FAA

AIRCRAFT ACCIDENT INVESTIGATOR'S

DESK REFERENCE GUIDE

CHAPTER __

STRUCTURES AND METALLURGY
TECHNIQUES AND PROCEDURES FOR STRUCTRUAL AND MATERIAL AIRCRAFT FAILURE INVESTIGATION

Prepared by
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The authors are involved in consulting engineering activities dealing with metallurgical and structural failure analysis activities, as well as studies concerned with maintenance and operations, design, performance and material testing, crashworthiness, and reconstruction.

The authors invite the comments of students during and after the classroom sessions so that we can all continue to learn from each other and continue to improve the quality of investigations.

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TECHNIQUES AND PROCEDURES FOR STRUCTURAL AND MATERIAL AIRCRAFT FAILURE INVESTIGATION

INTRODUCTION

This book has been developed for students at the Transportation Safety Institute in the advanced course dealing with accident investigation techniques and procedures. Both FAA and NTSB investigators use this book and are exposed to similar classroom training dealing with topics in mechanical behavior of aerospace materials and structural analysis of aircraft. This text is organized to complement the classroom training given on the subjects of structures and materials.

Structures and materials should be considered together whenever mechanical defects, failures, or malfunctions are encountered. The mechanical properties of materials are inseparable from the structural design aspects of aerospace vehicles. Adequate performance of aircraft relies upon careful design and lightweight, but strong materials to achieve an optimum balance of initial cost, maintainability and the lowest possible life cycle cost, while being highly reliable if operated with proper care.

These interrelated aspects of design, manufacturing, maintenance, and operational factors combine to create potential problems which can lead to service failures. This book is intended to give the experienced air safety investigator a comprehensive resource to better understand the root causes of problems that will likely be seen in roles as inspectors and investigators.

The materials section of this book concentrates on metals and alloys common to modern aerospace vehicles. The nature of metals, how they are processed and treated, and how they respond to applied stresses is the key to begin the study of the mechanical properties of materials. Sections dealing with wood, fabric, plastics and composites are also discussed since all these materials are used in aircraft design today.

The fundamentals of structural design are also covered in considerable detail. The concepts of external loading on the airframe, design principles of primary structural elements and responses to steady, cyclic and dynamic loads are covered. From these ideas, it is possible to understand the structural behavior of the airframe under normal and over stressed conditions and identify primary failures and secondary or consequential failures.

The structural concepts give a broad view of the aircraft strength properties, while the materials concepts allow detailed examinations of individual broken or damaged parts.

In the air transport field, relatively few accidents involving structural failure of a major component have occurred. Nonetheless, the few in-flight structural failures that have occurred have focused attention on design and maintenance problems and must be addressed. In the general aviation field, a greater number of major in-flight structural failures have occurred, most attributable to excessive flight loads imposed when the aircraft's operating limitations were exceeded by the pilot. Recently, several in-flight structural failures have occurred in transport aircraft involving both airframe and engine material failures. Both types of aircraft have experienced structural failures of landing gears because of landing accidents.

The higher performance of the modern aerodynamically clean airplane makes it easier for the pilot to exceed the design limitations of the aircraft. Although the record has been good and is expected to improve, the accident investigator must always consider in-flight structural failure a possibility and examine the structure to either eliminate or confirm it as a causal factor. The aircraft design engineering field is not static and new materials, new designs and new manufacturing processes are constantly being developed. These new ideas inevitably produce new problems for the aircraft investigator to consider during the course of each accident investigation. The use of higher strength aluminum alloys and the resultant higher stress levels in the structural components have highlighted the importance of fatigue and stress corrosion failures.

These problems challenge the accident investigator following an accident involving suspected in-flight structural failure. Which structural part or component initially failed in flight? Why did the part or component fail? The "what failed?" question dictates that the investigator first know how to look for a failure and second, know how to recognize the various fractures he will encounter. The first question requires a knowledge of investigatory procedures and techniques and the latter requires a knowledge of fracture and analysis.

The "why did it fail?" question is generally more difficult to answer and an understanding of aircraft loadings is a prerequisite in the search to provide the answer.

It is evident that a complete investigation of the structural failure problem should include four general topics:

- Basic Metallurgy & Materials
- Aircraft Loadings
- Fracture Analysis
- Procedures and Techniques

Only those procedures and techniques specific to structural failure investigations will be considered. It is presumed in this presentation that the investigator is familiar with standard investigation procedures, especially with those associated with witnesses and operational problems. The material on fracture analysis has been divided into sections on progressive failures and static failures. Recognition of different types of fractures is emphasized. This phase is of chief interest to the investigator whose main task is to ferret out the cause of failures from a pile of wreckage.

The overall problem of aircraft loadings is presented in considerable detail. A more thorough understanding of the loads imposed on an aircraft structure will produce better investigations, recommendations, and corrective actions.

The topics covered contain basic information and must be thoroughly understood before the investigator initiates examination of the wreckage at the accident scene. The section on aircraft loads can be considered necessary background material; the sections on progressive and static failures as basic tools for detecting primary and secondary failures; and general procedures of investigations as techniques employed to use the background information and basic tools effectively.

1.0 BASIC METALLURGY AND MATERIAL BEHAVIOR

1.1 CLASSES OF ENGINEERING MATERIALS

The materials encompassed in aircraft structures are primarily metals, plastics, composites, fabrics and wood.

The investigator should be familiar with the variety of materials used in aircraft construction. Texts are available in technical libraries under the titles of "Aircraft Materials and Processes", "Aircraft Structural Materials", "Materials for Aircraft Fabrication", etc. The objective is not to become an expert on aircraft materials, but to be aware of the complexity of such materials and recognize when technical assistance is required.

A variety of materials are used in modern aircraft. The specific material selected for a specific application is based upon a number of criteria. Included in these criteria are: 1) mechanical properties such as tensile strength, yield strength and ductility; 2) physical properties typified by density and chemical reactivity; 3) availability such as sheet, plate, structural shapes, forging and castings; and 4) economic considerations.

1.2 PRODUCTION AND PROCESSING OF MATERIALS

1.2.1 Origin of Grain Structure in Metals

All metals are crystalline in their solid state; that is, they are made up of crystals or grains in which the atoms are arranged in a repetitive three-dimensional pattern.

Crystallization is the process by which metals solidify from a liquid state. It takes place by a mechanism of nucleation and growth (Figure 1). As a liquid metal is cooled in the solidification temperature range, it begins to solidify by the formation of tiny crystallites around nuclei in the liquid. These tiny crystallites are referred to as dendrites. These crystallites grow larger and larger, eventually growing together as the solidification process is completed.

The places where grains meet are called grain boundaries. Within these grain boundaries, the atoms are not perfectly aligned with the crystals on either side. Grain boundaries are regions of higher internal energy and as a result, metallurgical reactions occur in the grain boundaries. In some alloys, such as the heat treatable aluminum alloys of the #2014 type, copper particles precipitate in the grain boundaries resulting in an opportunity for microscopic galvanic corrosion cells to develop. The root cause of intergranular corrosion is this chemical difference that develops in the grain boundaries.

The size of the grains can have a significant influence on the properties of the metal. Larger grain size metals are more brittle than fine grain materials. Not only are fine grain materials harder and stronger, they are also tougher. On the other hand, a larger grain size improves the high temperature creep properties of alloys that must withstand prolonged heating in service.

Cast metals, such as aluminum castings used for brackets, have relatively large grains and often internal pores and chemical segregation. Accordingly, castings can be, and often are, brittle. Most castings are therefore designed to be large and carry low stress.

By hot working metals, such as rolled or extruded shapes or forgings, the grains are refined and the part is much more ductile and tougher than an equivalent casting. Nearly all heavy duty parts are made by a wrought process to provide a good grain structure and best mechanical properties.

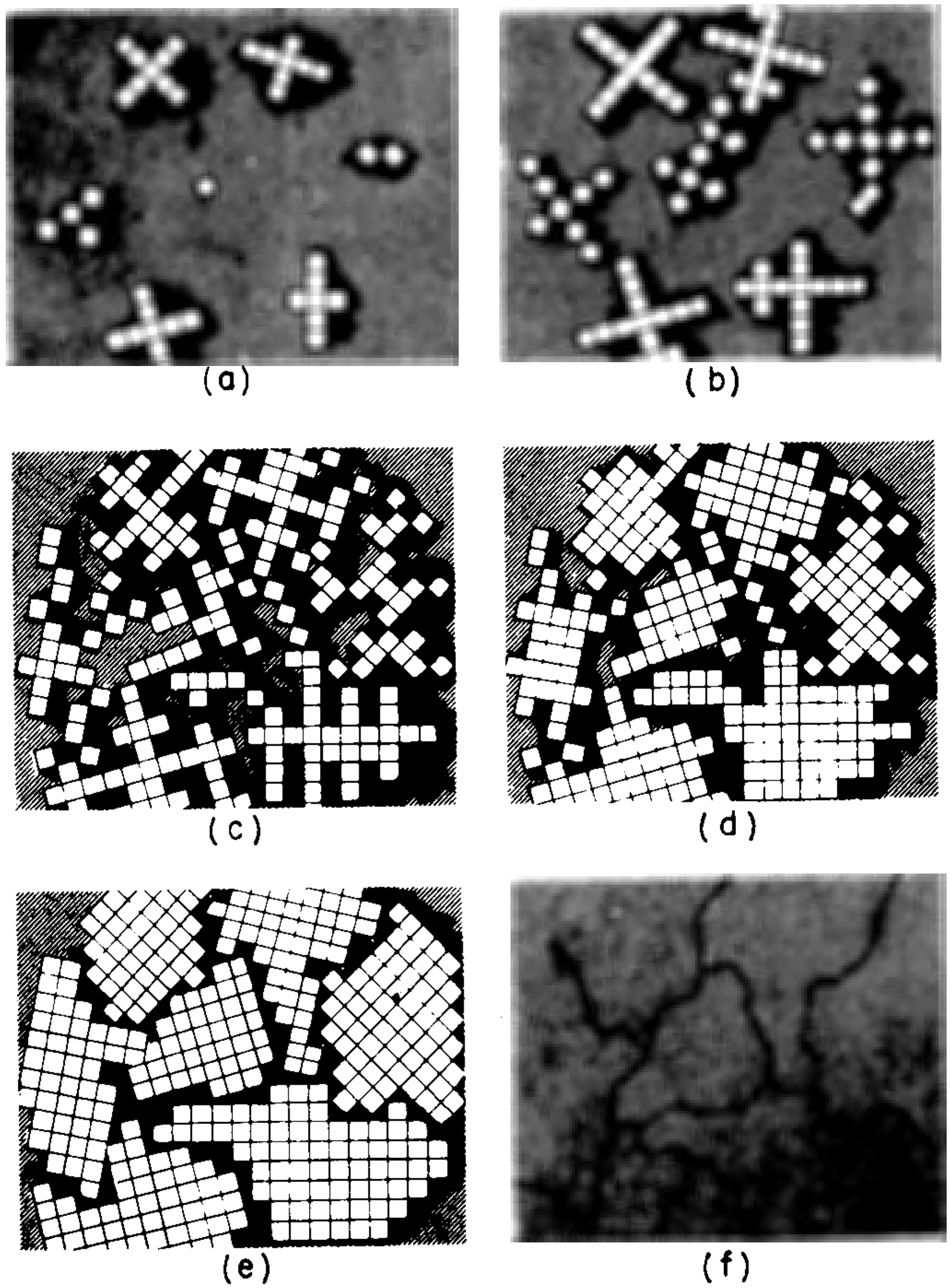


Figure 1

Stages (a) through (e) picture the crystallographic grain growth of a metal as suggested by Rosenhain. Junction lines of adjacent crystalline growths are represented in (f). The shaded portion of (e) represents the boundary area between crystals grains. Atoms at the boundary are subjected to different electrical attraction-repulsion forces than atoms at center of grain. Thus, grain boundary properties are not the same as for the crystal proper.

Steel has a BCC structure in the annealed or normalized condition. Once heat treated by quenching and tempering, the BCC structure is altered and forms a very hard and brittle structure called martensite. Tempering the martensite at between 400° and 1100° causes a softening to occur and the toughness increases as the hardness drops.

Nearly all of the naturally soft metals are FCC. Copper, gold, silver, and aluminum are examples of FCC metals. About the only common HCP metal is titanium.

The terms crystallization and recrystallization are two words greatly mistreated by the layman. These words are used when an untrained observer says, "The metal crystallized and broke," or, "it was a case of recrystallization." All metals are crystallized in the solid state!

Alloys are metals which are formed by mixing two or more metals together. Often the alloy consists of primarily one type of metal (the base metal) to which a relatively small amount of another element (the alloying element) has been added. The most common types of alloys are interstitial alloys or substitutional alloys. With the interstitial alloys, the atoms of the alloying element fit in the crystal structure in between the atoms of the base metal. Substitutional alloys are those in which atoms of the alloying elements replace or substitute for the base metal atoms in crystal structure.

1.2.2 Common Aircraft Structural Metals

There are numerous sources of information on metals used in aircraft structures. One of the most comprehensive sources of such information is the Metals Handbook series which is published by the American Society for Metals.

1.2.2.1 Steel

Steels are interstitial alloys of iron and carbon. Iron is the major constituent and carbon is the basic alloying element in steel. Small amounts of residual or impurity elements, such as phosphorus and sulfur, are always present in steel and many other alloying elements, such as nickel, chromium, molybdenum, vanadium and tungsten may be added in carefully controlled amounts. In general, the carbon content of steel determines the ultimate strength obtainable in steels that are hardened by quenching and tempering. Other elements, or combinations of elements influence: 1) the combinations of strength and toughness that can be obtained by heat treatment; 2) the thickness of sections that can be hardened completely through to the center of the section; and 3) the response of the steel to environmental conditions, such as corrosive atmospheres or high temperatures.

Ferrous (iron) alloy materials are generally classified according to carbon content.

TABLE I

Material	Carbon Content
Wrought iron	Trace to 0.08%
Low carbon steel	0.08% to 0.30%
Medium carbon steel	0.30% to 0.60%
High carbon steel	0.60% to 2.2%
Cast iron	2.3% to 4.5%

A numbering system for the identification of steel has been established by the American Iron and Steel Institute (AISI) and the Society of Automotive Engineers (SAE). The following table gives the AISI-SAE designations for carbon and low alloy steels. (In the complete designations the XX's would be replaced by numbers to indicate carbon content. For example, 1035 steel has 0.35% carbon and 53100 steel has 1.00% carbon):

TABLE II

10XX	Plain carbon steel
11XX	Carbon steel with additional sulfur for easy machining
13XX	Carbon steel with about 1.75% manganese
40XX	.25% molybdenum
41XX	1% chromium, .25% molybdenum
43XX	2% nickel, 1% chromium, .25% molybdenum
46XX	1.7% nickel, .2% molybdenum
48XX	3.5% nickel, .25% molybdenum
51XX or	1% chromium steels
51XXX	1% chromium, 1.00% carbon
52XX or	1.5% chromium steels
52XXX	1.5% chromium, 1.00% carbon
61XX	1% chromium steel with .15% vanadium
86XX	.5% chromium, .5% nickel, .20% molybdenum
87XX	.5% chromium, .5% nickel, .25% molybdenum
92XX	2% silicon steels, .85% manganese
93XX	3.25% nickel, 1.20% chromium, .12% molybdenum

There are numerous steels with higher percentages of alloying elements that do not fit into this numbering system. These include a large group of stainless and heat resisting alloys in which chromium is an essential alloying element. Some of these alloys are identified by three digit AISI numbers and many others by designations assigned by the steel company that produces them. The few examples below will serve to illustrate the kinds of designations used and the general alloy content of these steels:

TABLE III

EXAMPLES OF STAINLESS AND HEAT RESISTANT STEELS
NOMINAL COMPOSITION (PERCENT)*

Alloy Designation	Carbon	Chromium	Nickel	Other	General Class of Steel
302	0.15	18	9		Austenitic
310	0.25	25	20		Austenitic
321	0.08	18	11	Titanium	Austenitic
347	0.08	18	11	Columbium or Tantalum	Austenitic
410	0.15	12.5			Martensitic, Magnetic
430	0.12	17			Ferritic, Magnetic
446	0.20	25		Nitrogen	Ferritic, Magnetic
PH15-7 Mo	0.09	15	7	Molybdenum, Aluminum	Precipitation Hardening
17-4 PH	0.07	16.5	4	Copper, Columbium, or Tantalum	Precipitation Hardening

The strength and ductility, or toughness of steels is controlled by cold working or heat treating. In general, any process that increases the strength of a material will also decrease its ductility.

*Remainder is iron and normal impurities.

Cold working, or strain hardening, is basically a deformation process performed at normal room temperatures. It is used commercially to increase the strength of many materials, including low carbon steels and austenitic (FCC) stainless steel that cannot be hardened by heat treatment. Examples of this process are the cold rolling of sheet or strip material, cold drawing of wire, surface rolling and shot peening.

1.2.2.2 Heat treatment

Heat treatment is applied to steel either to prepare it for a subsequent manufacturing operation, or to produce material with properties suitable for a specific service application. Some information on basic heat treatment is given below.

Annealing is a process of softening and increasing the ductility of steel by a heating and cooling cycle. Slow cooling (usually in the furnace used for heating) is required in steels that are hardened by quenching. Rapid cooling is required for precipitation hardening of steel.

Normalizing is similar to annealing, except that steel is cooled in still air. This treatment is used mainly to refine the grain structure and improve the machinability of the steel.

Quenching involves rapid cooling of steel from a suitable elevated temperature. This is usually accomplished by immersion in oil or water, although air cooling can be used for some types of alloyed steel. Quenching is the first step in producing hardened quenched and tempered steels. Quenching leaves most steels very hard and brittle and with undesirable internal stresses, so they must be tempered before they can be used. In precipitation hardening steels, quenching is used after softening or annealing treatments to retain the alloy in solid solution. Subsequent hardening occurs when treated at an intermediate temperature which forms the strengthening precipitates.

Tempering (sometimes called Drawing): Reheating a steel that has been hardened by quenching to a suitable temperature (lower than the temperature from which it was quenched for hardening), holding it at that temperature for a suitable period of time and then cooling to room temperature. Tempering reduces the hardness, increases the ductility, relieves internal stresses and increases the toughness of the quenched steel.

