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FRACTURE CONTROL PROCEDURES FOR AIRCRAFT STRUCTURAL INTEGRITY

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SUMMARY

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.

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"Fracture Mechanics has, in fact, been a boon to the metal producing industry; it has made the finite crack in a structure reputable and even fashionable." (Quoted from A. M. Freudenthal, Miami Beach, Florida, December 1969.)

INTRODUCTION

Primary aircraft structural components generally contain flaws or defects of variable shape, orientation, and criticality which are either inherent in the basic material or are introduced during the fabrication or assembly processes.

From an industry survey (ref. 1) it was concluded that the majority of cracks found in aircraft structures were initiated from tool marks, manufacturing defects, and the like. When not detected, these flaws experience the combined 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. The final fracture process is most often sudden, unexpected, and almost totally devoid of gross plastic deformation or yielding. While this "brittlelike" behavior is most spectacular in the so-called high-strength alloys, it is seen to occur to some degree in most of the commonly used aircraft structural materials.

Recent cases of catastrophic failure in primary structure of first-line aircraft have emphasized the need for a "fresh" new look at the structural integrity process currently used to design and qualify structural systems. Under such an improved process, fracture control would insure the reduction in the probability of catastrophic failure due to the presence of undetected flaws and cracks. This assurance can best be achieved by the intelligent material selection based on fracture as well as common strength considerations and by assuming the existence of flaws in "new" structures and accounting for their probable growth during service.

Linear elastic fracture mechanics analysis and testing techniques have reached the state of development where they may be used with a moderate level of confidence to assess the degree of flaw criticality, to predict the extent of subcritical flaw growth prior to fracture, and to determine the resultant failure modes (ref. 2). Much of the basic groundwork for the current application of linear elastic fracture mechanics to "real" structures can be attributed to the investigation associated with fracture control of metallic pressure vessels for space applications (refs. 3 and 4). While attempts to translate this technology to aircraft usage have been moderately successful, limitations must be recognized which are due to the complex spectrum of loads, temperatures, and chemically aggressive agents that comprise the aircraft environment.

Fail-safe procedures in aircraft have resulted from civil requirements and from independent regulation within the particular airframe company. These efforts have been beneficial on many Air Force aircraft.

Application of fracture mechanics within the Air Force has been almost exclusively "after the fact" to determine remaining safe life with cracks, residual strength, and safe inspection intervals for older systems in which flaws have developed and progressed to near-critical dimensions. Some examples of service application in which the Air Force Flight Dynamics Laboratory (AFFDL) actively participated are summarized in table I (see refs. 5, 6, 7, 8, 9, and 10). In practically all cases, however, attempts to formulate reliable solutions were hampered by the lack of an adequate material-environmental data base and deficiencies in analysis techniques, particularly those techniques which must account for load interaction and environmental effects. One purpose of this paper is to review those areas of application where deficiencies in the technology exist and to offer suggestions for alleviating these deficiencies.

Under the F-111 Recovery Program (ref. 9), basic fracture mechanics data are currently being amassed for D6ac steel by the contractor and several laboratories. (See refs. 8, 11, and 12.)

Specific criteria, guidelines, or requirements for considering fracture mechanics principles in the design and procurement cycle for Air Force aircraft have not existed in the past. Only recently have requirements been levied for new systems. It is too early to assess their impact. In the proposed revisions to the Air Force Airplane Structural Integrity Program (ASIP) which is given in reference 13, damage tolerance considerations are outlined. These changes are currently being reviewed prior to being formally incorporated.

There exists a natural unwillingness amongst many to accept the "preexistent flaw" concept in aircraft design because of the weight penalties normally associated with damage-resistant structures. There are those who cite system performance degradation and the time and cost of implementing fracture requirements as deterrents. The imposition of arbitrary fracture requirements should be done cautiously under current state-of-the-art limitations in analysis methods and testing techniques are resolved and material-environmental behavior is better understood.

In this paper, recent structural material utilization cases are summarized to indicate those problems associated with the use of high-strength material. General fracture control procedures are reviewed with specific reference to the impact of safe crack growth and remaining strength requirements on system design. Examples are cited, including recent laboratory efforts in the analysis of crack growth under variable-amplitude spectrum loading. Limitations in basic design criteria, material data, and analysis are reviewed.

SYMBOLS

a	crack size, length or depth, inches
a_{cr}	critical crack size, inches
a_p	proof-test crack size, inches
Δa	change in crack size, inches
B,t	thickness, inches
c	one-half surface crack length, inches

E	modulus of elasticity, ksi
f	frequency of test load application, cycle/minute
K	stress intensity factor, ksi-$\sqrt{\text{in.}}$
K_c	critical stress intensity factor, ksi-$\sqrt{\text{in.}}$
K_{Ic}	plane strain fracture toughness, ksi-$\sqrt{\text{in.}}$
K_{ISCC}	critical stress intensity factor for stress corrosion cracking, ksi-$\sqrt{\text{in.}}$
K_{max}	maximum stress intensity factor, ksi-$\sqrt{\text{in.}}$
K_{min}	minimum stress intensity factor, ksi-$\sqrt{\text{in.}}$
$\Delta K = K_{max} - K_{min}$, ksi-$\sqrt{\text{in.}}$	
M,N	number of load cycles
ρ	material density, lb/in^3
$R = \frac{K_{min}}{K_{max}}$	
r_y, R_y	radius of crack tip yield zone, inches
σ	stress, ksi
$\Delta\sigma$	change in stress, ksi
σ_L	limit stress, ksi
σ_{ys}	yield strength, ksi
da/dN	fatigue crack growth
da/dt	environmental crack growth

Subscripts:

0,1,2,3,... reference values

A,B,C,D,E,F,G requirements

c critical

i initial

f final

max maximum

min minimum

MATERIALS UTILIZATION IN STRUCTURAL DESIGN – RESISTANCE TO FRACTURE

With the advent of higher performance air vehicles, weight minimization has necessitated optimum design and construction techniques and greater utilization of the high-strength, high-efficiency, and limited-ductility materials. The process also has evolved increased operating stresses and, thus, lower tolerance to flaws and cracks.

These applications have resulted in critical flaw dimensions of the order of the material thickness which make positive detection by current nondestructive inspection (NDI) practice questionable. Current trends in the structural design utilization of high-strength alloys for resistance to catastrophic fracture can be evaluated by examining trends in two basic material parameters, the plane strain fracture toughness index K_{Ic} and the conventional yield strength σ_{ys}.

For a specific application, the designer must select a material of reasonably high strength in order to meet static strength requirements and still achieve minimum weight. A parameter for evaluating structural efficiency (σ_{ys}/ρ) is mentioned later. In the selection process, however, fracture toughness must be a consideration. The achievement of maximum yield strength and maximum fracture toughness is often difficult as is illustrated in figure 1. It is generally recognized that within certain material groups, toughness decreases with increasing yield strength. This trend is illustrated in figure 1 for aluminum, titanium, and several selected steels where material data from table II have been plotted. Variations in K_{Ic} can be expected for any given alloy and strength level, and these variations are generally due to metallurgical aspects, impurities, or manufacturing processing. This variability makes the selection of a "design allowable" extremely difficult.

In specifying a particular material and strength level (minimum acceptable σ_{ys}), the designer usually would not be concerned about those quantities of material which possessed strength levels on the upper end of the normal range. However, because of the dramatic decrease in K_{Ic}, he must in many cases limit the upper bound of acceptable range of yield strength. This is current practice in specifying titanium alloys. In figure 1, K_{Ic} ranges for two common titanium alloys are noted. These data are shown at one yield strength value to illustrate the fallacy in specifying only σ_{ys} minimum. Recent F-111 experience with D6ac steel has indicated a similar phenomenon; however, the variation of K_{Ic} is dependent upon the heat treatment procedure (ref. 9). In this case, two specimens of material from different lots might possess the same measured σ_{ys} and yet have a two-to-one range in K_{Ic}.

The material selection process is therefore a trade-off procedure wherein many concurrent requirements must be satisfied. For the case in point, the designer must establish criteria for accepting either a reduced toughness or a reduced strength level. The choice might be dictated by overall flaw tolerance. This is illustrated in figure 2 where the ordinate $\left(K_{Ic}/\sigma_{ys}\right)^2$, a parameter indicative of crack size, is used. Since structures are designed to withstand (statically) a percentage of the yield strength, this parameter may be conveniently used to illustrate flaw tolerance sensitivity. Examination of figure 2 indicates a more dramatic reduction in the crack length parameter with increased yield strength.

The same trend is repeated in figure 3; however, the yield strength has been normalized with respect to the material density ρ. The parameter σ_{ys}/ρ is one form of structural efficiency used to select materials. Note that material ranking has changed, with titanium being superior to steel. One exception illustrated is that 18Ni-Co-Mo maraging steel and 9Ni-4Co-2C fall beyond the bounds illustrated. There are recognizable limits on the values of both $\left(K_{Ic}/\sigma_{ys}\right)^2$ and σ_{ys}/ρ for materials in use today. The bounds are illustrated in figure 3.

The data presented in figure 3 clearly illustrate the relationship of nondestructive inspection (NDI) capability and material selection to resist brittle fracture. For example, a through-the-thickness crack will experience plane strain fracture when $K = K_{Ic} = \sigma\sqrt{\pi a_{cr}}$. If fracture is assumed to occur at the design limit stress, the value of critical crack length a_{cr} can be computed. For many aircraft structures, design limit stress is of the order of $\sigma_L = 0.6\sigma_{ys}$ and $a_{cr} = \left(\frac{K_{Ic}}{0.6\sigma_{ys}}\right)^2 \frac{1}{\pi} \approx \left(\frac{K_{Ic}}{\sigma_{ys}}\right)^2$. Thus each point in figure 3 might be considered the critical characteristic flaw dimension for plane strain fracture and thus would describe the sensitivity level required for fleet inspection. For this type of selection criterion, many materials may be prohibited because of the extremely small flaws which must be detected. Limits of NDI practice are not well defined.

With the technological trend in material utilization growing toward greater strength-density ratios, it seems logical also to define more realistic limits on the material selection based on uncontrollable "human element" defects. Thus, the crack size definition of figure 3 might indicate limits produced by normal tool marks, scratches, or gouges produced during manufacture or maintenance. If these limits are recognized as sound, then more effective means of inspection may be required, such as proof testing, if use is to be made of these alloys (fig. 4).

All the data from table II has been plotted in figure 5 with both K_{Ic} and σ_{ys} normalized with respect to density ρ. This plot indicates an apparent technological limit which material producers might find difficult to exceed (ref. 2).

In the previous discussion it was assumed that plane strain fracture is dominant. Fortunately, this is not always the case because of the effect of thickness, plasticity, and geometry (figs. 6 and 7). The question does remain, however, as to what role K_{Ic} has in the material selection and analysis process.

It is perhaps safe to conclude that the selection of candidate materials for fracture considerations can be made on the basis of superior K_{Ic}, as long as the materials are similar. The decision, however, rests upon the thickness required to fulfill the task. In figure 7, the variation of critical stress intensity factor with thickness is illustrated for several alloys (ref. 2).

MATERIAL SELECTION – RESISTANCE TO FLAW GROWTH UNDER REPEATED LOADS

In the preceding discussion, K_{Ic} and σ_{ys} were shown to be effective parameters in selecting a material class and alloy to resist brittle fracture under plane strain conditions. Wide variations in strength and toughness were indicated within a given material. Toughness was also seen to vary within a given alloy group.

Material selection based on cyclic growth considerations is not as clearly defined, since observed trends in rate data for a nonaggressive environment indicate that materials within a group or class generally fall within a narrow scatterband, with little, if any, dependence on toughness. Average growth-rate curves have been included in figure 8 to illustrate the relative relationship between materials. Hahn (ref. 6) has observed that the rate da/dN can be approximated for many materials as

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

in the central or log linear portion of the growth-rate curve. Several points are shown in figure 8 which were obtained by using the Hahn expression. Because of the relationship of growth rate to modulus E, the data can be normalized with respect to the material density ρ as indicated in figure 9 where rate curves are seen to converge. It is apparent, then, that a material's advantage can only be assessed on an individual application basis. Growth under variable-amplitude spectrum loading, for example, may produce different trends in growth retardation due to the interaction of loads. Generally speaking, however, the time to failure from an initial flaw is dependent primarily upon the toughness K_{Ic}. This is illustrated in figure 10, with cutoffs for several levels of toughness. The relative effect, however, may be dependent upon the shape and severity of the spectrum.

While the preceding discussion has been concerned with the cyclic flaw growth behavior, the selection of materials for repeated load application in the presence of flaws may be seriously influenced by the chemical and thermal environments in which the structure must operate. No attempt is made in this paper to cover these trends. The reader is referred to several excellent publications (refs. 6, 11, 14, and 15).

FRACTURE CONTROL – BASIC CONSIDERATIONS

The traditional Air Force approach to structural integrity (ref. 13) requires that "safe life" be evaluated through the cyclic test program. The success of this approach in determining the overall fatigue resistance of full-scale structures has been well documented (refs. 6 and 16). The achievement of "fatigue quality" through careful workmanship, surface finishes, and detailed design (local stress levels) and the demonstration of resistance to crack initiation are basic and reasonable goals. Therefore, before presenting suggested procedures for fracture control, it is important that two basic tenets be stated:

(1) Damage tolerant design and fracture control philosophy should not be considered as substitutes for adequate fatigue considerations.

(2) Consideration must be given to the probable existence of flaws within all basic primary structures.

Crack initiation resistance and fracture resistance should be considered as complementary objectives.

