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DAMAGE TOLERANCE FOR COMMUTER AIRCRAFT

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SUMMARY

The damage tolerance experience in the United States Air Force with military aircraft and in the commercial world with large transport category aircraft indicates that a similar success could be achieved in commuter aircraft. The damage tolerance process is described for the purpose of defining the approach that could be used for these aircraft to ensure structural integrity. Results of some of the damage tolerance assessments for this class of aircraft are examined to illustrate the benefits derived from this approach. Recommendations are given for future damage tolerance assessment of existing commuter aircraft and on the incorporation of damage tolerance capability in new designs.

INTRODUCTION

The Aircraft Structural Integrity Program (ASIP) was initiated in 1958 by the United States Air Force in response to fatigue failures which occurred in service aircraft in the fifties (ref. 1). The original version of the ASIP recognized the need for a fatigue analysis and test program for fatigue life validation. The fatigue methodology adopted was a reliability based approach commonly referred to as the "safe life" method. This approach was used in the development of USAF aircraft in the sixties. It was found that this approach did not adequately address the threat of manufacturing and service induced damage. The safe life approach did not preclude the use of "brittle" materials at high stresses, and therefore, the structures were not tolerant to defects in the structure derived from the manufacturing process or from in-service maintenance of the aircraft. Consequently, aircraft designed and qualified by this approach suffered premature failures when operated in the service environment. There were many aircraft in this category, such as the F-5, KC-135, B-52 and F-111. Each of these aircraft suffered failures in service at times considerably less than the laboratory demonstrated safe life. Service failures resulted in costly redesign and modification programs on many aircraft. Further, the safe life approach did not include a means for establishing an inspection program. There was no rational process for determining where, when, how and how often inspections should be accomplished to maintain operational safety. These deficiencies in the safe life approach motivated the USAF to adopt the damage tolerance approach. After this approach was applied to the design of the B-1 bomber and to the modification program of the C-5 and F-4 aircraft, it was formally established in MIL-STD-1530A released in December of 1975. Since that time all of the major weapons in the USAF have had their Fleet Structural Maintenance Plans (i.e., the how, when and where to inspect or modify the aircraft to maintain safe and economical operations) updated by the damage tolerance approach.

There has been a somewhat different path to damage tolerance in the commercial world. The Martin 202 wing failure in 1948 and the deHavilland Comet fuselage failures in 1954 led to the adoption of the fail-safe design concept. It was concluded after the Avro 748 wing failure in 1976 and the Boeing 707 horizontal tail failure in 1977, where both aircraft had been designed to be fail-safe, that fail-safety alone was not adequate to protect an aircraft from catastrophic failure. It was further concluded that an inspection program based on damage tolerance should be added to the maintenance program. In the United States, this was accomplished in 1978 with the release of Amendment 45 to FAR Part 25.

The success of the damage tolerance approach for military aircraft and for large transport category aircraft has firmly established this process for maintaining safety. The question is: "should the damage tolerance approach be extended to the small commuter aircraft that are in fare paying passenger service?" A partial answer to this question can be obtained from an examination of the history of the development of commuter class aircraft. Some of these aircraft have been certified under FAR Part 23 requirements and some have been certified under FAR Part 25 requirements. However, not all of the Part 23 aircraft or the Part 25 aircraft have been certified under the same rules. Improvements were made in the basic certification requirements as a result of operational aircraft experience. Examples of these changes for Part 23 aircraft include the following: in 1953 the rule was modified to include the pressure cabin; in 1969 the wing and wing carry thru were added and in 1989 the empennage was added. Consequently, there are aircraft operating in commuter service that have little, if any, assessment of their resistance to cracking by either the safe life or damage tolerance method.

The application of the safe life approach for commuter aircraft has the same difficulties as those experienced by the Air Force - namely from the use of brittle materials in locally high stress applications that may be subject to accidental damage in production or service. In addition, there is no mandatory requirement for a fatigue test of commuter aircraft. In those cases where testing was not accomplished, the safe life was determined based on the results of a linear cumulative damage analysis that was divided by a scatter factor. The scatter factor is a number greater than one that is intended to account for uncertainty in the analysis as well as the material and manufacturing quality variability. As indicated above, some of the older aircraft developed under FAR Part 23 have been certified without even a fatigue analysis. There are no life limits or a rationally based inspection program on these older aircraft, and in many cases, there is currently no basis for establishing a life limit or a rationally based inspection program.

However, there have been few commuter aircraft accidents attributed to structural failure. This is believed to be the result of the fact that the generic stresses in these aircraft are generally low. The strength requirements for maneuver and gust combined with the normal conservatisms that are used in stress analyses cause these stresses to be low. The situation may be different for local areas where the absence of detailed stress analysis or adequate testing could lead to failure. At the present time, the risk of this occurring is unquantified.

It is the purpose of this report to discuss the damage tolerance approach for commuter aircraft and how the damage tolerance process could be used to develop an inspection program or alternatively life limits. The findings of the available damage tolerance assessment efforts will be summarized and recommendations made for damage tolerance analyses of the structural components of both existing and new commuter aircraft.