Stress Relieving (sometimes called Process Annealing) is a process similar to tempering, but used primarily to reduce internal stresses.

Solution Treatment is a treatment used primarily on precipitation hardening alloys (both ferrous and nonferrous) prior to precipitation treatment. The alloy is heated to a relatively high temperature and held for a predetermined length of time to redissolve precipitated particles back into a solid solution. It is then cooled fast enough or quenched to prevent the constituents of these particles in precipitating out of the solid solution.

Precipitation Heat Treatment involves reheating the solution treated alloy to a low or moderate temperature the alloy constituent to precipitate out in a controlled fashion in order to increase strength. Also called precipitation treating.

Case Hardening is a hardening treatment applied to steel, in which a surface layer (case) is made substantially harder than the inner portion (core). This operation may involve only heat treatment, but in most applications it combines heat treatment with a process for increasing the carbon or nitrogen content of the case. In some processes, both carbon and nitrogen are added to the surface layer of the steel. Typical processes used for case hardening are carburizing, nitriding, carbonitriding, cyaniding, induction hardening and flame hardening.

1.2.2.3 Casting, Forging, Rolling, Extrusion

Steel and aluminum components in aircraft structures are produced by means of casting, forging, extrusions and rolling. Casting involves the making of a mold, usually composed of a mixture of sand and binders, and then pouring liquid metal into the mold cavity allowing it to take the shape of the mold cavity and then solidify.

Forging starts with a hot metal billet of some pre-cut shape which is placed in the cavity of a die in a forge press or hammer. The forging press then closes, forcing the hot metal to flow into the shape of the die cavity. Forgings have greater toughness than castings, but have greater limitations on final shapes obtainable.

Extrusions are made by forcing a hot metal billet out through an extrusion die which has some definite cross-sectional shape. Usually long bar or rod-like lengths of the cross-section are made.

Rolling is the method used to produce sheet products. Large steel or aluminum ingots are progressively rolled down into thinner and thinner bands until a final thin coil is produced. This can then be cut or sheared into smaller lengths used for metal stamping or used directly as sheets.

Each of these processes develops a specific grain structure which will affect the mechanical strength, ductility, and corrosion resistance of the finished part.

1.2.2.4 Plating and Coatings

Due to environmental considerations, many times a metal component will require additional protection or require enhanced wear properties. Other than heat treatments discussed earlier, the metal surface may be covered by plating, metal spray, cermet coating, anodizing or paint coating.

Chromium plating is used on many aircraft components for both enhanced wear and corrosion protection. Cadmium plating is also used as a corrosion protective coating on many aircraft components. Because the electrolytic metal plating is done in an aqueous bath and the process is not one hundred percent efficient, evolved hydrogen from the solution can be charged into the metal components. In high strength steel parts, after plating is completed, this hydrogen must be baked out at low temperatures to prevent embrittlement of the base steel.

Metal and/or cermet spray coatings are applied to enhance wear or high temperature corrosion resistance on areas such as turbine blades in the hot section.

Anodizing is a process of producing a controlled oxide film on aluminum surfaces. These films can incorporate colored dyes and are used to enhance corrosion protection and wear resistance. Paints are used primarily for corrosion protection and appearance.

1.2.2.5 Aluminum Alloys

Aluminum and aluminum alloys are the main materials used in aircraft structures. Aluminum can be used in both cast and wrought forms, depending upon the designed loading patterns. Casting alloys are generally less ductile than wrought and therefore are used for complex shapes in areas with lighter loads and where ductility will not be a factor. Wrought forms such as forgings, extrusions and rolled products make up the bulk of aluminum used for structural components. These forms usually have greater strength ductility and toughness than cast products.

Aluminum alloys have a face centered cubic lattice structure and do not exhibit a ductile to brittle transition as some steel products, thus they maintain their ductility in very cold temperatures.

Wrought alloys use a four digit numerical designation to identify the chemical composition. The first digit indicates the main alloy group, the second digit indicates modifications of the original alloys and the last two identify the alloy or indicate the aluminum purity in non-alloyed materials. The following table gives the main designated alloy groups:

TABLE IV

1XXX	Commercially pure (99% aluminum or better). The last two digits reflect the purity, i.e., 1060 would be 99.60% Al.
2XXX	Prime alloy element is Copper.
3XXX	Prime alloy element is Manganese.
4XXX	Prime alloy element is Silicon.
5XXX	Prime alloy element is Magnesium.
6XXX	Prime alloy elements are Magnesium and Silicon.
7XXX	Prime alloy element is Zinc.
8XXX	Other elements.

Most of these alloys contain other elements that significantly affect their properties. The nominal compositions of a few of the more widely used alloys in aircraft structures are given below:

TABLE V

Nominal Chemical Composition
Percent of Alloying Elements*

Alloy	Silicon	Copper	Manganese	Magnesium	Chromium	Zinc
2024		4.5	0.6	1.5		
5052				2.5	0.25	
6061	0.6	0.25		1.0	0.20	
7075		1.6		2.5	0.30	5.6
7079		0.6	0.20	3.3	0.20	4.3
7178		2.0		2.7	0.30	6.8

In aluminum alloys, as with steels, the properties of any specific alloy can be altered by work hardening (often called strain hardening), heat treatment, or by a combination of these processes. Alloys in the 1XXX, 3XXX, 4XXX, and 5XXX series can be strengthened by various degrees of cold work, but not by heat treatment. Alloys that are strengthened by heat treatment are precipitation hardened after a preliminary solution treatment and include the 2XXX, 6XXX and 7XXX series. The basic temper designations for aluminum alloys are as follows:

F - As fabricated: Applies to products that acquire some temper from processing, but in which there is no special control of strain hardening or thermal treatment.

O - Anneal: wrought products only, lowest strength condition.

H - Strain hardened wrought products only.

W - Solution heat treated: An unstable temper applicable in finished products (only to alloys) that spontaneously age harden at room temperature after solution treatment.

T - Thermally treated to produce stable tempers other than F, O, or H.

Aluminum alloys for aircraft can be generally classified into two large groups: precipitation (or age hardening) and strain (or cold work hardening).

Precipitation hardening alloys are the 2XXX, 6XXX and 7XXX series. These alloys are given a solution anneal to dissolve all the alloying element and are then quenched. A low or room temperature aging or precipitation treatment is then performed to allow controlled precipitation of finely dispersed particles of the alloying element from the solid solution. This precipitation process strengthens the alloy. The basic subdivision of the "T" tempers are given below:

T1: Cooled from an elevated temperature shaping process (such as extrusion or casting) and naturally aged to a substantially stable condition.

T2: Annealed (cast products only). Applies to cast products that are annealed to improve ductility or dimensional stability.

*Remainder is aluminum and normal impurities.

T3: Solution heat treated and then cold worked.

T4: Solution heat treated and naturally aged at room temperature to a substantially stable condition.

T5: Cooled from an elevated temperature shaping process and then artificially aged at some preselected elevated temperature.

T6: Solution treated and artificially aged.

T7: Solution treated and then stabilized. This stabilization treatment provides additional control of some special characteristics of the material.

T8: Solution heat treated, cold worked and then artificially aged.

T9: Solution heat treated, artificially aged and then cold worked.

T10: Cooled from an elevated temperature shaping process, artificially aged and then cold worked.

In the above definitions, artificially aged means that the material was given an elevated temperature precipitation hardening treatment. Naturally aged means that precipitation hardening occurs spontaneously at room temperature.

Strain hardening alloys are the 1XXX, 3XXX, 4XXX and 5XXX series. The alloy designation is followed by an HX, HXX, or HXXX.

The first digit following the H indicates the specific combination of basic operations as follows:

H1 Strain hardened only. Applies to products which are strain hardened to obtain the desired strength without supplementary thermal treatment.

H2 Strain hardened and then partially annealed. Applies to products which are strain hardened more than the desired final amount and then reduced in strength to the desired level by partial annealing.

H3 Strain hardened and then stabilized. Applies to products which are strain hardened and then stabilized by a low temperature heating to slightly lower their strength and increase ductility.

The second digit following the designations H1, H2 and H3 indicates the degree of strain hardening. These range from 0 (annealed) to 8 (full hard).

The third digit, when used, indicates a variation of a two digit temper. It is used when the degree of control of temper or the mechanical properties are different from, but close to those for the two digit H temper designation to which it is added, or when some other characteristic is significantly affected.

1.2.2.6 Titanium Alloys

Titanium was discovered in 1789 by an English clergyman named Gregor. Five years later, a German by the name of Klaprath named the element titan because of its strong chemical bond with other elements. In the ingot form, free of scale and clean, titanium possesses a silver-gray color. It is non-magnetic (as are aluminum alloy and austenitic stainless steel) and thus precludes the use of a magnet for material separation and identification. Titanium has very high strength to weight ratio. It is lightweight like aluminum yet has the strengths of steel, however, it is quite costly.

Titanium alloying elements are aluminum, chromium, iron, manganese, vanadium, molybdenum, tin and tungsten, as an example, Ti-8AL-1MO-1V, Ti-6AL-4V and Ti-6AL-6V-2Sn are three structural alloys. These titanium alloys consist of aluminum, molybdenum, vanadium and tin in varying percentages.

Titanium was first used as a structural metal in 1952. Unalloyed titanium is sometimes used for firewalls, bulkheads, compressor cases, shrouds and tail pipes. The most common use of titanium alloys is in gas turbine engine parts, such as compressor discs, blades and spacers. However, some use has been made of titanium alloys, mainly in military aircraft, in primary structural members, skin and airframe forgings and fasteners. Use of titanium alloys is expected to increase in the future. For example, large quantities of titanium alloys are being used in the Boeing 747 for drag fittings and parts of the landing gear beams.

1.2.2.7 Copper

Copper is used in its basic form primarily as an electrical conductor and for oil and fuel line tubing. Copper fuel lines have almost been replaced by flexible lines at points of high vibration.

Copper in the annealed state possesses an ultimate tensile stress of 32,000 psi and a yield stress of 6,000 psi. It possesses the mechanical property of work hardening (strain hardening) under a condition of vibration and for this reason has been eliminated from the fuel system in the carburetor to firewall area. Curved lines and 360^{o} spirals are incorporated to reduce the stresses from vibration.

Copper is annealed by heating in a furnace to 1100^{o}F to 1200^{o}F and then quenching in water. Maximum softness and ductility is obtained if the high temperature is held no longer than five minutes.

A failed copper line, perhaps a causal factor in an accident, should be analyzed for mechanical properties, provided the aircraft did not burn. The characteristics of copper wire, when exposed to external heat versus electrical load, are covered under Systems Investigation.

Copper is the major constituent of brass and bronze. Zinc is the primary alloying element in brass. Tin, aluminum, or silicon is used as alloying elements in the various types of bronze.

1.2.2.8 Other Alloys

There are hundreds of alloys in addition to the three base metals previously discussed. The cobalt-base alloys are an example. Such alloys, high in cobalt content, are resistant to wear and oxidation-corrosion even under extreme conditions of elevated temperatures and in corrosive environments.

The development of the turbosupercharger at the beginning of World War II was dependent upon the discovery of metals and alloys which would maintain adequate strength at temperatures of 1300^{o}F and above. The cobalt-base alloys were found to possess superior performance as gas turbine blades. Many major improvements in turbine blade materials have taken place since World War II and coupled with improved design, have led to longer time between overhauls (TBO's).

Another series of alloys are nickel based. A number of these alloys are known by their trade names, such as Hastelloy and Inconel. The English counterpart of Inconel is Nimonic. Inconel only possess from 4.5% to 7% iron and therefore is not a steel alloy.

Inconel is used as the recording medium in certain flight recorders and is commonly miscalled stainless steel tape. Other flight recorders utilize an aluminum alloy recording medium. If it is necessary in a report to discuss the metal used as a flight recording medium, the correct material should be specified. Inconel is used since it can be fabricated in thin sheets and possesses good strength and physical properties at elevated temperatures.

1.2.2.9 Properties of Composites

Composite materials have been used for many years and are represented by many familiar engineering materials such as glass fiber reinforced plastic (fiberglass). Even plywood is a sort of composite material. In general, composite materials are designed to take advantage of enhanced material properties and in particular, arrange constituent materials into a component that has optimum strength at low weight.

High performance composites now being developed for modern aerospace structures are based on both polymeric and metallic matrix materials with glass, carbon, graphite, Kevlar, or boron fibers serving as the primary load carrying element. The design, testing and performance evaluation of these newer materials requires a new technology to describe and analyze composite material behavior. Appendix 1 contains an extensive glossary of terms related to composite material technology to help the reader become familiar with the language and processes involved.

1.2.2.9.1 Strength of Composites

Modern composites are expensive but efficient lightweight structural materials. One of their major attributes is providing both high stiffness and load carrying capacity while being lightweight. Their lightness is primarily due to the inherent low density of the constituent materials. Once designed and manufactured to take advantage of these custom properties, the weight of the final structural component is often much less than if constructed of solid metal. Accordingly, the strength of composite material will often be referred to as a "specific" strength. This means that the property being reported is divided by the material's density to provide a measure of the strength on a weight and size basis. In other words, a high specific strength implies a structural material is both strong and lightweight. Ultra high strength steels compare favorably with the best composite materials, but on a pound-for-pound basis, modern composite materials are superior.

Composite materials still require advances in the state-of-the-art to lower their cost, improve in service inspection techniques and assure adequate repair methods. The Appendix describes many of the different terms related to this advancing materials industry. Understanding of design and manufacturing methods will assist in the proper analysis of material performance studies.

Figure 2 illustrates specific strength (strength divided by density) versus specific modulus (stiffness divided by density) for several traditional materials and modern composite materials. Materials with high strength and stiffness along with light weight are desirable for optimum performance.

1.3 MECHANICAL BEHAVIOR OF MATERIALS

Aerospace materials are used to support air loads and provide the structural and mecahnical functions of the airframe and systems components. Applied loads create stresses in members. Stresses result in the material deforming under load. The basic formulas for stress and strain are discussed in this section.

Stress is defined as the acting load divided by the area of material supporting the load. Mathematically, this is given as:

$$S = \frac{F}{A}$$

where: S = stress (psi)
F = load (lbs.)
A = area (in^2)

Strain is a measure of the deformation caused by a applied load:

$$e = \frac{\Delta L}{L}$$

where: e = strain (%)
ΔL = change in size (in.)
L = original size (in.)

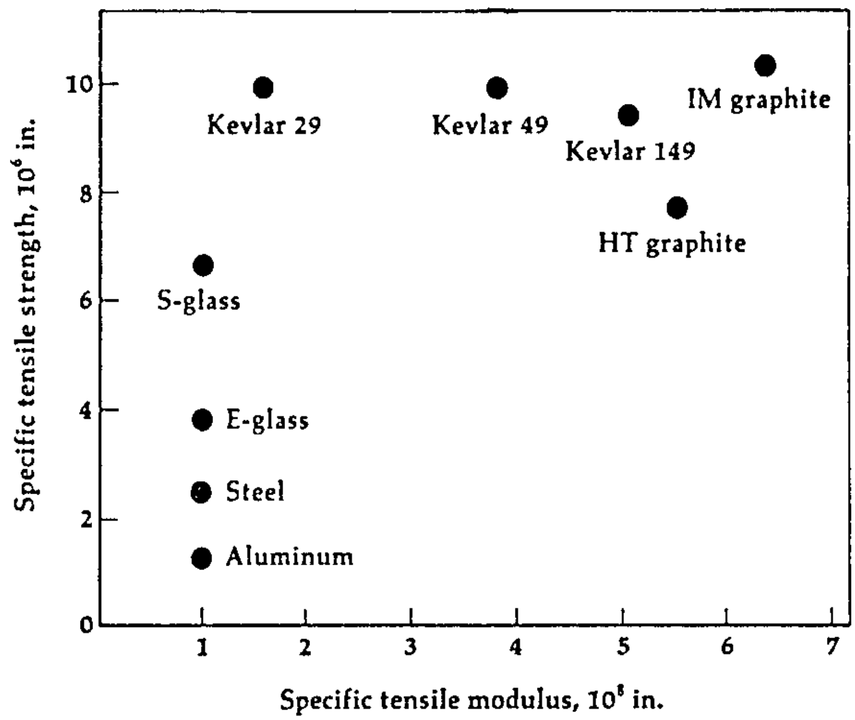


Figure 2

Specific tensile strength as related to specific tensile modulus for several composites and high strength alloys.

Experience shows that all solid materials can be deformed when subjected to external load. It is further found that up to certain limiting loads a solid will recover its original dimensions when the load is removed. The recovery of the original dimensions of a deformed body when the load is removed is known as elastic behavior. The limiting load beyond which the material no longer behaves elastically is the elastic limit. If the elastic limit is exceeded, the body will experience a permanent set or deformation when the load is removed. A body which is permanently deformed is said to have undergone plastic deformation.

The unique feature of truly elastic behavior is that strain is a linear function of stress and, when one is zero, so is the other. Hence stress and strain are related simply by a proportionality factor.

For simple loading where a normal stress S is applied to a body, the relationship between such normal stresses and the associated strains is given by Hooke's Law:

$$S = Ee$$

where: S = stress (psi)
E = modulus (psi)
e = strain (%)

For bodies loaded under shear Hooke's Law is expressed as:

$$\tau = G\,\delta$$

where: τ = shear stress (psi)
G = shear modulus (psi)
δ = shear strain (%)

When an elastic rod is stretched, not only does it lengthen in accordance with Hooke's law, but it contracts laterally as well. The lateral strain has been found by experience to be a constant fraction of the longitudinal extension. This fraction is called Poisson's Ratio and depends on the specific material. Poisson's ratio is given by the following equation:

$$\nu = \frac{-\text{(Lateral strain)}}{\text{(Axial strain)}}$$

The negative sign merely indicates that the longitudinal and transverse strains have opposite signs since extension in the one direction is accompanied by contractions in the other.

Value for Young's Modulus and Poisson's Ratio for a variety of materials are listed in Table VI.