By virtue of its complex nature and varied operational regimes, an airframe encounters a wide variety of natural and induced environments. While this makes the application of fracture theory a rather difficult task, the general overall goals which must be achieved are rather simply stated, as follows:

(1) Encourage the intelligent selection of fracture-resistant materials, manufacturing processes, and so forth

(2) Provide an incentive to design for inspectability with damage-resistant structural configurations (i.e., multiple load paths)

(3) Aid in establishing effective and realistic inspection procedures

(4) Assist in selecting and controlling safe operating stresses

In the Materials Utilization section, materials data were presented to illustrate how strength-density ratio (efficiency) could result in the selection of material with an undesirable level of toughness. Likewise, the choice based on fatigue alone might lead to serious difficulty since many high-strength materials (steels, for example) may have acceptable fatigue resistance but possess low resistance to brittle fracture and subcritical flaw growth (stress corrosion cracking, for example).

Structural configurations which possess multiple load paths, crack stoppers, and so forth, are necessary and desirable; however, their ability to function and meet specific preassigned goals must be demonstrated early in design.

Controlling design stress levels for common structural materials can have untold benefits from both the strength and fatigue points of view and can prevent costly field maintenance problems. For example, multiple load path, redundant, and fail-safe arrangements may effectively prevent the loss of aircraft, so long as adequate and frequent inspections are planned. The sole dependence on the fail-safe approach to achieving fracture control without regard to limiting design stresses may result in frequent member failures, costly unscheduled maintenance, and aircraft downtime. This situation can be alleviated by requiring each member in the multiple or redundant set to be inherently resistant to flaw growth within prescribed bounds (i.e., it must have a safe life with cracks).

The ability to detect and quantify flaws and cracks, both in the raw product form and the final assembled structural article, remains as the most significant measure in deterring catastrophic fracture. Instituting fracture control procedures is, in fact, a frank admittance that serious flaws can and often do go undetected. This fact was dramatically pointed out by Packman, Pearson, Owens, and Young (ref. 17) in a study for the Air Force Materials Laboratory. The data in figure 11 have been obtained from that report and depict the sensitivity and reliability of common NDI methods in controlled laboratory experiments. The results are quite surprising because relatively large flaws were not detected. This does not mean that all hope is lost of improving present methods and procedures. On the contrary, continued development of improved NDI techniques is mandatory.

Fracture control procedures are most beneficial if effectively implemented and managed. Implementation consists of satisfying specific requirements for analysis and test based on established ground rules and definitions of required strength, assumed damage, service life, and inspection intervals. A balanced design within the goals of damage tolerance is thus insured. It is important that the basic definitions, goals, and fracture requirements be established early in the design phase in order to impact trade studies. Implementation requires a firm material data base, knowledge of operational environments, design criteria, and an analytical capacity to perform complex flaw-growth and strength analyses.

If fracture control procedures are instituted early, they form a portion of the basic design criteria and no weight penalties can then be attributed to their existence. Weight penalties are only recognized if the requirements are levied after the design is frozen.

FRACTURE CONTROL – REQUIREMENTS

It should be acknowledged that the preparation of detailed step-by-step requirements for fracture control is a difficult task because of the numerous classes of aircraft (i.e., fighter bombers, trainers, etc.) in use today by the Air Force and because of the various types of structural arrangements which comprise these airframes. With regard to the structural aspects, the term "Damage Tolerant" is perhaps most common and is used within the Air Force (ref. 13) to describe those configurations "which will minimize the loss of aircraft due to the propagation of undetected flaws, cracks, or other damage."

Supplemental requirements for the ASIP (ref. 13) and various military specifications (ref. 18) are currently being formulated to insure the achievement of damage-tolerant design. Such requirements will be applicable to all primary structures, the failure of which would reduce the strength level below specified limits and endanger the safe operational flight characteristics of the aircraft.

In general, requirements to insure adequate fracture control take on the form of specific directives in the areas of (1) design, (2) analysis, and (3) test.

In the following discussion, a representative set of specifications for fracture control is described to indicate the relative levels of importance placed on structural arrangements, inspections, and so forth.

It is generally recognized that there are two major design steps which are required to produce a damage-tolerant structure:

(1) Controlled safe flaw growth (safe life with cracks)

(2) Positive damage containment (remaining or residual strength)

Neither of these should be considered separate and distinct, however, since it is the judicious combination of both that is required for effective fracture control.

Since the assumption is made that flaws do exist in new structures and can go undetected, full compliance with this philosophy requires that consideration be given to the probability that flaws will exist in any and/or all members, including each element of a redundant or multiple load path group. This is important because it is easy to rationalize that each member of the multiple set could be flawed. For example, 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. The first major requirement for fracture resistance must, therefore, dictate that any member must have a safe life with assumed cracks present.

For any given application, the overriding factors which govern the details and complexity of the fracture requirements and demonstrations are (fig. 12)

(1) The class or type of structure

(2) The quality of production and assembly NDI

(3) The accessibility of the structure

(4) The assurance that the member will be inspected in service

(5) The probability that a flaw of subcritical size would go undetected even though periodic inspections are made

Most structural members can be classified by load path (fig. 13):

(1) Single load path

(2) Single primary load path with auxiliary crack arrest features

(3) Multiple and redundant load path

Class 2 includes such items as pressure cabins and pressure vessels, where relatively large amounts of damage may be contained by providing tear straps, stiffeners, and the like. While some load shedding does take place, the primary load path is singular. Detection of damage for such cases is likely, because of fuel or pressure leakage.

Class 3 structures are generally designed so that some percentage of original strength is retained during and subsequent to the failure of one element (often called fail safe). Assurance of this capability should be mandatory by analysis and tests. The containment of damage is often produced by natural barriers such as production splices and so forth.

Accessibility and inspectability were indicated in the section on Basic Considerations for Fracture Control as major items in fracture control. This point cannot be overemphasized. Not only should the structure be inspectable, but assurance must be given that it will be inspected periodically after assembly. Because of recent experiences with high-strength materials, speculation has arisen whether or not subsurface cracks of near-critical size can be found in service by use of routine inspection procedures and equipment. A positive criterion such as "leak before break" may have to be levied in order to assure their detection. Otherwise, an inspectable structure would have to be classified as noninspectable. (See fig. 14.)

Engineering Criteria – Definitions

Before specific fracture requirements for design, analysis, and test can be levied, certain aspects of loading and service must be defined for each type of aircraft. In most cases, these items will be unique for each particular system and will be specified in the basic design criteria.

Strength limits.- The percentage of unflawed static strength which is to be maintained with prescribed amounts of damage must be established. This load is generally the limit load but may vary with aircraft types.

Dynamic factors.- The effect of dynamic load amplification due to the release of energy as the damage is introduced must be included.

Inspection intervals.- Inspection intervals shall be consistent with required safe crack growth intervals and the requirements for residual strength.

Damage limits.- The size of initial flaws which may be expected to slip by inspection must be established from NDI capability studies. Final damage limits will be based on fracture and inspection requirements. In addition, the number and locations of members which are to be considered failed for residual strength purposes must be identified. Damage limits should be established for each system based on individual requirements, materials applications, and so forth.

Design Trade Study Analyses

A primary function of the fracture control requirements during early design stages is to assist in the selection of damage-resistant materials and structures, with some incentive offered to those that are easily inspectable and those that include multiple or redundant load paths. In figure 15, key factors which influence these trade studies are summarized. Each member is first classified as to structural type, inspectability, and so forth, and a candidate material is selected. Limits of assumed initial damage size are assigned together with the engineering criteria for life, strength, and final damage size. The analysis is then performed by utilizing the appropriate cyclic and sustained loads and environments. The process is then iterated until a satisfactory combination of material and stress level is selected which fulfills the strength and life requirements. The resultant information is then incorporated with other design considerations until a satisfactory design is achieved.

Analysis – Detailed Requirements

The analysis consists of determining the growth rates of initial flaws under cyclic loading and environment and insuring that these flaws remain subcritical for the specified time period. Initial flaw sizes generally reflect the NDI capability but may be influenced by such criteria as proof tests and manufacturing processes. The flaws are generally assumed to be normal to the maximum principal stress field. The character and shape of the flaws are usually influenced by such aspects as

(1) Materials and processing

(2) Manufacturing and assembly

(3) Handling and service conditions

Experience has indicated that the flaw types shown in figure 16 are most representative in aircraft.

In table III, a set of hypothetical analysis requirements have been tabulated for the three classes of structures, based upon whether or not the assemblies will be inspected in service.[1] The information from table III has been translated into figures 17, 18, and 19 for clarity. As is indicated, each class is designed for a safe crack growth period from an initial flaw. The final fracture dimensions are governed by plane strain fracture at limit load unless conditions indicate that this mode of fracture is unlikely. Some motivation to design with inspectability and with high-toughness materials (and thus higher stresses) is offered for $(a_3 > a_5)$ and $(a_4 > a_5)$. The final crack dimensions a_3 and a_4 must truly be detectable however; otherwise, the structure should be reclassified as noninspectable. It was previously stated that subsurface flaws most likely should be put in the noninspectable class (for service inspections). However, in most cases, it is possible to achieve through-the-thickness cracks and thus "positive detection" with proper selection of materials and stresses.

A safe life period of two inspection intervals has been indicated for the class 1 and class 3 inspectable cases. This will result in a slight reduction in allowable design stresses but will offer more chance to detect the subcritical crack.

For the class 1, single load path, structure the requirement to satisfy a safe life with cracks is easily accepted because of the consequence of losing the member.

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[1] These requirements are presented for purpose of illustration only and do not represent USAF policy.

However, as previously stated, the preexistent flaw concept requires that all members, including each member of a multiple load set, be assumed flawed. It is not sufficient simply to design the multiple load path structure to a remaining strength criterion with one principal member failed. This does not insure that initial flaws in a member will not grow to critical size in a relatively short period of time and result in broken members and unscheduled, costly maintenance. Therefore, the safe life requirements C and E as listed in table III and indicated in figure 18 are applicable to every member of the structure. However, since there should be some incentive to design class 3 structures, the size of the initial assumed flaws in the class 3 structure is reduced from that in the class 1 structure for the noninspectable case $(a_1 < a_2)$. By doing this, the designer is admitting that the design is more comfortable and that he is willing to take a larger risk of operating with cracks.

Supplemental safe life (with cracks) requirements (F and G) for the class 3 structure are listed in table III and are applicable to the remaining structure after the one principal member has failed. In these requirements, the assumption is made that the element could fail at any time during the life (or inspection period) and go undetected. The remaining structure (assumed to be flawed) would then be required to carry the maximum load for the duration of the remaining specified time period. The stresses which result from requirements F and G most likely will dominate the design. In actual practice, studies would have to be conducted to determine the most appropriate time to assume the member failure. In requirement F, the remaining growth period would be one inspection interval regardless of when the member was assumed to have failed. As is indicated in figure 19, the total growth in any one member is equal to the amount which occurs prior to the failure of the principal element plus the amount which occurs subsequent to the failure at an increased stress level.

Alternate Scheme to Assess Remaining Life

In the previous section, requirements F and G (table III) were presented to satisfy the requirement for some remaining life in the multiple load structure after the failure of any principal member. An alternate scheme, and one which may be less restrictive, has recently been prepared for use in the Air Force. The principal difference is that the remaining structure is considered to be intact (unflawed) subsequent to the failure of the principal element. The requirement is stated as follows in reference 18:

> "Fail Safe. Primary structure that is designed fail safe shall be readily inspectable and meet the following requirements after failure of a principal structural element: (1) the remaining structure shall sustain without failure, the maximum expected load or limit load, whichever is greater, (2) the airplane shall be controllable within the design speed limits, and (3) catastrophic

failure of the remaining structure will not occur under repeated load conditions during the time period to the next opportunity to detect the failure. Verification of the ability of the remaining structure to withstand the repeated loads shall be accomplished by determining the crack growth period from an initial flaw to failure of the principal element, and then insuring that the life (including a factor of four) of the remaining structure will equal or exceed the time interval established for the next inspection. Inspection intervals shall be as agreed to by the procuring agency . . ."

Fracture Control – Verification and Demonstration

In the preceding discussion, requirements for analysis were presented. In certain instances, experimental verification or demonstration of compliance should be required.

Safe crack growth tests (class 1 and class 3).- Although basic growth-rate data will be generated to support analysis techniques, it is desirable to augment the constant-amplitude tests with spectrum crack growth tests conducted on a meaningful flight-by-flight basis. This is particularly true where reliance has been placed upon positive detection by surface flaws penetrating the member thickness. In most cases, these experiments can be conducted on representative coupons, or small specimens if stresses are well known. If the geometry is complex, it is more desirable to utilize prototype component structure and run the growth tests in conjunction with the static or cyclic preproduction tests.

Demonstration tests utilizing full-scale structures (i.e., complete aircraft) should not be necessary since it is generally quite easy to duplicate localized conditions surrounding the crack tip.

Damage arrest (class 2).- Demonstration of crack arrest capability and subsequent cyclic life should be required. These tests may be conducted on representative specimens or on the full-scale aircraft at the conclusion of the static or fatigue test. In most cases, critical damage is introduced mechanically to simulate service condition (battle damage, etc.).

Establishment of Inspection Procedures

An additional function served by the safe crack growth analysis is the establishment of inspection procedures for an individual structure or for all members in the aircraft which are manufactured from the same material. The use of fracture analysis procedures allows inspection or rejection with more confidence by classifying parts and regions within a part according to the required NDI sensitivity.