THE DAMAGE TOLERANCE APPROACH

The approach that has been used for damage tolerance assessments in the Air Force has not significantly changed over the last eighteen years. There are, however, differences in the emphasis that has been placed on the individual tasks among the aircraft that have been assessed for damage tolerance capability. This is a natural consequence of the differences among the aircraft and the differences in the data base available at the start of the assessment. As would be expected, many of the tasks that are performed for the damage tolerance assessment are precisely the same as that accomplished for the classical fatigue analysis which is the basis for establishing design stresses for aircraft that use the safe life method. The timing of the damage tolerance assessments that have been performed under sponsorship by the Air Force is shown in figure 1. As seen in this figure, both large and small aircraft have been subjected to an assessment. The effort required for an assessment generally increases as the size of the aircraft increases. Also, it depends heavily on the data base available at the time of the assessment and the criticality of the structure to the usage environment. For example, the assessment on the F-4 required more effort than normal because the loads data had not been verified by a flight loads survey. A flight loads survey was conducted on this aircraft to provide this needed data. Also, the assessment performed on the T-38 for operations in the severe usage associated with lead-in-fighter training took considerably longer than the assessment of the F-15 in a usage for which it was designed even though the structure of the F-15 is more complex than the T-38. The damage tolerance process and the details of many of the assessments performed by the Air Force are discussed in reference 2. As discussed in this reference the process was divided into the following six tasks:

-- Identification of critical areas

-- Development of the stress spectrum

- Establishment of the initial flaw criterion

-- Establishment of the operational limits

-- Development of the Force Structural Maintenance Plan

-- Development of a fracture mechanics based tracking program

The first five of these tasks are illustrated in figure 2.

It was found for all of the damage tolerance assessments that the identification of critical areas (i.e., those areas of the structure where an inspection or modification is expected to take place during the life of the structure) is augmented from a knowledge of the service problems, laboratory fatigue testing, stress analyses and strain surveys from ground and flight tests. For many of the older commuter aircraft there have been service cracking problems. The manufacturer and the FAA have addressed these areas through airworthiness directives. These areas would be a starting point for the analyst to locate potential adjacent area cracking points. The fatigue test data base, however, for most of the commuter aircraft is nonexistent. Also, in most cases there is little if any data on the stress distributions on the structural components. Therefore, the identification of critical areas of the structure will need to be determined from analytically derived stresses from areas of the structure where there are either stress concentrations or materials used that have poor fracture properties. The lack of a fatigue test or a strain survey data base for a complex military aircraft structure would be a significant detriment to the successful completion of an assessment. However, the inherent simplicity of most commuter aircraft structures makes this task considerably easier. Further, for many of the commuter aircraft, there has been a classical fatigue analysis performed. These fatigue analyses required that the critical areas of the structure for fatigue be located. The areas of the structure that are critical for safe life are also critical for damage tolerance. There could, however, be some areas that are critical for damage tolerance that are not critical for fatigue. Therefore, an existing fatigue analysis is extremely helpful, but not always sufficient for the damage tolerance analysis. Therefore, the analyst needs to initially identify the critical areas in somewhat conservative fashion and then reduce the number for a detailed examination by a preliminary screening process. This preliminary screening activity may involve some finite element modeling followed by a simplified evaluation of the stress spectra and a fracture analysis.

The stress spectrum development was typically the most difficult to perform of all the tasks for the damage tolerance assessments. Much of the difficulty arises because of the need for accuracy. Accuracy of the stress spectrum is important for the commuter aircraft whether the analysis is for safe life or for damage tolerance. As stated in reference 2, there are the following three phases in the development of the stress spectrum for an aircraft:

-- Determination of the aircraft usage
-- Determination of the external loads
-- Determination of the stresses

With some exceptions, there is considerable commonality in the usage of commuter aircraft. The maneuvering load factors are, in general, similar to the large transport category aircraft. Also, the maneuver loads, when compared to low level turbulence, are typically benign. The major usage differences that are significant are number of hours per flight and altitude flown. The number of hours per flight determines the number of ground-air-ground cycles and the altitude flown determines the turbulence spectrum. It is believed that the atmospheric turbulence has been characterized well enough to provide an adequate spectrum for these aircraft.

The second phase in the development of the stress spectrum for the airframe is to determine the external loads on the structure for a given maneuver or gust condition. This is a problem for many commuter aircraft because of the fact there has been little, if any, effort expended to perform a flight loads survey to determine the loads empirically. This may not be a problem for the wing since the wing configurations on commuter aircraft generally lend themselves to analysis. However, this problem may be particularly significant for the empennage and consequently on the aft fuselage. The interpretation of the results of the final analyses for obtaining operational limits should be done with a recognition of the possible shortcomings of this phase of the stress spectrum development.

The third phase of the stress spectra development is to determine the detail stresses in the critical areas resulting from the external loads. It was found in all of the military aircraft damage tolerance assessments that knowledge of the detail stresses was lacking. This was due primarily to the lack of computer capability when these aircraft were designed. The qualification process substituted laboratory testing for detailed analyses. It is essential for either a safe life or a damage tolerance analysis that there be an adequate understanding of the stresses in the structure. With the computational capability that exists today, this is not a significant inhibitor to the process. Further, the simplicity of the commuter aircraft structures makes these analyses economically viable.

The desired end product of the stress spectra development is a flight-by-flight spectrum of stresses for each of the critical areas. The damage tolerance assessments that have been performed have shown that the ordering of the flights in the spectrum is not significant if the ordering is random. This has also been found to be true for the commuter aircraft stress spectra. In addition to the development of the baseline spectra which are intended to represent average usage, it is desirable to generate variations of these spectra that represent the expected range of usage of the vehicle. The successful completion of the fracture analyses and tests with these spectra provide confidence that there is a sound data base for use in the development of a maintenance plan for the aircraft.