TABLE VI

YOUNG'S MODULUS AND POISSON'S RATIO OF A VARIETY OF MATERIALS

Material	Temp. (^{o}F)	E(psi)	μ
	+1000	38.8×10^{6}	
Beryllium	RT	41.7×10^{6}	0.05
	-300	42.7×10^{6}	
	+1000	56.1×10^{6}	
Tungsten	RT	59.3×10^{6}	0.30
	-300	60.4×10^{6}	
Wrought iron	+1000	25.6×10^{6}	
and steel	RT	30.8×10^{6}	0.28
	-300	32.2×10^{6}	
Cast iron (gray iron)	RT	$11\text{-}23 \times 10^{6}$	0.17
Aluminum	RT	$10\text{-}11 \times 10^{6}$	0.33
Concrete	RT	$3.5\text{-}5 \times 10^{6}$	0.19
Quartz fiber	RT	7.4×10^{6}	
Limestone	RT	8.4×10^{6}	
Rubber (soft)	RT	$0.15\text{-}2 \times 10^{3}$	0.49
Wood	RT	$0.9\text{-}2.2 \times 10^{6}$	
Nylon	RT	$0.18\text{-}0.45 \times 10^{6}$	
Epoxy	RT	$0.4\text{-}20 \times 10^{6}$	
Glass	RT	$9\text{-}10 \times 10^{6}$	0.25
Kevlar	RT	27×10^{6}	-
Carbon (graphite) Fiber	RT	55×10^{6}	-

1.3.1 Thermal Strain

When the temperature of an unrestrained body is uniformly increased, the body expands, and the size increase is:

$$\Delta L = \alpha L \Delta T$$

where: ΔL = change in size (in)
α = coefficient of expansion in/in-^{o}F
ΔT = temperature change (^{o}F)
L = original size

In this action, all dimensions increase and no shear strain occurs.

If a straight bar is restrained at the ends so as to prevent lengthwise expansion and then is subjected to a uniform increase in temperature, a compressive stress will develop because of the axial constraint. The stress is:

$$S = \propto \Delta T \; E$$

In a similar manner, if a uniform flat plate is restrained at all edges and also subjected to a uniform temperature rise, the compressive stress developed is given by the equation:

$$S = \frac{\propto \Delta T \; E}{1 - \gamma}$$

A thermal stress can also arise because of the existence of a temperature gradient in a body. Figure 3 shows the internal stresses within a slab of infinite dimensions during heating and cooling. During cooling, the maximum stress is the surface tension. At the same time, force equilibrium requires a compressive stress at the center of the slab. During heating, the external surfaces are hot and tend to expand but are restrained by the cooler center. This causes compression in the surface and tension in the center as shown.

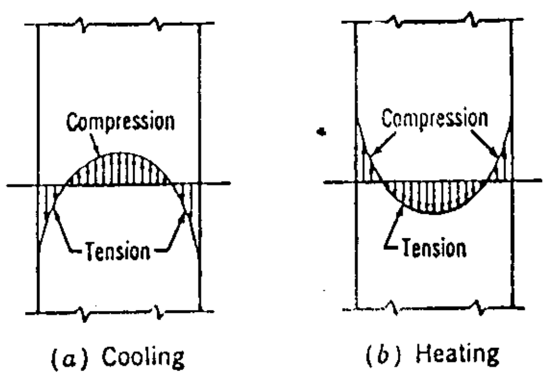


Figure 3

Thermal stresses in an infinite slab during heating and cooling.

Table VII lists approximate values of the thermal expansion coefficient for various engineering materials.

TABLE VII

COEFFICIENTS OF THERMAL EXPANSION (LINEAR MEAN COEFFICIENTS FOR THE TEMPERATURE RANGE 0-100C)

Material	Celsius Scale	Fahrenheit Scale
Aluminum	$23.9(10)^{-6}$	$13.3(10)^{-6}$
Brass, cast	$18.7(10)^{-6}$	$10.4(10)^{-6}$
Carbon steel	$10.8(10)^{-6}$	$6.0(10)^{-6}$
Cast iron	$10.6(10)^{-6}$	$5.9(10)^{-6}$
Magnesium	$25.2(10)^{-6}$	$14.0(10)^{-6}$
Nickel steel	$13.1(10)^{-6}$	$7.3(10)^{-6}$
Stainless steel	$17.3(10)^{-6}$	$9.6(10)^{-6}$
Tungsten	$4.3(10)^{-6}$	$2.4(10)^{-6}$

Notice that aluminum expands and contracts approximately twice as fast as steel; therefore, steel cables will slacken as an airplane cools off and vice versa.

1.3.2 STRESS - Strain Behavior of Materials

All structures deform under load. In fact, until a structure deforms, there is no resistance to load. The details of how loads and deformations relate to each other is the topic called engineering mechanics. It is important to have a basic understanding of how various materials respond to the loads applied to them in service.

1.3.3 Stress-Strain Curve

In order to intelligently utilize the capabilities of the various structural materials that go into the construction of an aircraft, it is necessary to test these materials. The stress-strain curve is the plot of a specific test that provides information about the mechanical properties of the material.

To understand the basics of the stress-strain curve, it is first necessary to understand stress and strain. Stress is force (in pounds) per unit area (square inches). If a rod of one square inch in cross-sectional area sustains a load of 50,000 lbs., the rod is under a tensile stress of 50,000 lbs. per square inch (psi).

In the following case, where the rod carries a tensile load of 50,000 lbs., the rod has stretched to a certain degree. All materials will deform under a load condition and because different materials deform differently, the purpose of plotting the stress-strain curve is to show this deformation. When stress is applied, material is deformed, hence resulting in strain. Strain is a result of stress. Strain is measured in units of inch per inch. In other words, how much does each inch of the material deform? If a ten inch rod is stretched 0.010 inch, it is under a strain of 0.001 (0.010/10) or stated differently it has strained 0.1%.

Referring to Figure 4 below, note that stress is plotted on the vertical axis and strain is plotted along the horizontal axis. This is not a curve of a specific material, but is representative of a rather ductile material such as low carbon steel or 2024 aluminum alloy.

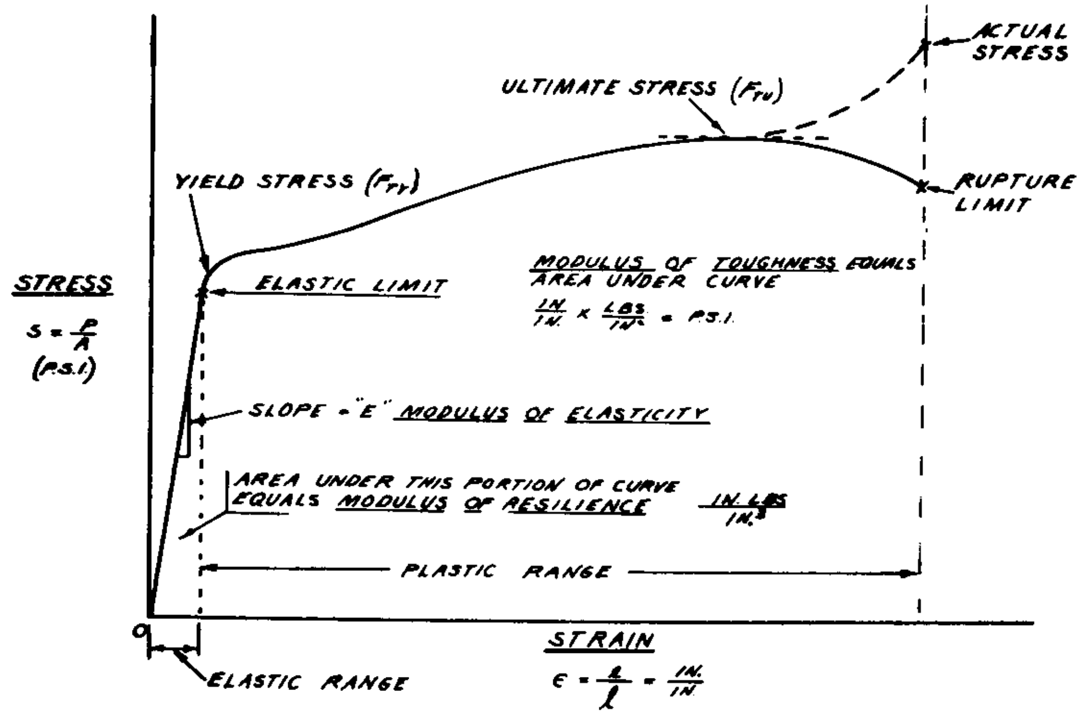


Figure 4

A typical stress-strain curve for steel illustrating the definitions for mechanical properties.

As a tensile load is applied, the material begins to elongate. As long as the stress does not exceed the elastic limit, the material will return to its original shape when the load or stress is removed. The vertical dotted line in the figure represents the deformation resulting when the stress has been raised to the elastic limit. The distance from zero deformation to the dotted line represents the elastic range of the material. According to Webster, "elastic" is defined as "having the property of immediately returning to its original size, shape, or position after being stretched, squeezed, flexed, or expanded".

In studying the elastic range and the elastic limit, it is obvious that an aircraft structure should operate at some point on the straight line portion of this plot, and not exceed the elastic limit. The investigator should associate the design term limit load factor with the straight line portion of this curve.

If an individual designed an aerobatic-type aircraft with a limit load factor of 6, he would have to decide what stress below the elastic limit to use for a good service life and the remaining portion down to zero stress would be divided into six equal spaces to denote each "G". As an example: Assume the elastic limit was 40,000 psi and the designer selected 36,000 psi as the maximum stress expected in service. Then, in straight and level unaccelerated flight, the particular part under consideration would be under a stress of 6,000 psi (1 G). In a coordinated level turn with a 60^{o} bank, this part would be under a stress of 12,000 psi (2 G's). In a 6 G pullout maneuver, the stress would be 36,000 psi.

Referring again to Figure 4, note that the material will continue to sustain higher loads without separating or breaking. Exceeding the elastic limit will cause permanent deformation. The material will locally neck down and begin separating when the ultimate stress is reached. More ductile materials will stretch plastically to a greater extent than a more brittle material. The area under the curve (a product of stress times strain) is a measure of the toughness of the material.

Hardness and strength are related such that higher strength materials have higher hardness when measured by the standard techniques. It is usually much simpler to measure the hardness of a component being evaluated than to prepare a tensile test bar from available material.

A hardness test is a simple way to verify the part has been properly heat treated. The engineering drawing for the part will normally call for either a hardness or strength range. Hardness tests can also be useful to check for loss of strength due to exposure to fire or abusive heating in overhaul. Understrength bolts can be sorted by checking the hardness across the hex flats. Although hardness tests are generally considered non-destructive, hardness testing does leave a small pit in the surface, therefore, hardness tests should not be used on sealing surfaces or fatigue critical parts, etc., that will be returned to service.

The following hardness tables depict the relationship between hardness and tensile strength:

TABLE VIII

HARDNESS RELATIONSHIP SCALES

ROCKWELL STANDARD C 150 kg Brale	ROCKWELL STANDARD A 60 kg Brale	ROCKWELL STANDARD D 100 kg Brale	ROCKWELL SUPERFICIAL 15N 15 kg Brale	ROCKWELL SUPERFICIAL 30N 30 kg Brale	ROCKWELL SUPERFICIAL 45N 45 kg Brale	Vickers DPH 10 kg	Knoop 500 gm & Over	Brinell 3000 kg	Tensile Strength Thousand lbs. per sq. inch
80	92.0	86.5	96.5	92.0	87.0	1865	—	—	—
79	91.5	85.5	—	91.5	86.5	1787	—	—	—
78	91.0	84.5	96.0	91.0	85.5	1710	—	—	—
77	90.5	84.0	—	90.5	84.5	1633	—	—	—
76	90.0	83.0	95.5	90.0	83.5	1556	—	—	—
75	89.5	82.5	—	89.0	82.5	1478	—	—	—
74	89.0	81.5	95.0	88.5	81.5	1400	—	—	—
73	88.5	81.0	—	88.0	80.5	1323	—	—	—
72	88.0	80.0	94.5	87.0	79.5	1245	—	—	—
71	87.0	79.5	—	86.5	78.5	1160	—	—	—
70	86.5	78.5	94.0	86.0	77.5	1076	972	—	—
69	86.0	78.0	93.5	85.0	76.5	1004	946	—	—
68	85.6	76.9	93.2	84.4	75.4	940	920	—	—
67	85.0	76.1	92.9	83.6	74.2	900	895	—	—
66	84.5	75.4	92.5	82.8	73.3	865	870	—	—
65	83.9	74.5	92.2	81.9	72.0	832	846	—	—
64	83.4	73.8	91.8	81.1	71.0	800	822	—	—
63	82.8	73.0	91.4	80.1	69.9	772	799	—	—
62	82.3	72.2	91.1	79.3	68.8	746	776	—	—
61	81.8	71.5	90.7	78.4	67.7	720	754	—	—
60	81.2	70.7	90.2	77.5	66.6	697	732	614	314
59	80.7	69.9	89.8	76.6	65.5	674	710	600	306
58	80.1	69.2	89.3	75.7	64.3	653	690	587	299
57	79.6	68.5	88.9	74.8	63.2	633	670	573	291
56	79.0	67.7	88.3	73.9	62.0	613	650	560	284
55	78.5	66.9	87.9	73.0	60.9	595	630	547	277
54	78.0	66.1	87.4	72.0	59.8	577	612	534	270
53	77.4	65.4	86.9	71.2	58.6	560	594	522	263
52	76.8	64.6	86.4	70.2	57.4	544	576	509	256
51	76.3	63.8	85.9	69.4	56.1	528	558	496	250
50	75.9	63.1	85.5	68.5	55.0	513	542	484	243
49	75.2	62.1	85.0	67.6	53.8	498	526	472	236
48	74.7	61.4	84.5	66.7	52.5	484	510	460	230
47	74.1	60.8	83.9	65.8	51.4	471	495	448	223
46	73.6	60.0	83.5	64.8	50.3	458	480	437	217
45	73.1	59.2	83.0	64.0	49.0	446	466	426	211
44	72.5	58.5	82.5	63.1	47.8	434	452	415	205
43	72.0	57.7	82.0	62.2	46.7	423	438	404	199
42	71.5	56.9	81.5	61.3	45.5	412	426	393	194
41	70.9	56.2	80.9	60.4	44.3	402	414	382	188
40	70.4	55.4	80.4	59.5	43.1	392	402	372	182
39	69.9	54.6	79.9	58.6	41.9	382	391	362	177
38	69.4	53.8	79.4	57.7	40.8	372	380	352	171
37	68.9	53.1	78.8	56.8	39.6	363	370	342	166
36	68.4	52.3	78.3	55.9	38.4	354	360	332	162
35	67.9	51.5	77.7	55.0	37.2	345	351	322	157
34	67.4	50.8	77.2	54.2	36.1	336	342	313	153
33	66.8	50.0	76.6	53.3	34.9	327	334	305	148
32	66.3	49.2	76.1	52.1	33.7	318	326	297	144
31	65.8	48.4	75.6	51.3	32.5	310	318	290	140
30	65.3	47.7	75.0	50.4	31.3	302	311	283	136
29	64.6	47.0	74.5	49.5	30.1	294	304	276	132
28	64.3	46.1	73.9	48.6	28.9	286	297	270	129
27	63.8	45.2	73.3	47.7	27.8	279	290	265	126
26	63.3	44.6	72.8	46.8	26.7	272	284	260	123
25	62.8	43.8	72.2	45.9	25.5	266	278	255	120
24	62.4	43.1	71.6	45.0	24.3	260	272	250	117
23	62.0	42.1	71.0	44.0	23.1	254	266	245	115
22	61.5	41.6	70.5	43.2	22.0	248	261	240	112
21	61.0	40.9	69.9	42.3	20.7	243	256	235	110
20	60.5	40.1	69.4	41.5	19.6	238	251	230	108

ROCKWELL STANDARD B 100 kg 1/16" ball	ROCKWELL STANDARD F 60 kg 1/16" ball	ROCKWELL SUPERFICIAL 15-T 15 kg 1/16" ball	ROCKWELL SUPERFICIAL 30-T 30 kg 1/16" ball	ROCKWELL SUPERFICIAL 45-T 45 kg 1/16" ball	E 100 kg 1/8" ball	Knoop 500 gm or over	Brinell 3000 kg / DPH 10 kg	Tensile Strength Thousand lbs. per sq. inch
100	—	93.0	82.0	72.0	—	251	240	116
99	—	92.5	81.5	71.0	—	246	234	112
98	—	—	81.0	70.0	—	241	228	109
97	—	92.0	80.5	69.0	—	236	222	106
96	—	—	80.0	68.0	—	231	216	103
95	—	91.5	79.0	67.0	—	226	210	101
94	—	—	78.5	66.0	—	221	205	98
93	—	91.0	78.0	65.5	—	216	200	96
92	—	90.5	77.5	64.5	—	211	195	93
91	—	—	77.0	63.5	—	206	190	91
90	—	90.0	76.0	62.5	—	201	185	89
89	—	89.5	75.5	61.5	—	196	180	87
88	—	—	75.0	60.5	—	192	176	85
87	—	89.0	74.5	59.5	—	188	172	83
86	—	88.5	74.0	58.5	—	184	169	81
85	—	—	73.5	58.0	—	180	165	80
84	—	88.0	73.0	57.0	—	176	162	78
83	—	87.5	72.0	56.0	—	173	159	77
82	—	—	71.5	55.0	—	170	156	75
81	—	87.0	71.0	54.0	—	167	153	74
80	—	86.5	70.0	53.0	—	164	150	72
79	—	—	69.5	52.0	—	161	147	—
78	—	86.0	69.0	51.0	—	158	144	—
77	—	85.5	68.0	50.0	—	155	141	—
76	—	—	67.5	49.0	—	152	139	—
75	99.5	85.0	67.0	48.5	—	150	137	—
74	99.0	—	66.0	47.5	—	147	135	—
73	98.5	84.5	65.5	46.5	—	145	132	—
72	98.0	84.0	65.0	45.5	—	143	130	—
71	97.5	—	64.0	44.5	100	141	127	—
70	97.0	83.5	63.5	43.5	99.5	139	125	—
69	96.0	83.0	62.5	42.5	99.0	137	123	—
68	95.5	—	62.0	41.5	98.0	135	121	—
67	95.0	82.5	61.5	40.5	97.5	133	119	—
66	94.5	82.0	60.5	39.5	97.0	131	117	—
65	94.0	—	60.0	38.5	96.0	129	116	—
64	93.5	81.5	59.5	37.5	95.5	127	114	—
63	93.0	81.0	58.5	36.5	95.0	125	112	—
62	92.0	—	58.0	35.5	94.5	124	110	—
61	91.5	80.5	57.0	34.5	93.5	122	108	—
60	91.0	—	56.5	33.5	93.0	120	107	—
59	90.5	80.0	56.0	32.0	92.5	118	106	—
58	90.0	79.5	55.0	31.0	92.0	117	104	—
57	89.5	—	54.5	30.0	91.0	115	103	—
56	89.0	79.0	54.0	29.0	90.5	114	101	—
55	88.0	78.5	53.0	28.0	90.0	112	100	—
54	87.5	—	52.5	27.0	89.5	111	—	—
53	87.0	78.0	51.5	26.0	89.0	110	—	—
52	86.5	77.5	51.0	25.0	88.0	109	—	—
51	86.0	—	50.5	24.0	87.5	108	—	—
50	85.5	77.0	49.5	23.0	87.0	107	—	—
49	85.0	76.5	49.0	22.0	86.5	106	—	—
48	84.5	—	48.5	20.5	85.5	105	—	—
47	84.0	76.0	47.5	19.5	85.0	104	—	—
46	83.0	75.5	47.0	18.5	84.5	103	—	—
45	82.5	—	46.0	17.5	84.0	102	—	—
44	82.0	75.0	45.5	16.5	83.5	101	—	—
43	81.5	74.5	45.0	15.5	82.5	100	—	—
42	81.0	—	44.0	14.5	82.0	99	—	—
41	80.5	74.0	43.5	13.5	81.5	98	—	—
40	79.5	73.5	43.0	12.5	81.0	97	—	—