The development of such an inspection procedure for a typical application is illustrated as follows. Spectrum crack growth information is plotted in figure 20(a) as a function of the initial crack size (only a_0 is shown) for various degrees of spectrum severity (maximum stress). In this example, the required safe growth period is N hours, and a_0 is the largest crack size that can be tolerated for this material application. The maximum expected spectrum stress is σ_4. NDI procedures must insure the reliable detection of a_0 during fabrication and assembly.

This spectrum growth information is translated into more meaningful form in figure 20(b) where, for any level of design stress, the largest tolerable flaw which would grow to failure in N hours is plotted. Rather than using fracture at N hours, a criterion based on positive detection could be substituted and produce a similar diagram.

Application of Requirements

While the full impact of the proposed fracture requirements can only be assessed through an extensive design application study on an existing system, the relative severity can be assessed by studying typical examples. The following example illustrates the values of design stress for a single material which would result under each requirement listed in table III:

Example: Tension cover; aircraft type, fighter

Material, 7075-T6

$K_{Ic} = 30 \text{ ksi-}\sqrt{\text{in.}}$

Thickness = 0.375 in.

Initial flaw assumptions (surface flaw) (a/2c = 0.5):

$a_1 = 0.050$ in. (for all inspectable cases)

$a_2 = 0.150$ in. (for all noninspectable cases)

Final flaw size:

a_4 = Minimum detectable size = 0.375 in.

a_3 = Minimum acceptable equivalent = 0.500 in. for single load path structure

Stress information:

The fighter spectrum information is contained in table IV in terms of a unit of maximum stress value $\sigma = 37$ ksi. These occurrences in table IV are the equivalent of 40 hours of flight. The maximum limit stress for design purposes is:

$\sigma_L = 1.5\sigma = 55.5$ ksi

Spectrum growth-rate data:

By utilizing constant-amplitude growth-rate data (ref. 19), the CRACKS computer routine (ref. 20), and the AFFDL crack growth retardation model (ref. 10), the stress spectrum (table IV) was translated into plots of crack depth a as a function of number of flights starting with an initial crack length $a_1 = 0.050$ in. (fig. 21) and $a_2 = 0.150$ in. (fig. 22). All levels of stress from table IV were increased or decreased proportionally to achieve the variation in growth due to spectrum severity.

Material toughness:

The cutoff line for $K_{Ic} = 30$ ksi-$\sqrt{\text{in.}}$ is indicated in figures 21 and 22. The effect of varying this parameter was not investigated in this example.

Life requirement:

Service life = 160 blocks = 160 × 40 = 6400 hours. Inspection intervals are planned each 1/4 lifetime of 40 blocks = 1600 hours.

<u>Requirement A</u>:

Initial crack depth:

$a_1 = 0.050$ in.

Final crack depth:

$a_3 = 0.500$ in. (based on positive detection)

Life requirement:

N_A = 80 blocks = Two inspection intervals

Design stress σ_A:

This goal cannot be achieved with this material since K_{Ic} is limited to 30 ksi-$\sqrt{\text{in.}}$ and the inspection requirement of 0.500 in. is not possible. A material change would most likely be required.

<u>Requirement C</u>:

Initial crack depth:

$a_1 = 0.050$ in.

Final crack depth:

$a_4 = 0.375$ in. (based on positive detection)

Life requirement:

$N_C = 80$ blocks

Design stress, maximum:

σ_C (allowable) = $1.27\sigma = 47$ ksi

Requirement D:

Initial crack depth:

$a_2 = 0.150$ in.

Life requirement:

$N_D = 160$ blocks = One lifetime

Final crack depth:

a_5 = Plane strain fracture > 1.0 in.

Design stress, maximum:

σ_D (allowable) = $0.81\sigma = 31$ ksi

Requirement E:

Initial crack depth:

$a_1 = 0.050$ in.

Final crack depth:

a_5 = Plane strain fracture = 0.58 in.

Life requirement:

$N_E = 160$ blocks

Design stress, maximum:

σ_E (allowable) = $1.08\sigma = 40$ ksi

Requirement F:

Coupled with requirement C is the additional requirement that the structure remaining after failure of the principal member will be capable of carrying limit load for one additional inspection period, or 1/4 lifetime. The lower portion of the growth data from figure 21 has been replotted in figure 23.

(a) Assume that the member breaks accidentally after the first flight and remains undetected until the next inspection interval. The stress is assumed to increase by 20 percent, with the requirement being no failure at limit load in 1/4 lifetime or 40 blocks. From figure 23, it can be seen that a stress level of approximately $1.6\sigma = 60$ ksi would grow to failure in 40 blocks. Therefore

$$\sigma_{F_a}(\text{allowable}) = \frac{60}{1.20} = 50 \text{ ksi}$$

(b) Assume the member failure to be at 1/4 lifetime (just subsequent to inspection). The crack in the remaining structure has grown an amount Δa during the first inspection period. Thus,

$$\text{New initial } a = a_1 + \Delta a = 0.050 + \Delta a$$

This condition can be satisfied by trial and error by using figure 23. The result indicates that $\sigma_{F_b} \approx 1.2\sigma = 44.4$ is appropriate for this condition. Failure at any other time could be checked to see whether a lower stress would result. Note that no criterion for positive detection was required since at the next inspection the broken member would be found.

Requirement G:

In a similar manner, requirement E should be checked for life after member failure.

(a) Assume failure on first flight (from fig. 21)

$$\sigma_E = 1.08\sigma = 40 \text{ ksi}$$

$$\therefore \sigma_{G_a} = \frac{\sigma_E}{1.2} = 33.3 \text{ ksi}$$

(b) Assume failure at 1/2 lifetime. The incremental growth during the first 1/2 lifetime must be added to a_1. The requirement for 1/2 remaining life shall then be determined. From figure 21, by trial and error, a stress level of $\sigma_{G_b} = 1.0\sigma = 37.0$ ksi is seen to satisfy the requirements.

Summary:

The following table is a summary of the previous example:

Requirement	Design stress, σ, ksi	Condition
A	Not satisfied	Inspectable class 1
C	47	Inspectable class 3
D	31	Noninspectable class 1
E	40	Noninspectable class 3
F_a	50	Inspectable class 1
F_b	44.4	
G_a	33.3	Noninspectable class 3
G_b	37.0	

The results clearly indicate the advantages offered by designing for inspectability since the allowable stresses for requirements C and F are greater than for requirement G. The incentive for multiple, in lieu of single, load path design is seen in the resultant allowable design stresses for requirements E and G being greater than for requirement D.

ANALYSIS AND DATA REQUIREMENTS FOR IMPLEMENTATION

The successful implementation of the fracture control analysis requires the analytical capability for cyclic and environmental flaw growth, aircraft usage information, and basic strength and fracture data for proposed candidate materials.

Criteria Requirements

Initial considerations for fracture resistance and control of subcritical flaw growth must be established during the criteria development stage and must reflect appropriate chemical, thermal, and operational loads environments. For example, recent materials usage has necessitated the generation of data on sustained-load flaw growth in aggressive environments such as fuel and water (fig. 24(a)). Because loading rate and dwell times are important in the assessment of environmental effects, it has become important also to generate load-time spectra of the type indicated in figure 24(b).

Material Data Requirements

The major material strength and fracture properties required to perform the analyses and trade studies for fracture considerations are illustrated in figure 25. In all cases (except K_{Ic}) no approved standard test methods exist to determine these properties. Through experience, however, various test techniques and specimens have evolved. (See fig. 25.) As is often the case, a specimen developed for one function or application is used to generate a multitude of data. Testing techniques and data interpretation may mask important material responses or indicate false reaction to stress and environment. For example, in a recent comparison of cyclic growth-rate behavior in D6ac steel (refs. 9, 11, and 12) comparative growth rates obtained from compact tension and surface-flawed specimens indicated a predominant stress-level effect for the surface-flawed specimen, whereas no clear dependency was observed for the compact tension case (fig. 26). These effects are currently being investigated.

Fracture Analysis Methods

Prediction of fracture and growth behavior requires a means of translating external applied loads into stresses in the region of the crack tip. Finite-element techniques offer a vast potential in the area, particularly in complex structural arrangements (refs. 21 and 22). A rather broad collection of stress intensity solutions exists (ref. 4); however, their use is limited in many cases and extrapolation is often required to provide the best estimate of K.

Considerable effort is being expended in the development of computer routines to "integrate" growth-rate (da/dN) data (ref. 20), for example, and to account for the retardation effect of overloads in variable-amplitude spectra. As an example of this type of activity, the AFFDL has recently developed a mathematical model for predicting the growth delay effect (ref. 10). The basic model is concerned with the effect of the overload plastic zone on the subsequent rate of growth as indicated in figure 27. A hypothetical residual or reduction stress is then computed which suppresses the subsequent cyclic loads. Retardation is accomplished in three modes, depending on the relative size of the overload in relation to the subsequent cyclic level (fig. 28). Effective ΔK and R values are computed and reduced rates obtained from normal da/dN and ΔK relationships. Note that growth can be completely stopped (fig. 28). An extensive testing program is being completed at AFFDL to evaluate the merit of the model. In figure 29 are some early correlations with single overloads in aluminum (ref. 6). Fairly good correlation is noted also with randomized block spectrum data for D6ac steel (fig. 30).

Growth analysis schemes need to be extended to include the effects of loading rate and delay time (sustained load growth). Free surface effects and flaw shape changes, including the transition of a surface flaw to a through crack, must be included.

SUGGESTED AREAS OF STUDY

The suggested areas of study for the application of fracture mechanics in structural integrity have been summarized and are presented as table V. This table is obtained from reference 23.

CONCLUDING REMARKS AND RECOMMENDED TOPICS FOR STUDY

The author has attempted to present the significant impact of fracture mechanics and fracture control in the overall program of airframe structural integrity. The true weight, cost, and performance trade-offs associated with the implementation of these or any requirement can best be judged by experience and application to existing systems. A fair assessment can only occur, however, if continued materials and structures development efforts are directed toward upgrading existing fracture mechanics and fracture analysis technology.

The author has summarized in tabular form a rather extensive "shopping list" of items which require attention. In many cases, a relatively high degree of proficiency exists and application experience is all that is necessary while others require new thought and new direction.

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TABLE I.- TYPICAL SERVICE APPLICATIONS OF FRACTURE MECHANICS

System	Problem	Material	Type of structure	Solution	Reference
C-130	Appearance of in-service cracks in center wing area. Extremely heavy usage of aircraft due to mission change. Structure readily inspectable. Plane stress case.	Aluminum sheet	Multiple load path, heavily stiffened "planks"	Analytical estimate of remaining strength with relatively large cracks. Stiffened case. In-house experimental verification of K solution for stiffened structure. Simulated and actual panels.	5
F-100	"Thunderbird" accident. Discovery of small cracks in fastener holes during inspection. Structure not readily inspectable.	Aluminum plate	Single load path skin	Inspection program to determine maximum probable flaw sizes. Analysis of spectrum growth (no retardation assumed). Estimate of residual strength. Laboratory tests of spectrum growth with actual flaws from service and manufactured cracks.	6
T-37	Main spar failure, plane strain, crack originated in fastener hole. Structure moderately inspectable.	Aluminum "TEE" extrusion	Single load path	Estimate of stress intensity factor for complex geometry. Growth estimate for safe inspection interval	7
F-111	Fatigue test failures. Plane strain cases. Service failure (A/C) # 94). Surface flaw during manufacture, loss of aircraft. Structure not readily inspectable.	Steel plate and forging	Single load path cases	Test failure analysis. Spectrum growth tests and analysis techniques for determination of inspection intervals following proof test.	8, 9, and 10

TABLE II.- TYPICAL MATERIAL PROPERTIES

[Source: Air Force Materials Laboratory]

Material	Yield strength, σ_{ys} (typical), ksi	Plane strain toughness, K_{Ic}, ksi-$\sqrt{in.}$	$2\left(\frac{K_{Ic}}{\sigma_{ys}}\right)^2$ (a)	$\left(\frac{K_{Ic}}{\sigma_{ys}}\right)^2$ (b)	$\frac{\sigma_{ys}}{\rho}$	$\frac{K_{Ic}}{\rho}$
Steel						
D6ac	205	50 to 90	0.120 to 0.38	0.06 to 0.19	724	177 (318)
4340	220	53	.12	.06	777	187
300M	247	69	.16	.08	872	244
18Ni-Co-Mo	285	89	.20	.100	1007	314
H-11	294	40	.04	.02	1039	141
9Ni-4Co-2C	180 to 190	110 to 170	1.22	.61	600	467
Aluminum						
7075-T73 (forging)	66	31	0.44	0.22	660	310
2024-T851 (plate)	58	23	.32	.16	580	230
2024-T851 (extruded)	58	28	.46	.23	580	280
2014-T6	66	35	.56	.28	660	350
7075-T651 (plate)	78	26	.22	.11	780	260
7175-T73 (forging)	75	35	.44	.22	750	350
7075-T6 (plate)	76	27	.26	.13	760	270
7075-T6 (forging)	75	27	.26	.13	750	270
7079-T6	69	30	.38	.19	690	300
Titanium						
Ti-6Al-4V (ann)	137	50 to 60	0.26 (0.38)	0.13 (0.19)	856	312 (375)
Ti-6Al-4V (STA)	158	41	.14	.07	988	256
Ti-6Al-6V-2Sn (ann)	150	35 to 50	.10 (.22)	.05 (.11)	937	219 (312)
Ti-6Al-6V-2Sn (STA)	163	34	.08	.04	1018	212
Ti-13V-11Cr-3Al (STA)	168	25	.04	.02	1050	156

[a] ASTM thickness required for plane strain fracture.
[b] Equivalent to a_{cr} for $\sigma_L = 0.6\sigma_{ys}$.

