4.2.2 Fail-Safe Helicopter Fuselage Structures

According to the fail-safe design criteria suggested in Section 3, a damage-tolerant structure may be designed by combining the following basic structural configurations:

(1) A single load path.

(2) A single primary load path with auxiliary crack-arrest features.

(3) Multiple and redundant load paths.

In addition to the best combination of the foregoing structural configurations, the design of the fail-safe structure is optimized by selecting the appropriate materials and considering their accessibility and inspection requirements. As mentioned above, the designs for fail-safe structures complement those for safe-life structures by including fatigue strength designs which resist crack initiation.

Of the three fuselage constructions listed in Section 4.2, the skin-stringer-frame construction is preferred, as indicated overwhelmingly by the industry and Government representatives, because of its reliability and inherent multiload path configuration. However, although the redundant characteristics of the skin-stringer-frame construction lend themselves very well to the fail-safe design principles (Reference 1 and a Sikorsky Aircraft report[18]), the redundancy in itself does not ensure a fail-safe structure. The rationale behind this statement is described in Section 3.

[18] Rich, M.J., and Linzell, L.E., "DAMAGED" STATIC AND FATIGUE STRESS ANALYSIS OF VTOL STRUCTURES, Sikorsky Aircraft, February 1969.

Normally a structure is considered redundant if after one or more members are removed, the structure can still safely sustain the applied load. Reference 18 cites that the semi-monocoque type of construction makes the skin-stringer-frame redundant. And yet when the individual members of this construction are considered, it is frequently advantageous to use some nonredundant load paths in the fail-safe design. For example, a sufficient number of stiffeners must usually be added to the fuselage skin in order to meet static strength requirements; these stiffeners also serve as crack arresters. Similarly, the skin-stringer-frame combination is considered to be inherently redundant, although the frames and stringers themselves may or may not be redundant. In essence, the skin-stringer-frame construction requires the incorporation of all three of the damage-tolerant design configurations. Redundant load path design is useful in constructing damage-tolerant frames and stringers. Although crack stoppers are often the most feasible means of supplementing the damage tolerance of the skin, the design technique best suited for a particular application may be determined after making all necessary cost and weight trade-offs.

Also inherently redundant, the sandwich construction has the crack-arresting characteristics of a core-face sheet combination. This construction consists basically of face sheets bonded to an inside core. Honeycomb panels, laminated panels, panels constructed of composite materials, and combinations of these configurations are forms of sandwich construction (typical sandwich construction configurations are illustrated in Figure 26).

Like the sheet skin-stringer-frame construction, the sandwich construction is readily adaptable to fail-safe principles, partly because of its inherent crack-arrest capability. As illustrated in Figure 26, the bonded honeycomb core will retard the propagation of a crack in the outer skin. When fabricated by laminating successive layers of boron filaments or graphite fibers with epoxy resins, composite structural materials will have similar inherent crack-arrest characteristics. Sandwich panels incorporating bonded stiffeners are feasible for monocoque structures.

To make a sandwich construction fail safe, all the damage-tolerant design requirements outlined in Section 3 (material selection, inspectability, stress allowables, etc.) apply.

The sculptured plate (integrally stiffened) structure is fabricated from a single piece of material by mechanical, electrical, or chemical means. Material is selectively removed so that only thin walls integral with heavier stiffening sections remain.

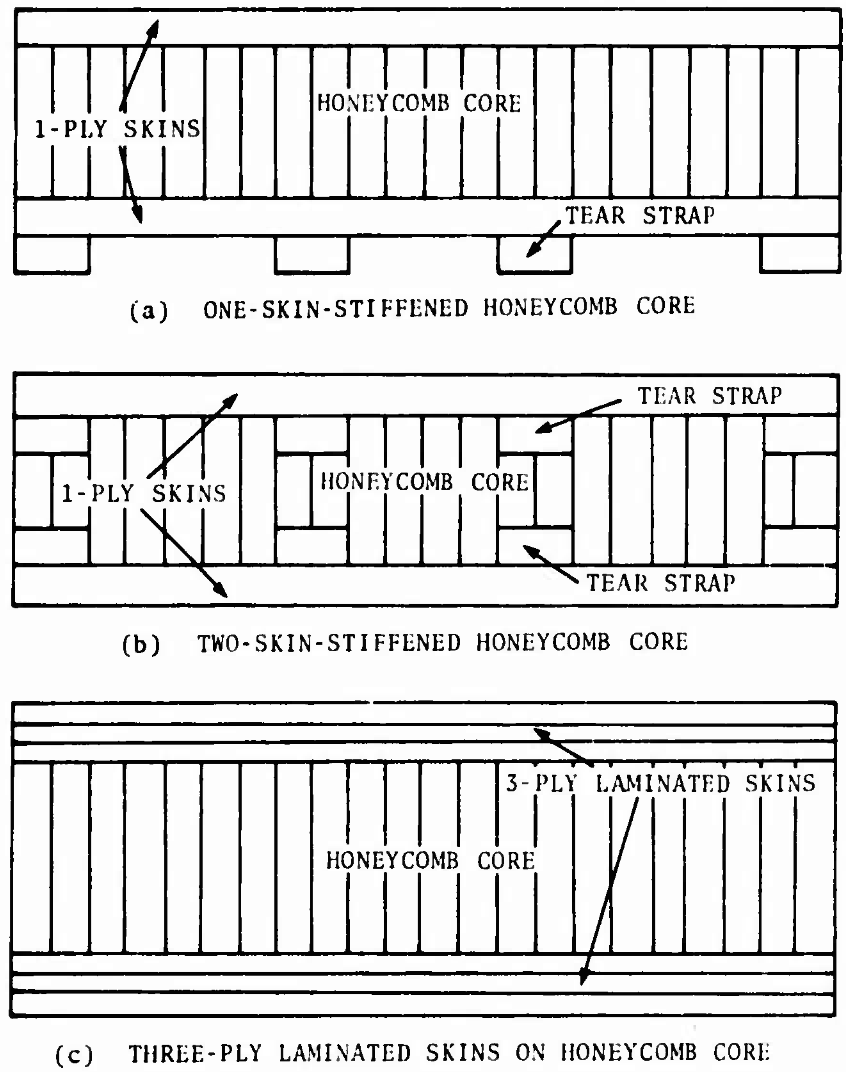


Figure 26. Fuselage Sandwich Construction Design Configurations.

This structure has evolved primarily because of the high premium placed on the lightness required for achieving the desired performance. Extensively sculptured parts manufactured from high strength materials allow high design stresses while minimizing the number of joints. Although this design reduces the structural weight, it makes the structure highly sensitive to fracture; however, the application of damage-tolerant principles can make this structure more resistant to flaw propagation.

Whether or not the sculptured plate construction may be practically applied to a particular fuselage and designed to fail-safe principles depends on at least three factors: (1) parts minimization, (2) high-stress allowables, and (3) low-fracture toughness.

Although parts minimization increases the inspectability of the sculptured plate structure, this structure must generally be replaced when a flaw is detected, whereas the sheet skin-stringer-frame and sandwich structures may be repaired with relative ease.

Much of the acceptance of low-fracture toughness is due to a high strength-to-weight ratio. But such a ratio requires high-strength materials and in turn special design provisions.

Again the necessity of using high-strength, low-fracture-tough materials usually means a small critical crack length, which in turn puts a heavy burden on the NDI method utilized and decreases the reliability of crack detection.

To improve the resistance of the sculptured plate construction to crack propagation, other designs, such as bonded crack stoppers, may be incorporated in the structure, but the resultant fail-safe status would have to be questionable.

### 4.2.3 Recommended Helicopter Fuselage Design Methodology

The current helicopter fuselage design methodology may be summarized as follows:

(1) Static strength is the prime criterion in sizing the fuselage structure.

(2) Skin-stringer-frame construction is generally preferred because of its inherent multiload-path and crack-arresting capabilities, acceptable cost and weight, easy maintenance, and reliability.

(3) The fuselage structure is designed according to the safe-life design principle, but its degree of fail-safe provision varies with the particular construction.

As a result of this methodology, no known catastrophic helicopter accidents can be attributed to the fuselage structure (as indicated by the Government/industry survey). Most accidents have been due to failures which can be greatly overcome by reducing the vibration level[19].

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[19] Veca, A.C., VIBRATION EFFECTS ON HELICOPTER RELIABILITY AND MAINTAINABILITY, Sikorsky Aircraft Division, United Aircraft Corporation, USAAMRDL Technical Report 73-11, Eustis Directorate, U.S. Army Air Mobility Research and Development Lab., Fort Eustis, Virginia, April 1973, AD-766-307.

Future helicopter fuselage structures will probably have more demanding fatigue strength requirements because of the projected increases in transmission and rotor-induced loads. In addition, the possibility of a flawed structure must be considered in the design phase, for as stated in Reference 1, "Primary aircraft structural components generally contain flaws or defects of variable shape, orientation, and criticality which either are inherent in the basic material or are introduced during the fabrication or assembly processes." Therefore, the fail-safe design criteria presented in Section 3 should be applied to the helicopter fuselage structure. Further, since the criteria for all structural configurations described in Section 3 can be satisfied, specific technical feasibility-cost-weight trade-off studies should be conducted for all potential design combinations according to the rationale in Section 3.

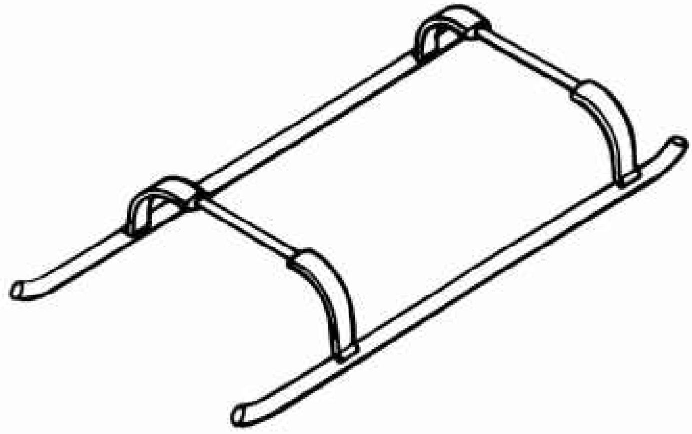
All information sources strongly recommend the semi-monocoque structure. Some combination of the design techniques of these sources would represent an optimum fail-safe structure for a given material. Materials with high fracture toughness properties are also preferred so that they may reveal large visible cracks prior to catastrophic failure, thus improving the probability of detection and the level of fail safety.

## 4.3 INTRODUCTION TO HELICOPTER LANDING GEAR DESIGN METHODOLOGY

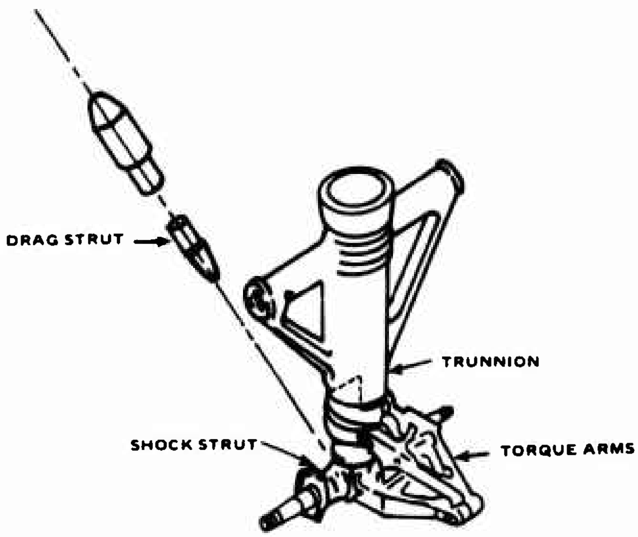
The helicopter landing gear structure must absorb the loads due to ground movement, which are generally insignificant, and those due to landing impact, which are a function of pilot technique and the relative condition of the landing surface.

Of the several effective landing gear designs, all are either the oleo-strut wheel or the skid type. Used for many years on lightweight helicopters, the skid type consists of two skids that are aligned longitudinally along the helicopter fuselage and connected to the fuselage through cross tube members (see Figure 27(a)). The oleo-strut wheel type is a strut-mounted, shock-dampened wheel assembly used in a conventional (tailwheel), tricycle, or quadricycle gear configuration. A typical oleo-strut wheel landing gear is illustrated in Figure 27(b).

Since the safe-life and fail-safe design philosophies are concerned primarily with cyclic loading, they are generally applied to those airframe structures whose failure could cause severe damage to the rest of the helicopter system and injury to its occupants. Therefore, the landing gear system must be distinguished from that landing gear structure which will be considered relative to the fail-safe and safe-life design philosophies. Except for the rolling gear (wheels, tires, brakes, and miscellaneous hardware), all primary load-carrying members of the landing gear structure (such as the shock strut on the oleo-strut gear and the cross tube member on the skid gear) are included in the following discussion.



(a) SKID TYPE GEAR




(b) WHEEL ASSEMBLY TYPE GEAR

Figure 27. Typical Oleo-Strut and Skid Types of Helicopter Landing Gear.

Compared with the available information on the other helicopter structures, very little data could be obtained on the fatigue aspects of the helicopter landing gear. Most of the available literature deals with (1) the extent to which the landing gear structure can improve the survivability and crashworthiness of the helicopter system, (2) the reliability and maintainability of the skid landing gear, and (3) the feasibility of incorporating composite materials in oleo-strut wheel landing gears. The Government/industry survey revealed that fatigue failures in helicopter landing gear are extremely rare and that most of the landing gear failures are due to landing impacts whose loads exceed the design limits.

Other significant aspects of the landing gear design are as follows:

(1) The landing gear structure is a prime means for improving the crashworthiness of the helicopter system.

(2) The static overload capability of the primary structures in the landing gear makes normal operational impact loads small relative to the design strength; consequently, the gear has adequate fatigue-strength characteristics because of the low operational stresses.

(3) The only military design specification with fatigue requirements for helicopter landing gear structures is AR-56. None of the military or commercial helicopter design specifications require fail-safe designs for landing gear structures.

(4) Although fatigue failures in landing gear structures are not likely, the application of fail-safe designs to this structure may improve its structural reliability.

The following paragraphs describe the helicopter landing gear design methodology and suggest the direction for future helicopter landing gear studies.

### 4.3.1 Landing Gear (Limit Load) Design Criteria

The helicopter manufacturers stated that the primary design specifications for helicopter landing gear structures are contained in MIL-S-8698, AR-56, and FAR Parts 27 and 29. **Except for MIL-S-8698, which is used by all manufacturers, the use of these specifications as design criteria varies with the individual manufacturer design philosophies.**

All three design specifications prescribe a sink rate, in combination with a specified rotor lift, as the basis for defining the magnitude of the landing impact loads which must be sustained in the helicopter design. For example, MIL-S-8698 specifies an 8-ft/sec sink speed at a rotor lift equivalent to two-thirds of the basic design gross weight.

Other helicopter design specifications include the spectrum of critical landing conditions which permit determining the minimum strength requirements relative to the vertical, drag, and side gear loads during the critical landing conditions. However, as pointed out in USAAMRDL-TR-72-61[20], MIL-S-8698 and the Federal Aviation Regulations do not specifically require an analysis procedure for determining the maximum vertical reaction or maximum load factor representative of the first landing impact. This parameter is usually determined by one of the four following methods:

In one method, the dynamic conditions may be readily estimated by assuming a realistic acceleration level, normally based on previous test results or design experience. Then with the maximum acceleration level and the mass of the vehicle known, the force that the landing gear must sustain is easily calculated.

A second method is based on the assumption that the fuselage design limit load factor is the peak fuselage acceleration allowed by the landing gear. This method also eliminates the need for a dynamic analysis since the peak gear force required may be calculated as in the first method.

A third method is based on the relationship between kinetic and potential energy before and after landing impact. Requiring a known or assured oleo efficiency, this method provides a means for computing the load factor from the sink rate, rotor lift, and gross weight parameters.

A fourth method involves a dynamic analysis to determine the gear loads normally occurring in a particular landing environment. This analysis is required in AR-56.

In summary, all four methods provide the means for determining the maximum gear load to be sustained in a given landing impact condition. Once the vertical gear load is determined, all related drag and side forces are easily computed. The landing condition which produces the most demanding loads on the gear structure is termed the limit condition, and these loads collectively represent the limit load.

The following sections discuss the oleo-strut wheel gear and the skid gear separately since the two systems absorb

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[20] Phillips, N.S., Carr, R.W., and Scranton, R.S., CRASHWORTHY LANDING GEAR STUDY, Beta Industries, Inc., USAAMRDL-TR-72-61, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1973, AD-765-489.

impact loads through slightly different energy-absorption systems. In these sections, it is assumed that the primary load-carrying landing gear members are sufficiently similar, so that their specific configurations need not be considered.

### 4.3.2 Design Considerations for Oleo-Strut Wheel Landing Gear

The primary purpose of the landing gear system is to absorb the loads imposed on the helicopter system during take-off, landing, and ground operations. Such loads are sustained by a combination of tire shock absorption, oleo shock absorption, and structural elastic action of the landing gear structure. Therefore, the design of the primary load-carrying members requires (1) providing an oleo-strut wheel system which functions predictably throughout the normal landing impact loading regime, and (2) giving the primary structural members adequate strength, as prescribed in the aforementioned design specifications, such that the probability of their static failure under the required loading spectra is extremely remote.

The landing gear static strength requirements are not given quantitatively in the primary design specifications. Essentially, MIL-S-8698, AR-56, and FAR Parts 27 and 29 require that the landing gear structure have adequate strength gear to carry the forces imposed by the limit landing condition without failure.

In summary, the design procedure for the oleo-strut wheel type of landing gear is basically (1) calculating the reaction forces on the basis of the specified critical landing conditions, (2) determining the structure to sustain these forces, and (3) selecting the hydraulic areas and orifice sizes for a shock strut compatible with the landing gear structure (Reference 20).

Additionally, MIL-S-8698 and FAR Parts 27 and 29 prescribe reserve energy requirements. MIL-S-8698 specifies that the landing gear structure will not fail in a drop with a sinking speed times the square root of 1.5, where the sinking speed is the limit value associated with the landing weights as specified in Section 3.4.2 (a statement of rotor lift equivalent to two-thirds of the design gross weight). FAR Parts 27 and 29 specify a drop height equal to 1.5 times the limit drop with rotor lift not to exceed 1.5 times two-thirds of the maximum gross weight. While AR-56 does not specify a reserve energy requirement, MIL-S-8698 contains a clause (applicable to the Navy Bureau of Aeronautics) which introduces the concept of "yield load factor" for helicopter structures. The similarity of the yield strength requirement and the reserve energy requirement in the resulting overload strength is obvious. Therefore, the secondary objective of the landing gear system is to provide increased capabilities for the crashworthiness of the helicopter system. The following discussion summarizes several innovative landing gear designs to provide this increased capability.