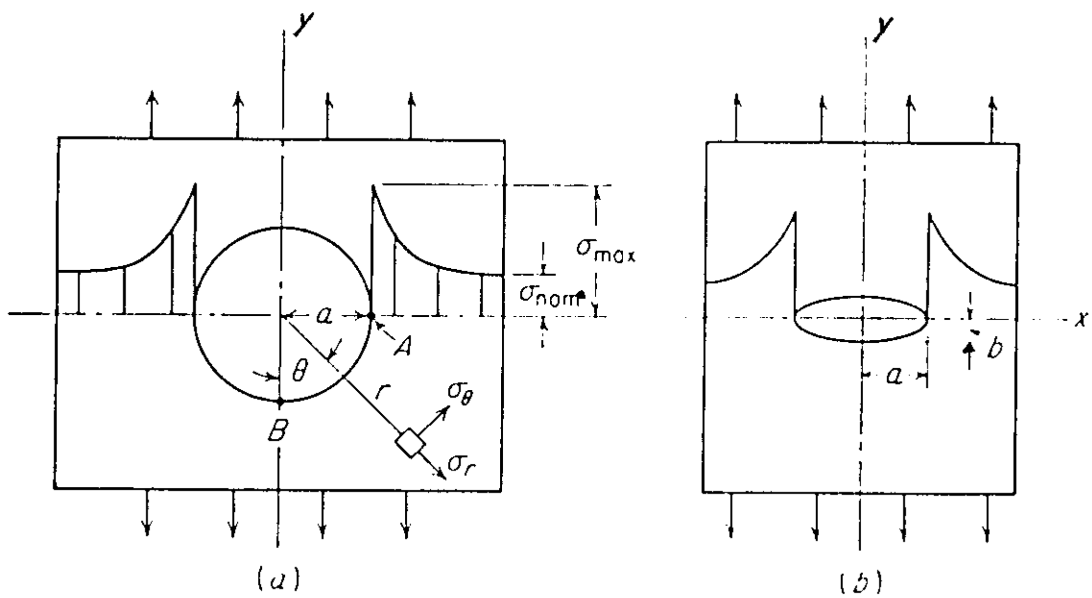
1.3.4 Stress Concentrations

1.3.4.1 Holes and Other Shape Changes

A geometrical discontinuity in a body, such as a hole or a notch, results in a non-uniform stress distribution at the vicinity of the discontinuity. At some region near the discontinuity, the stress will be higher than the average stress at distances removed from the discontinuity. Thus, a stress concentration occurs at the discontinuity, or stress raiser. Figure 5 shows a plate containing a circular hole which is subjected to a uniaxial load. If the hole were not present, the stress would be uniformly distributed over the cross-section of the plate and it would be equal to the load divided by the cross-sectional area of the plate. With the hole present, the distribution is such that the axial stress reaches a high value at the edges of the hole and drops off rapidly with distance away from the hole.

The stress concentration is expressed by a theoretical stress concentration factor K_t. Generally, K_t is described as the ratio of the maximum stress to the nominal stress based on the net section, although some workers use a value of nominal stress based on the entire cross-section of the member in a region where there is no stress concentrator.

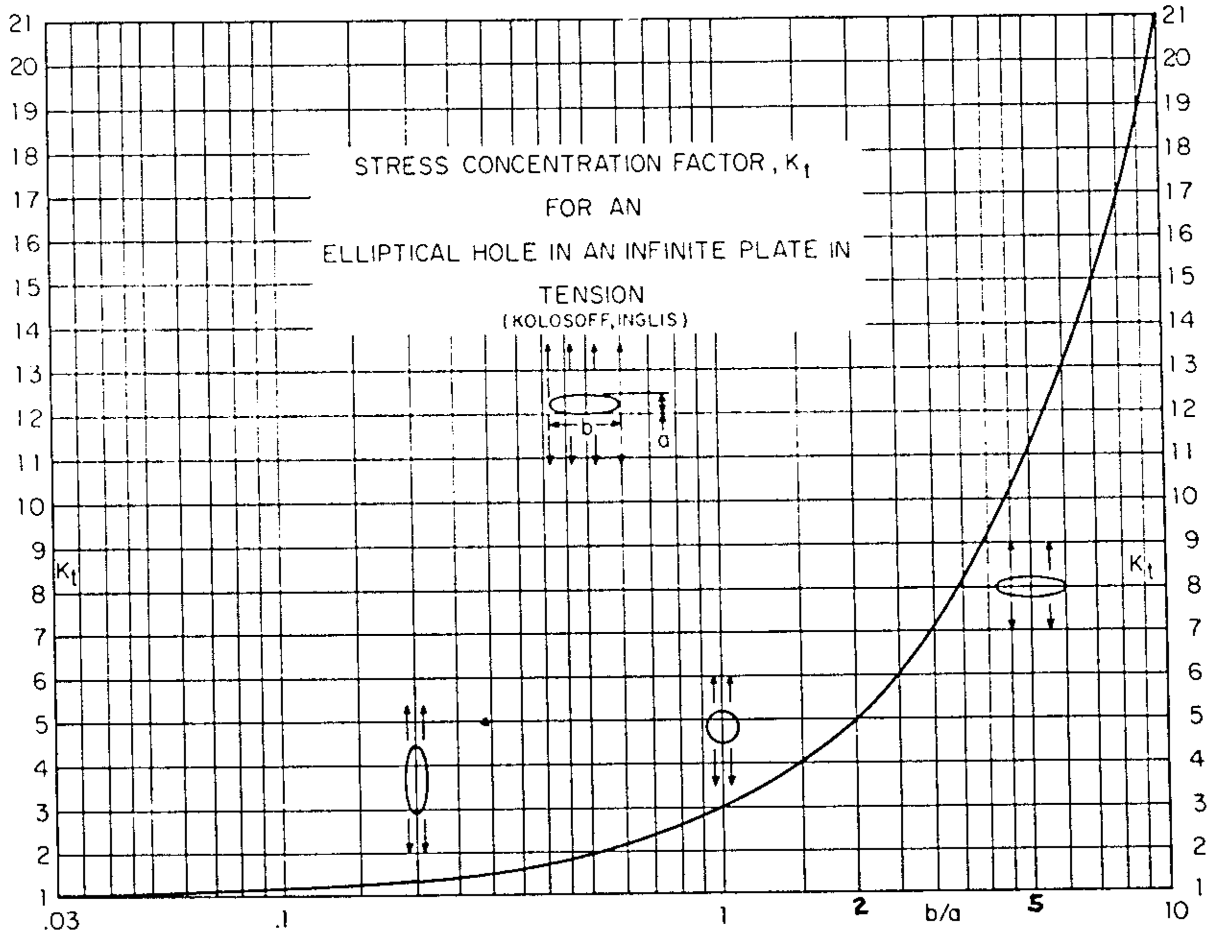
$$K_t = \frac{Smax}{Snominal}$$



<u>Figure 5</u>

Stress distributions are due to (a) circular hole and (b) elliptical holes.

Figure 6 illustrates the variation of stress concentration as the orientation of an elliptical hole changes with respect to the applied load. The stress concentration for a circular hole (b/a=1) is equal to 3.



<u>Figure 6</u>

Stress concentration of an ellipitical hole as a function of shape and orientation to the applied load. (after R. E. Peterson)

1.3.4.2 Notches

A notch can be broadly defined as any change of section which alters the local stress distribution. In this sense, the definition would include keyways, circumferential grooves, holes, contour change, threads, etc.

Theoretical stress concentration factors have been determined as a function of geometrical shapes. Figure 7 depicts theoretical stress concentration factors for various geometric shapes.

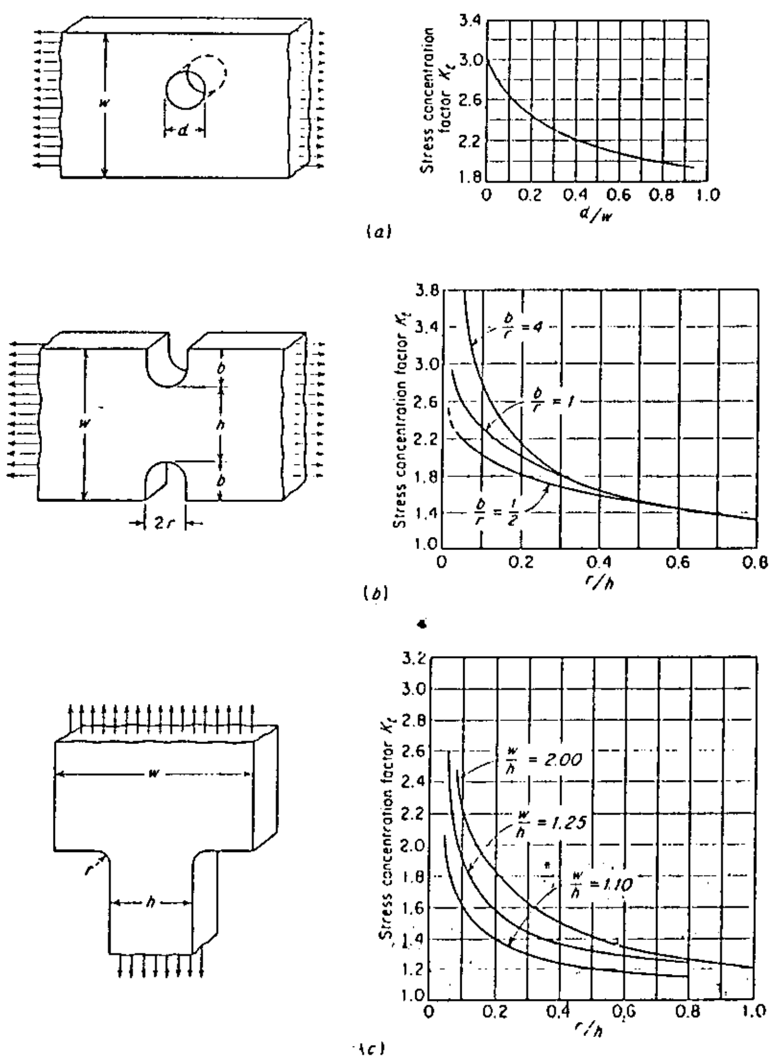


Figure 7

Stress concentration factors for various notch geometries.

In addition to obvious shape changes inducing stress concentrations, there are a variety of more subtle variations in local areas that depending upon orientation to the applied load may induce significant increases in stress. These variations are commonly referred to as notches and are typified by machining irregularities, stamp marks, internal cracks and certain metallurgical irregularities. Included in the metallurgical irregularities are forging laps and folds, decarburized areas in steels, inclusions and other metallurgical structure variations. Since notches can be quite sharp and have significant variations in dimensions, the theoretical stress concentration factors can be quite high.

Almost all structural materials are sensitive to notches and the fatigue strength of a part with a notch is less than for one without a notch.

There is evidence to indicate that the harder or higher tensile strength alloy is more notch sensitive than softer alloys. This high notch sensitivity of high strength steels makes it imperative that care be used in the design and maintenance of parts made from these steels.

Notches as such cannot be completely eliminated from any design, so we accept them and work toward lessening their deleterious effect. Generous filleting is a factor toward reducing the high stress concentration in some applications. Cold working the thread roots, rolling the threads and undercutting the last thread will improve the fatigue strength of threaded parts. Cold working or rounding the edges of holes will minimize the effect of the hole. In some instances, unloading notches adjacent to a design notch will reduce the stress concentration factor.

Many of the severe fatigue problems can be traced to notch effects. Since the area containing the notch has a lower fatigue strength than the unnotched area, failure will occur there first. For this reason, in examining the wreckage after a structural failure accident, the investigator should pay particular attention to fractures originating at changes in section, through bolt holes, etc. Not all of the fractures, of course, will be fatigue failures, but if there is such a failure, it is likely that it will occur at one of these locations.

1.3.4.2.1 Decarburization

Decarburization is the loss of carbon from the surface of a ferrous alloy as a result of heating in air which depletes the carbon at the surface. The end result is a soft skin on the surface of the part which reduces the fatigue properties considerably. Decarburization is an important consideration in spring design since the chromium-vanadium and silicon-manganese spring steels are especially susceptible to this effect. Decarburization does occur in other steels and resulting fatigue failures are found in bolts, forgings and other steel parts. The usual procedure to eliminate this difficulty is to machine the soft skin off the part or heat treat parts in a protective atmosphere. Decarburized surfaces are prone to fatigue crack generation.

1.3.4.2.2 Corrosion

When a corroded part is subjected to repeated loading, the fatigue life value for the metal is appreciably reduced. The pits on the corroded surface act as notches and produce the same deleterious effect as notches. If the repeated stress is applied at the same time as the corrosion is taking place, we have a special type of fatigue called corrosion fatigue. Corrosion effects in concert with cyclic stressing can promote the formation of cracks earlier and cause fatigue cracks to grow faster than if they corrosion were not present.

When corrosion fatigue occurs in combination with a stress concentration from a geometrical notch (drilled hole, scratch, change of section, etc.) the total reduction in fatigue strength is amazing. The following test data illustrate this very well. The rotating beam test endurance limit of an SAE 3140 (heat treated to 162,000 psi) specimen tested in air was 90,000 psi. When a specimen of the same material, but with a hole drilled in it, was tested in a stream of running water, the failing stress at 10,000,000 cycles was only 9,000 psi. In other words, the combination of corrosion fatigue and the notch had reduced the fatigue strength by a factor of ten.

There is a corrosion condition in which splitting along the grain lines takes place. This is quite commonly observed in extruded aluminum alloy members in which the material splits or flakes off. This is called exfoliation. Exfoliation is a specific form of intergranular corrosion where the end grain is attacked and the corrosion extends lengthwise in the part following the grain pattern in the part. The corrosion products work into a corrosion separation area and create a wedging action. This results in a stress concentration at the tip or sharp edge of the wedge. This damage may cause weakening of the member and initiate a fatigue crack.

1.3.4.2.3 Fretting

Fretting corrosion can be considered as a special type of oxidation and wear. Fretting corrosion occurs when two parts are clamped, press-fitted or shrunk together and subjected to vibratory loads. The low amplitude rubbing causes points on the surface to alternately weld, tear and oxidize. In steel, a red oxide powder is visible on the surface thus affected, while in aluminum or magnesium the fretting debris is black. Fretting corrosion roughens the surface, inducing a local stress concentration and early fatigue failure results.

1.3.4.2.4 Inclusions

All metals contain very small nonmetallic particles which are sulfides or oxides remaining from the refining of the metal. These particles are usually less than .001 inch in diameter and are called inclusions. Some people believe that inclusions in a material are responsible for a large number of service fatigue failures. Actually only a fraction of service failures result from this cause. All metals have inclusions. Before they can appreciably affect the fatigue life of a part, they must be large, relative to the inclusions normally found in the material and must be at or near the surface of the part. Generally speaking, inclusions at the center or near the center of a part will have little effect on the fatigue life. One researcher with one particular material and type of specimen concluded that large inclusions, or a cluster of many small ones might reduce the fatigue strength about 15 percent when the ratio of the depth of the inclusion in a radial direction to the diameter of the specimen was approximately 1 to 10. By and large, the effect of inclusions as inherent stress raisers is very minor compared to the imposed stress raisers in the form of poor fillets, oil holes and discontinuities of that character which are so common in neglected design and maintenance.

The investigator should examine the origin of fractures for unusually large defects. If suspicious areas are discovered, a laboratory metallurgical examination should be considered. Often a metallurgist will be able to determine whether a significant defect exists based upon a nondestructive microscopic study.

1.3.5 Residual Stress

Internal or residual stress can combine with the external applied stress to increase or to decrease the actual maximum stress. Certain heat treatment or fabrication methods can develop tensile stresses on the surface of a part. Locally heating a small area on the surface of a part will introduce tensile residual stresses when the part is cooled.

When the part is subjected to repeated service loading, the residual surface tensile stress adds to the design tensile stress and can produce a total stress higher than the designer anticipated. Early fatigue failure cam result. In examination of a fracture, however, there will be little evidence pointing to the existence of a residual stress. It is then necessary to study the history of the part from manufacturing to prior service records. Heat treatment, quenching, drawing or rolling, cold working, cold straightening and other fabrication processes may all result in residual stresses. A welded assembly that is not normalized or heat treated after welding is a typical process which can very often develop large residual stresses. Maintenance items that may have bent or heated the part are potentially significant.

1.3.5.1 Clamping and Press-Fit

Although the harmful effect of a sharp inside corner is generally appreciated, designers sometimes fail to realize that a similar condition exists when a shaft has a collar clamped to it, or when a press-fit assembly is made without any planned distribution of local stress (Figure 8). It has been shown experimentally that the endurance limit of a smooth shaft dropped from 88,000 psi to 45,000 psi when a collar was clamped to it. Clamping has caused a number of aircraft structure and component failures. In many cases, the initial crack is formed in the press-fit and cannot be detected until failure results. One authority has commented that the life of a fuel line tubing under vibration loading is primarily controlled by the fittings and clamps.