TABLE III.- FRACTURE CONTROL ANALYSIS REQUIREMENTS

Safe crack growth	Inspectable class (a)			Noninspectable class (a)	
	1	2	3	1	3
	Requirement A (fig. 17)	Requirement B	Requirement C[b] (fig. 18)	Requirement D (fig. 17)	Requirement E[b] (fig. 18)
Initial flaw	a_1		[b] a_1	a_2	[b] a_1
Shall not grow to critical size . . .	[c] a_3	[d] a_6	[c] a_4	a_5	a_5
In the specified time of	Inspection period	One flight	Inspection period	One lifetime	One lifetime
Critical stress[e]	Limit load stress	Limit load stress	Limit load stress	Limit load stress	Limit load stress

Remaining strength additional safe life. Class 3	Inspectable class 3	Noninspectable class 3
	Requirement F[b] (fig. 19(a))	Requirement G[b] (fig. 19(b))
Subsequent to failure[f] of one principal member, the remaining structure shall be capable of carrying	Limit[e] load stress	Limit[e] load stress
At the end of	One inspection period	One service lifetime

[a] Class 1 = Single load path.

Class 2 = Single load path (crack arrest features).

Class 3 = Multiple load path.

[b] Each member of class 3 structure shall be assumed to be flawed. The safe crack growth requirements shall be continuous throughout the specified time periods prior to and subsequent to the failure of the element.

[c] a_3 and a_4 are determined by fracture considerations but must be large and detectable (i.e., through-the-thickness cracks). Otherwise, classify as uninspectable. (Note: $a_3 >> a_4$.)

[d] a_6 must be readily inspectable (i.e., pressure loss). Otherwise, classify as noninspectable class 3 structure.

[e] Will vary with aircraft type.

[f] Member can fail at any time during specified period.

TABLE IV.- STRESS SPECTRUM FOR FIGHTER AIRCRAFT EXAMPLE [a]

Layer	σ_{min}, ksi	σ_{max}, ksi	Cycles	Layer	σ_{min}, ksi	σ_{max}, ksi	Cycles
1	0.06	16.6	63	30	7.94	34.9	2
2	7.04	27.0	76	31	3.64	16.1	37
3	.45	13.7	371	32	7.57	16.8	367
4	5.90	26.4	37	33	7.15	25.6	109
5	.79	17.5	111	34	7.91	37.0	1
6	10.60	25.4	2	35	1.63	6.3	265
7	.76	14.2	363	36	.79	20.8	34
8	4.02	28.7	5	37	7.81	20.2	318
9	3.64	10.7	1280	38	3.68	11.8	6
10	6.77	22.9	62	39	0.0	11.3	21
11	3.64	16.6	1	40	7.18	17.9	374
12	6.07	17.5	89	41	2.01	13.9	478
13	8.64	21.8	41	42	1.59	8.8	46
14	9.51	19.1	57	43	.06	11.9	300
15	3.78	14.0	491	44	1.59	11.3	10
16	0.0	13.9	6	45	7.91	31.7	4
17	3.81	17.5	74	46	0.0	16.4	4
18	7.88	13.4	682	47	7.57	14.5	306
19	.72	10.4	1376	48	8.26	24.9	15
20	9.37	16.0	66	49	7.98	26.1	5
21	.52	17.2	34	50	8.19	12.9	230
22	6.76	8.6	1621	51	7.98	10.7	1338
23	7.98	11.8	1589	52	.06	19.8	19
24	.45	10.6	1374	53	3.85	10.4	1546
25	0.0	8.8	67	54	0.0	6.4	238
26	7.08	28.5	1	55	.48	16.1	114
27	7.39	22.8	250	56	7.08	14.9	370
28	.06	22.1	8	57	3.85	20.8	7
29	1.63	13.9	2	58	2.01	13.9	478

[a] Single block is equivalent of 40 flight hours.

TABLE V.- SUGGESTED AREAS OF STUDY FOR THE APPLICATION OF FRACTURE MECHANICS IN STRUCTURAL INTEGRITY

PROGRAM: Implement rational fracture mechanics theory into the design criteria, material selection, analysis, qualification, and utilization of aircraft structural systems.

TOPIC AREAS:

I Criteria
II Data requirements and applications
III Fracture analysis methodology
IV Qualification for fracture resistance
V Utilization – structural concepts

SUBJECT BREAKDOWN:

I Criteria

a. Definition chemical and thermal environment for fracture requirements
b. Review past experience, structural failure, and so forth
c. Catalog critical structural materials arrangements and previous design considerations in order to establish which areas require extensive investigation
d. Establish fracture criteria for material selection and trade-off studies
e. Establish analogous "leak before break" criteria for aircraft application
f. Assemble design data and criteria for fracture applications
g. Definition of mission and analysis, including estimates of time at load factor
h. Incorporate criteria in basic specifications including ASIP modification

II Data requirements and applications

a. Establishment of measurable parameters K_c, K_{Ic}, K_{ISCC}, da/dt, da/dN, and others, including testing standards
b. Application of K_c and K_{Ic} in design
c. Fatigue crack growth data
d. Subcritical crack growth rate, environment, and temperature
e. Effect of loading sequence on cyclic growth or growth retardation
f. Nonpropagating crack study, threshhold of ΔK
g. Parametric growth data, mission segments
h. Extension of fracture mechanics testing standards to new classes of materials
i. Study of statistically derived crack sizes and shapes based on nondestructive inspection (NDI) and nondestructive testing (NDT)
j. Effect of stress state of fracture
k. Mixed mode fracture study

TABLE V.- SUGGESTED AREAS OF STUDY FOR THE APPLICATION OF FRACTURE MECHANICS IN STRUCTURAL INTEGRITY – Concluded

SUBJECT BREAKDOWN:

III Fracture analysis methodology

- a. Assemblage of currently applicable K factor relationships including application
- b. Guidelines for estimating K or approximate K for complex cases (including superposition)
- c. Development of K for complex cases, elastic solutions
- d. Finite-element studies, crack growth, subcritical growth development of K, model crack element for finite-element technique
- e. Plasticity and free surface effects
- f. Tabulation of equivalent cracks in complex flaw geometries
- g. Analytical crack model for growth under variable loading
- h. Routine for crack growth and life estimates including environment, rates, and load sequence effects
- i. Analytical study of variation of flaw shape and surface flaws
- j. Statistical analysis to establish confidence levels for toughness and life estimates, scatter factor for application to analysis results
- k. Residual strength and static considerations
- l. Handbook preparation and design guidelines
- m. Development of semiempirical methods for estimating K
- n. Fracture arrest, damage-tolerant analysis methods
- o. Study of the effect of crack bluntness on fracture behavior

IV Qualification for fracture resistance

- a. Real-time flaw growth testing including temperature and environment (specimens)
- b. Real-time flaw growth (structures)
- c. Crack growth resistance and crack arrest testing
- d. Damage tolerance or fail-safe testing
- e. Test time reduction for (a) and (b) above
- f. Proof testing:
  - Repeat work of Tiffany (Boeing) for typical aircraft structures
  - Extend knowledge and techniques to satisfy environment and requirements
  - Statistical assessment of the risks and merits of proof testing

V Utilization – structural concepts

- a. Concepts for flaw and crack arrest
- b. New material utilization
- c. Performance and weight trade-off studies
- d. Fabrication of structural concepts and full-scale testing
- e. Inspection, fracture mechanics interface, and flaw classification
- f. Proof testing, full scale

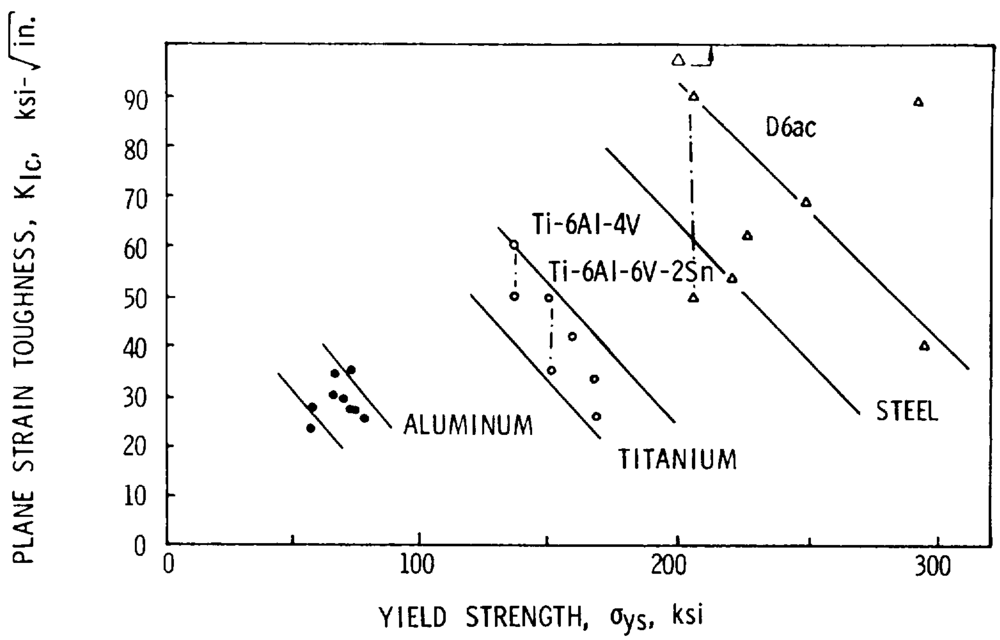


Figure 1.- Trends in toughness variation.

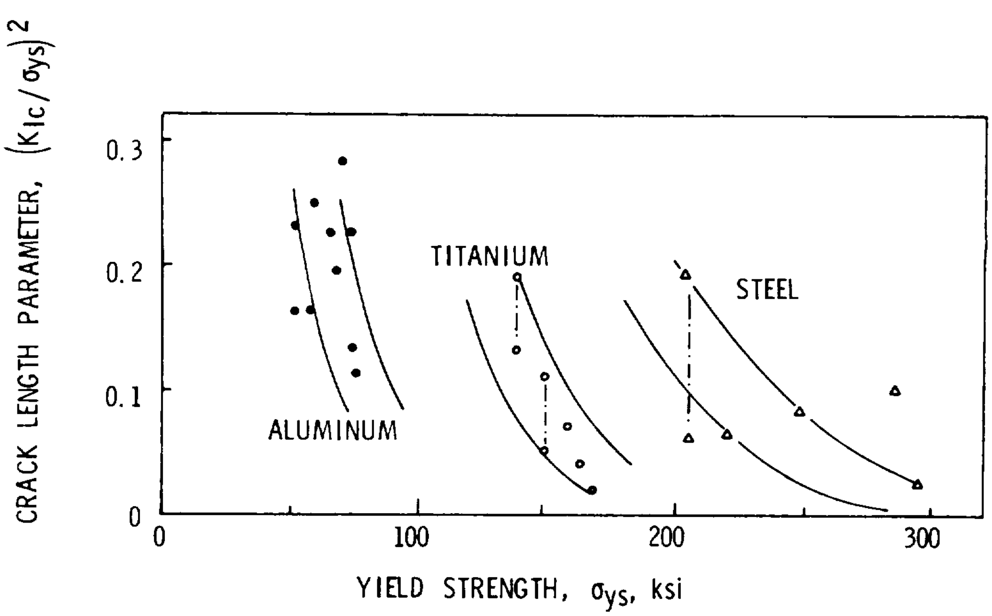


Figure 2.- Variation of crack length parameter with yield strength.

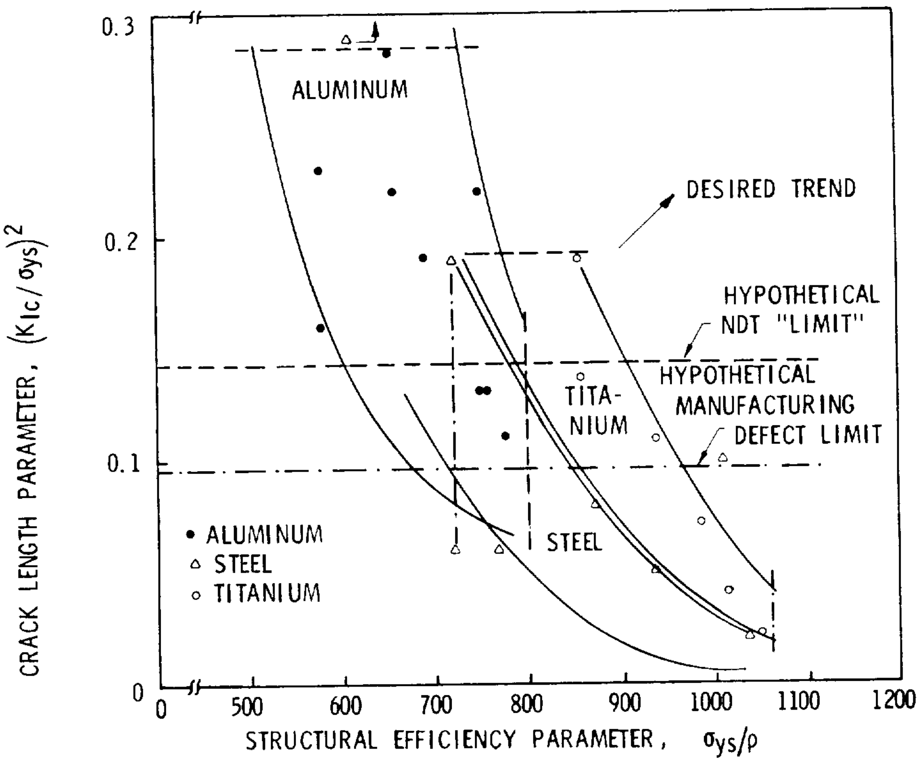


Figure 3.- Variation of crack length parameter with structural efficiency parameter.