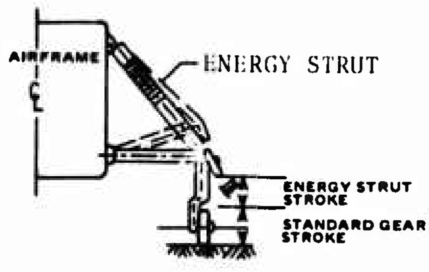
In severe landing impacts, the oleo-strut gear will hydraulically lock up because of the high initial strut-closing velocities. Since only the landing gear can absorb the landing impact before its transmittal to the airframe, auxiliary "one-shot" energy absorption devices have been incorporated in the landing gear system. As in the Table 4 listing of typical devices (Reference 20), most of the absorbers are complex mechanisms, involving metal fracture, plastic deformation of metal, and so on, where the energy strut, or load limiter, is activated at a predetermined level; for example, Figure 28 shows an operational energy strut which acts as a fully plastic device after the calibrated pins are sheared (Reference 17).

| TABLE 4. ENERGY ABSORPTION TECHNIQUES[20] |
| --- |
| Honeycomb Compression |
| Tube Flare |
| Inversion Tube |
| Rod Through Tube |
| S-Shaped Bar |
| Standard Cable |
| Metal Tube |
| Strap/Rod |
| Tension Pulley |
| Bar Through Die |
| Wire Through Pattern |
| Rolling-Torus |

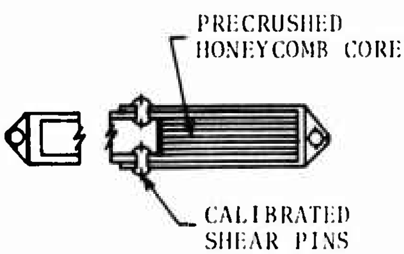
As in normal landing impacts, the primary load-carrying members must have sufficient static strength for the proper functioning of the energy-absorbing devices. Because of the close interrelationship between the static and low-cycle fatigue strengths, the high static strength requirement increases the fatigue-resistant capability of the landing gear.

AR-56 is the only one of the three primary design specifications to require a fatigue evaluation and specify a service life for helicopter landing gear structures. MIL-S-8698 only infers the need for an adequate fatigue strength in all structures, and the FAR document excludes the landing gear structure from its fatigue requirements.

The Government and industry representatives almost unanimously stated that helicopter landing gear structures, in general, are relatively insensitive to fatigue and that the primary cause of oleo-strut wheel gear failures is attributed to static overloads, not to fatigue. This statement is substantiated by the fact that no significant oleo-strut problems or deficiencies were revealed in the literature survey. The absence of such documentation is noteworthy since, in contrast, failures of the skid landing gear were found in the survey.




(a) LANDING GEAR SYSTEM WITH RESERVE ENERGY CAPABILITY




(b) ENERGY STRUT

Figure 28. Auxiliary Energy-Absorption Device.

As learned in the industry survey, the MIL-S-8698 and AR-56 criteria greatly influence most of the military helicopter designs; therefore, many of the current helicopter landing gear components (particularly the oleo-strut wheel) are designed according to the safe-life design philosophy and have a reserve energy absorption capability.

### 4.3.3 Design Considerations for Skid Landing Gear

As stated in Section 4.3, skid gear are normally configured with two skids aligned longitudinally along the helicopter fuselage. The skids are connected to the fuselage through cross tube members and supported with many shock strut assemblies. The primary contribution of skid landing gear designs is their provision for a low-cost, lightweight means of forming small static deflections to accommodate normal landing impacts and high-energy dissipation characteristics (comparable to those provided by oleo-strut wheel gear) during hard landings.

The basic design procedure for skid-type gear is to determine the stress-strain relationship of the tube material, to develop the force-displacement relationship, and to integrate these data into an energy relation or dynamic response equation for the eventual satisfaction of the landing gear design specifications (Reference 20). The conventional skid landing gear, therefore, absorbs the energy induced by normal impacts strictly through an elastic structural action. The more severe landing impact energy is absorbed through a structural elastic-plastic action.

Energy struts (similar to those used on oleo-skid wheel landing gear systems) are frequently incorporated in skid landing gear to provide reserve energy and thus reduce the strength requirement for the cross tube member.

The three primary helicopter design specifications, generally, do not differentiate between oleo-strut wheel gear and skid gear. It is therefore inferred that all landing condition requirements apply for both structural concepts with the following exceptions.

(1) AR-56 makes specific reference to skid gear only to eliminate this type of gear from the rough field condition (run-on) requirements.

(2) FAR Parts 27 and 29 state that the structural yielding of the elastic spring members under limit loads is allowed. (This might be interpreted as recognizing that normal operational landing impacts, as encountered by commercial helicopters in their predictably mild landing environment, are well below the limit landing conditions defined by the design specifications.) Parts 27 and 29 also define the ground loading conditions for the skid gear (which are generally equivalent to those specified for the oleo-strut wheel gear).

Skid-type landing gear failures in the form of bending, cracking, and collapsing of skid tubes and cross tubes were documented in Reference 15 and two other sources[21,22] since they represent a serious structural deficiency from a maintenance standpoint. However, these failures, which occurred at helicopter training bases (Fort Rucker in particular), were attributed to hard run-on landings on rough surfaces and generally to hard practice landings inherent in training operations. In essence, the documented landing gear structural failures were consistent with the industrial survey finding that landing gear failures are caused primarily by static overloads.

### 4.3.4 Fail-Safe and Safe-Life Design Considerations

As evidenced by the preceding discussion, helicopter landing gear are designed for static loadings in that the specifications define minimum static strength requirements based on limit landing conditions. In addition, reserve energy requirements are levied on the landing gear system to enhance the crashworthy capabilities of the helicopter system.

The design specifications also require an evaluation of the fatigue strength of the landing gear structure and the establishment of a safe service life. The high static strength required by the limit landing conditions and the reserve requirements are relatively easy to satisfy.

All the industrial and Governmental representatives agreed that the possibility of statistical fatigue failures must be considered in the systematic pursuit of safe helicopter landing gear structures. Yet, they could cite very few structural failures (for any aircraft system) which could definitely be classified as statistically predictable fatigue failures.

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21 Clark, M.W., Krauss, W.K., and Ciccotti, J.M., IDENTIFICATION AND ANALYSIS OF ARMY HELICOPTER RELIABILITY AND MAINTAINABILITY PROBLEMS AND DEFICIENCIES: VOLUME IV - LIGHT OBSERVATION HELICOPTERS (OH-6, OH-58), American Power Jet Company, USAAMRDL Tech. Report 72-11D, Eustis Directorate, U.S. Army Air Mobility Research and Development Lab., Fort Eustis, Virginia, April 1972, AD-901-459.

22 Clark, M.W., Krauss, W.K., and Ciccotti, J.M., IDENTIFICATION AND ANALYSIS OF ARMY HELICOPTER RELIABILITY AND MAINTAINABILITY PROBLEMS AND DEFICIENCIES: VOLUME II - UTILITY, ATTACK, AND TRAINING HELICOPTERS (UH-1, AH-1, TH-1), American Power Jet Company; USAAMRDL Tech. Report 72-11B, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1972, AD-901-457.

This is the factor that lends credibility to applying fail-safe design concepts to any aircraft structure. Undetected manufacturing flaws, maintenance errors, and ballistic damage all represent structural discontinuities which, although not initiated by fatigue, can propagate under cyclic loading and eventually fracture. The fail-safe design philosophy provides a rationale for dealing with these nonstatistical failure conditions.

The industrial representatives had mixed opinions on the necessity of applying fail-safe concepts to helicopter landing gear structures. However, the general response was interpreted as negative according to the following general statements:

(1) The helicopter landing gear structure is not considered a flight critical structure.

(2) Landing gear design improvements for increased crashworthiness and survivability are considered to be on the frontiers where "new" landing gear design concepts are most needed.

(3) In the light of the fail-safe design philosophy and the particular loading spectra experienced by helicopter landing gear structures, it would be very difficult to produce a fail-safe landing gear design which would be cost and weight effective (when using current safe-life landing gear as a baseline).

(4) Helicopter landing gears designed to current design criteria have proven to be relatively successful in accomplishing their intended purpose.

In summary, the industrial representatives believed that the incorporation of fail-safe concepts in the design of helicopter landing gear structure was not warranted when there are so many other fatigue sensitive, flight critical components (e.g., many of the dynamic components) whose structural reliability could be improved by applying fail-safe design principles to them.

However, the industry representatives did agree to offer opinions and suggestions on this design problem should the need for a fail-safe helicopter landing gear become real.

Nevertheless, the various representatives believed that the application of fail-safe design techniques to helicopter landing gear is well within the capability of modern technology. In addition, the concensus was that a control fracture structure would be most adaptable to the design of such a landing gear when considering the technical feasibility, cost, and weight.

Moreover, many of the industrial representatives ac--knowledged the potential value of examining the feasibility of applying the damage-tolerant concepts to current landing gear structural designs. Such applications would require modifying the configurations to satisfy the rigid static strength requirements while using high fracture-resistant materials with slow crack propagation rates. Here again, cost and weight restrictions may render the concept not feasible. For example, a report by Rich et al.[23] documents an initial design study to explore the possibility of applying composite materials (which display inherent damage-tolerant properties) to the CH-53 main landing gear. The report concludes that while this concept is technically feasible, it offers low potential cost effectiveness.

The current design criteria for landing gear structures are, for the most part, contained in MIL-S-8698, AR-56, and FAR parts 27 and 29. There is very little variance between these criteria which may be labeled as safe-life design criteria with accompanying reserve energy criteria. The industry survey indicated that while a fail-safe helicopter landing gear was technically feasible, the practicality, and in fact the necessity, of such a landing gear structure is questionable. Therefore, since it would be beyond the scope of this study to draft new criteria reflecting fail-safe designs applicable to helicopter landing gear structures, the landing gear will be excluded from any further cost and weight trade-off studies relative to safe-life and fail-safe design studies (other than those presented in the previous paragraphs).

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[23] Rich, M.J., Ridgley, G.F., and Lowry, D.W., APPLICATION OF COMPOSITES TO HELICOPTER AIRFRAME AND LANDING GEAR STRUCTURES, Sikorsky Aircraft, Division of United Aircraft Corporation, NASA-CR-112333, National Aeronautics and Space Administration, June 1973.

## 5. TECHNICAL FEASIBILITY, COST, AND WEIGHT TRADE-OFF STUDIES

There are various structural configurations and design techniques capable of satisfying the fail-safe/safe-life design criteria described in Section 3. Some are obviously more effective from a technical feasibility, cost, or weight standpoint than others. The trade-off studies described in this section evaluate the relative merits of different fail-safe design techniques in terms of potential fail-safe effectiveness, cost, and weight.

In discussions with the Government and industry representatives, the question of "how safe is fail-safe" came to light several times. Similarly, these representatives stated that the redundant structure is the preferred design configuration and should be utilized whenever feasible. These discussions led to the concept of "level of fail safety" and ultimately to a mechanism for determining a fail-safe index.

As stated in Section 3, fail-safe designs should complement, not replace, safe-life designs. Further, the fail-safe goal should be achieved by designing a damage-tolerant structure and specifying inspection procedures and intervals so that a crack may be detected before it reaches critical proportions. The technical merit trade-off model was structured to express this rationale in terms of specific design techniques.

A combination of redundancy, positive crack retardation devices, damage-tolerant materials, etc., is usually necessary to make a structure flightworthy with a high residual strength and a slow crack propagation rate. Therefore, while each structure in the trade-off study must satisfy the fail-safe/safe-life design criteria, the more redundancy and positive crack retardation capability it has, the more potentially damage tolerant (and therefore desirable) it is. Since these design features tend to increase the effective damage tolerance of the structure, the structures are rated accordingly.

Another very important factor fundamental to fail-safe designs is the relative ease of inspecting the corresponding structures as well as the effectiveness of their crack detection technique. That is, the more positive the crack detection technique (for example, a reliable dial-type condition indicator compared with nondestructive inspection (NDI) equipment) and the easier the inspection (for example, simply looking at a dial between flights compared with removing the structure from the aircraft periodically and examining it through visual or NDI means), the more desirable are the "detectability/inspectability" characteristics of a structure. This concept of the level, or degree, of fail safety is illustrated in Figure 29. This figure was intended to stress the following: (1) structures whose critical crack lengths are small and therefore require NDI techniques to detect the cracks have a relatively low level of fail safety (that is, the probability of not detecting a crack is high), (2) structures whose cracks are in accessible areas and their critical lengths are large and readily detected visually have a relatively high level of fail safety, and (3) structures incorporating reliable condition indicators have a marked increase in the level of fail safety.

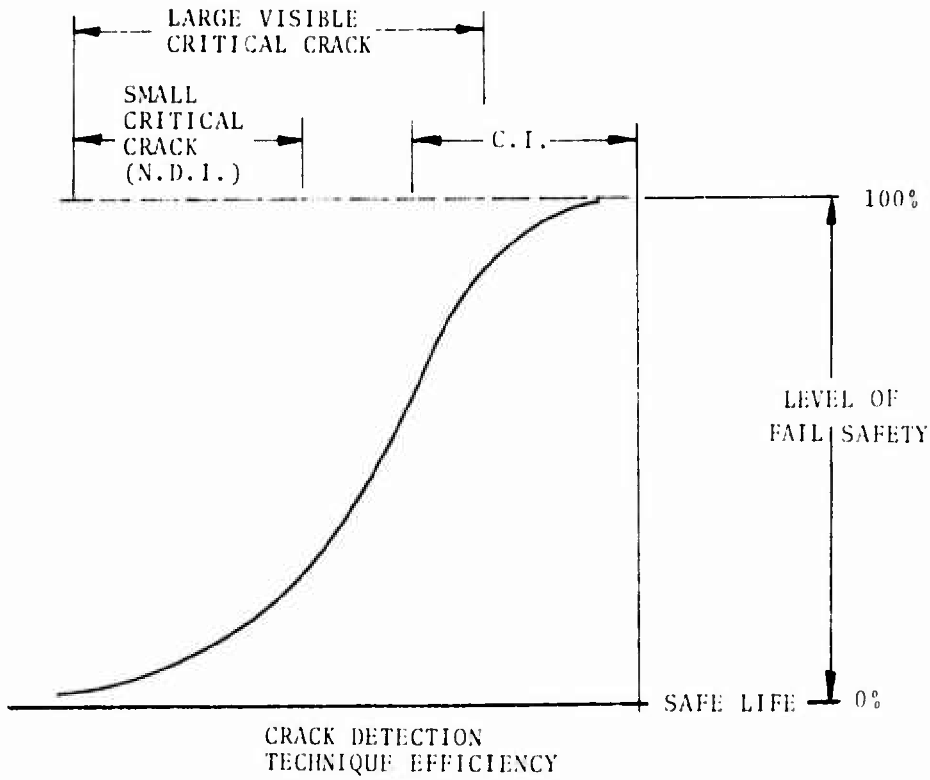


Figure 29. Concept of Level of Fail Safety.

In addition to the technical merit rating model, similar models were constructed for cost and weight factors related to the fail-safe design techniques. The cost and weight models are necessary since the greatest constraint in most design studies is cost and/or weight; as one of the representatives put it, "One can buy as much fail safety as he is willing to pay." Accordingly, rating equations for the cost and weight trade-off studies were established. The weighting factors which make up these equations were based primarily on experience gained in the current study and engineering judgment. This method of evaluating the fail-safe structures (representing various fail-safe criteria) relative to cost and weight was chosen so that the technical merit, cost, and, weight trade-offs would be sufficiently general for the purpose of this study.

The limitations of the technical merit, cost, and weight models described in Sections 5.2 and 5.3 are as follows: (1) the resulting ratings are based, for the most part, on those design features which are considered advantageous in a fail-safe structure, and (2) the resulting ratings and indices have meaning only relative to other structures rated by the same models. Consequently, in these ratings, a structure that satisfies the design criteria should not be compared with a structure that does not satisfy these criteria. Based on the factors most directly affecting the potential level of fail safety, these models provide a systematic means for making initial fail-safe design trade-off studies on fuselage structures and dynamic components.

## 5.1 FUSELAGE TRADE-OFF STUDY

The technical merit, cost, and weight rating tables developed for helicopter fuselage structures are presented in Appendix A. Table 5 summarizes the design features in the models.

TABLE 5. DESIGN FEATURES USED IN FUSELAGE TRADE-OFF STUDY

| Technical Merit Rating | Cost Rating | Weight Rating |
|---|---|---|
| Redundancy | Redundancy | Redundancy |
| Crack Retardation | Crack Retardation | Crack Retardation |
| Detection/Inspection | Detection/ Inspection | Detection/Inspection Material |
| | Material | Material |

The Government/industry survey revealed that the semi-monocoque structure is the preferred type of fuselage construction, from a fail-safe point of view, because of its inherent redundancy and crack-retardation characteristics. Therefore, this factor was weighted heavily in both the redundancy and the crack-retardation rating tables. Similarly, these two tables indicate the greater measure of inherent redundancy and crack retardation in sandwich construction compared with that in the sheet or the sculptured plate construction. Positive crack-arrest constructions (which include those design features incorporated explicitly to retard crack growth) generally have ratings higher than those for constructions with inherent crack-retardation designs.

"Leak-before-break" condition indicators are generally thought to represent the state of the art in incipient crack detection. However, to give the model more versatility, a hypothetical condition-indicating system for fuselage structures was included in the rating tables. The Government/industry survey indicated that only second to a reliable condition-indicating system would be a system in a fuselage structure with such detectability/inspectability (D/I) features that large cracks could be visually detected and that minimal structural teardown would be required for inspection. This important design feature is rated accordingly. In determining the D/I rating for fuselage structures, each identifiable substructure (e.g., frame, stringer, and skin) should be rated separately and then the individual ratings should be averaged to determine the D/I rating for the entire structure.

The technical merit rating (or fail-safe index) is derived by equally weighting (averaging) the redundancy (R), the crack retardation (C-R), and the detectability/inspectability (D/I) ratings.

With all the same design features considered, the cost and weight ratings were designed similarly as the technical merit ratings. The cost and weight ratings were based on the estimated costs and weights associated with those design features (redundancy, crack retardation, and detectability/inspectability) which improve the potential level of fail safety. Additionally, because of the increasing use of composite materials in fuselage structures, the costs and weights for composite and metallic materials were distinguished. Accordingly, designs with composite materials had a 5% cost penalty but a 5% weight reward (denoted by the subscript M). These percentages are not represented as actual values; rather, they were used in the general weighting process. (All relative costs discussed in this section refer to material, fabrication, and production costs, not life cycle costs.)

Finally, each of the four design feature categories was evaluated according to its relative effect on cost and weight. The following equations were developed to establish design feature weighting factors for computing the fail-safe cost and weight ratings for the fuselage structures:

$$\text{"W" Rating} = 0.35\ W_R + 0.20\ W_{C-R} + 0.20\ W_{D/I} + 0.25\ W_M$$

$$\text{"\$" Rating} = 0.35\ \$_R + 0.10\ \$_{C-R} + 0.30\ \$_{D/I} + 0.25\ \$_M$$

Three conceptual fuselage designs (all potentially effective from a fail-safe standpoint) were selected for the trade-off study. The structural descriptions of these designs are as follows:

STRUCTURE A

Structure A is a semimonocoque fuselage construction with a riveted attachment consisting of a skin-stringer-frame combination. A detectable crack length on the skin is discernable to the naked eye under careful examination. Detectable crack lengths on the stringers and frames are also visible, but they may require NDI equipment in some areas. A moderate degree of teardown is required to inspect the stringers and frames. Doubles are incorporated around all cutouts.

STRUCTURE B

Structure B is a semimonocoque fuselage construction which has bonded aluminum honeycomb panels with internal tear straps. The inspection of panels requires a combination of visual examination and NDI but no structural teardown. Detectable crack lengths on the stringers and frames are generally visible, but they may require NDI equipment in some areas (no teardown required).

STRUCTURE C

Structure C is a monocoque sandwich clam shell construction composed of covers, composite materials, and a Nomex core (honeycomb). The structure is fabricated in two halves which are connected by upper and lower splice plates. The design includes heavy peripheral and longitudinal internal straps. Flaws are normally detected visually.

The results of the trade-off study are shown in Table 6 and illustrated in Figure 30.