The third task of the damage tolerance assessment is to determine the initial flaw size to be used for the subsequent fracture analysis. The data derived from operational experience generated over the last eighteen years has shown that for fastener holes in airframe structure, an initial corner flaw of 1.27 millimeter radius is adequate to provide safety of flight. There is no known case where an aircraft has failed in less than the time required for this size defect to grow to critical size. It has also been found that for airframe structure that a semicircular surface flaw with a radius of 3.175 millimeter is similarly adequate. No studies have been accomplished to show that the size of the aircraft has an influence on the probability distribution of flaws. A large aircraft would logically have more opportunities per aircraft for a rogue defect. It is not clear, however, how to adjust the size of the rogue defect for the smaller aircraft, and consequently this has not been done. Therefore, the initial flaw sizes (i.e., at the time the aircraft is manufactured) above are suggested for a damage tolerance assessment of a commuter aircraft. Where needed, a continuing damage size of 0.127 millimeter corner flaw in a hole should be used. It is further recommended that the in-service inspection flaw size be based on a probability of detection of 0.9 with a 95 percent confidence. In-service inspections have been a problem for commuter type aircraft since the safe life certification process places no emphasis on inspectability, and consequently there are many details that are difficult to inspect. Some of these problems have been diminished because of the recent developments in low frequency eddy current that will permit reliable inspections of structure that would have previously been considered virtually noninspectable. This technique is commercially available and performs inspections without the removal of the fasteners. It is especially suitable for many locations in commuter aircraft because the depth of penetration required for second and third layer inspections is relatively small compared to its capability.

The operational limits for commuter aircraft are derived from the fracture analyses and tests based on the stress spectrum for each critical area in combination with the initial flaw to be used for that area. As reported in reference 2, the predominant problem with aircraft is fastener holes. Thus, there is no problem in obtaining adequate stress intensity solutions for the critical areas of the structure. The test verification of the analytical flaw growth is normally an essential part of the process to obtain the spectrum interaction effects which may be significant in the determination of the operational limits. This is initially accomplished at the coupon level where the test specimens have the same material, thickness, hole geometry and, if necessary, fastener load transfer. The full-scale fatigue tests are used, when possible, for final validation of the crack growth analysis.

The Fleet Structural Maintenance Plan (FSMP), as indicated previously, is the plan that describes the inspection and modification program for the commuter aircraft during its anticipated operational life. The damage tolerance process is shown graphically in figure 3. The inspection program is determined from the calculation of the operational or safety limits. The initial safety limit for a fastener hole, for example, is the unfactored crack growth life from an initial 1.27 millimeter crack to critical crack length at limit load. It is recommended that the safety limit be divided by a factor of two to determine the inspection threshold. For a structure that is noninspectable or is not planned to be inspected, the safety limit could be used as a life limit. This should be done with care on those structures where there is reason to question the analysis because there has not been a flight loads survey to validate the loading. In most of the damage tolerance assessments performed on military aircraft there were many parts found that were extremely critical. In many cases these critical areas were revealed through in-service failures and through full-scale fatigue tests. As indicated in reference 2, these problems were addressed individually and an approach was determined that preserved safety at minimum cost.

DAMAGE TOLERANCE EXPERIENCE WITH OLDER AIRCRAFT

B-52 Companion Trainer

In 1980, the USAF was interested in procuring a B-52 companion trainer aircraft. This aircraft would be used to reduce the training requirements in B-52 aircraft and therefore would be required to fly its missions. These missions included a considerable amount of severe low altitude usage. The candidates for this aircraft included aircraft whose structural integrity was unknown to the Air Force. Previous success with damage tolerance on other aircraft motivated them to make an initial damage tolerance assessment of these aircraft to see how well they performed against this criterion. The following aircraft were examined:

-- British Aerospace 125 Series 700
-- Lockheed S-3A
-- Israeli Aircraft Industries Westwind
-- Dassault Mystère 10
-- Dassault Mystère 20
-- Gates Learjet 35
-- Rockwell Saberliner 65

The contractors were asked to provide basic information on the material used and the geometry of critical details in the wing, fuselage and empennage as well as one g stress and stress per g. These aircraft were assessed against an executive mission which was representative of commuter aircraft operations and also against the severe B-52 mission. The maneuver and gust exceedance functions for these missions were converted to a flight-by-flight spectra for the subsequent fracture analysis. The findings of this assessment were that there was considerable scatter in the material selection and in the capabilities of these aircraft. Three of the candidates were found to have a damage tolerance derived safety limit of over 100,000 hours for the executive mission and these same three had an adequate safety limit for use in the B-52 mission. One of them had a small safety limit in the executive mission and three of the candidates had a less than desirable safety limit in the B-52 mission. One of these aircraft was unacceptable for the B-52 mission. This assessment showed that a modest effort could provide a significantly better basis for screening candidate aircraft for a given usage. It also demonstrated that the generic stress levels chosen for commuter aircraft provided a good damage tolerance capability. The reason that some aircraft fell short of the desired goal was because of the lack of damage tolerance capability in some of the design details.

T-37

As a result of the anticipated cancellation of the T-46, the Air Force initiated a review in April of 1986 of the T-37 aircraft to determine its potential for a life extension. The T-37 is an aircraft used by the Air Force for initial pilot training of its air crews. Consequently, it is subjected to severe usage as indicated by the load factor exceedance function shown in figure 4. The development of the aircraft was initiated by the Air Force with a contract with Cessna in January of 1953. The material selection included 7075-T6 forgings and extrusions, 7075-T73 extrusions, and 2024-T42 sheet. The first aircraft was delivered in 1955 with a gross weight of 6600 pounds. The initial fatigue life investigations were based on a required service life of 8,000 hours and 20,000 landings. In 1968, a front wing spar failed on a T-37 with approximately 6,000 flight hours, and another aircraft suffered a failure in the horizontal tail.

In addition, there have been numerous cases of stress corrosion cracking of the 7075-T6 forgings. As a result of structural modifications and a full-scale fatigue test, the life of the aircraft was extended to 18,000 hours based on the safe life approach. By April of 1986, the weight of the 644 remaining aircraft had been increased to 7200 pounds. The high time aircraft had 15,322 hours and the average aircraft had 11,748 hours. The oldest aircraft was approximately 28 years old and the average aircraft was approximately 23 years old.