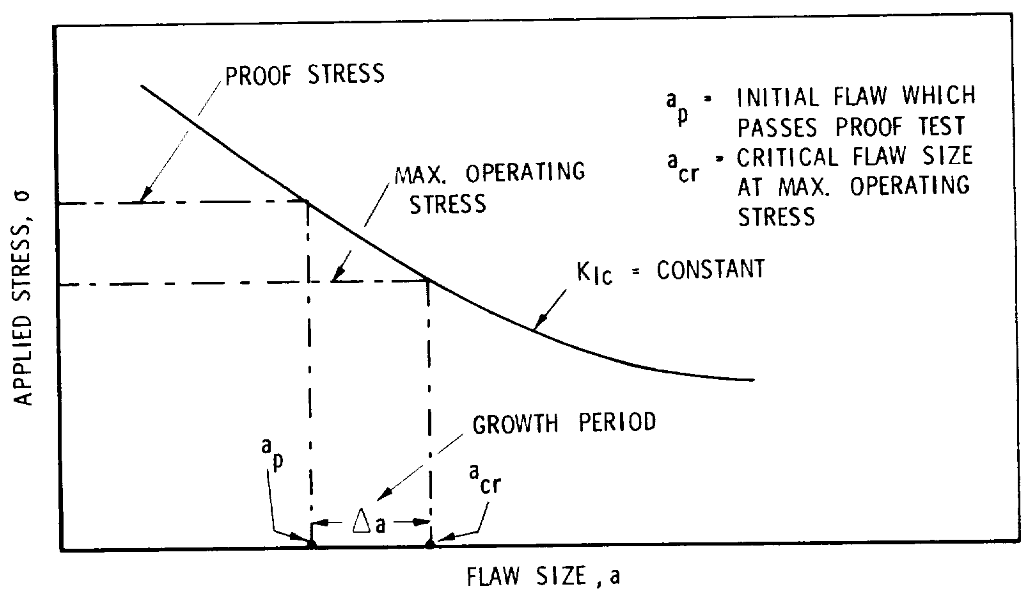


Figure 4.- Proof-test concept.

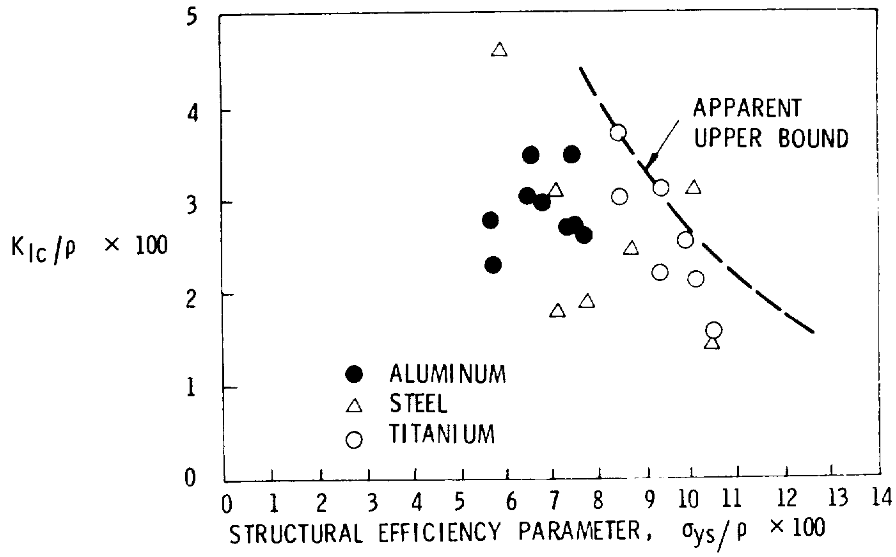


Figure 5.- Density-normalized variation of yield strength with fracture toughness.

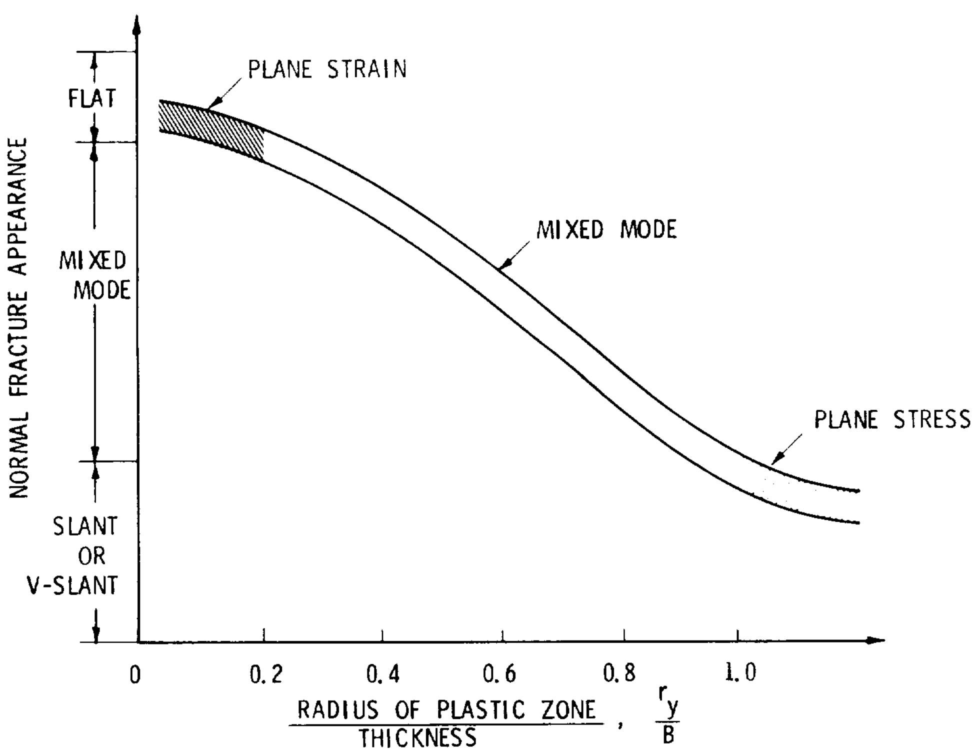


Figure 6.- Trend in fracture mode appearance as a function of crack tip plastic zone parameter (ref. 2).

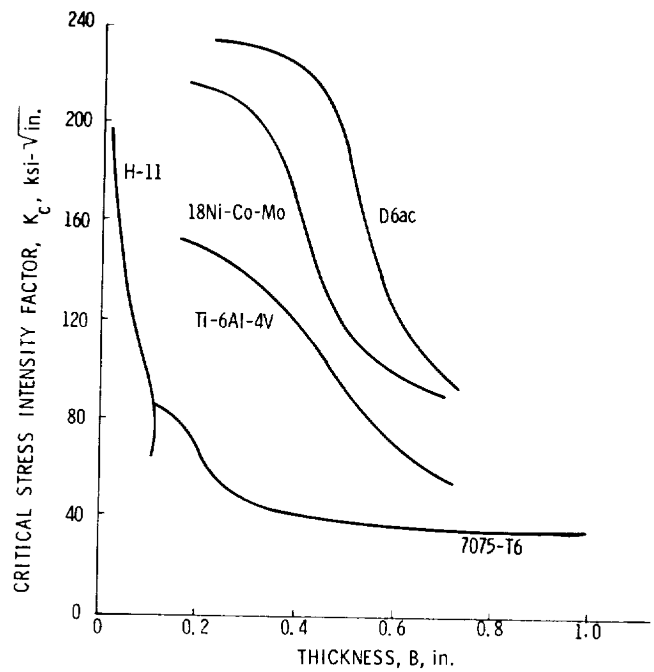


Figure 7.- Nominal critical stress intensities for several materials as a function of thickness (ref. 2).

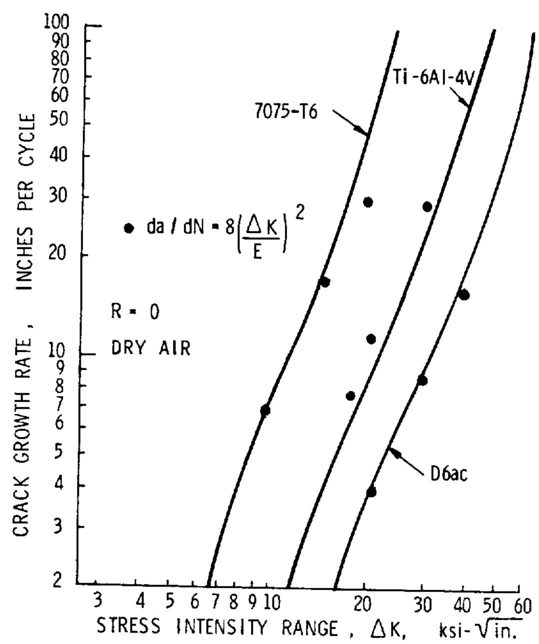


Figure 8.- Fatigue-crack-growth data for typical aircraft structural materials.

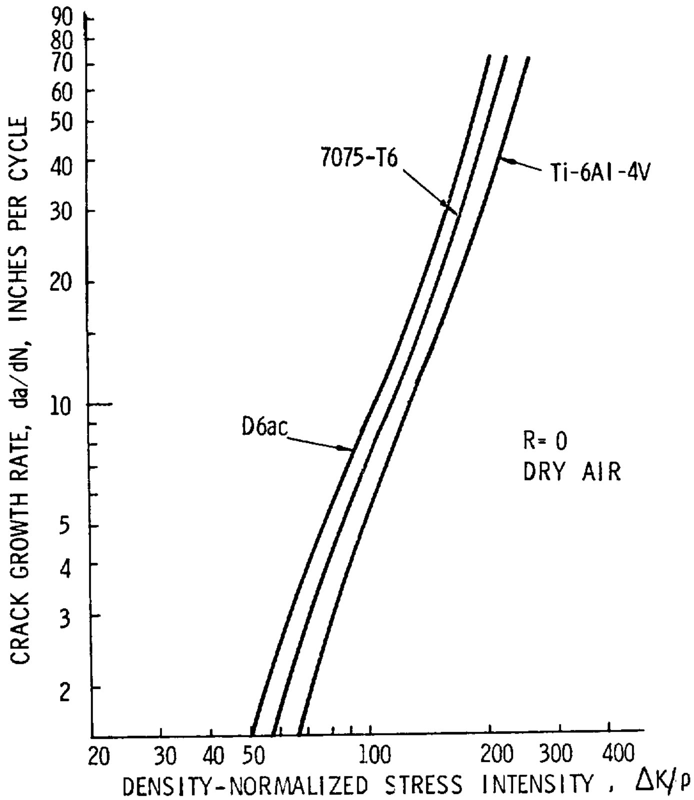


Figure 9.- Comparative crack growth data.

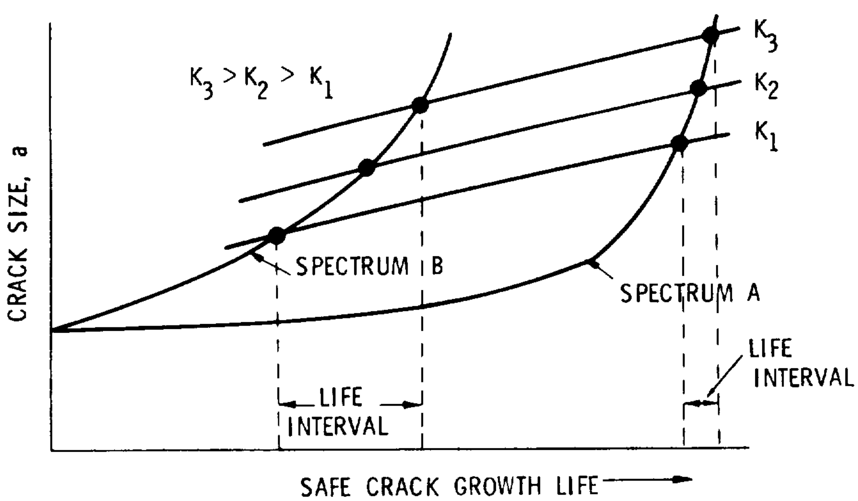


Figure 10.- Effect of fracture toughness life for various shape spectra.