TABLE 6. FUSELAGE STRUCTURAL DESIGN TRADE-OFF STUDY

| Fuselage Structural Description | Design Technique Technical Merit Rating | | | | Cost | | | | | Weight | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | R | C-R | D/I | Fail-Safe Index | $\$_R$ | $\$_{C\text{-}R}$ | $\$_{D/I}$ | $\$_M$ | $\$$ | $W_R$ | $W_{C\text{-}R}$ | $W_{D/I}$ | $W_M$ | $W$ |
| A | 6 | 6 | 5 | 5.67 | 9 | 9 | 4 | 6 | 6.75 | 4 | 7 | 6 | 4 | 5.00 |
| B | 9 | 10 | 5 | 8.00 | 7 | 7 | 5 | 6 | 6.15 | 7 | 8 | 6 | 4 | 6.25 |
| C | 4 | 8 | 7 | 6.33 | 6 | 3 | 7 | 4 | 5.50 | 10 | 10 | 9 | 6 | 8.80 |

NOTE: High rating in each category indicates desirability

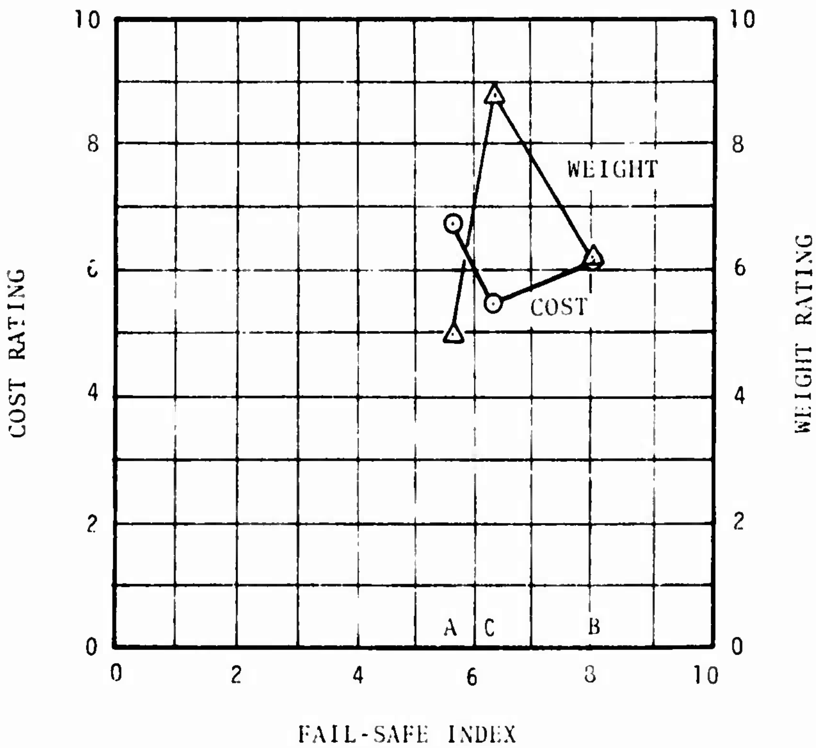


Figure 30. Fuselage Structural Design Trade-off Study.

Structure B offers the optimum in cost and weight relative to the level of fail safety. Structure A is the least costly but the heaviest, and Structure C is the lightest.

The high technical merit rating for Structure B was due primarily to the redundancy and crack-retardation features inherent in a semimonocoque sandwich construction. Considering the nominal cost and weight penalties for this structure, Structure B has the best potential as an effective fail-safe fuselage design.

## 5.2 DYNAMIC COMPONENT TRADE-OFF STUDY

Because of the similarity of their fail-safe designs relative to the technical merit, cost, and weight models, the rotor hub and rotating controls are considered together in the following structural trade-off study. The rotor blade must be considered separately.

The technical merit, cost, and weight rating tables developed for the helicopter dynamic component structures are presented in Appendix A. Table 7 summarizes the design features in the models.

| TABLE 7. DESIGN FEATURES USED IN DYNAMIC COMPONENTS TRADE-OFF STUDY | | |
|---|---|---|
| Technical Merit Rating | Cost Rating | Weight Rating |
| Redundancy | Redundancy | Redundancy |
| Detection/Inspection | Detection/ Inspection | Detection/Inspection |
| | Material | Material |

Since both the literature and the Government/industry survey indicated that the spar and the root-end attachment substructures are most critical, these two substructures were emphasized in the technical merit redundancy rating. The aft fairing assembly was considered the next most important component in the rating of a rotor blade design as an effective fail-safe structure.

Because of its high reliability, as evidenced by increasing industry-wide usage, the leak-before-break condition indicator for rotor blade spars was strongly weighted in the technical merit detectability/inspectability rating. In addition to this detection device, those designs whose detection and inspection procedures require the least amount of NDI equipment, component removal, and component teardown were assigned the more favorable ratings.

The technical merit rating for each of the rotor blade structures was derived by equally weighting (averaging) the redundancy (R) and the detectability/inspectability (D/I) ratings.

The rationale behind the cost and weight ratings for the fail-safe rotor blade designs is identical to that described for the fuselage structure in Section 5.1. As for the fuselage structure, designs with composite materials had a 5% cost penalty but a 5% weight reward.

The following equations were developed to establish design feature weighting factors for computing the fail-safe cost and weight ratings for the rotor blade structures:

$$\text{"W" Rating} = 0.40\ W_R + 0.35\ W_{C-R} + 0.25\ W_M$$

$$\text{"\$" Rating} = 0.40\ \$_R + 0.35\ \$_{C-R} + 0.25\ \$_M$$

The following three rotor blade designs were selected for the trade-off study:

STRUCTURE A

Structure A is a single spar-rib box construction. Primary construction materials are aluminum and steel. A D-shaped spar incorporates a pressure-differential condition indicator and the root end is attached by an integral flange.

STRUCTURE B

Structure B is a twin-beam honeycomb construction composed primarily of composite materials. Each glass fiber epoxy spar is adhesively bonded and mechanically attached to its own redundant titanium root end. The blade is fabricated in two halves which are bonded together at the chord line. The resulting redundant slow-crack-growth structure may be inspected visually.

STRUCTURE C

Structure C is a single spar/honeycomb construction composed of composite materials and compatible metals. Constructed of fiberglass, a D-shaped spar incorporates a pressure-differential condition indicator, and terminates in a multiple wraparound root-end retention system. A Nomex honeycomb core and fiberglass skins are bonded to the spar. A detachable aft fairing assembly affords easy spar inspection and repair.

The results of the trade-off study are shown in Table 8 and illustrated in Figure 31.

TABLE 8. ROTOR BLADE STRUCTURAL DESIGN TRADE-OFF STUDY

| Rotor Blade Structure | Design Technique Technical Merit Rating | | | Cost | | | | Weight | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| | R | D/I | Fail-Safe Index | $\$_R$ | $\$_{D/I}$ | $\$_M$ | $\$$ | $W_R$ | $W_{D/I}$ | $W_M$ | $W$ |
| A | 1 | 7 | 4.00 | 10 | 7 | 6 | 7.95 | 3 | 6 | 4 | 4.30 |
| B | 10 | 3 | 6.50 | 4 | 3 | 4 | 3.65 | 10 | 5 | 5 | 7.00 |
| C | 5 | 7 | 6.00 | 5 | 7 | 6 | 5.95 | 6 | 6 | 6 | 6.00 |

NOTE: High rating value indicates desirability

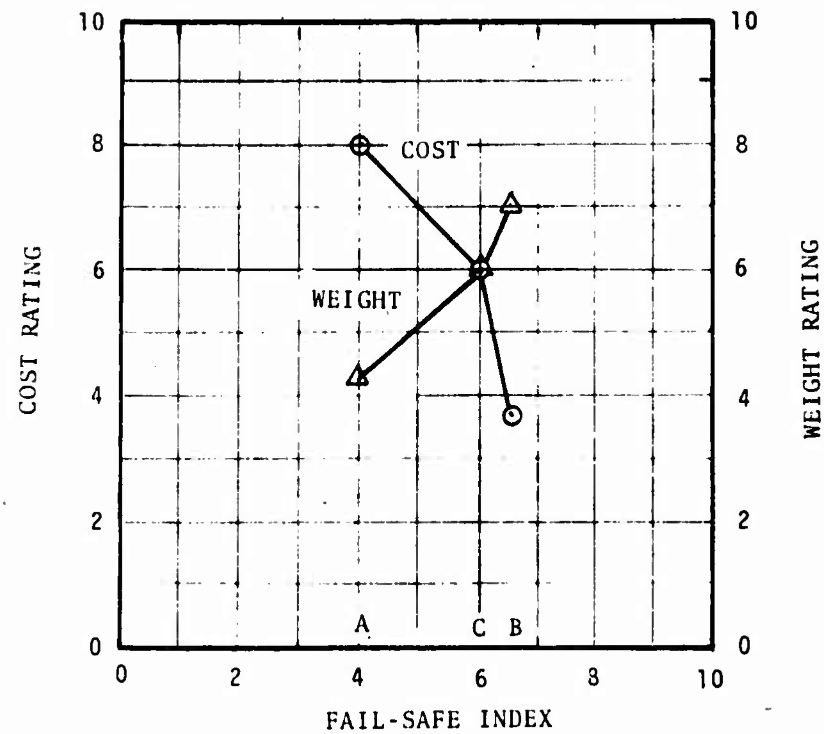


Figure 31. Rotor Blade Structural Design Trade-off Study.

The trade-off study indicated that Structure C would probably offer the highest level of fail safety with the detrimental effects of cost and weight being minimal. In addition, this structure had the highest fail-safe index and the highest weight rating (that is, the least weight). The extremely low cost rating for Structure C might improve in a more detailed design study.

As might be expected, Structure A had the highest cost rating, the lowest weight rating, and the least potential as a fail-safe structure.

As indicated in the two surveys, most of the design features that have a high potential for making a structure fail safe are being applied to the rotor hub and to a lesser extent to the rotating controls. Because of the difficulty in distinguishing specific substructures, the technical merit, cost, and weight ratings were based on (1) the redundancy rating, that is, the approximate percentage of the structure with a redundant load path, and (2) the D/I rating, that is the approximate percentage of a nonredundant structure monitored by a reliable condition indicator. The rating rationale is similar to that described previously.

The technical merit rating (or fail-safe index) is defined by equally weighting (averaging) the redundancy (R), the crack retardation (C-R), and the detectability/inspectability (D/I) ratings.

The following equations were developed to establish design feature weighting factors for computing the fail-safe cost and weight ratings for the rotor hub and rotating controls:

$$\text{"W" Rating} = 0.35\ W_R + 0.40\ W_{D/I} + 0.25\ W_M$$

$$\text{"\$" Rating} = 0.40\ \$_R + 0.35\ \$_{D/I} + 0.25\ \$_M$$

The following three conceptual designs for the rotor hub and rotating controls were selected for the trade-off study:

STRUCTURE A

In this structure, the rotor hub and rotating controls are constructed of titanium. Fail safety is provided primarily through redundancy and a slow crack-growth structure such that cracks are visible before they become critical. Pressure-differential condition indicators are incorporated in the pitch housing barrel and swash plate (redundancy not feasible in these structures).

STRUCTURE B

In this structure, the rotor hub and rotating controls are also constructed of titanium. Fail safety is provided by a slow crack-growth structure supplemented with pressure-differential condition indicators in most vital areas except the blade retention area where redundant load paths are provided.

STRUCTURE C

In this structure, the rotor hub and rotating controls are constructed of composite materials and compatible metals. Fail safety is provided by the slow crack-growth characteristics and inherent redundancy of composite materials. Extensive visual and nondestructive inspections are periodically required.

The results of the rotor hub/rotating controls trade-off study are presented in Table 9 and illustrated in Figure 32.

Structure A would probably offer the highest level of fail safety with the detrimental effects of cost and weight being minimal. This structure also had the highest fail-safe index rating.

TABLE 9. ROTOR HUB/ROTATING CONTROLS STRUCTURAL DESIGN TRADE-OFF STUDY

| Hub/Rotating Controls Structure | Design Technique Technical Merit Rating | | | Cost | | | | Weight | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| | R | D/I | Fail-Safe Index | $\$_R$ | $\$_{D/I}$ | $\$_M$ | $ | $W_R$ | $W_{D/I}$ | $W_M$ | W |
| A | 8 | 9 | 8.50 | 3 | 9 | 6 | 5.85 | 8 | 2 | 4 | 4.60 |
| B | 2 | 7 | 4.50 | 9 | 7 | 6 | 7.55 | 2 | 4 | 4 | 3.30 |
| C | 7 | 1 | 4.00 | 4 | 1 | 4 | 2.95 | 7 | 10 | 6 | 7.95 |
| NOTE: High rating value inicates desirability | | | | | | | | | | | |

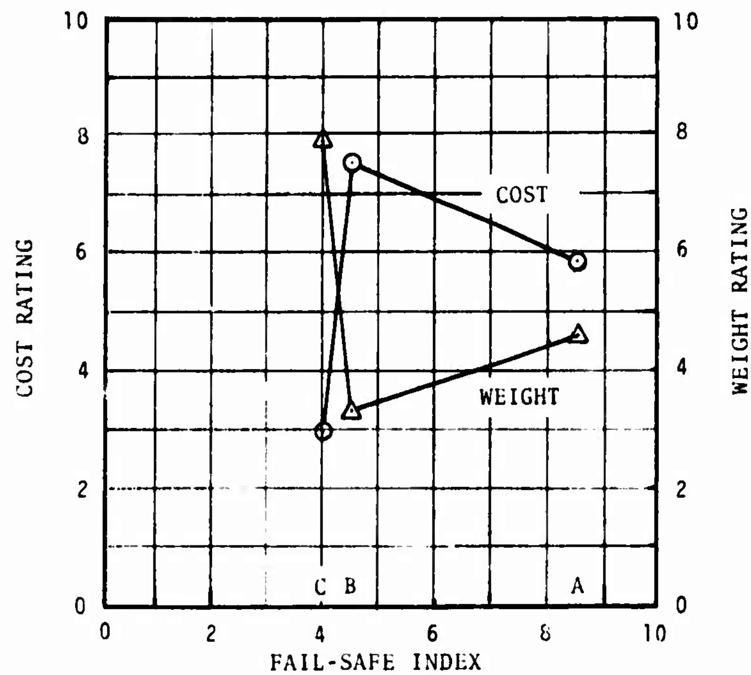


Figure 32. Hub/Rotating Controls Structural Design Trade-off Study.

Structure B had the highest cost rating, and Structure C had the highest weight rating.

## 5.3 TRADE-OFF STUDY SUMMARY

After technical merit, cost, and weight models were formulated for the helicopter fuselage, rotor blade, rotor hub, and rotating control structures, three designs for each structural category were evaluated relatively in terms of the design features that contribute to fail-safe performance.

The trade-off ratings are relative values; that is, the ratings for one design have significance only with respect to those for the other two designs. Further, the trade-off study was applicable only to the structures satisfying the fail-safe/safe-life design criteria described in Section 3.

# 6. RELATIONSHIP BETWEEN RECOMMENDED DESIGN CRITERIA AND HELICOPTER DESIGN SPECIFICATIONS

This section discusses the relationship between the recommended design criteria and the strictly military specifications, namely, MIL-S-8698 and AR-56.

## 6.1 MIL-S-8698 DESIGN CRITERIA

MIL-S-8698 (Military Specification Structural Design Requirements, Helicopters) specifies static and fatigue strength minimum requirements. In particular, it requires that the minimum ultimate factor of safety be 1.5, that the magnitude of stress reversals be minimized, and that materials and design details be such that the possibility of fatigue failure is minimal. In addition, it requires a minimum fatigue life of 1000 hours based on an approved fatigue loading schedule. Although the foregoing are safe-life design requirements, that is, they require adequate strength to resist fatigue failure, they do not include provisions for residual strength, slow crack growth, redundant load path, etc.

Paragraph 3.6.5.2.4.2 in MIL-S-8698 requires a fail-safe mechanism for the fuselage structure as follows:

> "Means shall be provided to prevent the complete separation of the power plant or rotor from the helicopter in case of failure of the isolator elastic material or its banding. In case of such failure, the displacement of the power plant shall not be sufficient to break fuel or oil lines, or result in rotor blades striking any part of the helicopter."

Although the context of this paragraph implicitly recognizes the need to apply fail-safe principles, it does not contribute to the state of the art in the fail-safe design described in Section 3. Moreover, it is not feasible to incorporate backup structures in most instances.

While MIL-S-8698 does not contribute to helicopter fuselage fail-safe design criteria, it does specify that helicopter designs meet a standard for fatigue-crack-initiation resistance that has stood the test of time. However, as nearly unanimously expressed by Government and industry representatives, this document should be updated to include provisions for the structural integrity of a flawed fuselage structure.

## 6.2 AR-56 DESIGN CRITERIA

AR-56 (Structural Design Requirements, Helicopters) outlines a safe fatigue life requirement. It requires static and fatigue strength minimums in terms of ultimate strength, limit strength, and fatigue strength. AR-56 prescribes that the design fatigue life for the critical helicopter structure in Class I helicopters (whose primary missions are attack, assault, utility, and training) be 6000 hours and that for the critical helicopter structure in Class II helicopters (whose primary missions are ASW, observation, reconnaissance, mine countermeasures, and cranes) be 5000 hours.

Additionally, AR-56 contains requirements for designing all critical structures to fail-safe design principles.

In essence, AR-56 prescribes that fail safety be effected by providing a redundant load path or by ensuring that a partial failure will be detected before a catastrophic failure occurs.

AR-56 also specifies that if the helicopter has a redundant load path structure, the failure of a single structural element or control element must not cause the ultimate factor of safety to be less than 1.0. Additionally, the failure of a single element will not cause the aircraft to have uncontrollable motions within its design performance limits.

As stated in AR-56, an alternative procedure for designing fail-safe structures on the basis of partial but detectable failures would be according to the following directives: (1) all partial failures must become readily detectable under prescribed and acceptable procedures; (2) the interval between the time when a partial failure becomes readily detectable and the time when such a failure will reduce the residual strength of the structure to the limit load must be determined; and (3) an inspection interval must be so defined with respect to the interval determined in (2) that the probability of catastrophic failure is extremely remote. This alternative design procedure is almost identical to that described for the procedure in criteria Case 5 (see Section 3).

AR-56 states the primary fail-safe design procedure as follows:

> "The complete airframe and all its components shall be constructed so that failure of a single structural element or control element will neither cause catastrophic failure nor preclude safe continuous flight to a normal destination where repairs/corrections can be made... Redundancy, such as alternate load-paths and systems, and

other fail-safe principles are required to achieve this capability."

This requirement may be somewhat incomplete as a fail-safe design criterion since the assurance of "safe continuous flight to a normal destination where repairs/corrections can be made" is open to interpretation. On the other hand, the design criteria in Cases 3, 4, 9, and 10 offer specific and likely superior fail-safe design criteria since they specify that any damaged structure be detected by an adequate inspection procedure and that the remaining structure have a safe fatigue life at least equal to the design service life or a crack life at least equal to the design service life (as determined by a residual life analysis such as that illustrated in Cases 9 and 10).

However, aside from the one potential weakness cited in the previous paragraph, the fail-safe and safe-life design requirements in Section 3.1.9 of AR-56 are on the whole very thorough and comprehensive. Because of their generality, these requirements may be applied to almost every aircraft structure.

## 7. SUMMARY AND CONCLUSIONS

Current fail-safe and safe-life design criteria, design philosophies, and specific design techniques have been reviewed and assimilated into a representative sample of the state of the art in fail-safe/safe-life design criteria. Also, current design practices relative to the dynamic components and the fuselage and landing gear structures of helicopters were examined to evaluate the relative necessity and applicability of the fail-safe design criteria to each of the structures.