The Air Force felt that there was not an adequate understanding of the structure of this aircraft from which to determine what modifications should be incorporated. Also, the inspection program for the critical details and the life remaining for the structure were not well established. To obtain a solution to these problems, the Air Force contracted with Cessna for a damage tolerance assessment. Although this aircraft was designed for acrobatic maneuvers that are outside the capability of the typical commuter aircraft, it was a small aircraft with construction that was similar to that used in commuter aircraft. Therefore, the damage tolerance assessment for this aircraft should encounter similar technical challenges as the damage tolerance assessment of any commuter aircraft. For this aircraft, the data base for the damage tolerance assessment was augmented by a flight loads survey to validate the loads on the wing and the empennage. There were forty-nine candidate critical areas identified by a screening process. Preliminary analyses were able to remove all but eighteen of the areas from consideration. Detail analyses were performed on these eighteen locations. Many of the critical locations in the aircraft were in the wing carry thru and in the fuselage. These areas had been subjected to a number of modification programs over the years, some of which involved the addition of steel straps. Some of these structural details were found to be extremely critical. The results of the T-37 assessment showed that the safety limits of the structure ranged from less than 1000 hours to over 60,000 hours. The safety limits of some of the details are shown in figure 5. It was found that all of these aircraft were operating with flight hours in excess of the safety limits. By policy, the Air Force would have grounded these aircraft until an inspection had been performed. Since some of these inspections required considerable downtime and continued operation of these aircraft was essential for meeting training requirements, the aircraft remained in service based on a risk assessment. The inspections were performed on an accelerated basis. The damage tolerance analysis effort clearly identified those areas that needed modification in order to extend the life of this aircraft.

Variable Stability Learjet

The test pilots school at Edwards Air Force Base in California uses an aircraft with a capability for variable stability as a part of its training program. The B-26 aircraft that had been used for this activity failed catastrophically from a crack in the wing structure in March of 1981. Consequently, the school procured a Gates Learjet model 24D to be modified to perform this function. To preclude a possible reoccurrence of a structural failure, the school contracted with Gates to perform a damage tolerance assessment of this aircraft. The Learjet was certified by the FAA under Part 25 of the Federal Air Regulations. Although fail safety was incorporated in the design, the damage tolerance analysis was performed on the basis of slow crack growth.

Since the usage was known to be more severe than the design usage for this aircraft, a program was initiated to collect data from actual operations of the aircraft. This was accomplished with an eight channel recorder that recorded both accelerations and strains for 334 flights (560 hours). The comparison of the normal acceleration for design and the variable stability usage is shown in figure 6.

Based on the recorded data, a flight-by-flight spectrum was developed for the damage tolerance analysis. The analysis included the wing, empennage and engine mounts. The wing analysis revealed that the initial safety limits for all of the points analyzed exceeded 100,000 flight hours. In addition, linear cumulative damage was used to calculate the mean lives of the wing critical locations. It was found that in most cases these mean lives were smaller than the safety limits. However, they were also approximately 100,000 hours. The contractor recommended that at 3,600 hours the wing be removed from the aircraft and subjected to visual, eddy current and X-ray inspections. The motivation for this inspection was apparently driven by a recommendation in the basic maintenance program for the aircraft. It was not derived from rational damage tolerance considerations.

The engine mount on the Learjet was a welded steel part manufactured from 4130 heat treated to 1,240-1,450 MPa. The linear cumulative damage analysis for the engine mount indicated that the part had a mean life of 764,000 hours. The damage tolerance analysis from a 1.5 millimeter corner flaw in a lightening hole revealed that the safety limit was less than 2,500 hours. This significant difference between the safe life and damage tolerance analysis is typical of materials that have an inherent sensitivity to flaws. The F-111 experience has shown that steel parts are subject to manufacturing damage as well as in-service damage. Unanticipated failures in cold proof tests have demonstrated this fact.

Piper PA-42

Another damage tolerance analysis of a commuter aircraft structural detail was conducted in 1990. This was a wing lower surface aluminum detail from the Piper (PA-42) Cheyanne IIIA. This detail was chosen because Piper had developed the data base from which this analysis could be easily accomplished. Piper had created a block spectrum of stresses for 15,000 hours of 1,000 hour blocks. This data was used to generate a flight-by-flight spectrum from which a damage tolerance assessment could be accomplished. The comparison of the stress exceedance function for the PA-42 usage and the C-5 usage is shown in figure 7. The similarity of these functions is remarkable.

It was found that stresses in this generic location in the wing were quite low and safety limit from a fastener hole in this aluminum structure was 52,500 hours. The crack growth function is shown in figure 8. This result was compared with results obtained with safe life methodology. First, the spectrum was analyzed by linear cumulative damage based on the S-N data in MIL-HDBK-5E. It was found that the unfactored life was 220,000 hours for a K_t of four in 2024-T3 aluminum. For a K_t of five for the same material, the unfactored life was found to be 26,400 hours. The decision on the selection of the effective K_t normally must be made from test data. It cannot be made on the basis of the geometric stress concentration only. This example illustrates the uncertainty that exists when test data is not available. The spectrum was also assessed based on the S-N curve from reference 3. It was found that the unfactored life was 29,200 hours. If a scatter factor of eight was used as recommended in reference 3, then the life would be 3,650 hours for this location. It is noted that the K_t of five data from MIL-HDBK-5E provides an unfactored life that is close to the unfactored life from reference 3. A study was performed to examine this location supposing that a 4340 steel strap had been used as a repair. The analysis of this steel part using an initial crack of 1.27 millimeters showed that the safety limit was 1,320 hours. The crack growth function for the steel is shown in figure 9. However, based on the S-N data in MIL-HDBK-5E for this same material at a K_t of 3.3, the unfactored life was found to be 121,000 hours and the factored life based on a scatter factor of eight was 15,125 hours. This illustrates the flaw sensitivity and the lack of damage tolerance for the steel. It is seen that the safe life approach would not have rejected steel in this location for an aircraft life of 15,000 hours with no inspections.