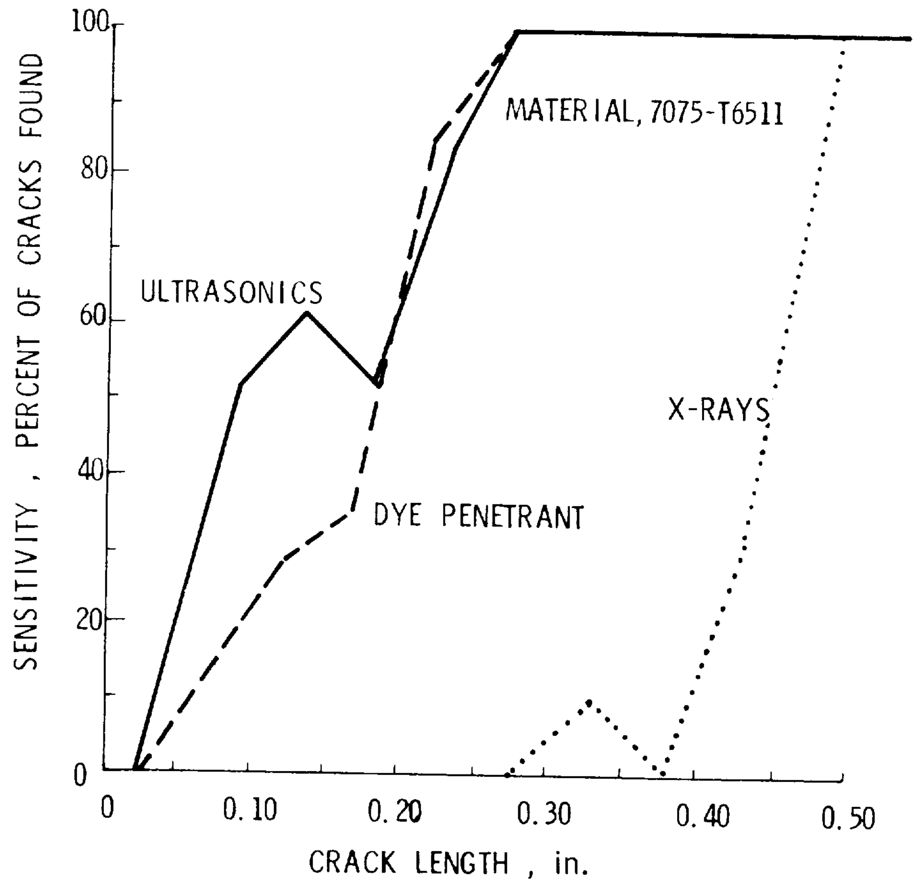


Figure 11.- Demonstration of flaw detection capability. (Data from ref. 17.)

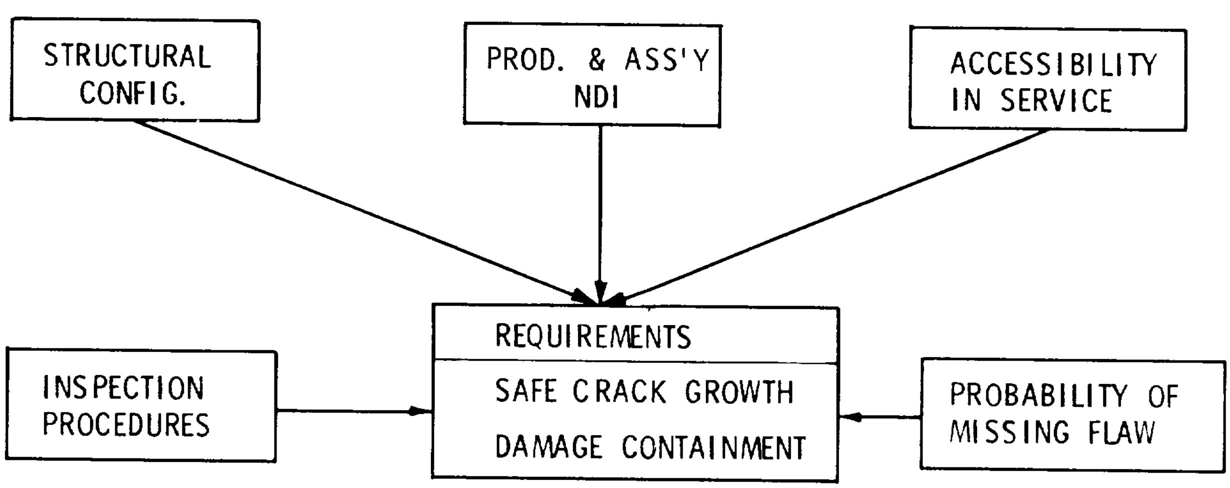


Figure 12.- Factors which affect requirements for fracture control.

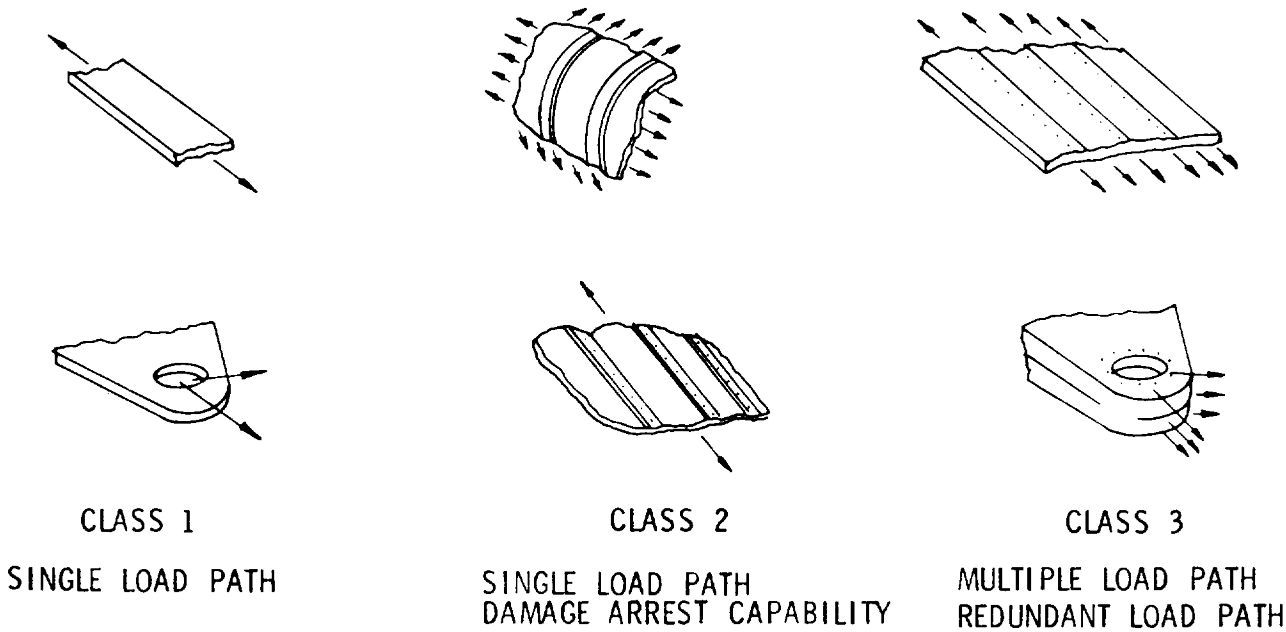


Figure 13.- Structural arrangements.

- POSITIVE CRACK GROWTH THROUGH THICKNESS INSURES DETECTION PRIOR TO CATASTROPHIC FAILURE

- MAY BE ACHIEVED BY SELECTING MATERIAL TOUGHNESS AND/OR GEOMETRY TO PRODUCE TRANSITIONAL GROWTH BEHAVIOR

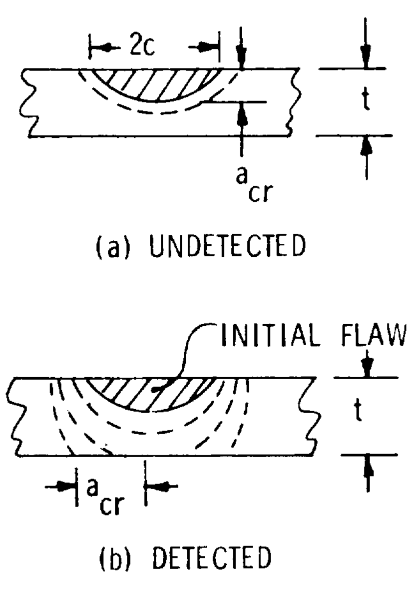


Figure 14.- "Leak before break" criteria for positive detection.

C.9

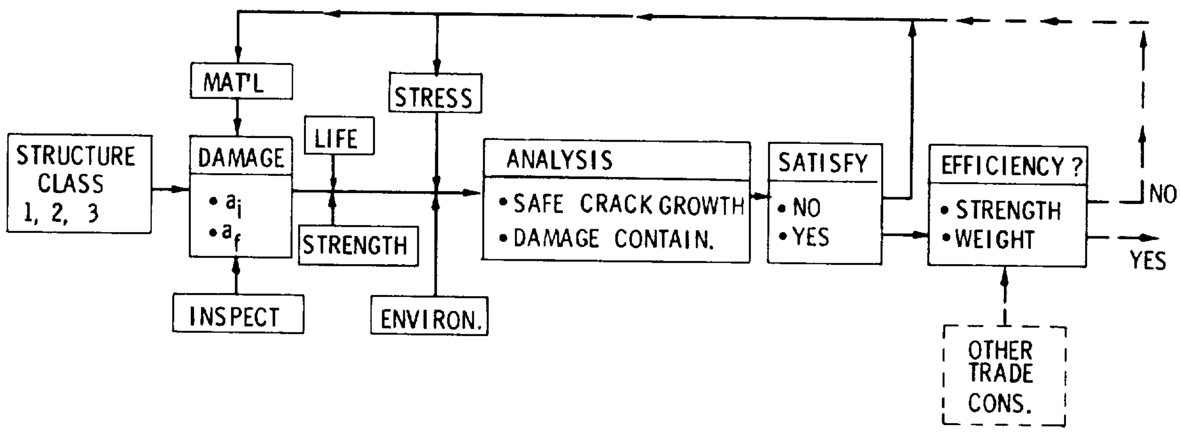


Figure 15.- Fracture control analyses for design trade studies.

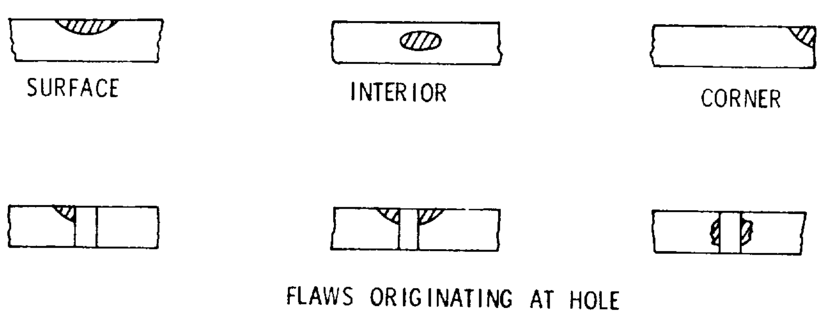


Figure 16.- Representative flaw shapes found in service.

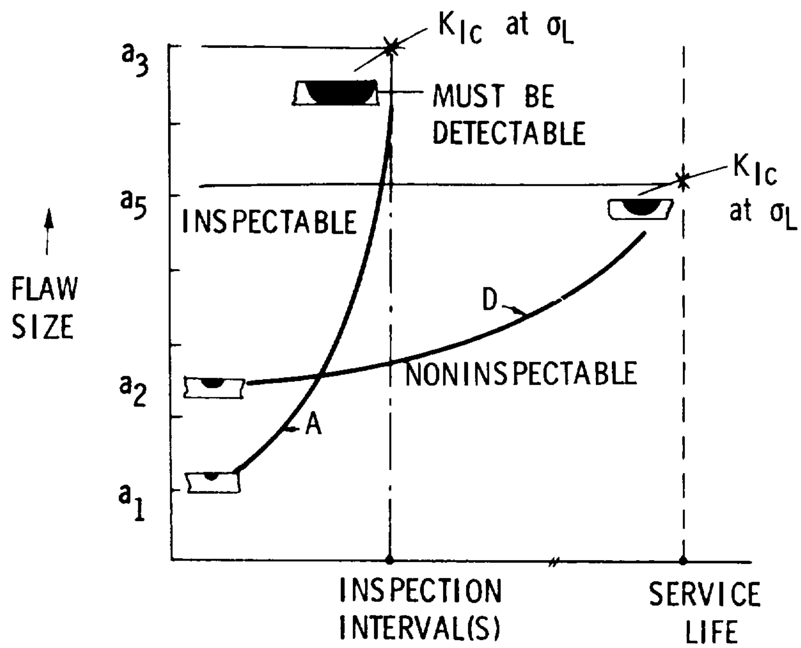


Figure 17.- Safe crack growth life requirements for class 1 structure.