Based on the state-of-the-art survey of current fail-safe and safe-life design criteria, the following conclusions were drawn:

(1) Fail-safe design principles must be considered as complementary to, not as substitutes for, safe (fatigue) life design principles.

(2) As discussed in Section 3, several different design techniques may be used to incorporate fail-safe provisions in helicopters.

(3) Wherever necessary, the fail-safe design principles may generally be applied to virtually any structure. However, the cost and weight penalties may be prohibitive unless the savings of innovative designs offset these penalties.

(4) As evidenced by the proposed HLH rotor blade and hub assemblies and the proposed UTTAS dynamic components, modern technology is capable of constructing fail-safe dynamic components with minimal cost and weight penalties.

(5) Since current helicopter fuselage structures (primarily those of semimonocoque construction) have an inherent capability of retarding crack growth, the fuselage is generally considered to be inherently fail-safe. Although this construction is inherently damage tolerant, design criteria such as those in Section 3 would ensure fail safety.

(6) The industry and literature surveys indicated that while fail-safe helicopter landing gear structures were technically feasible, the practicality, and in fact the necessity, of such landing gear structure is questionable.

(7) MIL-S-8698 does not contain any provisions for fail-safe designs. Such provisions have become essential to any military helicopter structural design specification.

(8) AR-56 contains a very comprehensive set of fail-safe design criteria.

## 8. RECOMMENDATIONS

Based on the results of the literature and Government/industry surveys, the following recommendations are proposed:

(1) Since the surveys revealed that fail-safe design principles have not as yet been applied to helicopter landing gear systems, preliminary design studies should be conducted to investigate the feasibility of developing potential fail-safe landing gear structures. At least such studies would advance the state of the art in helicopter landing gear design.

(2) Because of the difficulty in formulating and applying a model to conduct cost and weight trade-off studies on various fail-safe design techniques, a much more detailed model should be developed to aid the designer in evaluating the fail-safe effectiveness of these techniques.

(3) The effort to develop the above model should include designing and fabricating representative structures to derive standards for various levels and degrees of fail safety, cost, and weight.

(4) Any critical structure not as yet designed to fail-safe design criteria should be reevaluated to determine the feasibility of its being modified sufficiently to make it a fail-safe structure. Moreover, such studies should be oriented so that they may introduce innovative concepts for new fail-safe provisions, minimal retrofit requirements, and low cost and weight, such as the relatively simple concept for the leak-before-break condition-indicating system.

(5) Since many of the Government and industry representatives were indecisive in their definition of what constitutes a fail-safe structure, a short, concise document should be prepared to summarize the basic concepts of fail-safe philosophy, criteria, and methodology, to present representative illustrations of these concepts, and to stress the significance of these concepts in the light of the increasing importance and complexity of the state of the art in fail-safe design.

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# APPENDIX A

## TECHNICAL MERIT, COST, AND WEIGHT RATING TABLES FOR FUSELAGE AND DYNAMIC COMPONENTS

TABLE A-1. TECHNICAL MERIT (REDUNDANCY) - FUSELAGE

| "R" Rating | Structure Type | Construction Description |
|---|---|---|
| 1 | Monocoque | Sculptured plate |
| 2 | Monocoque | Sculptured plate with doublers |
| 3 | Monocoque | Sandwich panel/shell |
| 4 | Monocoque | Sandwich panel/shell with doublers and/or stiffeners |
| 5 | Semimonocoque | Monolithic skin/stringer/frame |
| 6 | Semimonocoque | Monolithic skin/stringer/frame with doublers and/or stiffeners |
| 7 | Semimonocoque | Sculptured plate skin/stringer/ frame |
| 8 | Semimonocoque | Sculptured plate skin/stringer/ frame with doublers |
| 9 | Semimonocoque | Sandwich skin/stringer/frame |
| 10 | Semimonocoque | Sandwich skin/stringer/frame with doublers and/or stiffners |

TABLE A-2. TECHNICAL MERIT (CRACK RETARDATION) - FUSELAGE

| "C-R" Rating | Structure Type | Construction Description |
|---|---|---|
| 1 | Monocoque | Construction displays neither positive nor inherent crack retardation characteristics |
| 2-4 | | Construction displays degree of inherent crack retardation capability |
| 5-7 | Semimonocoque | Construction displays degree of inherent crack retardation capability |
| 8 | Monocoque | Construction incorporates positive crack retardation mechanism |
| 9-10 | Semimonocoque | Construction incorporates degree of positive crack retardation mechanism |

TABLE A-3. TECHNICAL MERIT (DETECTABILITY/INSPECTABILITY) - FUSELAGE

| "D/I" Rating | Detection Technique | Relative Structural Teardown Required for Inspection |
|---|---|---|
| 1-2 | N.D.I | Extensive |
| 3-4 | Visual/C.I. | Extensive |
| 5-6 | N.D.I. | Minimal |
| 7-8 | C.I. | Minimal |
| 9-10 | Visual | Minimal |

| TABLE A-4. COST (REDUNDANCY) - FUSELAGE | | |
|---|---|---|
| "$\$_R$" Rating | Structure Type | Construction Description |
| 1 | Monocoque | Sculptured plate |
| 2 | | Sculptured plate with doublers |
| 3 | Semimonocoque | Sculptured plate skin/stringer/frame |
| 4 | | Sculptured plate skin/stringer with doublers |
| 5 | Monocoque | Sandwich panel/shell/frame |
| 6 | | Sandwich panel/shell with doublers and/or stiffeners |
| 7 | Semimonocoque | Sandwich skin/stringer/frame |
| 8 | | Sandwich skin/stringer with doublers and/or stiffeners |
| 9 | | Monolithic skin/stringer with doubler and/or stiffeners |
| 10 | | Monolithic skin/stringer |

| TABLE A-5. COST (CRACK RETARDATION) - FUSELAGE | | |
|---|---|---|
| "$\$_{C-R}$" Rating | Structure Type | Construction Description |
| 1 | Monocoque | Construction displays neither positive nor inherent crack retardation characteristics |
| 2-5 | Monocoque | Construction displays degree of inherent crack retardation capability and/or positive crack retardation mechanism |
| 6-7 | Semimonocoque | Construction incorporates degree of positive crack retardation mechanism |
| 8-10 | Semimonocoque | Construction displays degree of inherent crack retardation capability |

| TABLE A-6. COST (DETECTABILITY/INSPECTABILITY) - FUSELAGE | | | |
|---|---|---|---|
| "$\$_{D/I}$" Rating | Detection Technique | Inherent Inspectability | Relative Amount of Structural Teardown Required |
| 1 | C.I./N.D.I | No | Extensive |
| 2 | Visual | No | Extensive |
| 3 | C.I./N.D.I. | No | Minimal |
| 4 | Visual | No | Minimal |
| 5-6 | N.D.I. | Yes | Minimal |
| 7-8 | C.I. | Yes | Minimal |
| 9-10 | Visual | Yes | Minimal |

| TABLE A-7. COST (MATERIAL) - FUSELAGE | |
|---|---|
| "$\$_M$" Rating | Basic Structural Composition |
| 4 | Composite material |
| 6 | Metallic material |

| TABLE A-8. WEIGHT (REDUNDANCY) - FUSELAGE | | |
|---|---|---|
| "$W_R$" Rating | Structural Type | Construction Description |
| 1 | Monocoque | Sculptured plate |
| 2 | Monocoque | Sculptured plate with doublers |
| 3 | Semimonocoque | Monolithic skin/stringer/frame |
| 4 | Semimonocoque | Monolithic skin/stringer/frame with doublers and/or stiffeners |
| 5 | Semimonocoque | Sculptured plate skin/stringer/frame |
| 6 | Semimonocoque | Sculptured plate skin/stringer/frame with doublers |
| 7 | Semimonocoque | Sandwich skin/stringer/frame |
| 8 | Semimonocoque | Sandwich skin/stringer/frame with doublers and/or stiffeners |
| 9 | Monocoque | Sandwich panel/shell |
| 10 | Monocoque | Sandwich panel/shell with doublers and/or stiffeners |

| "$W_{C-R}$" Rating | Structural Type | Construction Description |
|---|---|---|
| 1 | Monocoque | Construction displays neither positive nor inherent crack retardation characteristics |
| 2-6 | Semimonocoque | Construction exhibits degree of inherent crack retardation capability |
| 7-8 | | Construction incorporates degree of positive crack retardation through design feature |
| 9 | Monocoque | Construction exhibits degree of inherent crack retardation capability |
| 10 | | Construction exhibits degree of inherent, as well as positive, crack retardation capability |

TABLE A-9. WEIGHT (CRACK RETARDATION) - FUSELAGE

| "$W_{D/I}$" Rating | Detection Technique | Inherent Inspectability | Relative Amount of Structural Teardown Required |
|---|---|---|---|
| 1 | C.I./N.D.I. | No | Minimal |
| 2 | Visual | No | Minimal |
| 3-4 | Visual | No | Extensive |
| 5-6 | C.I./N.D.I. | No | Extensive |
| 7-8 | Visual | Yes | Minimal |
| 9-10 | C.I./N.D.I. | Yes | Minimal |

TABLE A-10. WEIGHT (DETECTABILITY/INSPECTABILITY) - FUSELAGE

| TABLE A-11. WEIGHT (MATERIAL) - FUSELAGE | |
|---|---|
| "$W_M$" Rating | Basic Structural Composition |
| 4 | Metallic material |
| 5 | Metallic/composite |
| 6 | Composite Material |

| TABLE A-12. TECHNICAL MERIT (REDUNDANCY) - ROTOR BLADE | | | |
|---|---|---|---|
| "R" | | Rotor Blade (Primary) Components | |
| Rating | Spar | Aft Fairing Construction | Root-End Retention |
| 1 | Single | Rib | Integral flange |
| 2 | Single | Rib | Single "coke bottle" socket |
| 3 | Single | Rib | Double "coke bottle" socket |
| 4 | Single | Honeycomb | Integral flange |
| 5 | Single | Honeycomb | Wraparound with fitting |
| 6 | Single | Honeycomb | Single "coke bottle" socket |
| 7 | Single | Honeycomb | Double "coke bottle" socket |
| 8 | Single | Honeycomb | Fittingless wraparound |
| 9 | Twin | Honeycomb | Wraparound with a fitting |
| 10 | Twin | Honeycomb | Fittingless wraparound |

| TABLE A-13. TECHNICAL MERIT (DETECTABILITY/ INSPECTABILITY) - ROTOR BLADE | | | |
|---|---|---|---|
| "D/I" Rating | Detection Technique | Inspection Requirement | |
| | | Component Removal | Component Teardown |
| 1 | N.D.I./visual | Yes | Extensive |
| 2 | N.D.I./visual | Yes | Minimal |
| 3-5 | N.D.I./visual | No | Minimal |
| 6-8 | N.D.I./visual | No | None |
| 9-10 | C.I. | No | None |

NOTE: Average for spar and root-end attachment

| TABLE A-14. COST (REDUNDANCY) - ROTOR BLADE | | | |
|---|---|---|---|
| "$_R$" Rating | Rotor Blade Components | | |
| | Aft Fairing Construction | Spar | Root-End Retention |
| 1 | Honeycomb | Single | Double "coke bottle" socket |
| 2 | Honeycomb | Single | Single "coke bottle" socket |
| 3 | Honeycomb | Twin | Wraparound with fitting |
| 4 | Honeycomb | Twin | Fittingless wraparound |
| 5 | Honeycomb | Single | Wraparound with fitting |
| 6 | Honeycomb | Single | Fittingless wraparound |
| 7 | Honeycomb | Single | Integral flange |
| 8 | Rib | Single | Double "coke bottle" socket |
| 9 | Rib | Single | Single "coke bottle" socket |
| 10 | Rib | Single | Integral flange |

TABLE A-15. COST (DETECTABILITY/INSPECTABILITY) - ROTOR BLADE

| "$\$_{D/I}$" Rating | Detection Technique | Inspection Requirement | |
|---|---|---|---|
| | | Component Removal | Component Teardown |
| 1 | N.D.I./visual | Yes | Extensive |
| 2 | N.D.I./visual | Yes | Minimal |
| 3-5 | N.D.I./visual | No | Minimal |
| 6-8 | N.D.I./visual | No | None |
| 9-10 | C.I. | No | None |

Note: Average for spar and root-end attachment

TABLE A-16. COST (MATERIAL) - ROTOR BLADE

| "$\$_M$" Rating | Basic Structural Composition |
|---|---|
| 4 | Composite material |
| 6 | Metallic material |
| 6 | Composite/Metallic |

| TABLE A-17. WEIGHT (REDUNDANCY) - ROTOR BLADE | | | |
|---|---|---|---|
| "$W_R$" Rating | Rotor Blade Components | | |
| | Spar | Aft Fairing Construction | Root-End Retention |
| 1 | Single | Rib | Double "coke bottle" socket |
| 2 | Single | Rib | Single "coke bottle" socket |
| 3 | Single | Rib | Integral flange |
| 4 | Single | Honeycomb | Double "coke bottle" socket |
| 5 | Single | Honeycomb | Single "coke bottle" socket |
| 6 | Single | Honeycomb | Wraparound with fitting |
| 7 | Single | Honeycomb | Fittingless wraparound |
| 8 | Single | Honeycomb | Integral flange |
| 9 | Twin | Honeycomb | Wraparound with fitting |
| 10 | Twin | Honeycomb | Fittingless wraparound |

| TABLE A-18. WEIGHT (DETECTABILITY/INSPECTABILITY) - ROTOR BLADE | | | |
|---|---|---|---|
| "$W_{D/I}$" Rating | Detection Technique | Inspection Requirement | |
| | | Component Removal | Component Teardown |
| 1-3 | N.D.I./visual | No | Minimal |
| 4-6 | N.D.I./visual | No | None |
| 7 | N.D.I./visual | Yes | Extensive |
| 8 | N.D.I./visual | Yes | Minimal |
| 9-10 | C.I. | No | None |
| Note: Average for spar and root-end attachment | | | |

| TABLE A-19. WEIGHT (MATERIAL) - ROTOR BLADE | |
|---|---|
| "$W_M$" Rating | Basic Structural Composition |
| 4 | Metallic material |
| 5 | Composite/metallic |
| 6 | Composite material |

| TABLE A-20. TECHNICAL MERIT (REDUNDANCY) - HUB/ROTOR CONTROLS | |
|---|---|
| "R" Rating | Approximate Percentage of Structure Characterized by Redundant Load Path |
| 1-2 | 0% - 25% |
| 3-4 | 25% 50% |
| 5 | 50% |
| 6-7 | 50% - 75% |
| 8-9 | 75% - 100 % |
| 10 | 100% |

| TABLE A-21. TECHNICAL MERIT (DETECTABILITY/ INSPECTABILITY) - HUB/ROTOR CONTROLS | | |
|---|---|---|
| "D/I" Rating | Detection Technique | Approximate Percentage of Non-redundant Structure Monitored by Reliable C.I.* |
| 1- | N.D.I./visual | 0% |
| 3-4 | N.D.I./visual/C.I. | 0% - 25% |
| 5-6 | N.D.I./visual/C.I. | 25% - 50% |
| 7-8 | N.D.I./visual/C.I. | 50% - 75% |
| 9-10 | N.D.I./visual/C.I. | 75% - 100% |

* Assume most of remaining percentage of structure contains redundant load paths inspectable by NDI and/or visual means - off or on the helicopter.

| TABLE A-22. COST (REDUNDANCY) - HUB/ROTOR CONTROLS | |
|---|---|
| "$S_R$" Rating | Approximate Percentage of Structure Characterized by Redundant Load Path |
| 1 | 100% |
| 2-3 | 75% - 100% |
| 4-5 | 50% - 75% |
| 6 | 50% |
| 7-8 | 25% - 50% |
| 9-10 | 0% - 25% |

| TABLE A-23. COST (DETECTABILITY/INSPECTABILITY) - HUB/ROTOR CONTROLS | | |
|---|---|---|
| "$S_{D/I}$" Rating | Detection Technique | Approximate Percentage of Nonredundant Structure Monitored by Reliable C.I.* |
| 1-2 | N.D.I./visual | 0% |
| 3-4 | N.D.I./visual/C.I. | 0% - 25% |
| 5-6 | N.D.I./visual/C.I. | 25% - 50% |
| 7-8 | N.D.I./visual/C.I. | 50% - 75% |
| 9-10 | N.D.I./visual/C.I. | 75% - 100% |

*Assume most of remaining percentage of structure contains redundant load paths inspectable by NDI and/or visual means off or on the helicopter

| TABLE A-24. COST (MATERIALS) - HUB/ROTOR CONTROLS | |
|---|---|
| "$S_M$" Rating | Basic Structural Composition |
| 4 | Composite material |
| 6 | Metallic material |
| 6 | Composite/Metallic |

| TABLE A-25. WEIGHT (REDUNDANCY) - HUB/ROTOR CONTROLS | |
|---|---|
| "$W_R$" Rating | Approximate Percentage of Structure Characterized by Redundant Load Path |
| 1-2 | 0 - 25% |
| 3-4 | 25% - 50% |
| 5 | 50% |
| 6-7 | 50% - 75% |
| 8-9 | 75% - 100% |
| 10 | 100% |

| TABLE A-26. WEIGHT (DETECTABILITY/INSPECTABILITY) - HUB/ROTOR CONTROLS | | |
|---|---|---|
| "$W_{D/I}$" Rating | Detection Technique | Approximate Percentage of Nonredundant Structure Monitored by Reliable C.I.* |
| 1-2 | N.D.I./visual/C.I. | 75% - 100% |
| 3-4 | N.D.I./visual/C.I. | 50% - 75% |
| 5-6 | N.D.I./visual/C.I. | 25% - 50% |
| 7-8 | N.D.I./visual/C.I. | 0% - 25% |
| 9-10 | N.D.I./visual | 0% |

*Assume most of remaining percentage of structure contains redundant load paths inspectable by NDI and/or visual means off or on the helicopter

| TABLE A-27. WEIGHT (MATERIALS) - HUB/ROTOR CONTROLS | |
|---|---|
| "$W_M$" Rating | Basic Structural Composition |
| 4 | Metallic material |
| 5 | Composite/metallic |
| 6 | Composite material |