If the local stress is high in the area of fatigue crack initiation, several small cracks can form simultaneously. Once several small cracks link up, a series of steps are formed around the edge of the part. Figure 8 is an illustration of ratchet marks formed at a high stress fatigue origin.

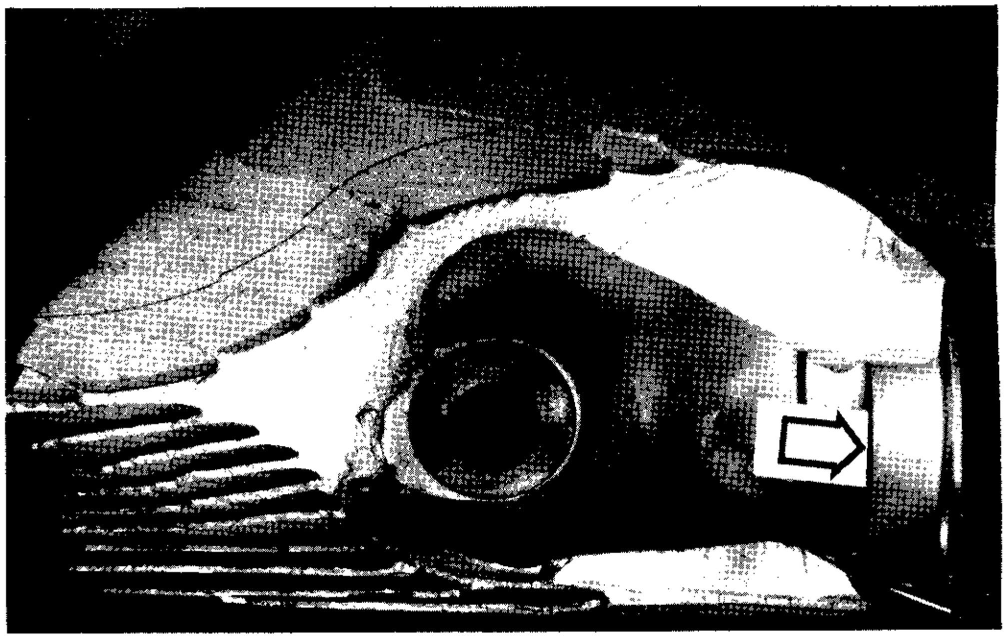


Figure 8

Pressed in steel collar contributed to fatigue of the aluminum cylinder head.

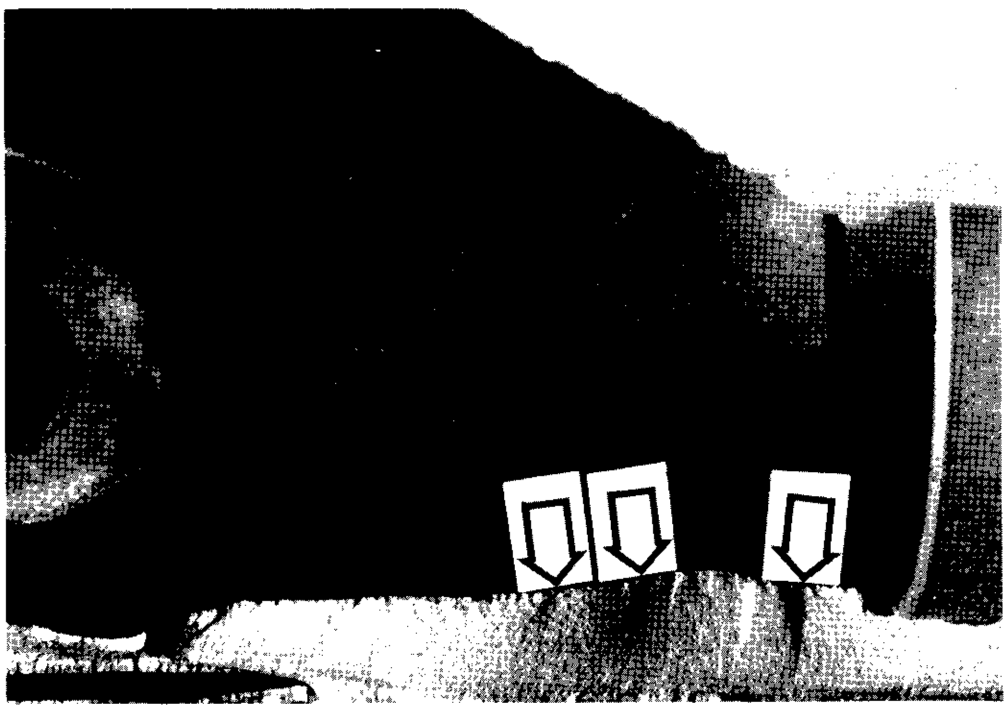


Figure 9

Ratchet marks (small offset steps) identify the multiple origins of a fatigue fracture.

Good design, assembly and maintenance procedures will eliminate a great number of such difficulties which turn up in service.

1.3.6 Surface Layers and Coatings

Surface layers may be intentionally designed and manufactured to resist fatigue and wear. In steel, a local outer layer or case can be produced by carburizing, nitriding, cyaniding, induction hardening or flame hardening. In each of these processes the benefit is two-fold. The harder outer layer, or case, has better fatigue resistance and the process induces beneficial surface compression stresses. In contrast, Aluminum Alclad sheet, on the other hand, has a lower fatigue strength than the unclad sheet, since the cladding of pure aluminum is weaker than the core material. If loaded excessively, even surface hardened parts may fatigue. The exact location of the origin will depend on the hardness of the case and core and the thickness of the case. The crack may start at the outside surface, the case-core interface, or internally.

Plating is applied to surfaces to prevent wear or as a protection against corrosion, but like most other surface treatments, it affects the strength of the member. The softer electroplating processes such as zinc, lead and copper have little effect on the fatigue strength of steel. Hard chrome or nickel plating, on the other hand, sometimes decreases the fatigue strength. Hard chrome plating is used successfully because the cyclic stresses are kept below the allowable stress for the material.

1.3.7 Cold Working (Rolling, Drawing and Shot Peening)

In cold working, the material is strained beyond the yield point and caused to flow plastically. Metal is cold worked in various ways. Sheet metal is cold worked by rolling or stretching. Rod material is drawn through dies. Shot peening is a form of cold working. In this process, the part to be worked is bombarded by small metal balls approximately 0.025" in diameter. Shot peening is often used in aircraft design to lessen the effects of stress concentration due to changes in section.

Insofar as fatigue is concerned, cold working produces two beneficial results. First, surface residual compression stresses are induced. Second, the cold working increases the endurance limit. Extreme cold working, however, can have an injurious effect on a part.

1.3.8 Toughness

It is highly desirable for critical structural components to be both strong and tough. Toughness relates to the energy absorbed by the material prior to fracture. In most ductile materials, a structural member will distort excessively before breaking. There may be no concern with its fracturing characteristics because failure in service will be by yielding. If, on the other hand, the body breaks after a slight amount of deformation, its fracture behavior becomes all-important, and as a consequence, properties that measure fracture resistance are primarily concerned with brittle fracture.

This type of failure occurs in either of two classes of materials. Glass and very hard parts are brittle under almost any service or testing condition. Other structural materials, of which steel is most important, perform in a far more complex fashion. These may be ductile under some conditions and brittle under others, making their behavior in service difficult to predict. They may be referred to as "frangible". Because of their dual nature and their great practical importance, these are the materials on which most brittle fracture studies are conducted.

Some values for toughness are easily obtained. If a tensile bar of, say, mild steel were stretched, its toughness would be found by measuring the area under its true stress-strain curve. Toughness is expressed in units of energy. If the stress-strain curve were plotted in pounds per square inch and strain in units of inch/inch, the energy value would be in-lbs/in^3.

Unfortunately, toughness measured by such a simple procedure, especially at room temperature, does not correlate with fracture resistance in service. Steel in uniaxial tension breaks in shear (i.e., shows a fibrous surface) while brittle fractures that occur in service are found to have granular surfaces, indicating a cleavage mode of failure. Apparently, this lack of correlation between the laboratory tests and field behaviors is due to different mechanisms of failure. Much of the fracture or toughness testing, especially on steels, is aimed at measuring the relative ease with which cleavage can be produced. By comparing the "cleavage tendency" of one steel with that of another on which service experience is available, it is possible to rate the new material for a particular application.

Impact tests methods have been developed to compare the toughness of various materials. These tests usually incorporate a notch of some type and the application of a load at a high rate.

Notches of almost every conceivable shape have been cut into tension or bending specimens to simulate either abrupt section size changes (e.g., at the hatch corners of ships) or accidentally formed cracks. One of the most common of these notched test bars has a square cross-section and is known as a Charpy V-notch specimen. These specimens are broken as simple beams in impact. The Charpy test is one of the most deeply rooted toughness tests and is included in the specifications for many products. It is performed on an impact machine equipped with a heavy pendulum hammer pivoted about a horizontal axis, which is raised to a fixed height prior to the test. In running impact tests, a specimen is placed in the path of the pendulum, and the energy required to break the sample is measured as the energy difference between the initial and final hammer swing heights. This absorbed energy, expressed in foot-pounds, is a measure of the material's toughness. The hammer is made to strike the Charpy bar behind the notch. Since the specimen is supported on its two ends, it breaks as a simple beam with the notch on the tension side.

Impact tests yield their most significant data when applied to materials that exhibit a transition temperature. The transition temperature is the temperature at which there is an abrupt change in the fracture mode from ductile fracture to brittle failure.

Materials subjected to excessive impact loading at temperatures below the transition temperature will fracture in a brittle manner and will absorb little energy in the process. Materials subjected to excessive impact loading at temperatures above the transition temperature will fail in a ductile manner and will absorb a significant amount of energy. Thus, there are situations that under a given impact load the material may or may not break depending upon temperature. To locate this change from tough and ductile to brittle, tests must be run over a range of temperatures.

There are three methods for evaluating the degree of brittleness (or its absence) at each testing temperature: (1) from the energy required to produce fracture; (2) from the fracture appearance, i.e., the relative amounts of shear and cleavage; and (3) from the ductility measured as lateral contraction at the notch bottom. Transition temperature values based on these are referred to as energy transition, fracture appearance, or fracture transition and ductility transition, respectively. Several methods have been used to define the transition temperature. There are:

a. The temperature at which the energy-temperature curve intersects the 15 ft.-lb. level. For example, a minimum impact energy of 15 ft.-lb. at -40 F. is frequently specified in acceptance tests.

b. The temperature at which the energy-temperature curve intersects the 40 ft.-lb. level.

c. "Average energy transition," the temperature at which the impact energy is midway between its values at high and low temperatures, i.e., upper and lower shelf heights.

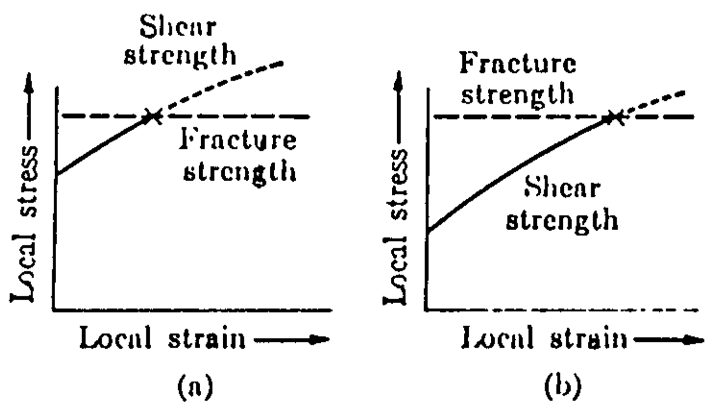


Figure 10

Deformation prior to fracture. (a) Low temperature or high rate of loading. (b) High temperature or slow loading

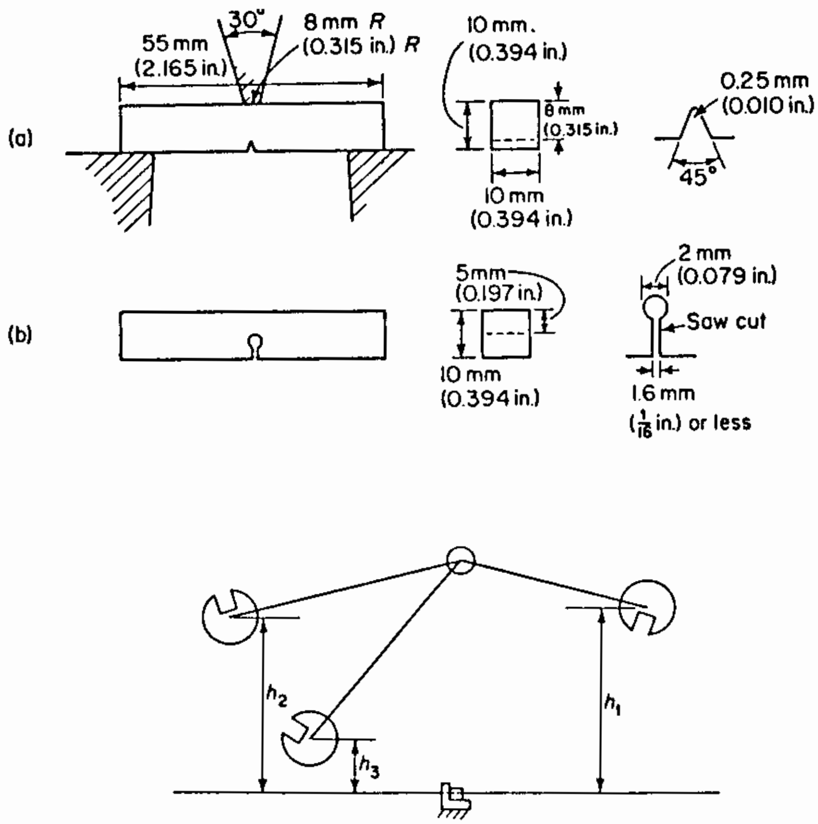
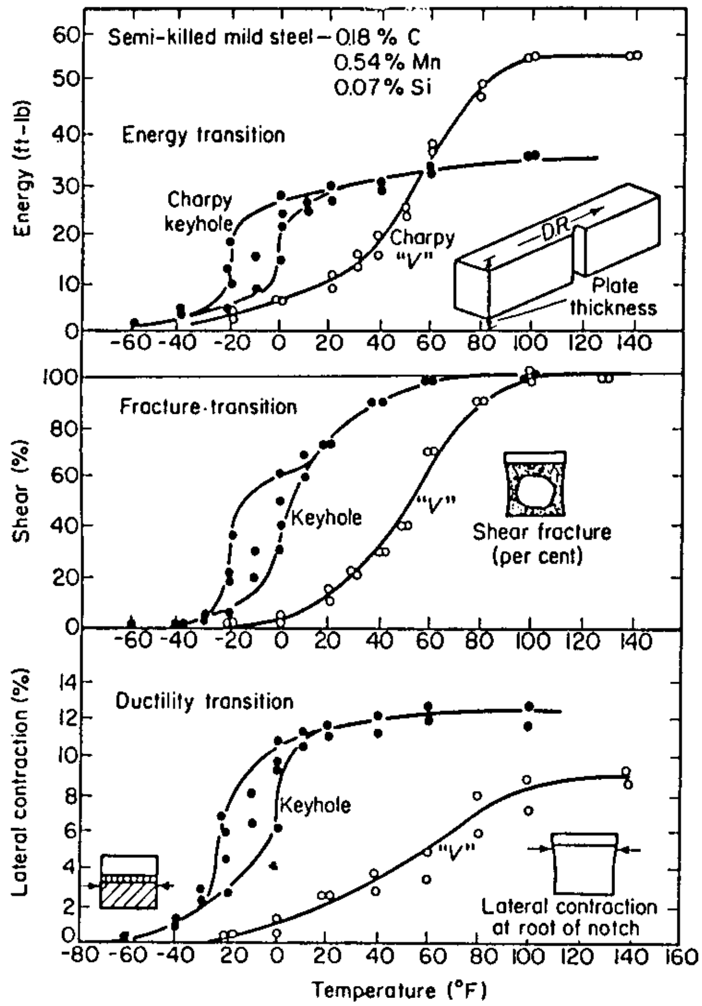


Figure 11

(a) "V" notch Charpy bar. (b) Keyhole notched Charpy bar. (c) Schematic diagram of impact machine.



<u>Figure 12</u>

Three methods for measuring the transition temperatures of Charpy "V" and Keyhole notched specimens.
From W. S. Pellini

1.3.9 Fatigue

Cyclic loading of structures is a condition which frequently arises and induces stresses which fluctuate between values. For example, the outer fiber on the surface of a rotating shaft, subjected to bending loads, undergoes both tension and compression with each revolution of the shaft. If the shaft is rotating at 3000 RPM, the fiber is stressed in tension and compression 3000 times every minute. This kind of loading produces stresses which are called repeated, alternating, or fluctuating stresses and can induce a progressive type failure.

Fatigue is simply the progressive failure of a part under repeated loading. Fatigue is not a new problem. In fact, it is a very old problem, dating back to the time when metallic materials were first used for structural components. As early as 1858, Wohler, one of the earliest researchers in the field, concluded after conducting many tests: "Wrought iron and steel will rupture at a stress not only less than the ultimate static strength of the material but even less than the elastic limit, if the stress is repeated a sufficient number of times." This finding is as true today as it was then.

A fatigue can begin as a minute crack or cracks. These initial cracks may be so minute that they cannot be detected by the naked eye and are even difficult to locate by dye penetrant, Magnaflux or X-ray inspection. These minute cracks will initiate at a discontinuity in the material such as changes in cross-section, keyways or holes. Less obvious initiation sites are machining irregularities, stamp marks, internal cracks and certain metallurgical irregularities, as discussed earlier.

Such discontinuities and irregularities produce stress concentration effects. Once a fatigue crack has initiated, it will grow larger under the actions of fluctuating stress. As the fatigue crack grows, the stress concentration effect associated with the growing crack becomes greater as the crack grows and the crack will progress more rapidly. As the crack gets larger, the remaining cross-sectional area gets smaller and the stress increases in magnitude until the remaining area fails suddenly in overload. A fatigue failure, therefore, has two distinct areas of failure (Figure 13). The first area is the progressive development of the crack while the second area is due to the sudden overload fracture.