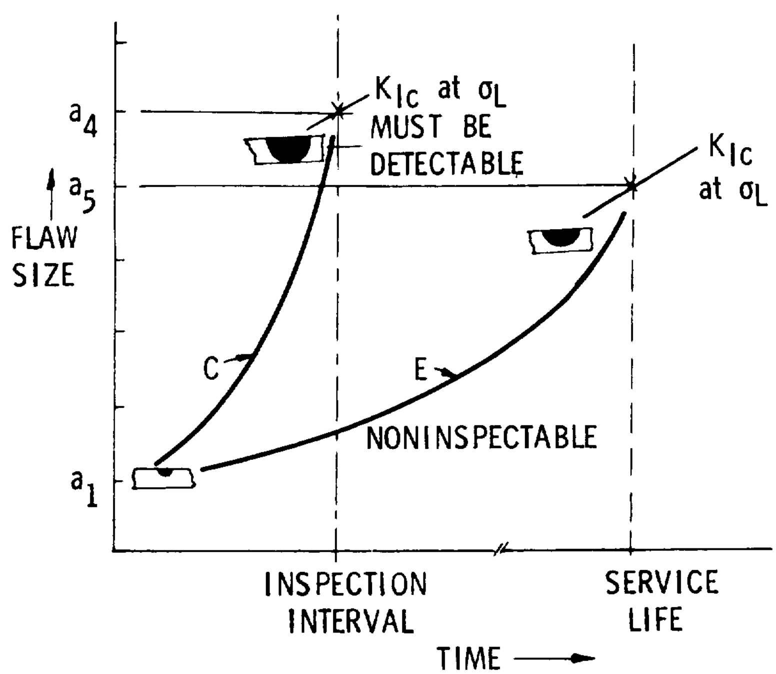
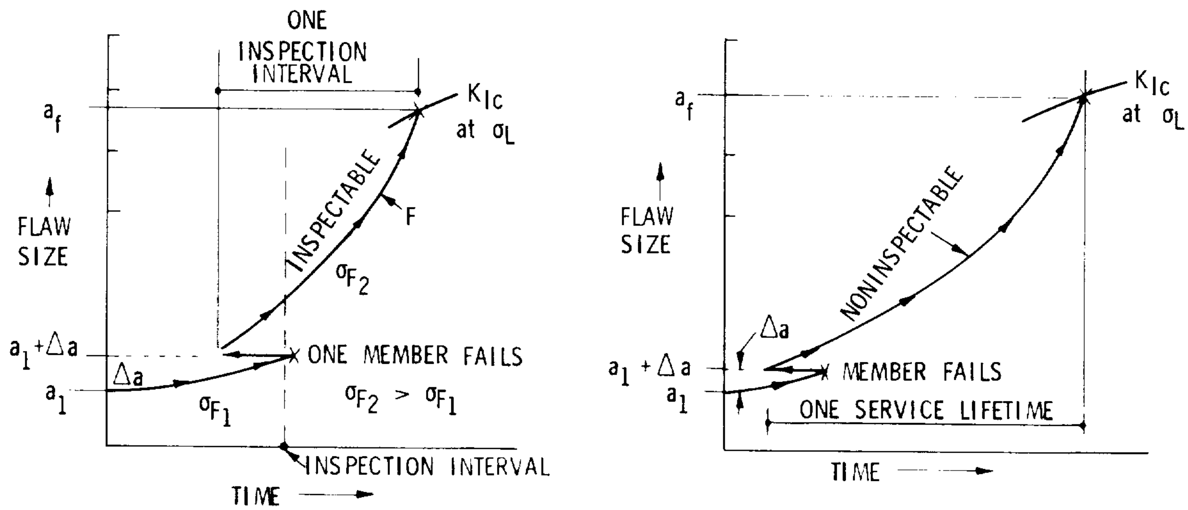


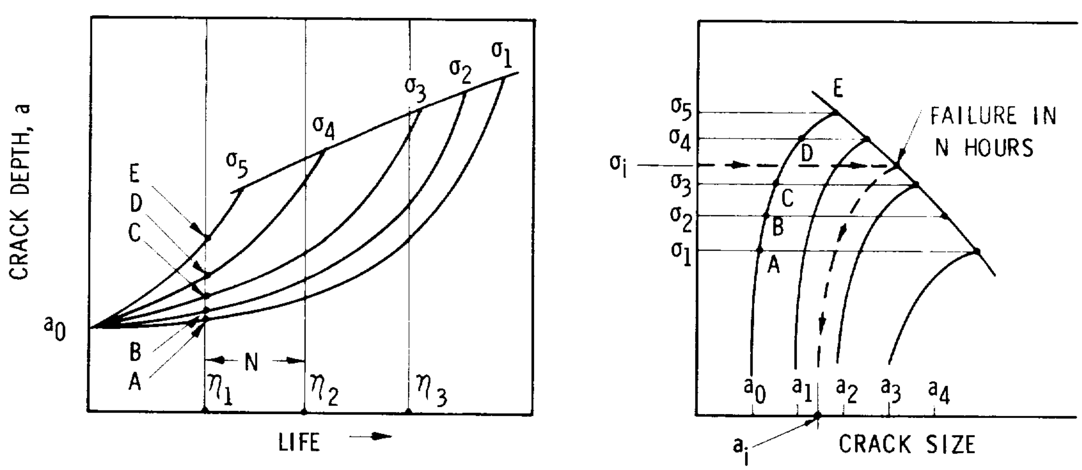
Figure 18.- Safe crack growth life requirements for class 3 structure (any member).



(a) Requirement F. Class 3 inspectable.

(b) Requirement G. Class 3 noninspectable.

Figure 19.- Safe life requirement for remaining structure after failure of single principal element.



(a)

(b)

Figure 20.- Illustration of inspection procedure based on safe crack growth analysis.

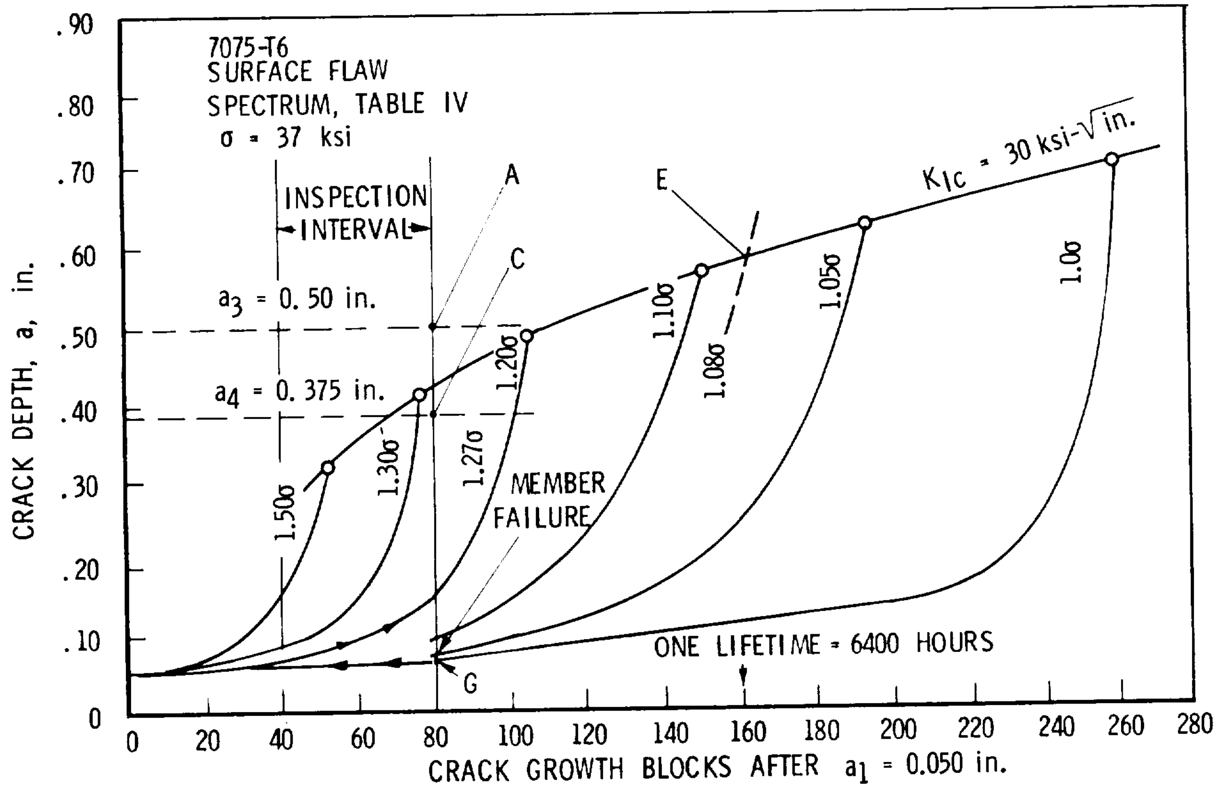


Figure 21.- Spectrum growth data. Fighter example. $a_1 = 0.050$ in.

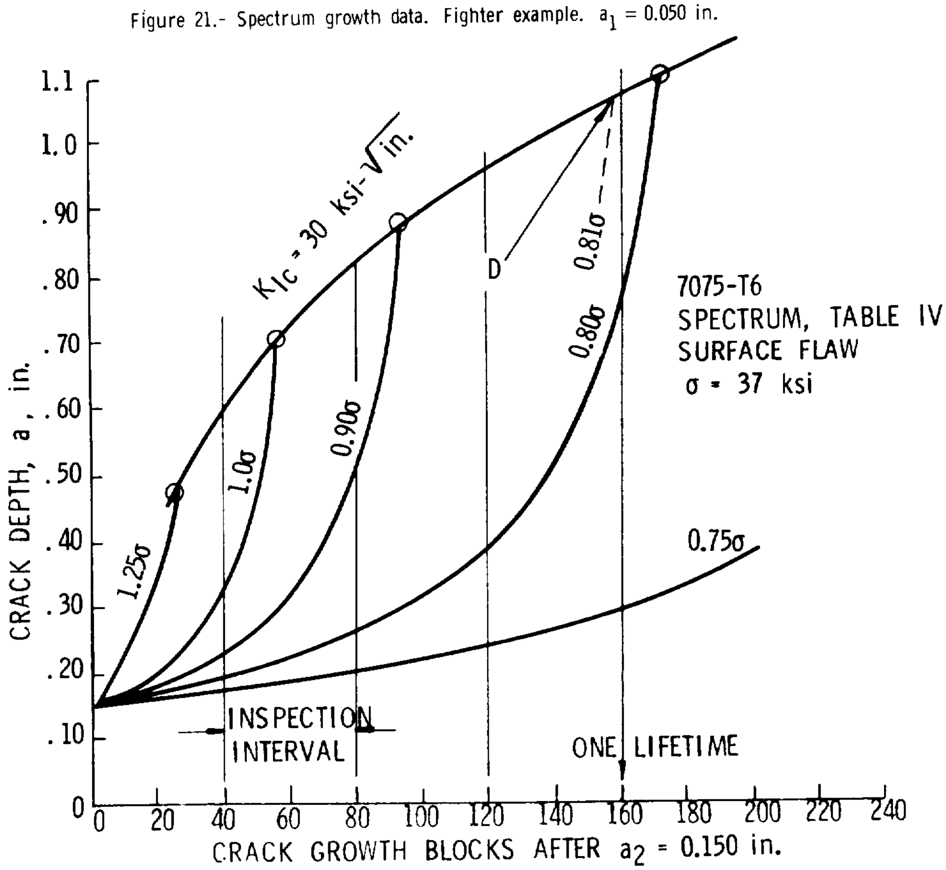


Figure 22.- Spectrum growth data. Fighter example. $a_2 = 0.150$ in.

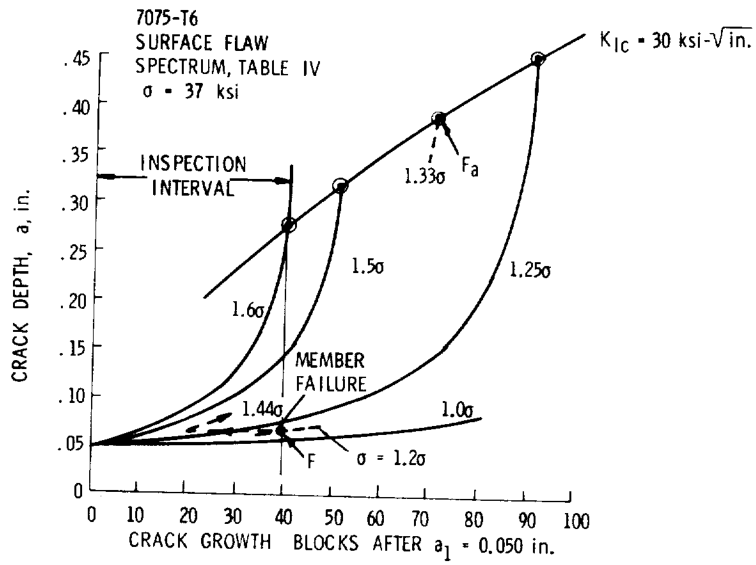
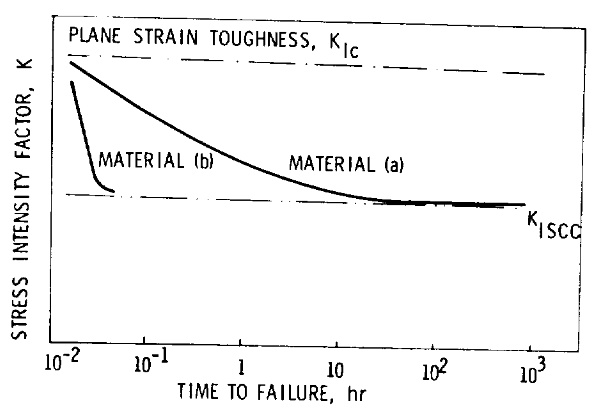
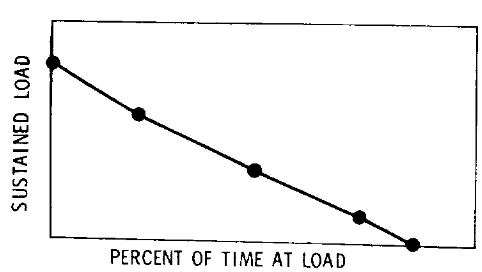


Figure 23.- Spectrum growth data. Fighter example. Data from figure 21.

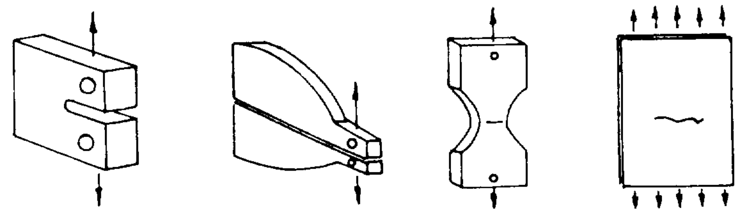


(a) Schematic of sustained load flaw growth.



(b) Schematic of sustained load spectrum.

Figure 24.- Sustained load growth and spectrum information requirements.