## APPENDIX B

## ABSTRACTS OF PRIMARY REPORTS IN LITERATURE SURVEY

TABLE B-1. SUMMARY OF PRIMARY REPORTS ACCORDING TO SUBJECT AREA

| SUBJECT AREA | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 | 22 | 23 | 24 | 25 | 26 | 27 | 28 | 29 | 30 | 31 | 32 | 33 | 34 | 35 | 36 | 37 | 38 | 39 | 40 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Helicopter Structures | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Rotor Blade | | | | | | | ● | | | | ● | | | ● | | | ● | | ● | | ● | ● | | ● | | | | | ● | ● | | | | | | | | ● | | |
| Rotor Hub | | | | | | | | | | | ● | | ● | | | | | ● | | | | | ● | | | | | ● | | ● | | | | | | ● | | ● | | |
| Rotating Controls | | | | | | | | | | | ● | | ● | | | | | | | | | | | | | | | | | ● | | | | | | | | ● | | |
| Fuselage | | | | | | | | | ● | | | | | | | | | | | ● | | | ● | | ● | | | | | ● | | | | | ● | | | | | |
| Landing Gear | | | | | | | | | | | | | | | | | | | | | | | | | | ● | | | | ● | | | | | ● | | | | | |
| ES/SL Design Methodology | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Proposed | ● | ● | | | ● | | | | | ● | | ● | | | ● | ● | | | | | ● | | | | | | | | | | | ● | | ● | | | ● | ● | ● | ● |
| Practice | | | ● | | ● | | ● | | ● | ● | ● | | ● | ● | ● | ● | ● | | ● | | ● | ● | ● | | | | | ● | | | ● | | ● | ● | | | ● | | ● | ● |
| Design Considerations | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Monolithic Structure | | | | | | | | | | | | | | ● | | | ● | | ● | | ● | | | ● | | | | | | | | | ● | | | | ● | | | |
| Controlled Fracture Struc | | | | | ● | | | | ● | | | | ● | | | | | | | | | | | | | | | | | | | | ● | ● | | | ● | | | ● |
| Redundant Structure | | | | | | | | | | | | | ● | | | | | | | | | | | | | | | | | | | ● | ● | | | | | | | |
| Design Material | | | | ● | | | | | ● | | | | ● | ● | ● | | | ● | ● | ● | ● | | | | | | | ● | | | | | | | | ● | | | | |
| Residual Strength | | ● | ● | | ● | | | | | | | | | | | | | | | | | | | ● | | | | | | | | | | | | ● | | | | |
| Crack Propagation | ● | ● | | | ● | | | ● | | | | | ● | ● | | | | | | | | | | | | | | | | | | | ● | ● | | | | | | ● |
| Condition Indicators | | | | | | | | | | | | | ● | | ● | | ● | | | | ● | ● | | | | | | | | | | | | | | | | | | |
| Cost | | | | | | | | | ● | | | | | ● | | | | | | | | | | | | ● | | | | ● | | | | | ● | ● | | ● | | |
| Weight | | ● | | | | | | | ● | | | | ● | | | | ● | | | ● | | | | | | ● | | | | | | | ● | | | ● | | | | |
| Other | | | | | | | ● | | ● | | | | | ● | ● | | ● | ● | ● | | | | | | ● | | | | | | | | ● | | | | | | | |
| Related Analyses | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Fracture Mechanics | | ● | | | ● | | | | | | | | ● | | | | | | | | | ● | | ● | | | ● | | ● | | | | ● | ● | | | ● | | | ● |
| Damage Tolerance | | ● | | | ● | | | | | | | | ● | | | | | | | | | ● | | ● | | | ● | | ● | | | | ● | ● | | | ● | | | ● |
| Ballistic Tolerance | | | | | | | | | | | | | | | | | | | | | | | | | | | | | ● | | | | | | | | | | | |
| Testing | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Instrumentation | | | | | | | | | | | | | | | ● | | | | | | | | | | | | | | | | | | | | | | | ● | | |
| Material | | | | ● | | ● | | ● | | | | | | | | | ● | | | | | | | ● | | | | | | | | | | | | ● | | | | |
| Component Design | | | ● | | | | ● | | | ● | | | ● | | ● | | | | ● | ● | ● | ● | | | | | | ● | | | | | | | | ● | | ● | | |
| Flight Loads | | | | | | | | | | ● | | | | | ● | | | | ● | | | | ● | | | | | ● | | | | | | | | | | | | |
| Service History | | | | | | | | | ● | ● | | | | | ● | | | | | | | | | | | | | | | | | | | | | | | | | |
| Miscellaneous | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Fabrication Methods | | | | | | | | | | | | | | ● | | | ● | | ● | ● | | | | | | | ● | | | | | | | | | ● | | | | |
| Maintenance | | | | | | | | | | | | | ● | ● | | | | ● | | | | | | | | | | | | ● | | | | | ● | | | ● | | |
| Inspection | | | | | | | | | ● | | | | | | | | | | | | | | | | | | | | | ● | | | | | | | | | | |
| Accident Investigation | | | | | | | | | | | | | | | | | | | | | | | | | | ● | | | ● | | | | | | | | | | | |
| Regulations | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | ● | | | | | | | | ● | | ● |

TABLE B-1. (Continued)

| SUBJECT AREA | LIBRARY REPORT NUMBER | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | | | | | 50 | | | | | | | | | | 60 | | | | | | | | | | 70 | | | | | | | | | | 80 |
| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 0 | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 0 | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 0 | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 0 |
| Helicopter Structures | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Rotor Blade |●| | | | |●|●|●|●| |●| | | |●|●| |●| | |●| | | | |●| | | | | | | | |●| | | |●| |
| Rotor Hub | | | | | | |●| | | |●| | |●|●|●| | | | |●| | | | |●| |●| | | | | | | | | | |●| |
| Rotating Controls | | | | | | |●| | | | | | | |●|●| | | | |●| | | | |●| | | | | | | | | | | | |●| |
| Fuselage | | | |●|●| |●|●|●| |●|●|●| | |●| | |●| |●|●| | | | | | | | | | | |●| |●| | |●|●|
| Landing Gear | | | |●| | |●|●|●| |●| | | | |●| | | | | | | | | | |●| | | | | | | | | | | | | |
| FS/SL Design Methodology | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Proposed |●|●| | | | |●| |●| | | | | |●| |●| | | | | |●|●| | | | |●|●|●| | |●| | | | | | |
| Practice | |●| | | | |●| |●| | |●| |●|●| | | |●|●| |●|●| | | |●| | |●|●| | |●|●| | |●|●| |
| Design Considerations | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Monolithic Structure | | | | | | | |●|●| |●| | | | | | |●| | | | | | | | | | | | | | | | |●| | | | |●|
| Controlled Fracture Struc | | | | |●| | | | | |●| | | | | | | | |●| |●| | | | | | |●| |●| | | | | | | | |●|
| Redundant Structure | | | | | | | | |●| | | | |●| | | | | | | | | | | | | | |●| | | | |●| | | | | |●|
| Design Material | | |●|●|●|●| |●|●|●|●| |●| | | | |●|●| | | | |●|●| |●|●| |●| | | | | |●| | |●|●|
| Residual Strength | |●| | | | | | |●| | |●| |●| | | | | |●|●|●| | | | | | |●| |●| | |●|●| | | |●| |
| Crack Propagation | | | | | | | | |●|●| | | |●| | | | | |●| | | | |●| | | |●| | |●|●|●|●| | | |●| |
| Condition Indicators | | | | | | |●| | | | | | |●| | |●| | | | | | | | | | | | | | | | |●| | | | | |
| Cost | | |●|●| | | | | | | | |●| | |●| | | | | | | | | | | | | | | | | | | | | | | |●|
| Weight | | |●|●|●|●| | | | | | |●| | | | | | | | | | | | |●|●| | | | | | | | | | | | |●|
| Other | | |●| | | | | | | | | | | | | |●|●| | | | |●| |●| |●| |●|●| | | | | | | | |●|
| Related Analyses | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Fracture Mechanics |●| | | | | | | | |●| | | |●| | | | | |●|●|●| | |●|●| | |●|●|●|●|●|●|●| | | |●| |
| Damage Tolerance |●| | | | | | | | |●| | | |●| | | | | |●|●|●| | |●|●| | |●|●|●|●|●|●|●| | | |●| |
| Ballistic Tolerance | | | | | | | |●|●| | | | |●| | | | | | |●| | | | | | | | | | | | | | | | | |●| |
| Testing | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Instrumentation | | | | | | | | | | | | | | | |●| | | | | |●| | | | | | | | | | | | | | | | | |
| Material | | | | | | | | | | | | |●| | |●| |●| | | |●| |●| | | | |●| | |●| | |●| | | | |
| Component Design | | | | |●|●| | | | | | | |●|●| |●|●|●| | |●| | | |●|●|●| | | | | | |●| | | | |
| Flight Loads | | | | | |●| | |●| | | | | |●| | | | | | |●| | | | | | | |●| | | | | |●| | | |
| Service History | | | | | | | | | | | | | | | | | | | | | |●| | | | | | | | | | | | | |●| | | |
| Miscellaneous | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Fabrication Methods | | | | |●|●| | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |●|
| Maintenance | | | | | | |●| | | | | | | | |●| | | | | | | | |●| | | | | | | | | | | | | | |
| Inspection | | | | | | |●| | |●| |●| |●| |●| | | | | | | | |●| | | |●|●| | | |●| | | | | |
| Accident Investigation | | | | | | | | |●| | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| Regulations | |●| | | | |●| | | | | | | | | | | | | | | | | |●| | | | | | | | | | | | | | |

1. Whittaker, I.C., and Saunders, S.C., Exploratory Development on Application of Reliability Analysis to Aircraft Structures Considering Interaction of Cumulative Fatigue Damage and Ultimate Strength, Boeing Company, Technical Report AFML-TR-72-283, Air Force Materials Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Dayton, Ohio, January 1973, AD-757-529.

An analysis method for determining the reliability of airplane structures, subjected to the cumulative and maximum operational loads and the resultant interaction of fatigue damage and strength, has been investigated. The design variables include the central tendency values of the fatigue performance; that is, the average lives to initiation and the growth of a major crack, and the effect of the crack on structural strength. Other variables include the standard operational procedure of periodic inspection of the structure and its repair when found to be damaged. Functions, based on the length of the fatigue crack, are used to describe both the residual strength of the structure and the probability of the crack being detected and the cracked structure being repaired. The times to initiation of a crack and the later time when the crack becomes critical, i.e., unstable, are taken as random variables. The derived reliability model considers that at any time the structure is either failed or unfailed. If unfailed, the structure may be uncracked, or cracked and undetected, or detected and repaired. Monte Carlo simulation is used to determine the cumulative distribution functions of the statistics describing these conditions. Results of an application of the developed reliability analysis system to an arbitrary situation are presented.

2. Whittaker, I.C., and Besuner, P.M., A Reliability Analysis Approach to Fatigue Life Variability of Aircraft Structures, Boeing Company, AFML-TR-69-65, Air Force Materials Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Dayton, Ohio, April 1969, AD-853-263.

The application of reliability analysis methods to the estimation of probable aircraft structural fatigue performance was investigated. Order statistics were used to assess the fatigue performance reliability of a fleet or number of fatigue-exposed details. A reliability analysis plan for application to aluminum alloy structural fatigue performance was developed and compared with the current fixed-scatter-factor procedure for determining the safe life of a structural detail. Both the two-parameter Weibull distribution and the log-normal distribution with empirically defined shape parameters were used to make the reliability plan tractable as compared to a distribution-free approach. Maximum-likelihood estimators, including one that considers only the first two-ordered failures, were employed to examine the many variables that might influence fatigue scatter, to quality fatigue data that represented aluminum structural scatter, and to establish shape parameter values that typified structural fatigue scatter. The sampling distributions of these estimators were required to work the problem and were calculated by means of existing theory or Monte-Carlo simulation. More than 2,000 groups of fatigue performance data were collected, analyzed, and used to demonstrate the feasibility of establishing a shape-parameter value. Based on this estimate, scatter factors have been generated to account for the penalty of limited input information, the degree of desired reliability, and the size of the exposed fleet. Using these factors, the possible effects of the reliability analysis on structural weight, payload, or range were explored for a jet-engined military tanker/transport-type airplane.

3. Schijve, J., The Accumulation of Fatigue Damage in Aircraft Materials and Structures, North Atlantic Treaty Organization Advisory Group for Aerospace Research and Development, AGARD-AG-157, National Aerospace Laboratory NLR, Amsterdam, The Netherlands, January 1972, AD-737-398.

The available literature is surveyed and analyzed. Physical aspects of fatigue damage accumulation are discussed, including interaction and sequence effects. Empirical trends observed in variable-amplitude tests are summarized, including the effects of a high preload, periodical high loads, ground-to-air cycles, and the variables pertaining to program loading, random loading and flight-simulation loading. This also includes results from full-scale fatigue test series. Various theories on fatigue damage accumulation are recapitulated. The significance of these theories for explaining empirical trends as well as for estimating fatigue properties as a design problem is evaluated. For the latter purpose, reference is made to the merits of employing experience from previous designs. Fatigue testing procedures are discussed in relation to various testing purposes. Emphasis is on flight-simulation tests.

4. Pinckney, R.L., and Freemand, R.B., Determination of Physical and Structural Properties of Mixed-Modulus Composite Materials, Vertol Division, The Boeing Company, D210-10196-1, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, June 1971, AD-732-489.

The objective of this program was to determine the physical and structural properties of mixed-modulus composite materials using combinations of graphite and S-glass fibers under static and fatigue loading conditions. The data indicates that the mixed-modulus system of S-glass and graphite is compatible with the structural and failure mode requirements of helicopter rotor blades.

5. Broek, D., Concepts in Fail-Safe Design of Aircraft Structures, March 1971, AD-723-317.

In order to obtain an appraisal of the state of the art of fail-safe design, the author made an inventory of fail-safe design methods applied by various aerospace companies and of research work relevant to the engineering approach of fatigue-crack propagation and residual strength. This memorandum is based on information from discussions with personnel of several companies and research laboratories, with the main emphasis on plane stress and transitional fracture behavior. The memorandum presents a brief description of the general approach to the fail-safe problem, an analysis of several of the existing methods that use this approach, including their shortcomings, and a summary of the data required for a good fail-safe design. A specific approach proposed for the presentation in MIL-HDBK-5 of data pertinent to the fail-safe design concept is evaluated in terms of its applicability to that concept.

6. Freudenthal, A.M., Fatigue Mechanisms; Fatigue Performance and Structural Integrity, Department of Civil Engineering and Engineering Mechanics, Columbia University, NONR-266(91), Office of Naval Research, December 1969, AD-701-415.

The paper recommends an interdisciplinary approach to fatigue based on the close interrelationship between the fatigue performance of the structure and the changes in the microstructure of the metal.

7. Mittenburgs, A.A., Fondriest, F.F., and Grover, H.J., Study and Analysis of Factors Affecting Fatigue Strength of Rotor-Blade-Retention Assemblies, PB-165-244.

A study and evaluation of current practices in the design and evaluation of rotor-blade-retention assemblies was made, followed by an experimental program investigating mainly the effects of geometric variables on the fatigue strength of lug-type steel fittings. The broad objectives of these investigations were to establish criteria applicable to the design of rotor-blade-retention fittings and to determine the optimum geometric relations of the design elements.

8. Schijve, J., Jacobs, F.A., and Tromp, P.J., The Effect of Load Sequence on Fatigue Crack Propagation Under Random Loading and Program Loading, National Aerospace Laboratory NLR, NLR-TR-71014-U, National Aerospace Laboratory NLR, The Netherlands, January 1971.

Crack propagation was studied in 2024-T3 Alcolad sheet specimens under two types of random loading and under program loading with a short period and a long period. In the program tests Lo-Hi, Lo-Hi-Lo and Hi-Lo sequences were employed. The loads were based on a gust spectrum. The crack rates were about the same under random loading and program loading with a short period. Under program loading with a long period, the crack rates were 2.5 times slower on the average, while a significant sequence effect was observed in these tests. Fractographic observations indicated different cracking mechanisms for the random tests and program tests with a short period on the one hand and the program tests with the long period on the other hand.

9. Sandoz, P.L., The Next Careful Steps in Commercial Aircraft Structures N72-13897, July 1973.

The report expounds on benefits to be derived, in future years, from utilization of advanced materials and fabrication processes while maintaining the durability and "fail safeness" of contemporary transports.

10. Nord, C.E., Estimating the Reliability of Fatigue Loaded Rotorcraft Structures, Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut, May 1967.

The purposes of this paper are to point out the severe limitations inherent in estimating high levels of structural reliability; to emphasize needs for information on rotorcraft structural environments and on fatigue properties of structures; to stress the importance of using past experience; and to point out those reliability disciplines, particularly the use of fail-safe concepts, that require greater emphasis and application.

11. Peck, W.B., A Survey of Helicopter Current Practices Relative to Fatigue, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

The fatigue loading of helicopter and VTOL rotor and drive system components is a dominant factor in the evolution of current designs. This paper outlines the nature of the fatigue problem and traces the history of design solutions which have been applied by the helicopter industry. A review is made of variations in current analysis methods for predicting the "safe life" of rotor and drive components. The significance of these variations is discussed in light of the extensive operating experience with turbine-powered military and commercial helicopters.

12. Schauble, J.J., and Maloney, P.F., An Approach to Helicopter Structural Reliability and Fatigue Life, Kaman Aircraft Corporation, Bloomfield, Connecticut.

The general subject of fatigue substantiation of aircraft components is discussed and some of the problem areas are explored. The uncertainties associated with two of the common techniques, fail-safe and safe-life, are reviewed and prepared to a safe-strength approach. A new simulation technique is introduced which relieves some of the uncertainty of the safe-life method. This technique also permits the inclusion and evaluation of factors which can be important in the determination of fatigue life, but are difficult to evaluate by other methods. The application of the technique is illustrated in a numerical example.

13. Reddick, H., McCall, C.D., and Field, D.M., Advanced Technology As Applied to the Design of the HLH Hub, Boeing Vertol Company, Philadelphia, Pennsylvania and U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, May 1973.

A heavy lift helicopter advanced technology component program featuring development and testing of critical components is being conducted. Innovative design approaches, new material systems and advanced technology concepts are being demonstrated for use on the Heavy Lift Helicopter (HLH). During the rotor hub design development, data has been generated to define the characteristics of a reduced size spherical elastomeric flap-lag-pitch bearing, a centering bearing which reacts blade shear forces, and material properties of large $\alpha + \beta$ Ti 6Al-4V forgings. Fail-safe design criteria have been established and are being employed in structural component designs. Fracture mechanics and finite element technologies have been used to size main hub components. To improve damper performance in mixed frequency environments, a frequency selective blade damper is being developed and tested. Data pertaining to each of these developments as well as related items such as reliability, maintainability and weight are also included in this paper.

14. Salkind, M.J., The Twin Beam Composite Rotor Blade, Sikorsky Aircraft Division of United Aircraft Corporation, Stratford, Connecticut, May 1973.

This report documents the design and fabrication of a twin beam rotor blade constructed of advanced fibrous composites. Study concludes that the twin beam rotor blade represents an improvement in this technology with respect to both structural efficiency and reliability and low-cost manufacture.

15. Jensen, H.T., The Evolution of Fail-Safe Concepts Rotorcraft, Sikorsky Aircraft, Division of United Aircraft Corporation, June 1965.

Fail-safe structural design has been the preferred method and an acceptable procedure for designing fixed-wing commercial transports for some time. This has not been true in the rotary-wing field. The reasons for this and the adherence to safe-life requirements with associated design and verification procedures will be discussed. The problems associated with rotary-wing safe-life procedures are presented. Changes are required to existing regulations and procedures which have motivated the safe-life approach so that the structural designer is not constrained to thinking in terms of a service life or a fatigue life as he now is. Practical considerations involved in the design and test verification of both safe-life and fail-safe structures are described.

16. Immen, F.H., Fail-Safe vs. Safe-Life Philosophy in Vertol Design, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

V/STOL aircraft combine the technological features of rigid-wing and rotary-wing aircraft, incorporating many of the structural features of helicopters. A review indicates that structures analysis of conventional aircraft have concentrated on fail-safe design, while helicopter design engineers have generally been content to rely on component safe-life design. The two philosophies are examined and an approach to the rational marriage of helicopter and airplane fatigue technologies is suggested, in order to provide an integrated, economically feasible, statistically rational combination of fail-safe and safe-life methodology for tilt-wing and tilt-rotor V/STOL aircraft.

17. Scarpati, T., Sanford, R., and Powell, R., The Heavy Lift Helicopter Rotor Blade, The Boeing Vertol Company, Philadelphia, Pennsylvania and U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, May 1973.

The Boeing Heavy Lift Helicopter rotor blade is an application of advanced technology encompassing improved airfoil and geometry distribution and composite materials. The selected airfoils, twist distribution and the use of control system pitch damping result in a considerable reduction in rotor system size and weights. The design consists of a fiberglass and titanium spar and provides a fail safety by means of a closed spar delta pressure system. This paper describes the blade's aerodynamic and structural features, fabrication methods and design support tests and discusses the considerations important to its development.