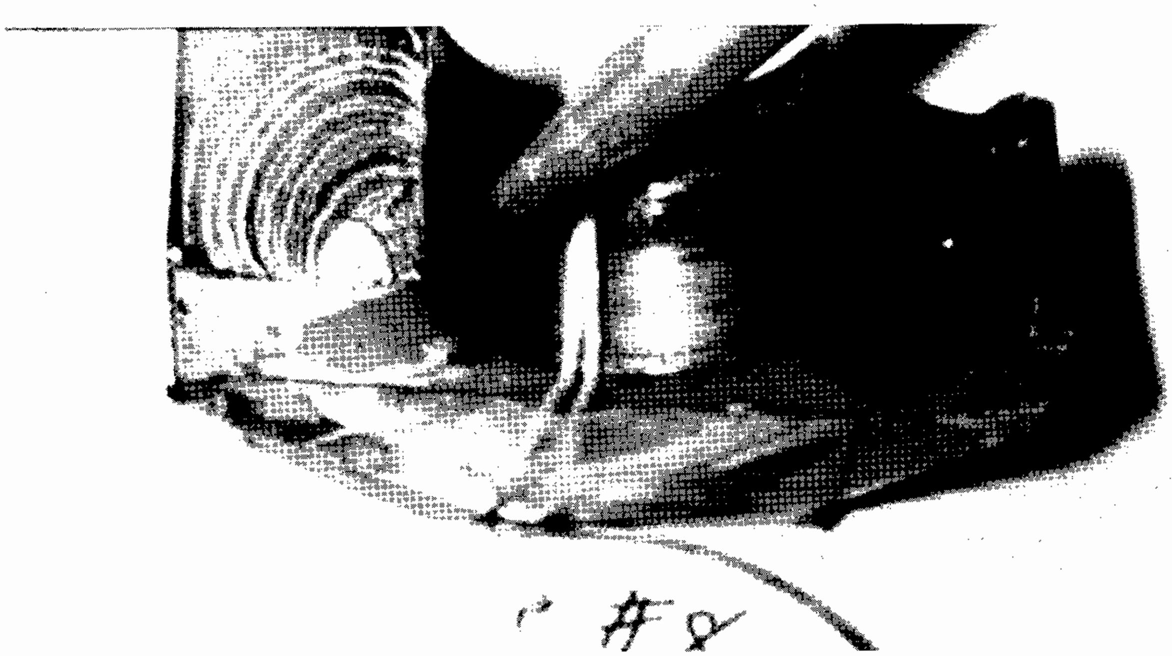


Figure 13

Fatigue fracture of an engine mount that originated at the indicated corner. At the later stages, the crack was "popping" from periodic high stress events.

1.3.9.1 Fatigue Strength and Endurance Limit

To determine the strength of a material under the action of fatigue loads, specimens are subjected to repeated or cyclic forces of specific magnitudes and the number of cycles or stress reversals required for failure are counted.

To establish the fatigue strength of a material, quite a number of tests are necessary because of the statistical nature of fatigue. For the rotating-beam test a constant bending load is applied, and the number of revolutions (stress reversals) of the beam required for failure is recorded. The first test is made at a stress which is somewhat under the ultimate strength of the material. The second test is made with a stress which is less than that used in the first. This process is continued, and the results plotted as an S-N diagram (Figure 14).

The ordinate of the S-N diagram is called the fatigue strength S_f; a statement of this strength must always be accompanied by a statement of the number of cycles N to which it corresponds. In the case of the steels, a knee occurs in the graph, and beyond this knee failure will not occur no matter how great the number of cycles. The strength corresponding to the knee is called the endurance limit S_e, or the fatigue limit. The graph of Figure 14 never does become horizontal for nonferrous metals and alloys, and hence these materials do not have an endurance limit.

Experimental data indicates the fatigue limit or endurance limit of steel is approximately equal to one-half the ultimate tensile strength.

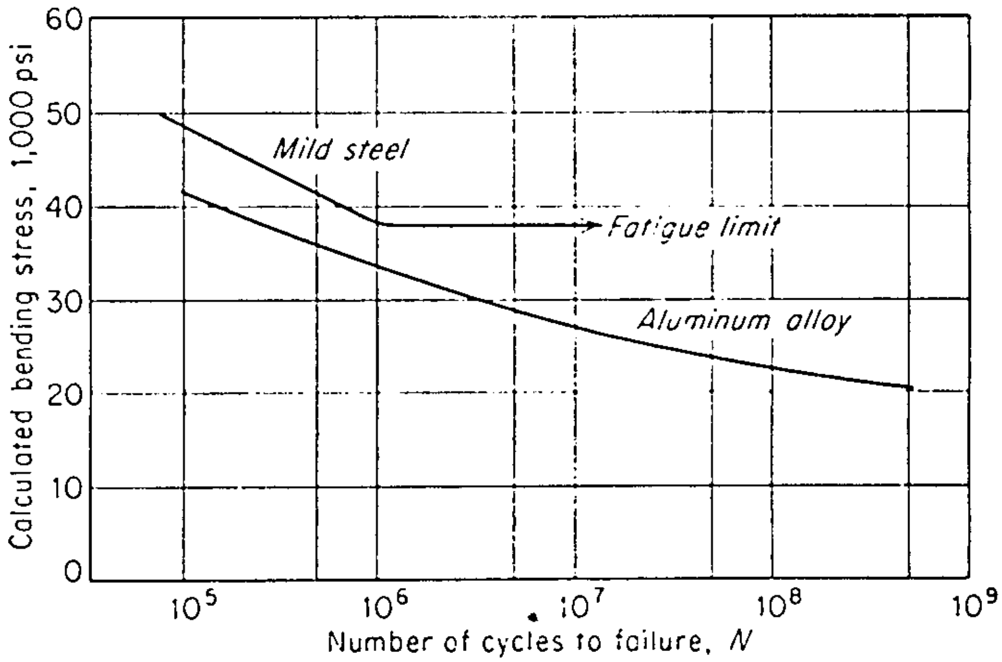


Figure 14

Typical fatigue curves for ferrous and nonferrous metals.

1.3.9.2 Fatigue Crack Propagation

The life of structural components that contain cracks or that develop cracks early in their lives may be governed by the rate of subcritical crack propagation. Inspection procedures are usually used to establish upper limits on the size of undetectable discontinuities rather than actual crack size. These upper limits are determined by the maximum resolution of the inspection procedure. Thus, to establish the minimum fatigue life of structural components, it is reasonable to assume that the component contains the largest discontinuity that cannot be detected by the inspection method. The useful life of these structural components is determined by the fatigue-crack-growth behavior of the material. Therefore, to predict the minimum fatigue life of structural components and to establish safe inspection intervals, an understanding of the rate of fatigue-crack propagation is required. The most successful approach to the study of fatigue-crack propagation is based on fracture-mechanics concepts.

The general nature of fatigue crack propagation using fracture mechanics techniques is summarized in Figure 15. A logarithmic plot of the crack growth per cycle, da/dN, versus the stress-intensity factor range, ΔK. Results of fatigue crack growth rate tests for nearly all metallic structural materials have shown that the da/dN versus ΔK curves have three distinct regions. The behavior in Region I (Figure 15) exhibits a fatigue crack growth threshold, ΔK_{th}, which corresponds to the stress-intensity factor range below which cracks do not propagate.

At intermediate values of ΔK (Region II in Figure 15), a straight line is usually obtained on a log-log plot of ΔK versus da/dN. This is described by the power-law relationship:

$$\frac{da}{dN} = C\Delta K^m$$

where a = crack length.
N = number of cycles.
ΔK = stress intensity factor fluctuation.
C and m are constants.

Fatigue crack growth rate data for some steels show that the primary parameter affecting growth rate in Region II is the stress-intensity factor range and that the mechanical and metallurgical properties of these steels have negligible effects on the fatigue crack growth rate in a room-temperature air environment.

For some steels, the stress ratio and mean stress have negligible effects on the rate of crack growth in Region II. Also, the frequency of cyclic loading and the waveform (sinusoidal, triangular, square, trapezoidal) do not affect the rate of crack propagation per cycle of load for some steels in benign environments. Crack growth rates in corrosive environments often show accelerated growth and are affected by cyclic frequency and stress cycle wave forms.

At high ΔK (Region III in Figure 15), unstable behavior occurs, resulting in a rapid increase in the crack growth rate just prior to complete failure of the specimens.

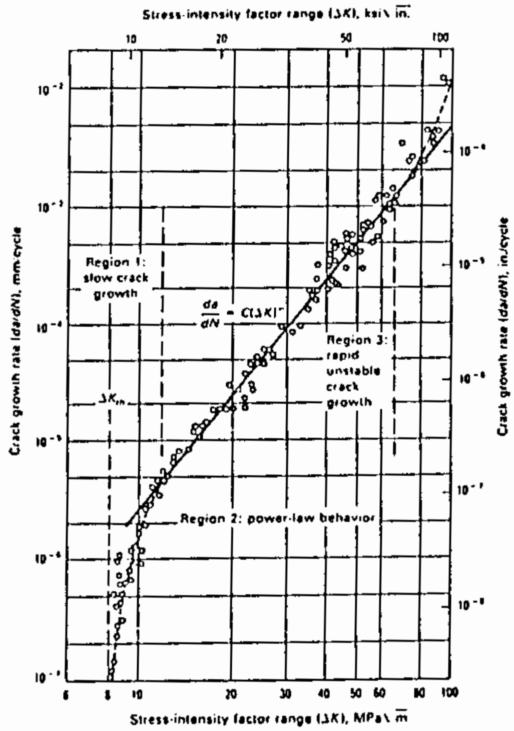


Figure 15

Typical crack growth behavior for steel.

1.3.9.3 Cumulative Fatigue Damage

Instead of a single reversed stress over its life, suppose a part is subjected to periods of high cyclic stress and other periods of lower cyclic stress. Under these conditions, how is the fatigue life of a part subjected to these variations in reversed stresses estimated? A search of the literature reveals that this problem has not been solved completely. Therefore, the approach presented here should be employed as a guide. An approach consistently in agreement with experiment has not yet been reported in the literature of the subject.

The theory which is in greatest use at the present time to explain cumulative fatigue damage is the Palmgren-Miner cycle-ratio summation theory, also called Miner's rule. This theory combines the fraction of fatigue life consumed at each stress level.

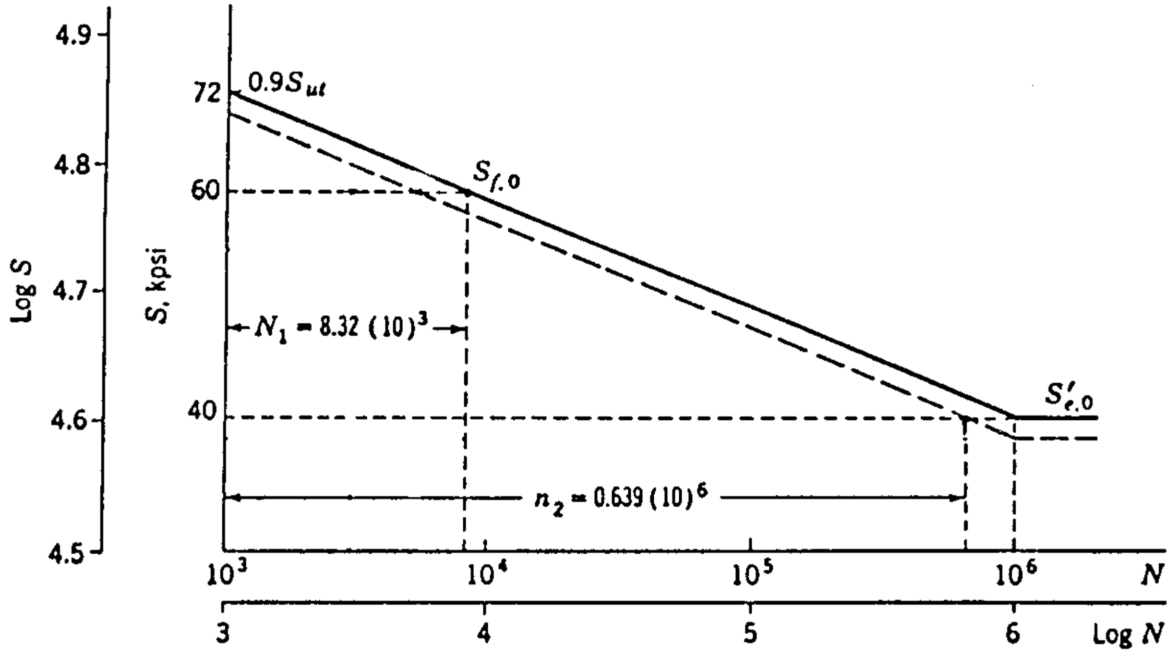


Figure 16

Use of Miner's rule to predict the endurance limit of a material that has been over stressed for a finite number of cycles.

In the above graph, we can see that this material can withstand 8000 cycles of a stress of 60,000 psi. If in actual service the affected part only sees 3000 cycles at this stress, 37½% of its life has been consumed (3000/8000 x 100%). Treating other stress levels in a similar fashion, we would expect failure when the combined fatigue damage equals 100%.

2.0 STRUCTURES - GENERAL

2.1 AIRCRAFT LOADS AND STRUCTURES

In flight and on the ground the aircraft is subject to various loads which result in stresses in the structural members of the aircraft. The investigator, in order to seek out the causative factors involved in an aircraft accident, must possess a basic understanding of the loads to which an aircraft is subjected to both during ground operations and throughout the flight regime. The investigator must also understand basically how the flight and ground loads are absorbed by the structure within the operational envelope defined by the manufacturer.

The remainder of the chapter is not intended to make an aerodynamist or a structures engineer out of the investigator, but to provide a basic understanding of aircraft loads and structures. This chapter is subdivided into types of loading (loads in flight and on the ground), a basic description of typical aircraft structural elements, and discussion of how loads are carried by primary structural members.

2.1.1 Aircraft Loadings - Variations in Loading Application

The mode or method of load application to an aircraft structure has an extremely important bearing on the way in which a part fails in service. Any categorizing of variations in loading applications is arbitrary since, in general, the difference between types is only one of degree. Thus, one mode of load application blends into another as rate of loading is decreased or increased. Changes in the frequency of loading will result in a change in the mode. No hard or fast rule can be applied.

For this reason, in the following discussion the various modes or methods of loading are arbitrarily divided into three basic types: static, repeated and dynamic.

2.1.1.1 Static Loading - Static loading of a structure can be subdivided into short time static loading and long time static loading:

2.1.1.2 Short time - In short time static loading, the load is applied so gradually that all parts are, at any instant in time, essentially in equilibrium, i.e., the simple, conventional stress formulas can be used directly. In testing, the load is increased progressively until failure of the structure results and the total time required to produce failure is not more than a few minutes. In service, the load is increased progressively to its maximum value, maintained at that maximum value for a limited time, and not reapplied often enough to make fatigue a consideration. The ultimate strength, elastic limit, yield point, yield strength and modulus of elasticity of a material are usually determined by short time static tests. As will be explained more fully later, this is the type of loading application used in conjunction with present day aircraft design criteria. Loads imposed upon the aircraft by various maneuvers, or by isolated peak gusts are generally considered for aircraft design purposes as static loads.

2.1.1.3 Long time - Under conditions of long time static loading, the maximum load is applied gradually to the structure, but the load is maintained for a significant time period. In testing, it is maintained long enough to enable its probable final effect to be predicted. In service, it is maintained continuously or intermittently during the life of the structure. The creep or flow characteristics of a material and its probable permanent strength are determined by long time tests at the temperature prevailing under service conditions. This type of loading application is generally only important to structures operating at elevated temperatures. When a part is loaded for a relatively long time at higher than normal temperatures, it will begin to creep or distort at a more or less uniform rate. The strength of the structure or part is reduced from its room temperature value. At the present time, there are few applications of this type of loading in civil aircraft. However, as aircraft speeds increase and skin temperatures are sufficiently high, this type of loading will take on increased significance. Turbine engine components, such as disks and blades, are life limited due to a process called creep stress rupture. Stress corrosion cracking and hydrogen embrittlement occur after a period of time which involves nucleation and propagation of cracks under long duration steady loads. These will be discussed in some detail in Chapter 3.0.

2.1.1.4 Repeated Loading - In repeated loading, the load is applied and wholly or partially removed or increased many times. This is the type of loading application which is commonly associated with fatigue. Generally speaking, repeated loading implies a large number of load applications. Under certain conditions, repeated loading of relatively few cycles can produce an effect similar to a large number of cycles. This point is explored further in the discussions of fatigue in Chapters 1.0 and 3.0.

Remember that the strength of a part is reduced from its static strength value when the part is loaded repeatedly. The actual reduction varies with the stress level and the number of repetitions. A typical example illustrates this point: A stress of about 70,000 pounds per square inch (psi) is required to break a round bar of 2014-T6 aluminum alloy under static tension loading conditions.

This same bar would fail after about 100,000 cycles under conditions of a reversed bending load that produces a maximum repeated stress of about 20,000 psi. Cycles of this order of magnitude can be encountered within the lifetime of an aircraft. In aircraft, atmospheric gusts, vibration and ground-air-ground cycles produce a repeated type of loading.

2.1.1.5 Dynamic Loading - In the two types of loading discussed above, a state of equilibrium existed, i.e., the external loads were in balance with the internal loads at each point in time.

In dynamic loading, the loaded structural member is in a state of vibration and static equilibrium does not exist for a period of time. Loosely speaking, there are two classes of dynamic loading: sudden loading and impact loading.

2.1.1.6 Sudden Loading - Sudden loading occurs when a weight or "dead load", not in motion, is suddenly placed upon a structural member of an aircraft structure. A beam would be loaded in this manner if a weight were suspended by a cord which allowed the weight to just touch the beam, before the cord was cut. The stress and deflection produced upon sudden load application would be approximately twice as great as if the weight were eased gradually onto the beam as in static loading. Any force will cause approximately twice as much stress and deformation when applied suddenly as when applied progressively. The actual magnitude of the "magnification factor" depends upon the type of force or load being considered and upon the stiffness of the system. In aircraft, gust loads are forms of sudden loading, although as will be seen later, they are handled during analysis of the structure as static loads.