	COMPACT TENSION	DOUBLE CANT. BEAM	SURFACE FLAW	CENTER CRACKED SHEET
FATIGUE CRACK GROWTH, da/dN	●	●	●	●
PLANE STRAIN TOUGHNESS, K_{Ic}	●	●	●	
PLANE STRESS TOUGHNESS, K_c				●
ENVIRONMENTAL CRACK GROWTH, da/dt	●	●	●	
FATIGUE GROWTH SPECTRUM EFFECTS			●	●

Figure 25.- Material fracture properties required for analyses and trade studies.

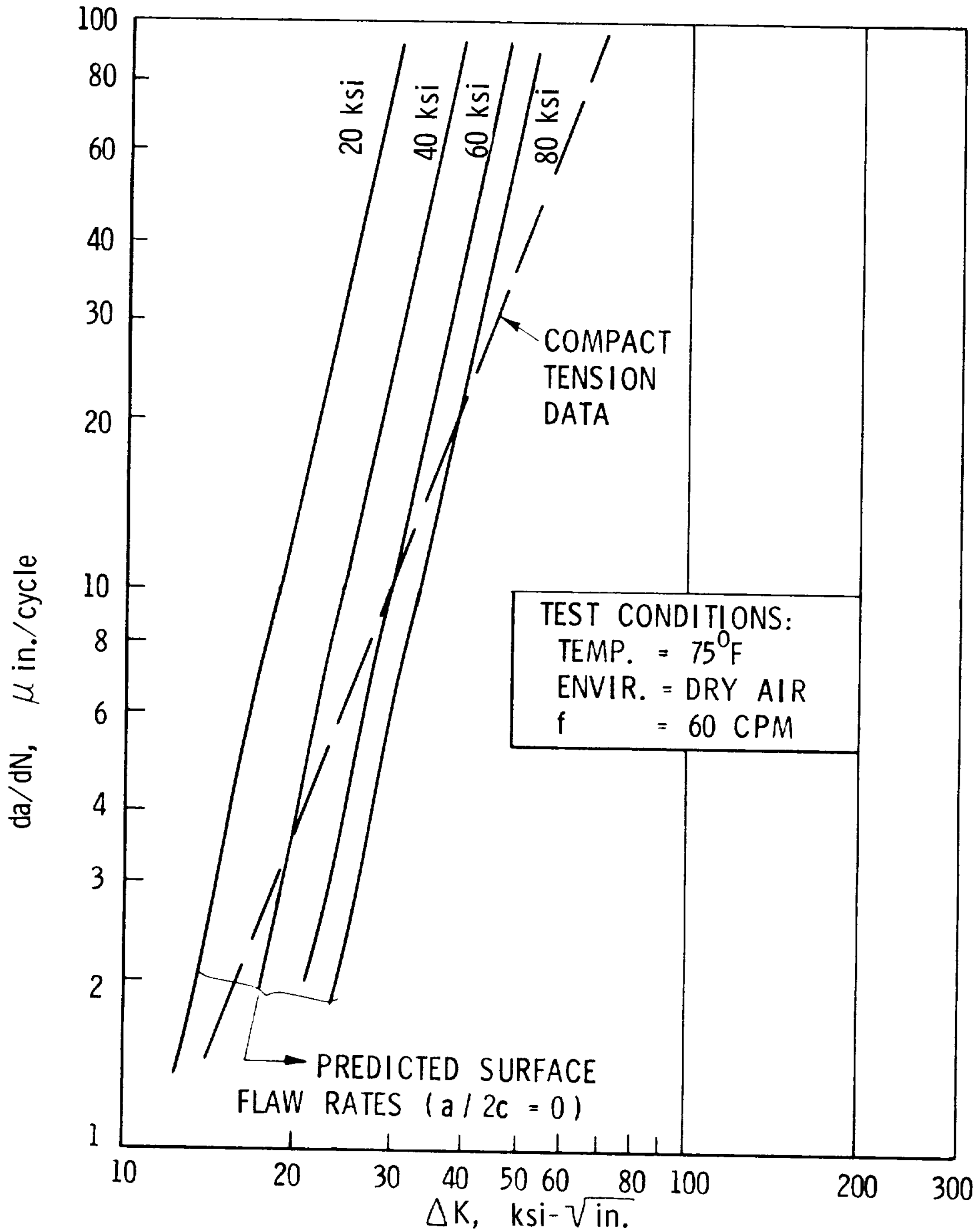
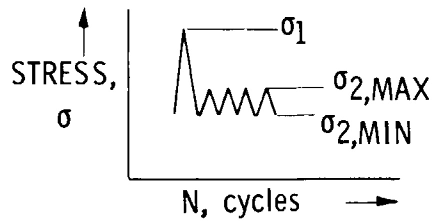
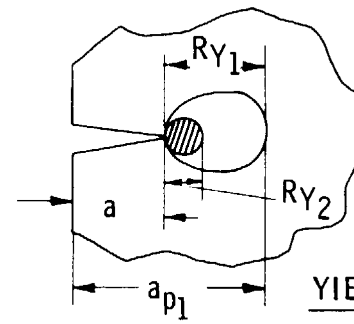


Figure 26.- Surface flaw data adjusted to $a/2c = 0$ and compared with compact tension data from reference 11.



LOADING



YIELD ZONE SCHEMATIC

SUMMARY OF AFFDL MODEL (REF. 10):

OVERLOAD σ_1 RETARDS da/dN for $\Delta\sigma_2$

R_Y = SIZE OF YIELD ZONE

a = INITIAL CRACK SIZE

a_c = CRACK SIZE AT ANY TIME AFTER THE OVERLOAD

a_{p_1} = EXTENT OF THE PLASTIC ZONE R_{Y_1} DUE TO OVERLOAD σ_1

$$= \frac{K_1^{\,2}}{2\pi(\sigma_{YIELD})^2} + a \qquad \text{(PLANE STRESS)}$$

σ_{ap} = STRESS REQUIRED TO PRODUCE A PLASTIC ZONE OF EXTENT a_{p_1} FOR ANY CRACK SIZE $a_c + R_{Y_c} < a_{p_1}$

$$= \sigma_{YIELD}\sqrt{\frac{2(a_{p_1} - a_c)}{a_c}}$$

σ_{red} = EFFECTIVE RESIDUAL STRESS CAUSED BY OVERLOAD – VARIABLE WITH a_c

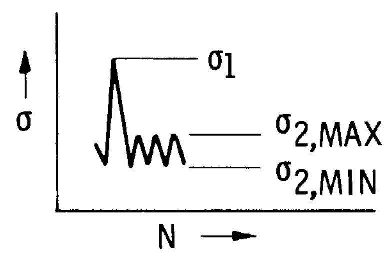
$$= \sigma_{ap} - \sigma_{2,MAX}$$

EFFECTIVE MAXIMUM AND MINIMUM STRESSES AND LOAD RATIO R COMPUTED FROM

$$\sigma_{2,MAX,EFF} = \sigma_{2,MAX} - \sigma_{red}; \quad \sigma_{2,MIN,EFF} = \sigma_{2,MIN} - \sigma_{red}$$

Figure 27.- Yield zone concept for retardation.

MODE 1



$\sigma_1 > \sigma_2$

$\sigma_{2,MIN,EFF} = \sigma_{2,MIN} - \sigma_{red} < 0$

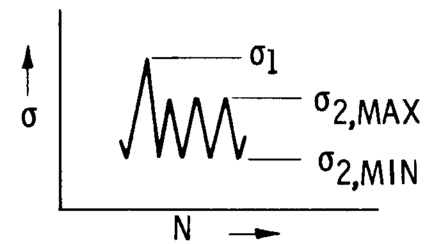
$\therefore \quad \sigma_{MIN,EFF} = 0$

$\Delta K_{2,EFF} = K_{2,MAX,EFF}$

$R_{2,EFF} = 0$

RETARDATION DUE TO REDUCED ΔK, R

MODE 2



$\sigma_1 \approx \sigma_2$

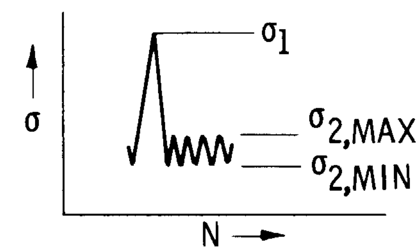
$\sigma_{2,MAX,EFF} \neq \sigma_{2,MIN,EFF} \neq 0$

$\therefore \quad \Delta K_{2,EFF} = \Delta K_2$

$$R_{2,EFF} = \frac{K_{MIN,EFF}}{K_{MAX,EFF}}$$

RETARDATION DUE ONLY TO REDUCED R

MODE 3



$\sigma_1 >> \sigma_2$

$\sigma_{2,MAX,EFF}$ AND $\sigma_{2,MIN,EFF} < 0$

$\therefore \quad \sigma_{2,MAX,EFF} = \sigma_{2,MIN,EFF} = 0$

$\Delta K_{2,EFF} = R_{2,EFF} = 0$

RETARDATION MAXIMUM - GROWTH STOPPED

Figure 28.- Modes of retardation - AFFDL model (ref. 10).

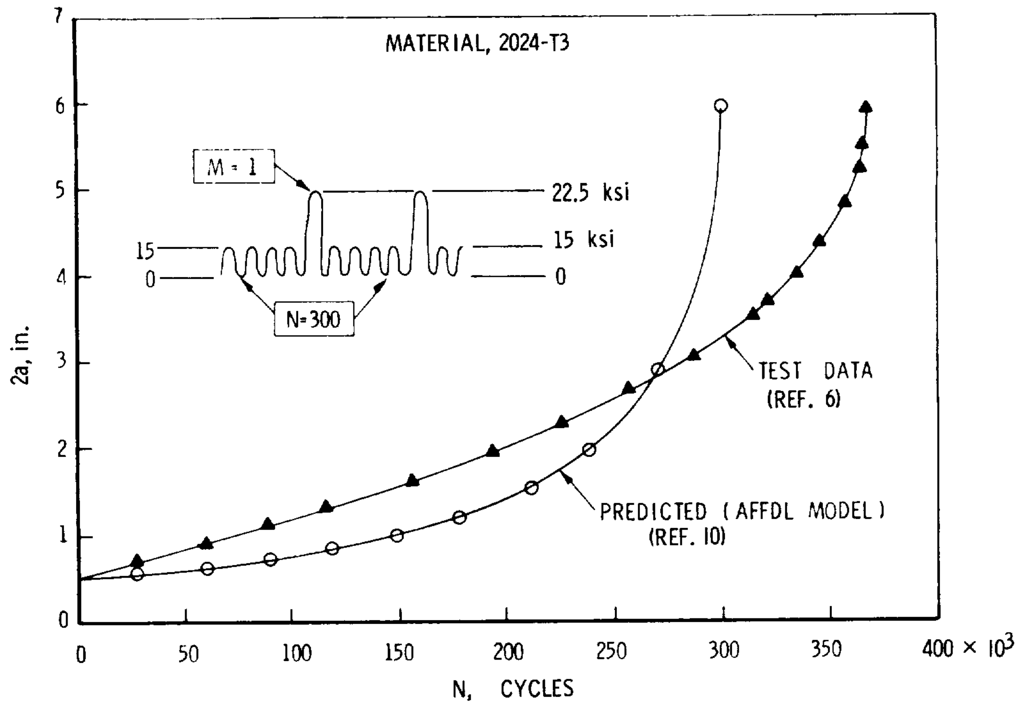


Figure 29.- Comparison of test and predicted crack growth. Single overload.

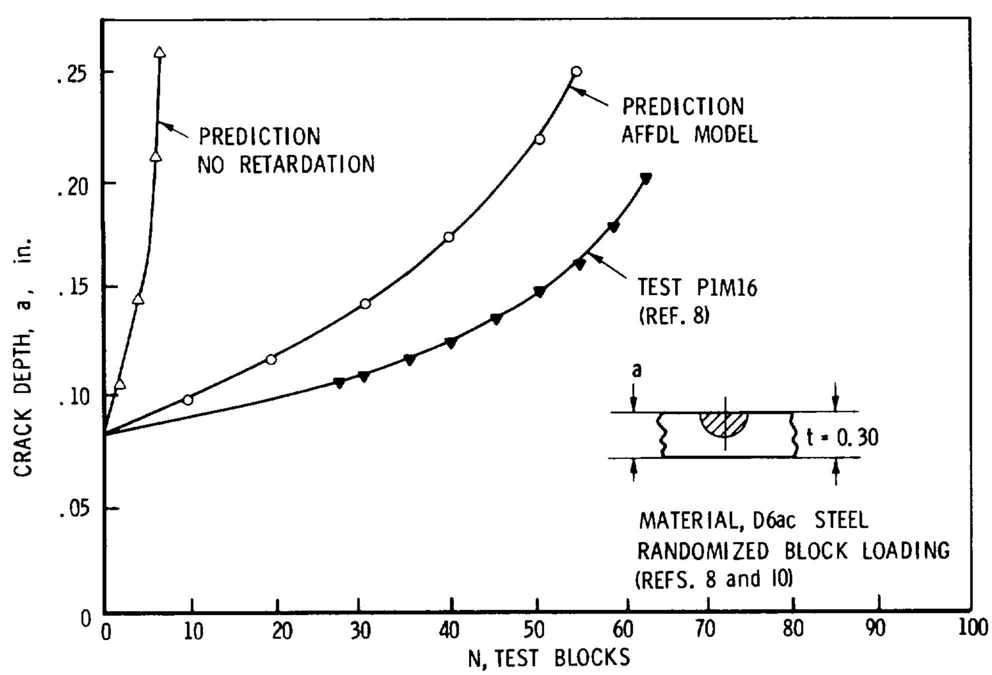


Figure 30.- Comparison of test and predicted crack growth. Randomized block spectrum loading.