18. Dutton, W.J., Development of the H-53 Elastomeric Rotor Head, Sikorsky Aircraft, Stratford, Connecticut, May 1973.

Under the sponsorship of the Naval Air Systems Command, Sikorsky Aircraft has successfully designed, built, and tested an improved nonlubricated production main rotor head for the model CH-53D helicopter. Interchangeable with the existing oil lubricated rotor head, the new elastomeric rotor head has been tested with both the present production aluminum blade and the new high-performance titanium main rotor blades.

19. Bettino, J., Tracy, R., and Zincone, R., Development of the CH-53D High Performance Titanium Main Rotor Blade, Naval Air Systems Command, Washington, D.C., and Sikorsky Aircraft Division, United Aircraft Corporation, Stratford, Connecticut, May 1973.

An Improved Rotor Blade (IRB) for production H-53 helicopters has been designed, manufactured and tested by the Naval Air Systems Command and Sikorsky Aircraft. The new blade has a continuous titanium spar, fiberglass cover and a Nomex honeycomb core. The titanium spar technology, first developed for the Sikorsky ABC rotor, is the key to the improvements of the rotor. The blade has a 11-1/2% wider chord and torsional rigidity 40% higher than the standard CH-53D aluminum blade with no blade weight increase. The Sikorsky SC 1095 cambered airfoil and a high nonlinear twist has been incorporated. This paper traces the history of the new blade through design, fabrication and test. The impact of the test results on the H-53 performance and blade structural reliability are summarized.

20. Zinberg, H., An Advanced Composite Tailboom for the AH-1G Helicopter, Bell Helicopter Company, Fort Worth, Texas, May 1973.

This paper describes a Bell IR&D program for the design, manufacture, and testing of an advanced composite tailboom for the AH-1G Cobra helicopter. The program was undertaken to gain experience in the design and manufacture of a major primary structure in advanced composite materials. It evaluated several materials and structural configurations, and chose a honeycomb sandwich of Nomex core and Modmor III graphite faces of $[0/\pm45/0]_T$. Two tailbooms were fabricated: one for structural test, one for possible flight test. The former has been tested and found to be about 11 percent stiffer than predicted. It failed at 127 percent of ultimate load, with the failure occurring in tension and emanating from a fastener hole adjacent to a nonstructural door. The location of the failure, the load at which it occurred, and the mode of failure were predictable.

21. Field, D.M., Finney, R.H., and Stratton, W.K., Achieving Fail Safe Design in Rotors, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

Safe-life design, which deals with the predictable, is frequently not adequate to assure a truly safe rotor blade, since failure is usually due to the unpredictable. Fail safety is a design approach which provides for the unpredictable. It requires designs which continue to function with partial failures, incorporate methods of detecting incipient failures, and provide accurate prediction of remaining life at detection. This paper discusses several means of meeting these requirements for safe rotor blades.

22. Thompson, G.H., and Weiss, W.L., Fail-Safety for the H-46 Rotor Blade, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania.

In July 1969, the Vertol Division of The Boeing Company contracted to design a system which would make the H-46 helicopter rotor blade fail safe - Integral Spar Inspection System (ISIS). ISIS is a continuously functioning inspection system, integral to the rotor blade, which will detect spar cracks in the early stages. A ground inspection at each rotor shutdown will provide ample warning of impending blade fracture. The development and qualifications of the system are discussed from the viewpoint of the structures engineer. The analytical methods used to assess the system's ability to detect cracks in the blade spar and provide sufficient warning time before failure are summarized.

23. Immen, F.H., Some Structural Considerations in the Design of the Chinook Helicopter, The Boeing Company, Vertol Division, AD-660-667.

The U.S. Army CH-47A Chinook is a transport helicopter developed by Boeing. Major considerations in the development of its structural integrity are discussed. Included is a description of fatigue analysis techniques which were developed to ensure safe life of its critical components. This technique includes a mission profile, component fatigue strengths, use of top-of-scatter flight loads data, use of Miner's rule of cumulative damage, and evaluation of possible anomalies on fatigue strengths and flight loads.

24. Stratton, W.K., and White, R.S., The Application of Fracture Mechanics to the Fail Safety of Rotor Blades, Vertol Division of The Boeing Company, Philadelphia, Pennsylvania.

The report recognizes that an essential element of establishing fail safety for many components is a means of predicting the time available from detection to failure of the part. This is especially true in a helicopter where many primary structures are monolithic. Of the several approaches evaluated, fracture mechanics was selected as best meeting the simultaneous requirements of accurate predictions, minimization of testing, ability to extrapolate to other designs, and establishment of a "tool" applicable to the design of new, as well as the modification of old, components.

25. Pual, W.F., Development and Evaluation of the Main Rotor Bifilar Absorber, Sikorsky Aircraft Division, United Aircraft Corporation, Stratford, Connecticut, 1969.

Vibration continues to be one of the major technical problems of the helicopter. The requirements of higher combat aircraft availability, lower maintenance costs and a more comfortable commercial operation demand a low vibration helicopter environment. This paper describes the development and reduction to practice of a highly successful vibration reduction concept: the main rotor bifilar vibration absorber.

26. Smith, H.G., and McDermott, J.M., Designing for Crashworthiness and Survivability, Hughes Tool Company, Aircraft Division, Culver City, California, May 1968.

The prospects of intentionally designing helicopter structures to provide maximum practicable crashworthiness and survivability are discussed. The pros and cons of the various approaches and design criteria are compared. The simultaneous interplay of load factors and deformations in comparison with indicated human tolerance to impacts leads to certain near optimum design criteria for crashworthiness. Areas where the specification of high design load factors alone may actually increase the severity of injury for a given crash impact are pointed out. Finally, the implications derived from helicopter crash experience are analyzed and compared with the proposed design criteria for crashworthiness.

27. Sipes, W.A., Metallurgical Investigation of UH-1 Stabilizer Bar Failure, Aero Materials Department, Naval Air Development Center, NADC-MA-7144, Naval Air Systems Command, Department of the Navy, Washington, D.C., September 1971, AD-888-923L.

Failed UH-1 stabilizer bar parts were forwarded to the Naval Air Development Center along with a request from the Naval Air Systems Command that a metallurgical examination be made as to the cause of failure. On the basis of the metallurgical examination of the failed parts, it was concluded that corrosion due to inadequate interior surface protection was the primary cause of failure. This corrosion caused pitting and intergranular penetration of the inner surface of the stabilizer bar assembly in the tube areas. At the critically loaded section of the assembly, inboard of the flash butt weld, cyclic service stresses caused crack growth by the fatigue mode from the multiple sites provided by the corrosion effects. The lateral growth, mutual abuttment, and penetration of these cracks into the tube cross-section produced the final systems to the outer surface.

28. Gewehr, H.W., Final Report on Design and Substantiation of an Increased Service Life Main Rotor Hub for the H-2 Helicopter, Kaman Aerospace Corporation, T-542, Naval Air Systems Command, May 1971, AD-885-544L.

The report documents a static strength stress analysis of the K613036-1 Titanium Hub assembly used on the main rotor assembly of UH-2 helicopters.

29. Fletcher, A.R., Metallurgical Analysis of Failure of Projectile Damaged Rotor Blade P/N AOZR1502, S/N A-1-796 (AISI 4340 Steel), Naval ADR Development Center, Aero Materials Department, NADC-MA-6842, Naval Air Systems Command, Department of the Navy, Washington, D.C., July 1968, AD-840-191.

The crash of a CH-46D helicopter, A/C 152569, occurred as a result of the fracture of the forward rotor blade. Examination of the rotor blade revealed a projectile hole in the joggle area of the undersurface of the spar. Fracture was preceded by the progression of a fatigue crack from either side of the hole through 45% of the spar cross-section.

30. Personnel of the Directorate for Product Assurance Systems Performance Assessment Division, Management Summary Report, OH-58A, U.S. Army Aviation Systems Command, Directorate for Product Assurance, USAAVSCOM Technical Report 73-2, U.S. Army Aviation Systems Command, St. Louis, Missouri, January 1973, AD-756-415.

This report presents the results of an independent assessment of the reliability, availability, and maintainability

attained by the fleet of subject Army aircraft presently deployed. Emphasis is placed on basic areas requiring management coordination. Problem identification in the areas of aircraft operations and maintenance as related to inadequacies of the reliability and maintainability aspects of the equipment is based on the analysis of data available from several sources, including the TAERS/TAMMS system, Army Aircraft Inventory and Flight Status Reports, Crash Fact Messages, Field Service Reports, and Product Quality Inspection Summaries.

31. McNair, W.J., Factors of Safety and Fail Safe Strength Criteria, Federal Aviation Agency, Defense Documentation Center, Defense Supply Agency, November 1966, AD-667-144.

This paper briefly traces the origin and use of the term "fatigue" in civil aviation. Sections of the current Federal Aviation Regulations pertaining to factors of safety and fail safe strength criteria for fixed-wing transport aircraft are briefly reviewed. Emphasis is also focused on the importance of adequate maintenance inspection intervals and procedures for aircraft.

32. Slinozuka, M., and Itagaki, H., On the Reliability of Redundant Structures, Department of Civil Engineering and Engineering Mechanics, Columbia University, AFML-TR-66-158, Air Force Materials Laboratory, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Dayton, Ohio, June 1966, AD-488-308.

General expressions for the probability that a simple redundant structure of material with a statistical yield point can sustain the applied load, even when yielding occurs in some of its members, are obtained. It is pointed out that, in the fail-safe design, the conditional probability of survival of the structure, under the hypothesis that yielding has occurred in at least one of the members, is as significant as the expected life of the damaged structure. Numerical examples indicate that while the structure utilizing the yielding material is appreciably better, from the point of view of fail-safe design, than that using a brittle material, caution should be exercised to apply the notion of fail-safe design to redundant structures against failure due to yielding, since the conditional probability of survival is low.

33. Hooke, F.H., The Fatigue Life of Safe Life and Fail-Safe Structures, A State-of-the-Art Review, Department of Supply, Australian Defense Scientific Service, Aeronautical Research Laboratories, ARL/SM 334, Commonwealth of Australia, June 1971, AD-894-041.

A review is made of the theory and practice of assessment of safe-life and fail-safe structures, and of determinations of safe inspection intervals for fail-safe structures. The problem is one involving statistical variability, and estimates are made of the intrinsic uncertainty in defining, for example, scatter factors, safe lives, or probabilities of failure. Consideration is given to defining the optimum test load history to represent service conditions.

34. Committee on Application of Fracture Prevention Principles to Aircraft, National Materials Advisory Board, NMAB-302, Department of Defense.

The elements of current fracture control plans and associated technologies were reviewed. After reviewing the status, applicability, and potential of the elements and technologies, it was concluded that fracture control plans and development of related technologies not only afford an opportunity to reduce catastrophic failures of aircraft structures and structural maintenance but also can help to quantify many structural material, design, NOE, and maintenance decisions that now are made on a relatively qualitative basis.

35. Veca, A.C., Vibration Effects on Helicopter Reliability and Maintainability, Sikorsky Aircraft Division, United Aircraft Corporation, USAAMRDL Technical Report 73-11, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1973, AD-766-307.

In this study, differences in reliability and maintainability data were examined on two groups of USAF H-3 helicopters with distinctly different vibration characteristics. One H-3 helicopter group was equipped with the rotor-mounted vibration absorber, a device which reduces helicopter vibration induced by the rotor; the second aircraft group did not have the absorber. The aircraft were alike in all other respects. The analyses performed on these data show a significant reduction in the failure rate and direct maintenance for the H-3 helicopter with absorbers and with reduced vibration levels.

36. Levenetz, B., Composite-Material Helicopter Rotor Hubs, Whittaker Corporation, Research and Development Division, USAAMRDL Technical Report 73-14, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, July 1973, AD 771973.

This report describes the development work conducted by the Whittaker Corporation for construction of helicopter rotor hubs from fibrous composite materials. The prototype hubs were designed to be structurally and functionally equivalent to the metallic hub used on the Sikorsky CH-54B helicopter. The design is based on the principle of filament-wound tension loops in combination with laminated shear panels. The report describes the design elements, the structural analysis, the construction methods, and the experimental evaluation of rotor hubs subjected to static as well as cyclic loads. Design and construction problems are discussed, and the potential of the composite hub concept is outlined.

37. Fatigue Life Prediction for Aircraft Structures and Materials, Advisory Group for Aerospace Research and Development, North Atlantic Treaty Organization, AGARD-LS-62, May 1973, AD-762-718.

The evaluation of the fatigue quality of an aircraft involves several steps: (1) determination of the fatigue load environment, (2) response of the aircraft structure, (3) internal load distributions, and (4) estimation of the fatigue properties. The fatigue properties comprise fatigue life, crack propagation and residual strength. The latter two items together with inspection procedures qualify the fail safety. The above aspects are discussed in the paper with reference to the contributions of design efforts, calculations, testing, inspections and fatigue load monitoring.

38. Rummel, K.G., Helicopter Development Reliability Test Requirements Volume I - Study Results, The Boeing Company, Vertol Division, USAAMRDL Technical Report 71-18A, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1971, AD-725-595.

This report covers a study to identify optimum reliability problem identification and demonstration test concepts for helicopter dynamic components, in order to facilitate formulation of cost-effective reliability test programs for future helicopters. Detailed failure mode test technique problem identification capability and cost data are presented from CH-47 helicopter development experience to aid in calculating specific test costs for future development programs. Sample test plans are presented for two helicopters representing size extremes. A plan is outlined for revising selected existing design and test military specifications and supplementing them with additional handbooks and specifications.

39. Jensen, H.T., The Application of Reliability Concepts to Fatigue Loaded Helicopter Structures, 1962, AD-284-471.

The report presents a method of analysis for determining safety-of-flight structural reliability - time relationships for parts whose mode of failure begins with fatigue cracking in primary structure and ends in an uncontrolled landing.

40. Wood, H.A., Fracture Control Procedures for Aircraft Structural Integrity, Air Force Flight Dynamics Laboratory/FBR, AFFDL-TR-71-89, Air Force Flight Dynamics Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Dayton, Ohio, July 1971, AD-731-565.

This report reviews the application of applied fracture mechanics in the design, analysis and qualification of aircraft structural systems. Recent service experiences are cited.

41. Gran, R.J., Oraxio, F.D., Paris, P.C., Irwin, G.R., and Hertzberg, R., Investigation and Analysis Development of Early Life Aircraft Structural Failures, Universal Technology Corporation and Del Research Corporation, AFFDL-TR-70-149, Air Force Flight Dynamics Laboratory (FBR), Air Force Systems Command, Wright-Patterson AFB, Dayton, Ohio, March 1971, AD-884-790.

An investigation and analysis of aircraft structural failures was conducted to assess the condition surrounding early life failures and to initiate improved methods for the structural analysis of such failure problems. The primary objective was to identify critical structural component areas and to define an analysis approach which would consider the useful life of a flawed or damaged structure.

42. Manning, S.D., Lemon, G.H., and Bouton, I., Study of Structural Criteria for Composite Airframes, Volume II - Current Criteria/Selected Rationale Review and Evaluation, General Dynamics, Convair Aerospace Division, AFFDL-TR-73-4, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Dayton, Ohio, April 1973, AD-767-707.

Criteria and design practices currently used for aircraft structures are examined and evaluated for applicability to composite structures. Selected probabilistic or statistical rationales are also reviewed and evaluated for possible applications. From these studies a plan was developed for acquiring understanding and data from which structural criteria and design practices applicable to composite airframes may be written. The basic characteristics of filamentary composites that are unique in comparison to those of metals are defined and explored. Special areas investigated include laminates, joints, and cutouts.

43. Glen, T.O., and Kock, L.C., Structural Composites on Future Fighter Aircraft, McDonnell Aircraft Company, A73-393-71, American Institute of Aeronautics and Astronautics, New York, New York, August 1973.

The study presented herein addresses the use of composite materials in future fighter aircraft. Included are an assessment of current and projected composite materials/structures technologies and trade-off studies through which least-weight design concepts might be identified. In the technology assessment phase of the study, it was determined that composite material allowables will increase during the next 5 years. These increases, coupled with design concepts which utilize composites to their best advantage, will result in substantial decreases in future fighter aircraft size and weight. Further, due to increased usage and automated fabrication methods, the cost of composite raw material and finished structure will decrease substantially. Areas requiring further development effort were identified in the study to include fastening and joining methods, environmental resistance, and damage tolerance. In the design concept trade-off phase of the study, attractive concepts were developed with composite materials for the wing torque box, fuselage, and inlet duct. Substantial weight savings were shown with these concepts. Producibility aspects of the designs were given special consideration, and assembly techniques for the concepts were addressed. It is concluded that composites indeed have the potential for reducing the weight and cost of future fighter aircraft.

44. Rich, M.J., Ridgley, G.F., and Lowry, D.W., Application of Composites to Helicopter Airframe and Landing Gear Structures, Sikorsky Aircraft, Division of United Aircraft Corporation, NASA-CR-112333, National Aeronautics and Space Administration, June 1973.

A preliminary design study has indicated that advanced composite helicopter airframe structures can provide significant system cost advantages in the 1980's. A 7-percent increase in productivity and a 5-percent reduction in life cycle cost are projected.

45. Bryson, L.L., and McCarty, J.E., Analytical and Experimental Investigation of Aircraft Metal Structures Reinforced With Filamentary Composites: Phase III, Major Component Development, Boeing Commercial Airplane Company, NASA-CR-2122, National Aeronautics and Space Administration, Washington, D.C., November 1973.

Analytical and experimental investigations, performed to establish the feasibility of reinforcing metal aircraft structures with advanced filamentary composites, are reported.

Aluminum-boron-epoxy and titanium-boron-epoxy were used in the design and manufacture of three major structural components. The components evaluated were representative of subsonic aircraft fuselage and window belt panels and supersonic aircraft compression panels. Both unidirectional and multidirectional reinforcement concepts were employed. Blade penetration, axial compression, and in-plane shear tests were conducted. Composite reinforced structural components designed to realistic airframe structural criteria demonstrated the potential for significant weight savings while maintaining strength, stability, and damage containment properties for all metal components designed to meet the same criteria.

46. Hoffstedt, D.J., Advanced Geometry, Glass-Fiber-Reinforced Plastic Rotor Blade Test Program, The Boeing Company, Vertol Division, USSAMRDL Technical Report 71-42, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, September 1971, AD-783-203.

This report presents the results of a program to design, tool, fabricate, bench test, and flight test glass-reinforced epoxy rotor blades. The program demonstrated the feasibility of using composite materials for primary members in helicopter rotor blades of nonuniform geometric characteristics. It also proved the capability of achieving sound structure with improved strength-to-weight ratios.

47. Wierenga, B.B., Blake, D.O., Hanson, R.E., and Cook, T.N., Analysis of Army Helicopter Inspection Requirements, RCA/Government and Commercial Systems, Aerospace Systems Division, USAAMRDL Technical Report 72-35, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, September 1972, AD-754-642.

Preventive maintenance scheduled inspection requirements were analyzed to select one inspection concept which could be applied effectively to all typical types of Army helicopters. A computer model was developed for comparison of alternate, practicable inspection schemes. The modeling and engineering evaluations result in the selection of the phased-inspection concept with 100-hour interval and 800-hour cycle times as the recommended inspection system for Army helicopters. This concept provides a high figure of merit based upon reliability and availability considerations and indicates substantial cost advantage over the other concepts. In addition, phased inspection involves less severe disruptions to aircraft operating schedules since each inspection point represents a shorter, more manageable work package than in other concepts.

48. Dotseth, W.D., Survivability Design Guide for U.S. Army Aircraft, Volume I - Small-Arms Ballistic Protection, North American Rockwell Corporation, USAAMRDL Technical Report 71-41A, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, AD 891122L.