2.1.1.7 Impact Loading - The term impact is generally associated with a moving body striking either another moving body or a fixed object. The result of a moving body striking a fixed object can result in the generation of very large impact related forces as is the case when an aircraft strikes the ground. The angle of the impact and the velocity have a large bearing on the magnitude of the forces developed. A small angle and low velocity allows the aircraft to dissipate the energy over a long distance minimizing the impact forces developed. The larger the impact angle and the higher the velocity, the greater the impact forces generated. The aircraft structure deforms under the above conditions absorbing energy, thereby resulting in the development of load factors significantly less than would be developed if two unyielding rigid bodies were to collide. The generation of large forces is a product of the body weight, velocity just prior to impact, and the stopping distance.

Materials used for the construction of aircraft normally would be expected to fail in a ductile manner under static loading conditions. However, under impact loading conditions, the same materials can be made to fail in a brittle manner if the rate at which the load is applied is sufficiently high.

2.2 DESIGN LOADS

Present day aircraft are designed essentially for static loads under steady state conditions. In the next section, the loads used in the design are discussed in some detail. These loads are assumed to act on the aircraft in such a manner that static equilibrium exists for applied external loads and the resulting internal loads. These loads fall into the short time static loading category. At the present time, methods for determining the life of parts or components under repeated loading are developing rapidly. Considerable research is under way to devise suitable methods to handle repeated loading conditions. In the past, the designer has intuitively proportioned the size of the structural members so that the effects of repeated loading are reduced in importance by reducing the internal stress levels. Dynamic loading has not been too critical in the past because conservatisms in the design criteria and relatively high stiffnesses in the primary structure have been such that the magnification factors developed by this loading have been either incorporated into the static design criteria, or they have been small enough to neglect.

2.2.1 Future Prospects

Future trends are always difficult to predict. It can be said however, that repeated loading and dynamic loading are assuming more and more importance. Catastrophic failure of a primary structure due to repeated loading (fatigue) is of constant concern to the designer. Dynamic effects are being given closer attention today. Lower design load factors, higher speeds and decreased stiffnesses due to thinner wing designs all aggravate the situation and make these two types of loading more significant. Whether the present static load criteria will be replaced by more complex criteria, which would include repeated loading and dynamic loading effects, depends to a great extent on the future development in these fields. Until such time as aircraft are designed for these effects and even afterwards perhaps, the aircraft accident investigator must examine all structural failures to determine if some other cause aside from exceeding the static strength is responsible for the failure. Fatigue failures often involve parts initially damaged in service or improperly maintained resulting in fracture of well designed components.

2.2.2 External Aircraft Loadings

The actual loads imposed upon the aircraft structure during in-flight and ground operation are extremely complicated in nature and an analysis of the infinite number of possible loading combinations would be an impossible task. For practical reasons, the aircraft structure is designed for a relatively small number of loading conditions, selected to bracket all the critical flight conditions likely to be encountered by the particular aircraft during its useful operational lifetime. These loading conditions or requirements are set forth in the Federal Aviation Regulations (FAR's) for all aircraft. Aircraft in excess of 12,500 pounds design gross weight are currently designed and certified according to FAR Part 25 requirements. Aircraft of 12,500 lbs. and less in design gross weight are currently designed and certified according to FAR Part 23 requirements. The different loading conditions specified in these regulations are, for the most part, of the static type, i.e., the aircraft is assumed to be balanced or in state of equilibrium.

In the following subsections the two main types of loading, air loads and ground loads, and the significance of vibration loads in aircraft design are briefly discussed.

2.2.3 Air Loads

Two separate types of air loads are imposed upon the aircraft in flight: maneuvering loads and gust loads. Although these loads often occur in combination, the design conditions are so selected that these loads can be considered separately.

2.2.4 Maneuvering Load Factors

When an aircraft is flying in straight and level flight, the lift forces on the wing and tail are in balance with the weight of the aircraft and the aircraft is said to be operating at a one "G" level. If the aircraft is maneuvered, the forces acting on the aircraft are either raised or lowered and the G level, or load factor, is accordingly increased or decreased. The exact load factor developed in any particular maneuver of a certain airplane depends, for the most part, on the rate of change on control surface displacement and the speed of entry into the maneuver. The higher the rate of change, or the higher the speed, the higher will be the resultant load factor developed by the aircraft in response to the control displacement. If at any instant in a maneuver the acceleration forces are balanced by equal and opposite inertia forces, the airplane can be considered to be in a state of equilibrium and a static type of structural analysis may be conducted.

For design purposes, then, the airplane is assumed to be subjected to certain load factors or acceleration units and the structure is designed to incorporate strength capability to react the define loads. For transport type aircraft where maneuverability is secondary, strength is provided for a limit positive load factor of 2.5 G's. The load factor used in light plane design depends upon the intended use category as defined by FAR Part 23.

Aerobatic type aircraft, for which all maneuvers are permitted, are designed for a positive maneuvering limit load factor of 6.0 G's.

Aircraft designed to the utility category requirements of FAR Part 23 are provided with a strength of 4.4 G's limit load factor. This type of aircraft is intended for only "mild" maneuvering and certain maneuvers are prohibited. Light planes designed under the FAR Part 23 normal category requirements are intended mainly for transportation purposes in which only ordinary maneuvers are permitted and, hence, the limit load factor is 3.8 G's. Some of our modern light planes are designed to meet both normal and utility category requirements by reducing the aircraft gross weight to allow flight operation at the higher limit load factor of the utility category. From the above, it can be seen that an aircraft is designed for an arbitrary load factor, the exact value depending upon the intended use of the airplane. Any of these values can be exceeded by the pilot if he fails to observe the operating limitations of the aircraft as defined in the limitations section of the aircraft owners manual. Very often people will ask "Why can't you design the airplane so that the pilot cannot fail the structure during any flight maneuver?" The answer, of course, is that the aircraft structure could be designed in this manner, but that the cost in weight would be prohibitive. With some of our modern light planes, while operating at relatively high speeds, the design limit load factor would have to be of the order of 20 G's or more to preclude structural failure within the aircraft maximum speed limitations.

2.2.5 V-n Diagram

The basic maneuvering flight conditions for any aircraft could be given by stating the limiting values of acceleration and speed for which the airplane is to be designed. It has been found more convenient however, to represent these conditions graphically on a diagram which is referred to as a V-n diagram. Two V-n diagrams are shown in Figures 17a and 17b. Figure 17a shows the operating envelope for typical aircraft design conditions and is also useful in illustrating the structural operating limitations for the aircraft. The lines AB and CED represent the restricted positive and negative maneuvering load factor which are limited to speeds less than the line BD, the design dive speed. These restricted maneuver lines intersect the airplane $C_{N,MAX}$ curves at A and C. At speeds between A and B, the pilot must be careful not to exceed the limiting maneuvering load factor since, in general, it would be possible for him to manipulate the controls to exceed these values. At speeds below A and C, the pilot need not worry particularly, since the $C_{N,MAX}$ of the airplane will be reached before the limiting values given by lines AB and CED can be developed and the airplane will stall. Generally speaking, if the airplane is designed for the air loads resulting from the conditions A, B, C, E and D, or the corners of the V-n diagram, it will be safe from a structural strength standpoint if it is flown within the specified boundries of velocity and acceleration. Conditions A and C are generally referred to as high angle of attack conditions and B and D as low angle of attack conditions.

With the design air speed and acceleration conditions specified on the V-n diagram as a basic criteria, the designer balances the airplane for the various forces and computes the air loads deveoped on individual components, such as the wing, tail, etc. These are then used by designer to develop the type of structure and to size the various individual structural members and components. From an investigation viewpoint, a thorough understanding of the significance of the V-n diagram is often useful in evaluating in-flight structural failures. Different aircraft, or different models of a particular aircraft are designed to meet the requirements of different categories and an effort should be made to determine if the aircraft was being operated in accordance with its limitations.

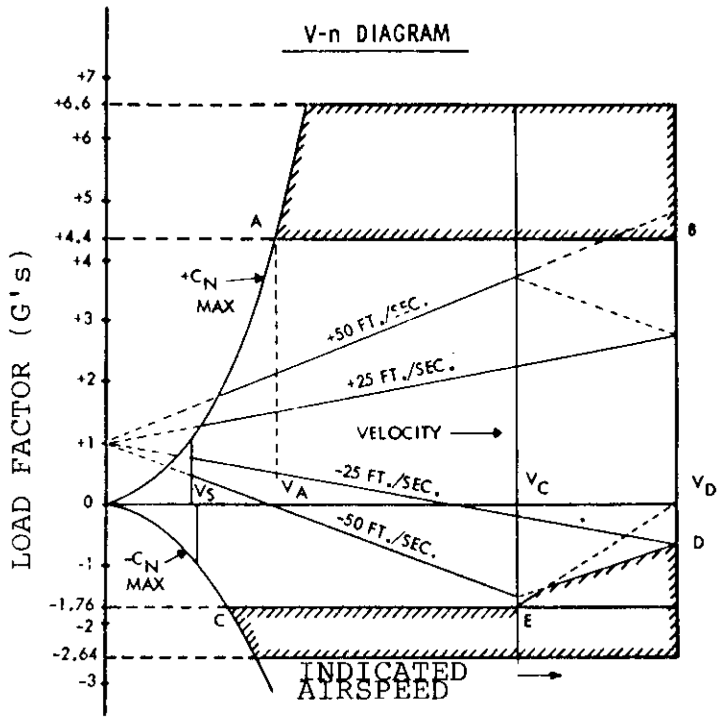


Figure 17a

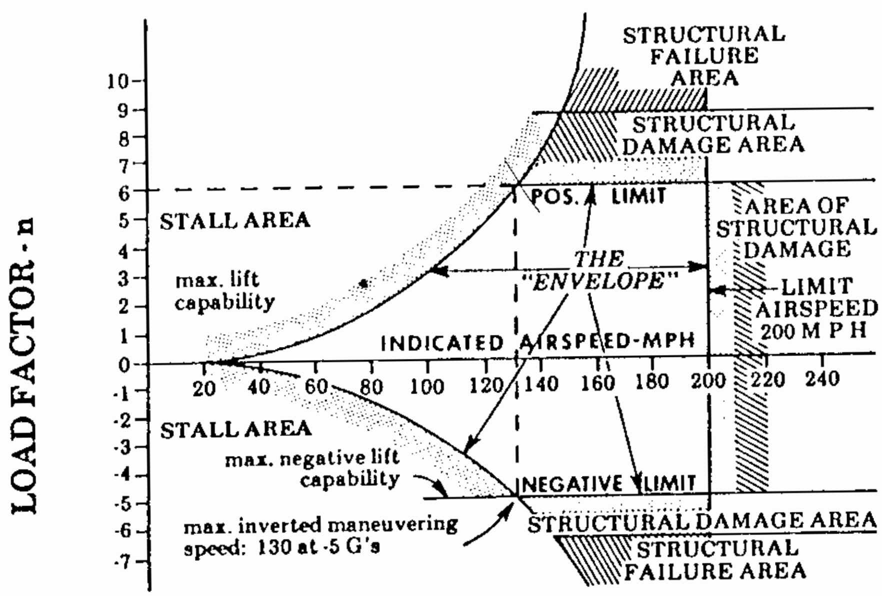
V-n diagram for maneuvering loads for an aircraft designed to +4.4 G's and -1.76 G's limit load. Ultimate load factors of +6.6 G's and -2.64 G's are 150% of the limit load factors. Vertical gusts of 25 fps and 50 fps create load factors as shown even without maneuvering the aircraft.

2.2.6 Gust Load Factors - Gust Envelope

On the V-n diagram, Figure 17a, V_C is the design cruise speed for maximum gust intensity. V_C is often referred to as the "rough air speed".

The specified gust velocities are vertical velocities and their effect on the aircraft is to produce a sudden change of angle of attack relative to the airstream which increases or decreases the air load generated by the wings (depending upon whether the gust is positive or negative). The gusts produce incremental accelerations on the aircraft structure. These incremental accelerations are added or subtracted as the case may be, to the one G load factor. The aircraft is designed for the resulting acceleration if the gust point falls outside the maneuvering envelope presented by the V-n diagram loads. The diagonal lines on Figure 17a are a graphical representation of the "gust lines" and show the load factor developed when a gust of an intensity defined by the appropriate FAR is encountered at various speeds. For this design, these gust lines fall within the manuevering envelope and are not critical. Whether gusts are critical or not in flight depends upon the wing loading and other aerodynamic characteristics at the time they occur. In general, it can be said that the higher the design maneuvering load factor, the less likely it is that gusts will be critical. For this reason, gusts are more significant in transport aircraft design (where the maneuvering load factor is 2.5 G) than in aerobatic aircraft (where the maneuvering load factor is 6.0 G). In any event, the designer must account for gust loads for the particular design.

Figure 17b shows how structural damage may occur when the aircraft is operated in excess of the limit load factors (the envelope). Structural failure may occur at any load factor in excess of the ultimate load factors. These statements are true so long as the aircraft is in an airworthy condition. In cases where prior damage such as fatigue is present in primary structural elements, failure may occur <u>inside</u> the operating envelope. One major reason to carefully examine post-accident fractures is to discover evidence of any progressive failure that might have caused in-flight failure. Any part that appears to be the primary failure location should be carefully examined to be sure it complies with the type design and conforms to the proper material and strength requirements. A primary failure that is a replacement part, or has been modified in maintenance, should be closely studied.



<u>Figure 17b</u>

V-n diagram for an aircraft designed to limit load factors of +6 G's and -5 G's. Structural damage is possible if the aircraft is flown outside the operating envelope. Structural failure is possible when the aircraft is flown beyond the ultimate load factors.

2.2.7 Wing Air Load

The lift developed by an airfoil is a result of the differential pressure between the upper and lower surfaces as a result of air flow over the wing as shown by Figure 18. The net effect is a force acting generally in the upward direction for all positive angles at attack up until the point of stall. Two important considerations exist for the generation of air load by an airfoil. The first is velocity of air flow over the airfoil and the second is the angle of attack of the airfoil. Each contribute to the generation of lift or the ability to generate the load factors presented by the V-n diagram.

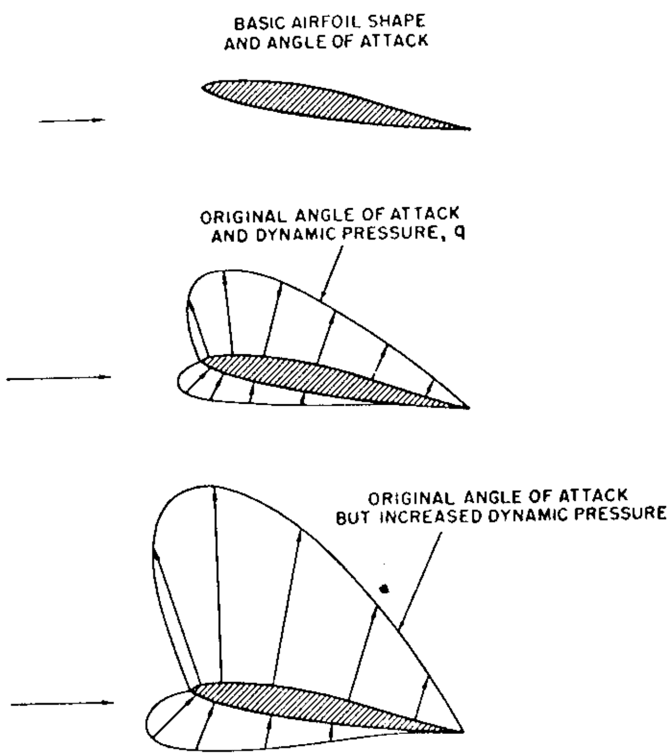


Figure 18

Development of lift from airflow over an airfoil section.

The point at which the center of pressure or lift acts on an airfoil shape is dependent upon the airfoil shape, and the angle of attack. At high angles of attack, the center of pressure is generally located forward on the airfoil and moves rearward as the angle of attack is decreased as shown by Figure 18. In order to generate the load factors described by the V-n diagram, the pilot must maneuver the aircraft in such a manner as to achieve the combination of angle of attack and airspeed to generate the described load factors.

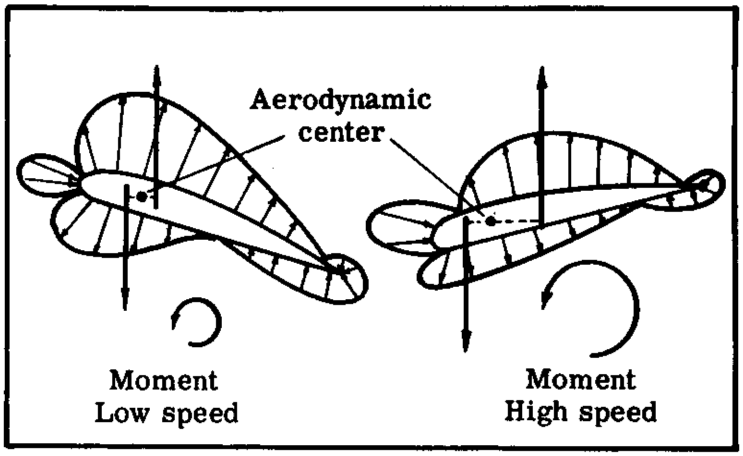


Figure 19

Schematic representation of the lift distribution on a cambered airfoil at low and high speeds.

In the design environment, once the airfoil shape has been selected, the lift characteristics for that airfoil shape are available based on past wind tunnel testing. The test data is used by the designer to generate wing air loads in support of the aircraft load factors defined by the V-n diagram. The air loads developed are distributed on the wing both in a chordwise direction and in a spanwise direction.

The chordwise air load distribution occurs on the moving airfoil as a function of airfoil shape, angle of attack relative to the air stream and control system or flap displacement. Chordwise air load distribution is the entity contributing to wing torque. The air load is considered as acting at the aerodynamic center for the airfoil. Also contributing to wing torque are displacement of the aileron and flap surfaces along with propeller or jet engine thrust if the engines are wing mounted. The torque loads created are reacted by the closed cell created by the combination of skin, stringers, spars and ribs.