DAMAGE TOLERANCE ANALYSES OF OLD AIRCRAFT

A preliminary effort has been made to establish the urgency for a damage tolerance assessment of some of the older commuter aircraft. Although some aircraft in this category appear to have maintenance plans and retirement times that are adequate to protect their safety, there are others where the risk of failure is not known even in qualitative terms. The aircraft in this latter category should be subjected to damage tolerance assessments as soon as possible. However, all of the commuter aircraft should have their maintenance plans based on the damage tolerance approach. For structures that are noninspectable, the retirement time should be based on one half of the safety limit as defined above unless there has been a validation of the loads through a flight loads survey. In this case, the retirement time should be based on the safety limit.

DAMAGE TOLERANCE ANALYSES OF NEW AIRCRAFT

The emphasis in the structural design of new commuter aircraft should be to eliminate, as far as practical, the need for in-service inspections to be performed to protect flight safety. However, it should be the intent that these aircraft be designed to be inspectable. There should be inspections to search for corrosion damage, in addition to inspections performed on a sampling basis to examine the structure for unexpected cracking.

It appears viable to design a commuter aircraft to an initial flaw concept and select the materials and establish the stresses such that no inspections are required. The stresses for the current population of commuter aircraft are such that if this were done there would be little, if any, weight impact of designing the structure to be free of inspections. The safety limit should be set at twice the desired service life of the aircraft. If life extension is required, then the inspection process would be initiated.

CONCLUSIONS

The results shown above for commuter aircraft indicate that many of the parts of the structure are damage tolerant based on the current certification program. However, as seen in the T-37 assessment, the Gates Learjet assessment and the sensitivity study on the Piper PA-42, there could be details in the structure that could pass the safe life requirements and still be a potential safety hazard. The technology for evaluating existing commuter aircraft is well established. The capability exists to assess each of the aircraft in this category and make a determination on life limits or an inspection program to extend the life. The nondestructive inspection capability exists such that it is likely that an inspection program can be identified that will permit "retirement for cause" rather than retirement by flight hours.

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3. Fatigue Evaluation of Wing and Associated Structure on Small Airplanes. AFS-120-73-2, Airframe Branch, Engineering and Manufacturing Division, Federal Aviation Administration Flight Standards Service, May 1973.