An extensive literature and information search was conducted to identify military aircraft small-arms protection enhancement techniques developed during the past ten years. This data was analyzed and used to develop a comprehensive survivability design guide for incorporation of ballistic protection features in U.S. Army aircraft. The design guide is structured for use by aircraft configuration and subsystem design organizations. It provides guidance for overall survivability design considerations and detailed information on specific enhancement techniques.

49. Rich, M.J., Vulnerability and Crashworthiness in the Design of Rotary Wing Vehicle Structures, Sikorsky Aircraft Division, United Aircraft Corporation, 680673, 1968.

The structural design and material usage of rotary-wing vehicle airframe and dynamic components are investigated with regard to reducing vulnerability from small-arms fire and improving crashworthiness.

50. Krupp, W.E., and Hoeppner, D.W., Fracture Mechanics Applications in Materials Selection, Fabrication Sequencing and Inspection, Lockheed California Company, Burbank, California, November 1973.

The activities of the technology assessment groups have culminated in revision of military requirements for aircraft structural integrity, in the form of a new military standard. This revision introduced fracture mechanics, damage tolerance, slow crack growth, and quantitative flaw detection capability concepts into early design, specification, qualification and life-cycle considerations. These aspects are discussed in this paper, which outlines a procedure for using fracture mechanics and damage-tolerant concepts for establishing the safety and reliability of critical aircraft structural components. This paper illustrates the use of fracture mechanics concepts to quantitatively compare the various alternatives involved in design, manufacturing, assembly and quality assurance, and to evaluate the possibility of premature failure for critical components.

51. Impact of Composite Materials on Aerospace Vehicles and Propulsion Systems, North Atlantic Treaty Organization Advisory Group for Aerospace Research and Development, AGARD-CP-112, North Atlantic Treaty Organization, May 1973, AD-762-686.

Data and information are presented on high-strength, high modules reinforcing fibers and organic resin composites fabricated from these fibers. Glass, boron, graphite, various metallic, PRD-49-III and silicon carbide fiber and composite properties are discussed. Combined fiber or hybrid composites containing boron and S-glass, and berylium fibers and S-glass are discussed. The properties of the various forms of asbestos reinforcements are presented along with the mechanical properties of several asbestos reinforced epoxy-resin composites.

52. Wells, H.M., and King, T.T., Air Force Aircraft Structural Integrity Program: Airplane Requirements, Aeronautical Systems Division, ADD-TR-66-57, May 1970.

This report summarizes requirements for the airplane portion of the Aircraft Integrity Program based upon the results of experience and events since the inception of the program in 1958. It supplements the detailed structural specifications for Air Force airplanes and updates Aeronautical Systems Division Technical Report 66-57, dated January 1968. Applicable military specifications are referenced throughout.

53. Rich, M.J., Ridgley, G.F., and Lowry, D.W., Application of Advanced Composites to Helicopter Airframe Structures, Sikorsky Aircraft Division of United Aircraft Corporation, May 1974.

A preliminary design study has been conducted on the applicability of composite materials to helicopter fuselage structure. The CH-53D, a large, high-speed cargo and troop transport helicopter, was used as a baseline design for the study. The construction of the CH-53D airframe, primarily aluminum skin/ stringers/frames, is representative of present generation helicopters. The study indicates that composite materials can be cost-effectively applied to primary helicopter airframe structures in the 1980's.

54. Weiss, W.L., and Zola, J.C., The Application of Fracture Mechanics to the Design of Damage-Tolerant Components for the UTTAS Helicopter, Boeing Vertol Company, May 1974.

The principles of fracture mechanics have been used extensively in the design of damage-tolerant structural components for the YUH-61A, UTTAS helicopter which is being designed, built, and tested by the Boeing Vertol Company under Army contract. The requirements for a high level of structural reliability, considering both combat and noncombat environments, led to establishment of a damage-tolerant criterion. The design of major structural components was often controlled by this criterion, which required specific residual fatigue life and static strength after sustaining initial damage of a given magnitude.

55. Woods, G.W., Rotorcraft Dynamic Component Life Factors, Aero Structures Department, Naval Air Development Center, NADC-ST-6901, Naval Air Systems Command, Department of the Navy, Washington, D.C., October 1969, AD-861-396L.

To provide the Naval Air Systems Command with a reliable method for predicting the structural life of the VTOL/rotorcraft dynamic components, a standarized analytic technique was established by analyzing the prediction-method variations in current contractor techniques. The standardized technique is useful in predicting the fatigue lives of developmental and operational VTOL/rotorcraft components and in evaluating proposals for off-the-shelf designs.

56. Personnel for the Directorate for Product Assurance Systems Performance Assessment Division, Management Summary Report, AH-1G, U.S. Army Aviation Systems Command, Directorate for Product Assurance, 72-28, U.S. Army Aviation Systems Command, St. Louis, Missouri, AD-756-377, July 1972.

This report presents the results of an independent assessment of the reliability, availability, and maintainability attained by the fleet of subject Army aircraft presently employed. Emphasis is placed on basic areas requiring management coordination. Problem identification in the areas of aircraft operations and maintenance as related to inadequacies of the reliability and maintainability aspects of the equipment is based on the analysis of data available from several sources including the TAERS/TAMMS system, Army Aircraft Inventory and Flight Status Reports, Crash Fact Messages, Field Service Reports, and Product Quality Inspection Summaries.

57. Clark, M.W., Krauss, W.K., and Ciccotti, J.M., Identification and Analysis of Army Helicopter Reliability and Maintainability Problems and Deficiencies: Volume IV - Light Observation Helicopters (OH-6, OH-58), USAAMRDL Tech. Rept. 72-11D, American Power Jet Company and Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April, 1972, AD-901-459.

As Volume IV of a four-volume report, this volume discusses a series of reliability and maintainability problems related to Army Light Observation Helicopters (OH-6, OH-58). Volume I details the standard format for the problem presentation and describes the various analysis elements within this format.

58. Maloney, P.F., Clark, F.B., and McIntyre, H.H., Application of Directed Glass Fiber Reinforced Plastic to Helicopter Tail Rotor Assembly, Kaman Aircraft Division, Kaman Corporation, USAAVLABS-TR-68-29, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, June 1968, AD-674-252.

This report documents an engineering program conducted for the purpose of investigating the feasibility of a new rotor concept utilizing a monolithic spar of directed glass fibers supported in an epoxy matrix. This concept uses the anisotropic property of the material to eliminate pitch bearings and thereby reduce maintenance requirements. The program included design, fabrication development, analysis and test phases. The latter two phases were of limited scope since the intent of the program was a basic feasibility investigation. The rotor has satisfactorily completed all phases, including a 25-hour whirl test. It was concluded that rotors of this general configuration are practical, offer significant advantages, and can be fabricated in a production environment.

59. Stone, M., Structural Reliability Through Detail Design and Development Testing, Douglas Aircraft Company, McDonnell Douglas Corporation, Long Beach, California.

This paper describes an approach to design dependable long-life aircraft through the use of proper detail design and structural development testing. In discussing the aspects of structure fatigue, it is necessary to recognize past design practices based on static strength and present requirements on designing for fatigue strength due to advancements made in testing and analysis.

60. Smith, S.H., Porter, T.R., and Engstrom, W.L., Fatigue Crack Propagation Behavior and Residual Strength of Bonded Strap Reinforced, Lamellated and Sandwich Panels, The Boeing Company, Commercial Airplane Group, Renton, Washington.

An exploratory testing program supplemented by linear elastic fracture mechanics was conducted to evaluate the fail-safe capability of bonded structures. The fatigue crack propagation behavior and residual strength of bonded strap reinforced simple, lamellated, and honeycomb sandwich panels were evaluated. Cyclic and static stress intensity factor reductions were the key parameters evaluated through variation of the geometrical, material, and test conditions of the structural panels.

61. Rich, M.J., Vulnerability Considerations in the Design of Rotary Wing Aircraft Structures, Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut, 1969.

The purpose of this program was to examine the aspects of vulnerability in the structural design of rotary-wing aircraft. The first step was to evaluate the requirements and to present, at least on a tentative basis, a rational conservative vulnerability design criterion. The aspects of such a vulnerability criterion must encompass both static and dynamically-loaded structures. Essentially, the requirement is a criterion for the residual strength and life for combat-damaged flight critical structures that affect the survivability of the vehicle. The next step was to review the means available to assess the effects of combat damage, particularly in regard to the use of fracture mechanics methods, for assessing the residual static strength and the rate of damage growth in determining residual life. In reviewing the available data and methods, the information was catalogued in a usable manner with the viewpoint of applying such existing information.

62. Swift, T., and Wang, D.Y., Damage Tolerant Design - Analysis Methods and Test Verification of Fuselage Structure.

The paper describes a method of analysis to determine the effects of stiffening elements on the stress distributions in cracked panels of typical fuselage structure. The analytical results are combined with material fracture toughness constants to determine the residual strength of the damaged structure. A method for analyzing fatigue crack propagation in the stiffened fuselage panels is developed. A fatigue crack propagation and residual strength test program was conducted on a variety of test specimens to experimentally verify the analytical methods.

63. King, T.T., Some Developments in the Air Force Aircraft Structural Integrity Program ASIP.

This paper includes: (1) an outline of the current documentation for the Air Force ASIP; (2) a discussion of the ASIP Flow Diagram which depicts current Air Force structural development requirements, with particular emphasis on reviewing; (3) recent changes in structural testing practices and policies; (4) the actual usage monitoring efforts; and (5) the Air Force Fatigue Certification Program.

64. Clark, M.W., Krauss, W.K., and Ciccotti, J.M., Identification and Analysis of Army Helicopter Reliability and Maintainability Problems and Deficiencies: Volume II - Utility, Attack, and Training Helicopters (UH-1, AH-1, TH-1), USAAMRDL Tech. Rept. 72-11B, American Power Jet Company and Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1972, AD-901-457.

As Volume II of a four-volume report, this volume discusses a series of reliability and maintainability problems related to Army Utility, Attack, and Training Helicopters (UH-1, AH-1, TH-1). Volume I details the standard format for the problem presentation and describes the various analysis elements within this format.

65. Wells, R.R., New Alloys for Advanced Metallic Fighter-Wing Structures, Northrop Corporation, American Institute of Aeronautics and Astronautics, American Society of Mechanical Engineers, Society of Automotive Engineers, April 1974.

This paper summarizes the materials test portion of a program sponsored by the U.S. Air Force Flight Dynamics Laboratory (AFFDL) to develop lightweight/low-cost fighter-wing structures. Fatigue-crack propagation-rate curves are presented which compare aluminum alloys 7475-T7651, 7050-T73651, 7050-T7654 plates and 7050-T736 forgings, and titanium alloys Ti-6-4 βMA. Ti-6-2-1-1 A plates, Ti-6-4 A and STA castings, and Ti-6-22-22 STA forgings. Standard mechanical properties are also reported. For most aircraft applications, these new aluminum alloys appear better than 7075. Except for Ti-6-2-1-1 A plate and Ti-6-4 STA castings, the titanium alloys tested are better than conventional, wrought Ti-6-4 and Ti-6-6-2.

66. Cook, T.N., Young, R.L., and Starses, F.E., Maintainability Analysis of Major Helicopter Components, Kaman Aerospace Corporation, USAAMRDL-TR-73-43, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1973, AD-769-941.

This report examines the factors responsible for the high man-hour cost of maintaining current-inventory Army helicopters. Major components of six helicopter models were analyzed to identify the significant man-hour consumers on each aircraft. Causes for maintenance were established in terms of failure modes, maintenance frequency, and average repair time. Major component replacement tasks were structured in terms of specific time elements, and important factors affecting maintenance task performance were established.

67. Phillips, N.S., Carr, R.W., and Scranton, R.S., Crashworthy Landing Gear Study, Beta Industries, Inc., USAAMRDL-TR-72-61, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, April 1973, AD-765-489.

The report documents a two-fold effort: (1) to develop rotary-wing landing gear concepts and criteria which, when applied, would lessen the magnitude of crash forces transferred to occupiable areas of helicopters involved in severe yet survivable accidents; and (2) to use the concepts and criteria to design, fabricate, and test an experimental prototype skid landing gear system.

68. Faiz, R.J., A Design Analysis of a CH-54B Main Rotor Hub Fabricated From Composite Materials, Sikorsky Aircraft Division, United Aircraft Corporation, USAAMRDL Technical Report 73-49, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, October 1973, AD 771966.

A flightworthy CH-54 main rotor hub has been designed using the composite tension strap concept developed by the Whittaker Corporation. The study showed that the basic tension strap hub concept could not adequately react out-of-plane hub moments and rotor torque. The redesign resolved these problems by adding external vertical shear webs, tapering and deepening the hub arms, and adding clamped ring elements in the central hub regime. The weight of the redesigned hub was equal to the original concept weight, but 300 pounds heavier than the CH-54 production hub. The rotor hub is designed to be flightworthy. Fabrication, fatigue tests, and whirl and flight tests can now proceed in discrete steps to confirm this.

69. Fitch, G.E., Application of Fracture Mechanics to Aircraft Structure, North American Rockwell, B-1 Division, Los Angeles, California.

The report documents the development of a criterion for design and analysis of critical airframe structures. The criterion emphasizes subcritical flaw growth from an initial size to critical crack length defined at limit load. With empirically determined fracture properties for each structural material, analytical procedures based on the stress intensity factor concept of linear fracture mechanics are adequate to calculate the crack-growth life of structures with preexisting flaws. Application of the procedures has resulted in weight impacts to critical structural components of the B-1 airframe. The fracture mechanics requirements, in general, closely compete with and often supplant the crack-initiation fatigue requirements for design impact. The most sensitive parameter in the process appears to be the initial flaw size that is to be determined by nondestructive inspection techniques.

70. Hardrath, H.F., Fatigue and Fracture Mechanics, NASA Langley Research Center, April 1970.

This presentation is organized according to the steps that must be taken to design an efficient aircraft structure to operate with minimum danger of fatigue failures in a real environment for some specified life. The first of these steps (preliminary design) is concerned with establishing design requirements and satisfying the static strength criteria that are chosen. The second step is a fatigue analysis with its many component parts. The third step includes fracture and fatigue crack propagation analyses. In each step many tests are required to check the adequacy of the design, and periodic inspections are required during service to identify and correct damage before catastrophic failure occurs. The frequency and sensitivity with which these inspections must be performed are discussed.

71. Hardrath, H.F., Structural Integrity in Aircraft, January 1973.

The paper reviews briefly the current design philosophies for achieving long, efficient, and reliable service in aircraft structures. The strengths and weaknesses of these design philosophies and their demonstrated records of success are discussed. The state of the art has not been developed to the point where designing can be done without major test inspection and maintenance programs. A broad program of research is proposed through which a viable computerized design scheme will be provided during the next decade.

72. Hardrath, H.F., Fracture Mechanics, NASA Langley Research Center, January 1974.

This paper provides a brief historical sketch of the fracture mechanics discipline and a cursory summary of the current analytical procedures. An attempt is made to assess the current status of the discipline, to suggest some engineering applications, and to recommend directions for future study.

73. Degnan, W.G., Dripchak, P.D., and Matusovich, C.J., Fatigue Crack Propagation in Aircraft Materials, Sikorsky Aircraft Division of United Aircraft Corp., USAAVLABS-TR-66-9, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, March 1966.

The influence of metallurgical, chemical, and geometric variables on fatigue crack propagation rates was investigated in alloys of aluminum, magnesium, steel, and titanium. Some limited fatigue crack propagation was done in laminated plastics. A possible correlation between fatigue crack propagation, fracture toughness, and tensile strength was also investigated.

74. Rich, M.J., and Linzell, L.E., "Damaged" Static and Fatigue Stress Analysis of VTOL Structures, Sikorsky Aircraft, February 1969.

An engineering stress analysis method is developed for determining the residual strength and fatigue crack life of "damaged" or initially cracked structures. The analysis is divided into two different techniques: one for redundant and the other for the nonredundant structure types. Two problems are encountered for the nonredundant structures. These are the static residual strength and the number of cycles to failure of the cracked member. The static residual strength and crack propagation are determined using the fracture toughness technique. An analytical procedure is developed for determining the crack propagation of a member subjected to an arbitrary stress distribution. The analytical procedure is applied to VTOL components and is shown to have good correlation with controlled experimental test results.

75. Rich, M.J., Crack Propagation in Helicopter Rotor Blades, Sikorsky Aircraft, Division of United Aircraft Corp., 1971.

Design criteria are presented for the residual strength and life of fatigue loaded helicopter structures. The crack propagation rate methods and data are reviewed, and a bilinear semilog method is shown to be most accurate for predicting residual life. The methods developed are compared with full-scale rotor blade fatigue data. The good correlation with test data demonstrates the value of fracture mechanics analysis for fail-safe design.

76. Rich, M.J., and Welge, R.T., Design, Analysis and Test of a Boron/Epoxy Reinforced Airframe, Sikorsky/United Aircraft Corporation, American Institute of Aeronautics and Astronautics, American Society of Mechanical Engineers, Society of Automotive Engineers, April 1972.

The airframe of a large helicopter generally requires additional stiffening for dynamic tuning to prevent resonance with the rotor vibratory forces. Investigations showed that aluminum stringers reinforced with boron/epoxy offered substantial weight saving for the CH-54B Skycrane helicopter to achieve the required airframe stiffness. As a result, a program has been conducted under a NASA contract to design, test, and evaluate the static and fatigue strength characteristics of the composite reinforcement.

77. Rich, M.J., Israel, M.H., Kenigsberg, I.J., and Cook, P.P., Power Spectral Density analysis of V/STOL Aircraft Structures, Sikorsky Aircraft Division, United Aircraft Corporation, May 1968.

The Power Spectral Density method of random loads analysis is presented for utilization in the structural analysis of V/STOL aircraft.

78. Berrisford, R.S., Structures Technology and the Helicopter, Structures Technical Area, Technology Applications Division, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia.

The report summarizes the state of the art in helicopter structures technology.

79. Wood, H.A., Fracture Control Procedures for Aircraft Structural Integrity, Air Force Flight Dynamics Laboratory, U.S. Air Force, NASA SP-309.

This report reviews the application of applied fracture mechanics in the design, analysis, and qualification of aircraft structural systems. Recent service experiences are cited.

Current trends in high-strength materials application are reviewed with particular emphasis on the manner in which fracture toughness and structural efficiency may affect the material selection process. General fracture control procedures are reviewed in depth with specific reference to the impact of inspectability, structural arrangement, and material on proposed analysis requirements for safe crack growth. The relative impact on allowable design stress is indicated by example. Design criteria, material, and analysis requirements for implementation of fracture control procedures are reviewed together with limitations in current available data techniques. A summary of items which require further study and attention is presented.

80. Swatton, S., Study of Advanced Structural Concepts for Fuselage, Boeing Vertol Company, A Division of The Boeing Company, USAAMRDL-TR-73-69, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, October 1973.

This report presents the results of a study conducted to develop advanced structural concepts and the application of fiber-reinforced composite materials for the Cobra AH-1G helicopter tail section. The study results recommend the Monocoque Sandwich Clamshell as the optimum design concept.

## APPENDIX C
## RESULTS OF QUESTIONNAIRE AND ORGANIZATIONAL INTERVIEWS

The results of the questionnaires completed by organizations concerned with fail-safe/safe-life design criteria are presented below. Notes taken during the organizational meetings were consulted in assembling this information.

Many of the questions are answered as either yes or no. This reflects the opinion of a majority of the organizations surveyed. Questions preceded by alphanumerical indices were directed at specific types of organizations: A for aircraft manufacturers, and G for Government organizations. Questions are grouped according to structural category.

### FUSELAGE

1. Are there critical components or assemblies on your fuselage structure where it is impractical to utilize fail-safe design methods?

   YES __*__ NO _____

   NOTE: Cost, weight, and inspections problems were cited as reasons in most instances.

2. Should fail-safe objectives be incorporated in design criteria for the fuselage to preclude catastrophic accidents caused by maintenance errors?

   YES __*__ NO _____

3.

(A) For fuselage components or assemblies constructed of composite materials do you employ specialized

| | YES | NO |
|---|---|---|
| Inspection procedures | * | |
| Maintenance procedures | * | |
| Quality control procedures | * | |

4. Do you consider condition indicators for fuselage components to be vital to your structural reliability approach?

YES _*_ NO ___

5.

(A) Indicate which of the following specifications you use in designing fail-safe fuselage structures.

| | | |
|---|---|---|
| a) | MIL-S-8698 | * |
| b) | AR-56 (Navy) | * |
| c) | FAR Part 27 (Normal) | |
| d) | FAR Part 29 (Transport) | |
| e) | Company specifications | |
| f) | Other | |

6. Are current fail-safe regulations and specifications for fuselage design restrictive as regards usage of

| | | YES | NO |
|---|---|---|---|
| a) | Condition indicators | | * |
| b) | Redundant structure | | * |
| c) | Backup structure | | * |
| d) | Control fracture structure | | * |
| e) | Monolithic structure | | |
| f) | Monocoque structure | | * |
| g) | Composite structure | | * |
| h) | Built-up structure | | * |
| i) | Sandwich structure | | * |
| j) | Semimonocoque structure | | * |

NOTE: Responses to question 6. e) were equally divided between yes and no.

7. Do your current inspection techniques for fuselage components provide sufficient warning of possible failure due to ballistic impact?

YES _____ NO _____

NOTE: Responses to question 7 were equally divided between yes and no. Rotary-wing manufacturers for the most just answered yes.

8.

(A) Do you presently employ condition indicators for fuselage components constructed from composite materials?

YES _____ NO __*__

9. Do your current designs for fuselage structure include on-board condition indicators of critical components subject to ballistic damage?

YES _____ NO __*__

10. Order, from one (1) to seven (7), the relative importance of the following factors on fuselage component failure as indicated by your failure and accident investigations.

| | | |
|---|---|---|
| a) | Ballistic impact | 7 |
| b) | Maintenance | |
| c) | Material | |
| d) | Quality control | 2 |
| e) | Residual strength | |
| f) | Structural type | |
| g) | Operational loading | 1 |

NOTE: Responses to questions 10-b), c), e), and f) were equally distributed between maintenance, material, residual strength, and structural type.

11. Indicate which of the following structure types you favor for fuselage construction.

| | | |
|---|---|---|
| a) | Backup structure | 5 |
| b) | Control fracture structure | 2 |
| c) | Monolithic structure | 5 |
| d) | Redundant structure | 1 |
| e) | Composite structure | 4 |
| f) | Built-up structure | 1 |
| g) | Sandwich structure | 3 |
| h) | Monocoque structure | 6 |
| i) | Semimonocoque structure | 2 |

NOTE: The numbering sequence in the above question indicates structure preference. Repeated numerals indicate ties.

12. Would your strength vs. cost and strength vs. weight trade-offs for a specific fuselage component be greatly affected by the knowledge that a reliable condition indicator was monitoring the component?

YES ____ NO __*__

13. Does minimizing the number of parts employed in fuselage assemblies, in general,

| | | YES | NO |
|---|---|---|---|
| a) | Increase stress allowables | | * |
| b) | Increase cost | | * |
| c) | Increase residual strength | | * |
| d) | Limit design methods to stop crack propagation | * | |
| e) | Decrease weight | * | |
| f) | Improve structural reliability | | * |
| g) | Enhance inspection | * | |
| h) | Favor the usage of composite materials | * | |
| i) | Improve maintenance | * | |

14. Do you believe that fail-safe concepts need to be applied to the fuselage structure?

YES __*__ NO ____

15.

(A) Have you employed condition indicators on your fuselage to assess crack propagation rates?

YES ____ NO __*__

16. Order, from one (1) to three (3), how you favor the following types of fuselage structure as regards each of the four categories.

| STRUCTURE | (a) FAIL-SAFE DESIGN | (b) COST | (c) WEIGHT | (d) RESIDUAL STRENGTH |
|---|---|---|---|---|
| Backup | 3 | 3 | 3 | 1 |
| Redundant | 1 | 1 | 2 | 2 |
| Control fracture | 2 | 2 | 1 | 3 |

17. Considering inspection of fuselage as a distinct procedure, do you favor

| | | YES | NO |
|---|---|---|---|
| a) | Regular inspections | * | |
| b) | Random inspections | | * |
| c) | Detailed inspections | * | |

18. Order, from one (1) to four (4), how you favor the following techniques for discovering possible fuselage failures.

| | | |
|---|---|---|
| a) | Condition indicators | 3 |
| b) | Inspection | 1 |
| c) | Maintenance | 2 |
| d) | Quality control | 4 |

19. In your fuselage designs, have you employed composite materials to

| | | YES | NO |
|---|---|---|---|
| a) | Fabricate fracture structure | | * |
| b) | Provide ballistic fail safety | | |
| c) | Fabricate backup structure | | * |
| d) | Retard crack propagation | | |
| e) | Fabricate redundant structure | | * |
| f) | Reduce weight | * | |

NOTE: Responses were equally divided on questions 19-b) and 19-d).

20.

g) In determining crack severity in fuselage components, is length more important than propagation rate?

YES ____ NO *

## LANDING GEAR

1.

(A) Indicate which of the following specifications you use in designing fail-safe landing gear structures.

| | | |
|---|---|---|
| a) | MIL-S-8698 | * |
| b) | AR-56 (Navy) | * |
| c) | FAR Part 27 (Normal) | |
| d) | FAR Part 29 (Transport) | |
| e) | Company specifications | |
| f) | Other | |

2. Are current fail-safe regulations and specifications for landing gear design restrictive as regards usage of

| | | YES | NO |
|---|---|---|---|
| a) | Condition indicators | | * |
| b) | Redundant structure | | * |
| c) | Backup structure | | * |
| d) | Control fracture structure | | * |

3. Do your current inspection techniques for landing gear components provide sufficient warning of possible failure due to ballistic impact?

YES _____ NO _____

NOTE: Responses were about equally divided.

4. In determining crack severity in landing gear components, do you place more importance on length than propagation rate?

YES _____ NO __*__

5. Do your current designs for landing gear structure include on-board condition indicators of critical components subject to ballistic damage?

YES _____ NO __*__

6. Order, from one (1) to seven (7), the relative importance of the following factors on landing gear component failure as indicated by your failure and accident investigations.

| | | |
|---|---|---|
| a) | Ballistic impact | 7 |
| b) | Maintenance | 4 |
| c) | Material | 2 |
| d) | Quality control | 3 |
| e) | Residual strength | 6 |
| f) | Structural type | 5 |
| g) | Operational loading | 1 |

7. Indicate which of the following structure types you have used for landing gear construction.

| | | |
|---|---|---|
| a) | Backup structure | * |
| b) | Control fracture structure | * |
| c) | Monolithic structure | * |
| d) | Redundant structure | * |
| e) | Composite structure | * |
| f) | Built-up structure | * |
| g) | Sandwich structure | |

8. Would your strength vs. cost and strength vs. weight trade-offs for a specific landing gear component be greatly affected by the knowledge that a reliable condition indicator was monitoring the component?

YES ______ NO __*__

9. Do you believe that fail-safe concepts need to be applied to the landing gear structure?

YES ______ NO ______

NOTE: Responses were about equally divided.

10. Are there critical components or assemblies on your landing gear structure where it is impractical to utilize fail-safe design methods?

YES __*__ NO ______

11. Should fail-safe objectives be incorporated in design criteria for landing gear to preclude catastrophic accidents caused by maintenance errors?

YES __*__ NO ______

NOTE: Prevailing methods are strictly inspection oriented.

12. Do you consider condition indicators for landing gear components to be vital to your structural reliability approach?

YES ____ NO __*__

13. Does minimizing the number of parts employed in landing gear assemblies, in general,

| | | YES | NO |
|---|---|---|---|
| a) | Increase stress allowables | | * |
| b) | Increase cost | | * |
| c) | Increase residual strength | | * |
| d) | Limit design methods to stop crack propagation | * | |
| e) | Decrease weight | * | |
| f) | Improve structural reliability | * | |
| g) | Favor the usage of composite materials | | * |
| h) | Improve maintenance | * | |

14. Order, from one (1) to three (3), how you favor the following types of landing gear structure as regards each of the four categories.

| STRUCTURE | (a) FAIL-SAFE DESIGN | (b) COST | (c) WEIGHT | (d) RESIDUAL STRENGTH |
|---|---|---|---|---|
| Backup | 3 | 2 | 3 | 2 |
| Redundant | 2 | 3 | 2 | 1 |
| Control fracture | 1 | 1 | 1 | 3 |

15. Considering inspection of landing gear as a distinct procedure, do you favor

| | | YES | NO |
|---|---|---|---|
| a) | Regular inspections | * | |
| b) | Random inspections | | * |
| c) | Detailed inspections | * | |

16. Order, from one (1) to four (4), how you favor the following techniques for discovering possible landing gear failures.

| | | |
|---|---|---|
| a) | Condition indicators | 4 |
| b) | Inspection | 1 |
| c) | Maintenance | 2 |
| d) | Quality control | 3 |

TESTING

1.

(A) Do you conduct damage-tolerant tests during the

| | | YES | NO |
|---|---|---|---|
| a) | Construction and development stage | * | |
| b) | Prototype stage | * | |
| c) | Production and service stage | * | |

2. Are current fail-safe regulations and specifications restrictive as regards testing of critical components?

YES ____ NO * 

3.

(A) Do you employ specialized testing techniques during

| | | YES | NO |
|---|---|---|---|
| a) | Inspection | * | |
| b) | Maintenance | * | |
| c) | Quality control | * | |

4. Should fail-safe testing objectives be incorporated in design criteria to preclude catastrophic accidents due to

| | | YES | NO |
|---|---|---|---|
| a) | Maintenance errors | * | |
| b) | Inspection errors | * | |
| c) | Quality control errors | * | |

5.

(A) Do you conduct, during the design stage, comparative tests for

| | YES | NO |
|---|---|---|
| a) Notched specimen fatigue behavior | * | |
| b) Crack propagation | * | |
| c) Residual static strength | * | |
| d) Fracture toughness | * | |
| e) Stress corrosion susceptibility | * | |

6.

(A) Do you have specialized testing techniques for components constructed of composite materials?

YES __*__ NO _____

7. Order, from one (1) to four (4), the degree of difficulty in testing, in general, of the following structure types as regards residual dynamic strength.

| | |
|---|---|
| a) Backup structure | 2 |
| b) Control fracture structure | 3 |
| c) Redundant structure | 1 |
| d) Monolithic structure | 4 |

8. Are there critical components or assemblies on your aircraft that are impractical to conduct fatigue testing on?

YES _____ NO __*__

9. Does the incorporation of fail-safe features reduce the requirement for flight and ground tests?

YES _____ NO __*__

10. Do you believe that reliability testing is strictly a fail-safe objective?

YES _____ NO __*__

11.

(A) Do you employ testing procedures to demonstrate the fail safety of a design?

YES __*__ NO _____

12. Do you have established procedures as regards the following types of testing?

| | | YES | NO |
|---|---|---|---|
| a) | Strain survey testing | * | |
| b) | Constant amplitude bench testing | * | |
| c) | Whirl testing (tower) | * | |
| d) | Operational load surveys | * | |
| e) | S-N testing | * | |
| f) | Load spectrum testing | * | |

13. Does minimizing the number of parts employed in your aircraft fail-safe assemblies, in general, reduce your testing requirements?

YES _____ NO __*__

DYNAMIC COMPONENTS

1. Do your current inspection techniques for dynamic components provide sufficient warning of possible failure due to ballistic impact?

YES _____ NO __*__

2. In determining crack severity in dynamic components, do you place more importance on length than propagation rate?

YES _____ NO __*__

3. Do you presently employ condition indicators for dynamic components constructed from composite materials?

YES __*__ NO _____

NOTE: Condition indicators discussed included chip detectors, pressure/vacuum devices, and various mechanical actuation devices. See questions 4, 7, and 11.

4.

(A) Do your current designs for dynamic components include on-board condition indicators of critical components subject to ballistic damage?

YES __*__ NO _____

5. Order, from one (1) to seven (7), the relative importance of the following factors on dynamic component failure as indicated by your failure and accident investigations.

| | | |
|---|---|---|
| a) | Ballistic impact | 7 |
| b) | Maintenance | 3 |
| c) | Material | 2 |
| d) | Quality control | 1 |
| e) | Residual strength | 4 |
| f) | Structural type | 6 |
| g) | Operational loading | 5 |

6.

(A) Indicate which of the following structure types you have used for dynamic component construction

| | | |
|---|---|---|
| a) | Backup structure | * |
| b) | Control fracture structure | * |
| c) | Monolithic structure | * |
| d) | Redundant structure | * |
| e) | Composite structure | * |
| f) | Built-up structure | * |

7.

(A) Have you employed condition indicators on your dynamic components to assess crack propagation rates?

YES _____ NO __*__

8. Are there critical dynamic components where it is impractical to utilize fail-safe design methods?

YES __*__ NO _____

9. Should fail-safe objectives be incorporated in design criteria for dynamic components to preclude catastrophic accidents caused by maintenance errors?

YES __*__ NO _____

10.

(A) For dynamic components or assemblies constructed of composite materials do you employ specialized

| | YES | NO |
|---|---|---|
| a) Inspection procedures | * | |
| b) Maintenance procedures | * | |
| c) Quality control procedures | * | |

11. Do you consider condition indicators for dynamic components to be vital to your structural reliability approach?

YES ____ NO  *

12.

(A) Does minimizing the number of parts employed in dynamic component assemblies, in general,

| | YES | NO |
|---|---|---|
| a) Increase stress allowables | | * |
| b) Increase cost | | * |
| c) Increase residual strength | | * |
| d) Limit design methods to stop crack propagation | * | |
| e) Decrease weight | | |
| f) Improve structural reliability | | |
| g) Enhance inspection | * | |
| h) Favor the usage of composite materials | * | |
| i) Improve maintenance | * | |

NOTE: Responses on 12e and f were equally divided.

13.

(A) Order, from one (1) to three (3), how you favor the following types of dynamic component structure as regards each of the four categories.

| STRUCTURE | (a) FAIL-SAFE DESIGN | (b) COST | (c) WEIGHT | (d) RESIDUAL STRENGTH |
|---|---|---|---|---|
| Backup | | 3 | 3 | 1 |
| Redundant | 2 | | 2 | 2 |
| Control fracture | | | 1 | 3 |

NOTE: Responses on 13-a) and b) were equally divided.

14. Considering inspection of dynamic components as a distinct procedure do you favor

| | YES | NO |
|---|---|---|
| a) Regular inspections | * | |
| b) Random inspections | | * |
| c) Detailed inspections | * | |

15. Order, from one (1) to four (4), how you favor the following techniques for discovering possible dynamic component failures.

| | |
|---|---|
| a) Condition indicators | 4 |
| b) Inspection | 1 |
| c) Maintenance | 2 |
| d) Quality control | 3 |

16.

(A) In your dynamic component design, have you employed composite materials to

| | YES | NO |
|---|---|---|
| a) Fabricate control fracture structure | | |
| b) Provide ballistic fail safety | | |
| c) Fabricate backup structure | | * |
| d) Retard crack propagation | * | |
| e) Fabricate redundant structure | | |
| f) Reduce weight | * | |

NOTE: Responses on 16-a), b), and e) were either equally divided or nonexistent.

17.

(A) Indicate which of the following specifications you use in designing fail-safe dynamic components

| | |
|---|---|
| a) MIL-S-8698 | * |
| b) AR-56 (Navy) | * |
| c) FAR Part 27 (normal) | * |
| d) FAR Part 29 (transport) | * |
| e) Compony specifications | * |
| f) Other | * |

18.

(A) Is the maintenance of dynamic components, in general, hampered by condition indicating devices?

YES ____ NO * ____

19.

(A) Are current fail-safe regulations and specifications for dynamic component design restrictive as regards usage of

| | | YES | NO |
|---|---|---|---|
| a) | Condition indicators | | * |
| b) | Redundant structure | | * |
| c) | Backup structure | | * |
| d) | Control fracture structure | | * |
| e) | Monolithic structure | | * |
| f) | Composite structure | | * |
| g) | Built-up structure | | * |

GENERAL

1. Indicate which of the following structural areas presently employ on-board condition indicators.

| | | |
|---|---|---|
| a) | Main rotor | 4 |
| b) | Tail rotor | 2 |
| c) | Fuselage | 1 |
| d) | Transmission | 3 |
| e) | Landing gear | 5 |

2. Order, from one (1) to five (5), where cracked parts occur most often in your aircraft.

| | | |
|---|---|---|
| a) | Main rotor | * |
| b) | Tail rotor | * |
| c) | Fuselage | |
| d) | Transmission | * |
| e) | Landing gear | |

3. Are there current regulations and specifications restricting the full utilization of existing fail-safe concepts?

YES ____ NO __*__

4. Do you maintain lists of critical components for each aircraft type?

YES __*__ NO ____

NOTE: Organizations, in general, indicated extensive records on failure frequency, inspection intervals, damaged parts, materials used, quality control, and operational loadings in addition to lists of critical components. Several organizations voiced concern as to quality of record keeping.

5. The following list of subjects can be associated with the fail-safe and/or safe-life approaches. Indicate how you associate the subjects.

| | SUBJECT | FAIL-SAFE | SAFE-LIFE |
|---|---|---|---|
| a) | Crack propagation testing | * | * |
| b) | Residual static strength | * | * |
| c) | Fracture toughness | * | * |
| d) | Stress corrosion | * | * |
| e) | Ballistic impact | * | |
| f) | Maintenance | * | * |
| g) | Quality control | * | * |
| h) | Flight loads surveys | * | * |
| i) | Backup structure | * | |
| j) | Residual dynamic strength | * | * |
| k) | Control fracture structure | * | |
| l) | Composite material | * | * |
| m) | Redundant structure | * | |
| n) | Fracture mechanics | * | * |
| o) | Condition indicators | * | * |
| p) | Parts minimization | | * |
| q) | Accident investigations | * | * |
| r) | Inspection | * | * |
| s) | S-N testing | * | * |
| t) | Structural reliability | * | * |

## LIST OF SYMBOLS

| | |
|---|---|
| $a$ | Crack size, length or depth (in.) |
| $a_{cr}$ | Critical crack size (in.) |
| $a_i$ | Initial crack size (in.) |
| $\Delta a$ | Change in crack size (in.) |
| $E$ | Modulus of elasticity (psi) |
| $K$ | Factor for multiplying calculated fatigue life to obtain safe service life |
| $K_{Ic}$ | Plane strain fracture toughness (in.$^{-3/2}$) |
| $K_{MAX}$ | Maximum stress intensity factor |
| $K_{MIN}$ | Minimum stress intensity factor |
| $K_T$ | Stress intensity factor |
| $\Delta K$ | $K_{MAX}-K_{MIN}$ |
| $N$ | Number of load cycles |
| $T_i$ | Reference time at which crack first becomes detectable |
| $T_{cr}$ | Reference time at which crack reaches critical length |
| $\sigma$ | Stress or standard deviation (psi) |
| $\sigma_A$ | Alternating stress (psi) |
| $\frac{da}{dN}$ | Fatigue crack growth (in./cycle) |



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