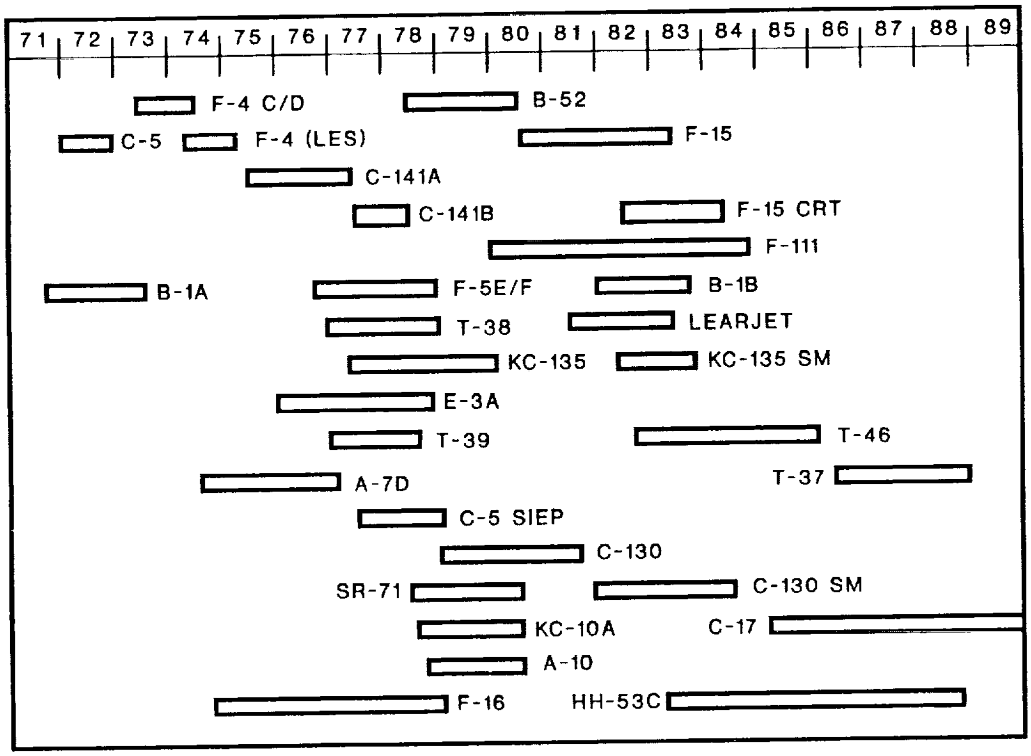

Figure 1. - Damage Tolerance Assessments of Aircraft

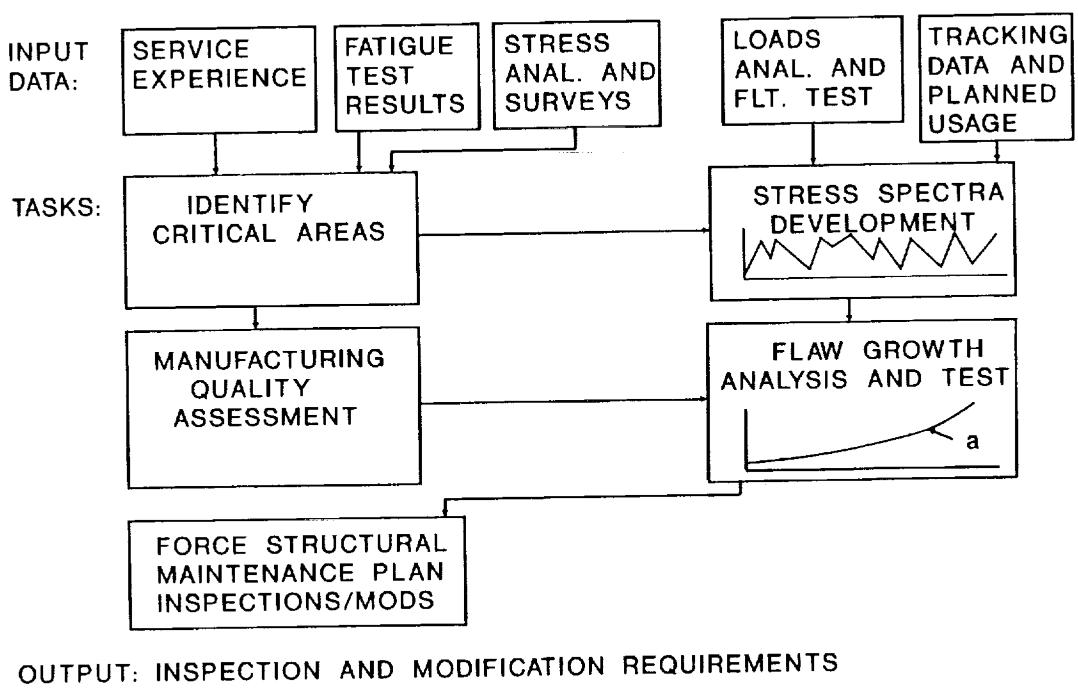

Figure 2. - Damage Tolerance Assessment Approach

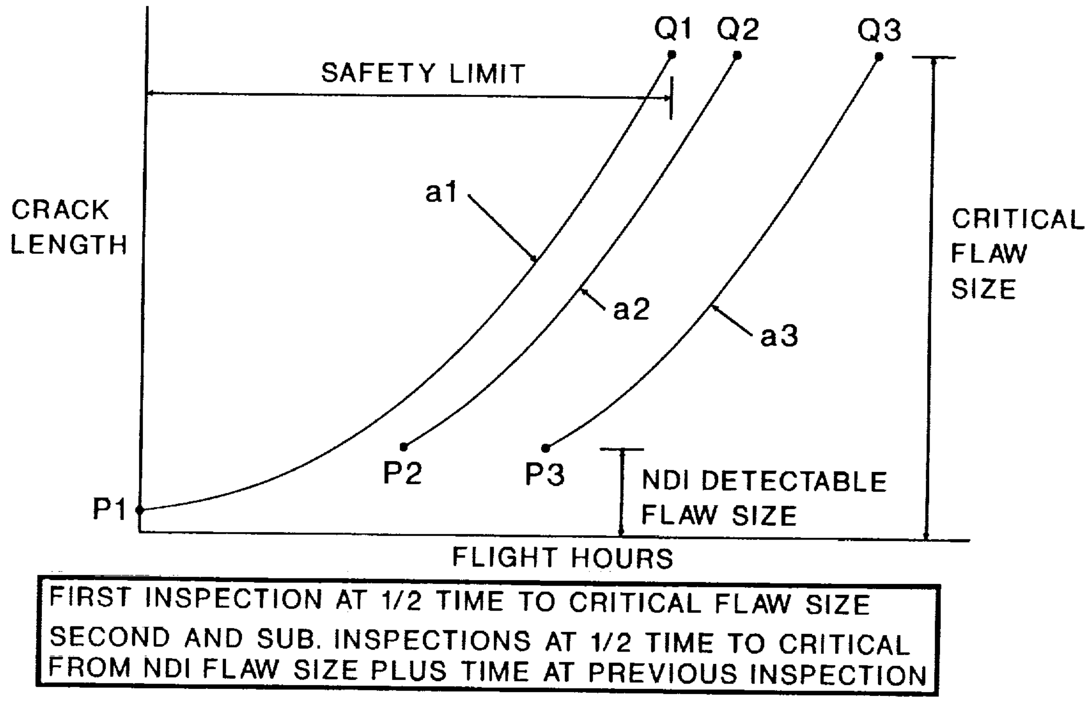

Figure 3. - Damage Tolerance Process - Slow Crack Growth

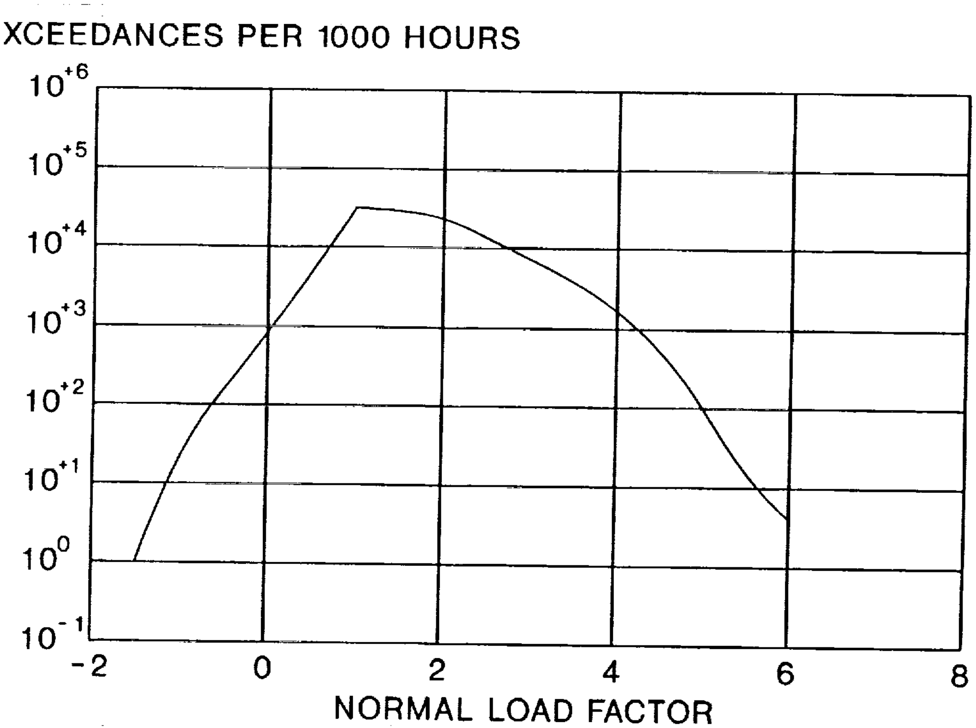

Figure 4. - Load Factor Exceedance for the T-37

LOCATION	MATERIAL	SAFETY LIMIT
WING CARRY THRU		
FORWARD UPPER CAP	2024-T42	›60,000
FORWARD LOWER CAP	2024-T42	9,822
FORWARD FORGING	7075-T6	1,930
FORGING LUG HOLE	7075-T6	874
WING FRONT SPAR		
LOWER CAP LUG	7075-T73	2,152
LOWER CAP FLANGE	7075-T73	32,150
LOWER CAP LUG HOLE	7075-T73	2,747
HORIZONTAL STABILIZER		
FRONT SPAR CAP	7075-T6	2,172
FUSELAGE CANOPY RAIL		
AFT LATCH CUT-OUT	2024-T42	›60,000
ATTACHMENT HOLE	2024-T3	›60,000

Figure 5. - Safety Limits for T-37 Details

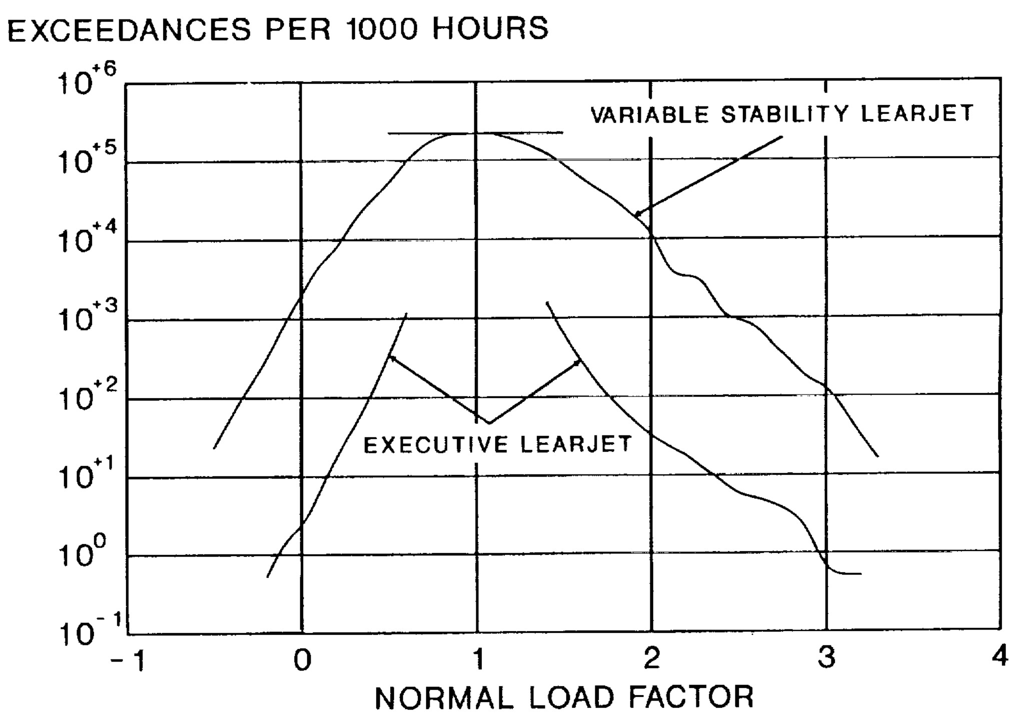

Figure 6. - Load Factor Exceedance for the V/S Learjet

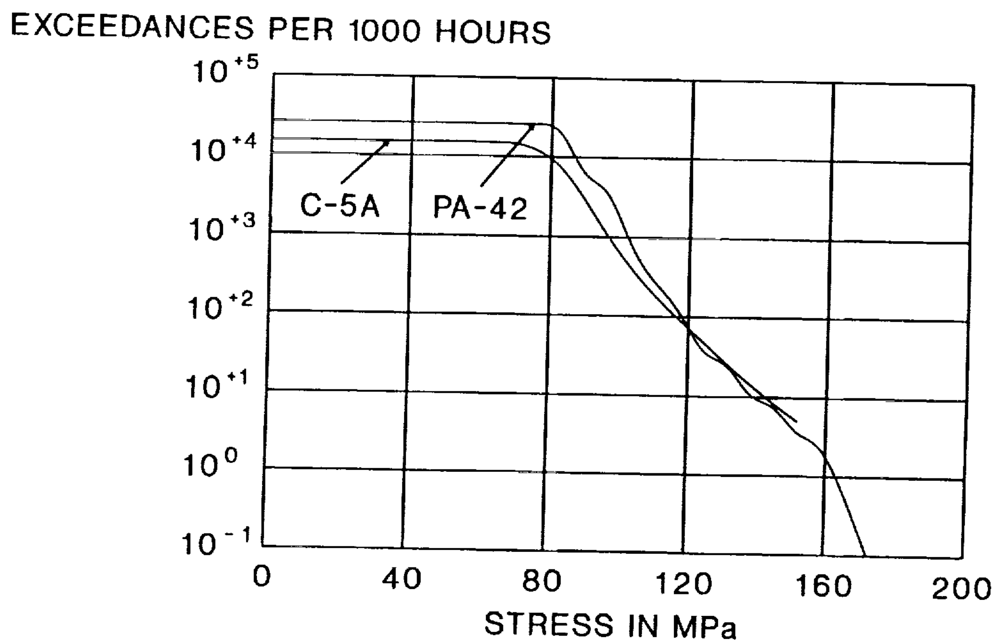

Figure 7. - Stress Exceedance Comparison for C-5A and PA-42

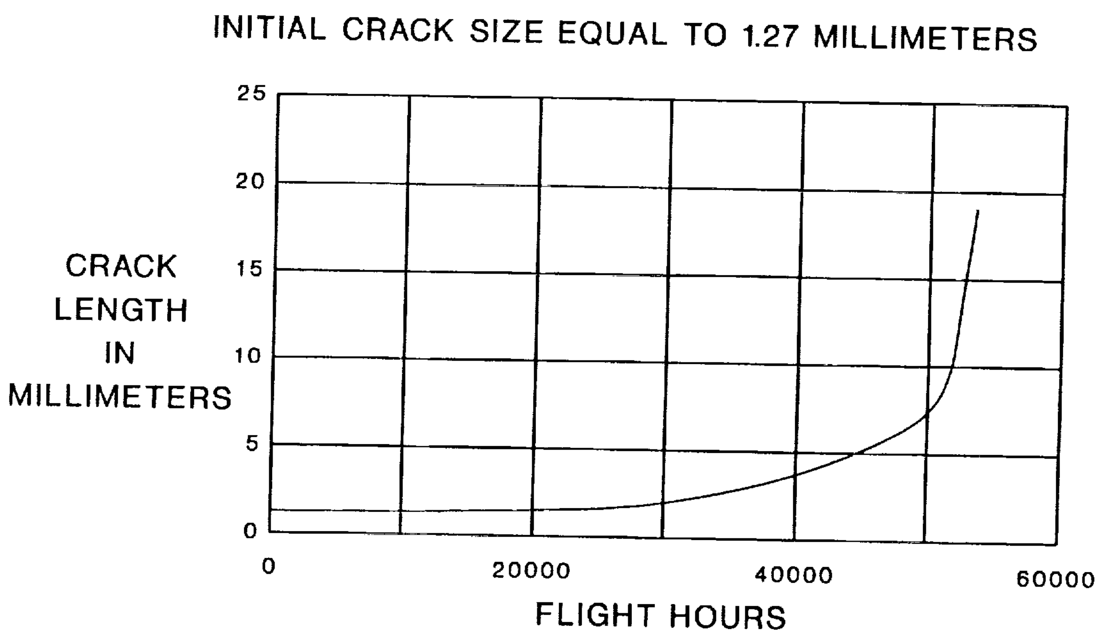

Figure 8. - Crack Growth Function for PA-42 - Aluminum

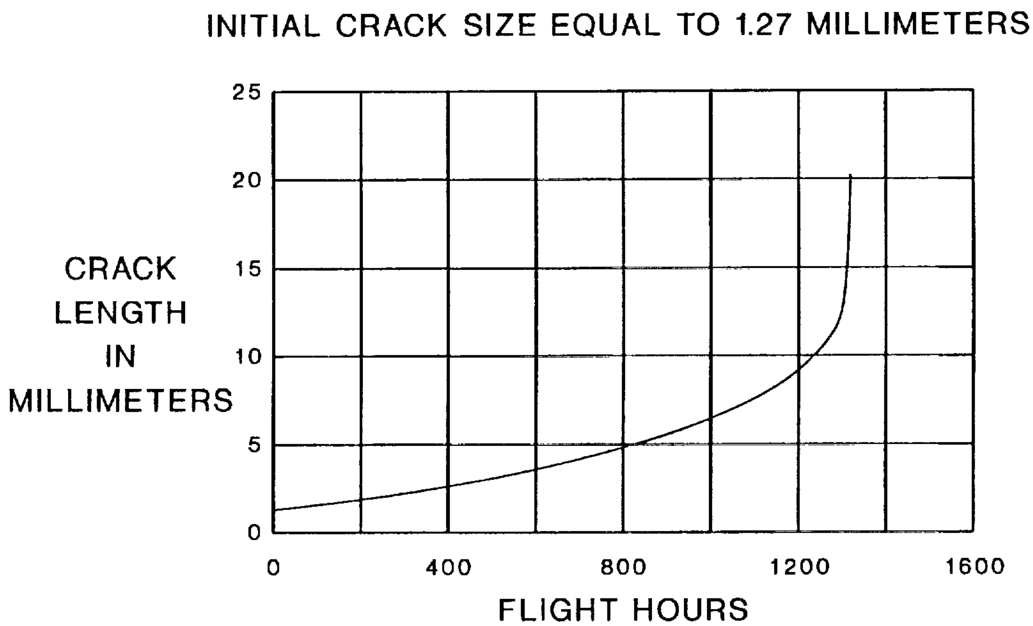

Figure 9. - Crack Growth Function for PA-42 - Steel