The spanwise air load distribution is the summation of the chordwise air load distribution beginning at the wing tip and ending at the wing root. For purposes of analysis, the wing spanwise air load is distributed in a rational manner from the tip to the wing root as shown by Figure 20. The wing shear in the vertical direction is summed from the tip to the root; shear thereby being a maximum at the root. In the same manner, the wing bending moment increases incrementally from the wig tip to the wing root. The wing shear is transferred in general directly to the wing spars by means of the wing skins and stringers. The wing bending is carried by the combined action of the spars, stringers and skins.

The applied air loads on the wing are relieved or lessened by the weight of the wing structure along with other wing items such as fuel, engines and landing gear. This occurs during flight under all conditions.

2.2.8 Air Loading of the Flight Controls, Wing & Tail

The horizontal tail loads are determined by the need to balance the aircraft for the various flight conditions described by the V-n diagram and are referred to as balancing tail loads. In addition to these balancing tail loads, maneuvering tail loads are prescribed in the pertinent regulations. The horizontal tail, for example, is designed for an abrupt upward or downward deflection of the elevator at the design maneuvering speed, V_a. These maneuvers are called "unchecked maneuvers". The horizontal tail is also designed for the loads imposed by a "checked maneuver" where the surface is deflected suddenly in one direction and then "checked" or reversed so that the maximum maneuvering load factor on the airplane is developed. Similarly, the vertical tail is designed for definite maneuvers involving displacement of the rudder at V_a with varying amounts of yaw. Maneuvering tail loads generally result in a distribution of loading which increases from near zero at the leading edge to a maximum at the hinge line and then to zero at the trailing edge. In other words, the center of pressure under maneuvering conditions moves aft on the surface. For gust loads or balancing loads, the center of pressure is near the leading edge and there is a concentration of load in this area. Examination of separated control surfaces will show evidence of distortion corresponding to the nature of the loads imposed.

In general, the horizontal tail, the vertical tail and the ailerons are designed for full displacement at the maneuvering speed, V_a. This means that these surfaces can be moved as abruptly as the pilot desires at speeds below V_a without danger of participating structural failure. For a large number of modern light planes, the designer has selected V_a to coincide with the speed at point A on the V-n diagram, so that if abrupt control displacement is restricted to speeds at or below V_a, no danger of failure of the entire aircraft will result. The development of the air loads for both the horizontal and vertical tail are developed in a manner similar to that of the wing.

The flaps are designed for the loads imposed upon them when extended to any angle up to the maximum velocity specified for use. The ailerons are designed for the balancing loads while in the neutral position, and also the maximum displacement at spced V_a.The distribution of loading on flaps, ailerons and tabs is approximately trianagular with the maximum unit loading occurring at the hinge line and dropping off to zero at the trailing edge.

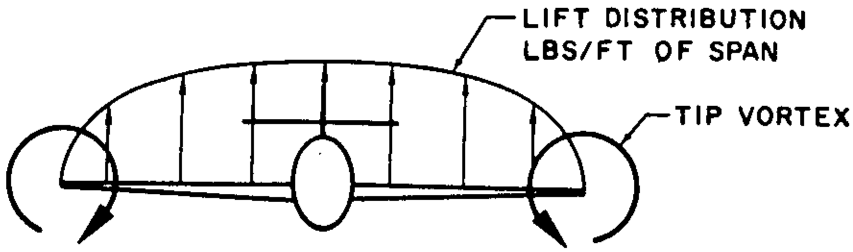
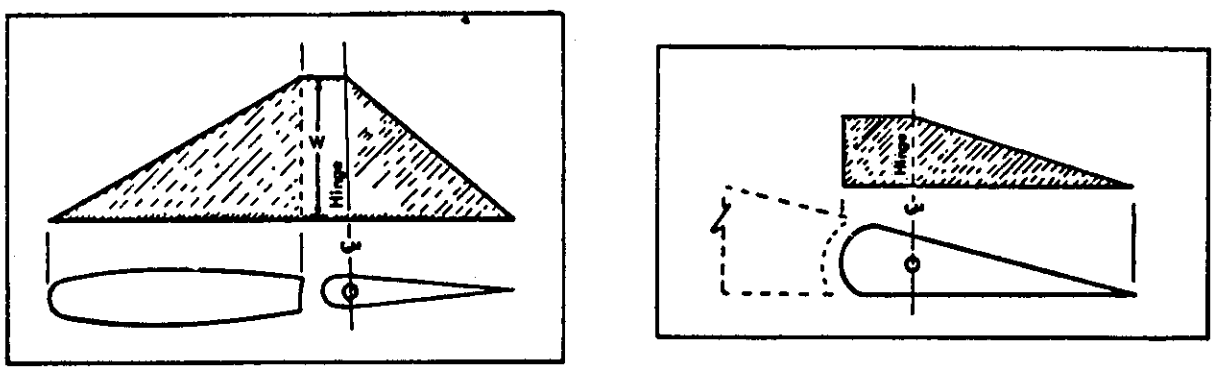


Figure 20

Typical elliptical spanwise lift distribution and tip vortecies.

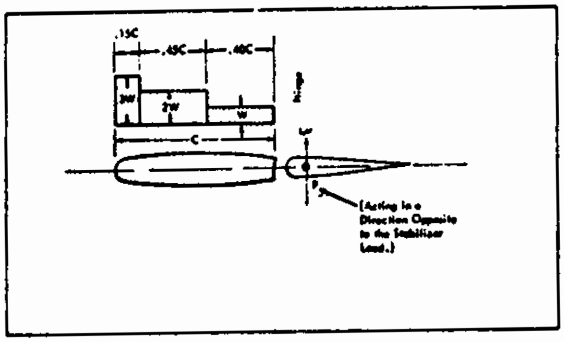


Tail surface load distribution.

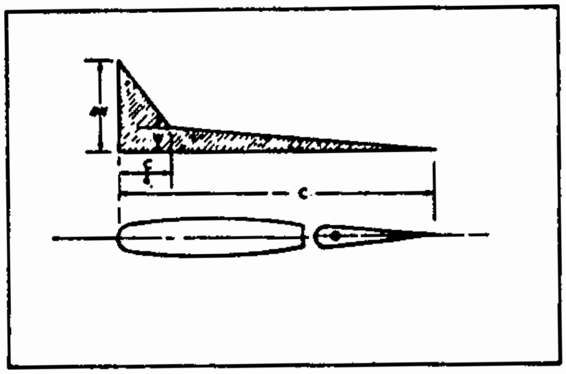
Aileron load distribution.

Figure 20a

Tail surface load distribution from maneuvering.



Tail surface load distribution.



Tail surface load distribution.

Figure 20b

Tail surface load distribution from angle of attack and gusts.

2.2.9 Fuselage Aerodynamic Loading Considerations

In general, air loads are not a criteria for the design of the fuselage itself, but may be a factor in design of closures such as windows, doors, and access panels. The fuselage loads can be considered as first generated in flight by weight of the structure, loads carried, balancing tail loads and loads generated by the vertical tail in response to the operation within the flight envelope described by the V-n diagram. Secondly, there are loads generated both in flight and on the ground by the landing gear. In addition, in pressurized aircraft, the cabin structure is subjected to a differential pressure load which may be the highest loading condition for the pressure cabin portion of the fuselage. Longitudinal seams are subjected to hoop tension stress from pressurization cycles. Fatigue cracks can form along riveted seams. The sudden link-up of multiple small fatigue cracks was responsible for the sudden failure of an Aloha Airlines 737, resulting in one death and numerous injuries.

2.2.10 Ground Loads

Ground loads are those loads imposed on the aircraft structure during landing, taxiing, braking, turning, etc. These loads are an important consideration of the overall design requirements. Whether a particular component or member will be designed by ground loads or air loads cannot generally be predicted for all components until both are computed and compared.

In general however, ground loading conditions are often critical for the wing structure inboard of the gear and for portions of the fuselage structure. In all cases, the landing gear attachment points are designed for the pertinent ground loads.

2.2.11 Landing Loads

In designing the aircraft for landing loads, the aircraft is assumed to contact the ground in several arbitrary attitudes, so selected that they will bracket all possible attitudes likely to be encountered.

For a tail-wheel type of aircraft, two basic conditions are used: the level landing and the tail down landing. For a nose wheel type of aircraft, a third condition, level landing with nose wheel just clear of the ground, is also specified. The vertical load acting through the center of gravity is in all cases, equal to the landing weight of the aircraft multiplied by the landing load factor and may be combined with a factor representing residual wing lift. The landing load factor is selected by the designer and he is required to demonstrate that this factor will not be exceeded by the landing gear when the aircraft is landing at the descent velocity specified by the regulations.

Descent Velocity Considerations

For transport type aircraft, a value of 10 ft./sec. is specified in the regulations for the design descent velocity. For light airplanes, the specified value of descent velocity is given by a formula,* but it need not be greater than 10 ft./sec., nor can it be less than 7 ft./sec. With these values of descent velocity in mind, the designer selects a value of load factor and then proceeds to design a shock strut or gear system which will absorb the landing energy corresponding to the descent velocity. Drop tests are conducted on the gear to verify that it will absorb the energy required within the selected load factor limitations.

From an investigation viewpoint, it is well to note that the specified descent velocities are high and that considerable static strength and shock absorbing characteristics are designed into the landing gear. A landing gear designed for a 10 ft./sec. descent velocity can absorb the shock of an unflared landing with the aircraft descending at 600 ft./minute. This is a hard landing. When landing gear failures occur, they are usually associated with fatigue failures, running over obstructions, or with free drops of the airplanes from considerable heights where the aircraft is "dropped" in from a stalled condition.

Spin-Up Spring-Back

When an aircraft initially touches down, the wheels are not rotating and a considerable drag force is required to accelerate the wheel assembly up to the speed of the aircraft. This drag load is referred to as the wheel spin-up load and formulas are available for calculating its magnitude. The wheel spin-up load is a very important design condition both for the landing gear and the attachment. Oleo drag struts, as used on some aircraft, are examples of one method used successfully to reduce the effect of wheel spin-up loads.

The wheel spin-up load builds up to a maximum value and in doing so, deforms the landing gear rearward. When the wheel assembly has been brought up to the speed of the aircraft, the wheel spin-up load drops off to zero and the gear springs forward so that at the instant of reaching dynamic spring-back, load may be considered to consist of the inertia of the effective mass at the axle bending the landing gear forward. The spring-back load is an important design consideration. Some aircraft must have the gear overhauled after a fixed number of landings to inspect for potential progressive damage from this type of cyclic loading.

2.2.12 Brake Loads, Taxiing Loads

In addition to the loadings described above, the landing gear is designed for such special conditions as braking, ground turning, taxiing, nose wheel yawing, etc. For each of these conditions, the loadings are specified in detail in the regulations for the particular type of airplane. During an investigation of a landing gear failure accident, it is important to attempt to relate the failure with the motion of the aircraft at the time. Unless this is done, a proper evaluation of the evidence may not be possible. Thus, if the failure occurred when the brakes were first applied, this may be significant and should be pointed out in the accident report.

2.2.13 Distribution of Loads

In some of the landing gear design conditions, the loads on the landing gear are balanced by linear or translational inertia forces and no angular motion of the aircraft is involved. In other conditions, the airplane is assumed to rotate or pitch about the main gear and angular inertia forces are generated which must be considered in the design. In either case however, the wing, fuselage and other components are subjected to inertia forces, usually in a downward direction. Of course, the upload on the gear as well as the spin-up spring-back loads must be reacted by the supporting structure through the structural connections. Worn or improperly maintained landing gear components can magnify the effect of these dynamic forces. The designer balances the airplane for all of the forces involved and analyzes the strength of the structure required to support these loads. In many cases, the landing conditions are more critical for certain components than flight conditions.

2.2.14 Vibratory Loads

Vibratory loads are the most troublesome type encountered in aircraft design. In general, their occurrence and magnitude are difficult to predict. Generally speaking, vibratory loads are considered only indirectly in the basic design and service testing is the most frequently used method to determine potential sources of trouble and define procedures to eliminate difficulties. A large portion of the minor fatigue cracking found in service is attributable to vibratory loads.

Vibratory loads are divided into two general types: resonant and forced vibrations. Cracked engine cowlings, isolated skin panels, ribs, etc. are possible examples of resonant type vibration failures. When the natural frequency of the particular part or component approaches the frequency of the normal engine RPM, high amplitude vibrations of the unit develop and early cracking results. For this reason, it is customary for the designer to keep the size of unsupported panels within maximum limits found by experience to be satisfactory. Forced vibrations are generally associated with slip-stream effects in the wake of propellers. These vibrations arise from the small but frequent changes in the local air velocity which produce corresponding effects on the aerodynamic loads. In many cases, it has been found necessary to reduce panel sizes in the propeller wake below that normally used for supported panels. Forced vibrations from small pressure changes in the wake of jets have produced failures on adjacent fuselage skin panels and on tail surfaces.

It is important to keep in mind the basic difference between vibratory loads and "overall" structural loads. Although in many instances it is difficult to distinguish clearly between the two types of loads, there are certain differences. Vibratory loads are generally of high frequency and are more or less of constant amplitude. Overall loads, on the other hand (aerodynamic, maneuver, gust and landing), are generally of lower frequency and of varying amplitude.

2.2.15 Internal Aircraft Loadings

In the previous section, the types of external loads and the procedures for calculating their magnitude were briefly discussed. In this section, the methods and procedures used by the designer and stress analyst to work the external loads such as the described air loads and landing gear loads into internal structural loads will be discussed.

2.2.16 Ways in Which External Loads are Supported

Basically, the external loads are resisted internally by tension, compression, shear, torsion and bending forces in specific structural members. In many applications, bending moment can be converted into couple loads of equal and opposite tension and compression forces. In most instances, the various types of loading occur in combination with one another, but loadings can almost always be considered separately and their effects added together.

2.3 STRUCTURAL ANALYSIS

The air loads generated during all phases of flight along with loads developed during all ground operations and emergency landings must be reacted by the aircraft structure. The loads defined as limit loads in the V-n Diagram (Figure 17) must be reacted without detrimental structural deformation while loads up to and including the ultimate load factor and the resulting loads must be reacted, but structural deformation of a permanent nature can occur. The ultimate load factor must be maintained for three seconds without total jeopardy of the aircraft structure. In most cases, aircraft structures have structural strength capabilities beyond that of ultimate load capability required by the FAR's. The ability of an aircraft structure to react to design ultimate load is dependent upon the condition of the structure which will be discussed in another section of the text.

The airframe of most general aviation fixed wing aircraft consists of five principal subassemblies: fuselage, wings, tail or stabilizer, flight control surfaces and the landing gear. The rotary wing aircraft have four principal subassemblies: fuselage, main rotor and gear box, tail rotor (if so equipped), and the landing gear.

The airframe components which make up the total aircraft are constructed of a wide variety of components consisting of aluminum, steel, plastics, composite materials and, on older aircraft, wood and fabric.

A typical wing structure for many of today's light, and all commercial aircraft is of the cantilever design; that is with no requirement for external bracing. Other aircraft use external bracing such as wing struts. The internal wing structure consists of wing spars, stringers running spanwise and ribs or formers running chordwise. The wing spars are the primary structural element making up the wing structure. The flaps, ailerons, engines and wings mounted landing gear are attached to the wing spars. The wing skins and stringers transfer air loads to the ribs which in turn transfer loads to the wing spars.

The number of spars in a wing design is dependent upon the design concept used by the aircraft manufacturer. Three typical designs are used: a single spar or monospar, a multispar or the box beam. The first two configurations are self-explanatory. The box beam is where two or more longitudinal members are interconnected by multiple spaced bulkheads which provide additional strength along wing airfoil contour. This type of construction is more typically used on large commercial aircraft.

The torsion loads are resisted by the skin covering and by the spar stringer and ribs. The torsion loads are distributed to the varius inter-rib boxes in the wing or tail and the resulting stresses are compared with the allowable stresses in order to evaluate the margins of safety. Around cutouts, such as landing gear wells, nonstressed door openings, inspection panels, etc., special design methods are used to maintain the structural strength, but the basic concepts are essentially the same. Doublers around openings are often used to reinforce edges and replace material missing from the openings which transfer loads to adjacent structure. The skin covering and spar webs are almost always designed as tension field beams. This means that the sheet is permitted to buckle under load but it should be remembered that even in this buckled state, the sheet is still carrying the applied loads. If the applied load is less than the design allowable for the sheet, the buckles will come out when the load is relieved. If the load is above the allowable, permanent wrinkles or buckles result and these will not completely disappear when the load is relieved. This is often a useful indication of excessive loads and the direction of the buckles will indicate the direction of the applied load which produced the buckles. Thus, if permanent wrinkles on the top surface of the wing or tail are found to run toward the trailing edge as they go outboard, this will indicate a nose-up torque loading. The direction of the wrinkles on the bottom surface would be in the opposite direction (i.e., they would run toward the leading edge as they went outboard). The investigator must be able to recognize these tension field wrinkles and distinguish them from the deep creasing which results from major crushing at ground impact.

The vertical and horizontal shear forces are also resisted by the skin covering and the spar webs. Various methods are available for determining their distribution to the different skin panels.

Fuselage stress analysis is not too different from wing and tail analysis. The fuselage is essentially a simple beam supported at the wing spar attachment points with a cantilever overhang forward and an overhang rearward. The section of the fuselage aft of the rear spar attachmment can then be considered as a cantilever beam and the shear, bending moment and torsion summed up for each station. As in the wing design, the torsion and shear loads are resisted by the skin covering, while the bending moment is resisted by the stringers or longerons and effective skin. Special methods are used for the stress distribution around window and door cutouts.

Three typical wing placements or configurations are used on today's aircraft: low wing, mid wing and high wing. The wing may incorporate dihedral, which is done for aircraft stability. Dihedral indicates the wing is not projected in a level manner from the fuselage but installed with the tip higher than the root. A typical dihedral angle is 5 degrees.

The fuselage is the body of the aircraft providing accommodations for the passengers, controls, cargo and other equipment such as the engine for single engine aircraft. There are two general types of fuselage construction: truss or semi monocoque. The truss is a rigid framework generally made out of tube which is then covered with fabric or an aluminum skin. The semi monocoque is the prevalent method used in today's aircraft.

The semi monocoque fuselage consists of formers, bulkheads and longerons or stringers covered by an aluminum load bearing skin. The formers and bulkheads are used to give the fuselage shape and allow the application of concentrated loads such as wings, power plants, horizontal stabilizer or vertical fin. The longitudinal longerons stringers stiffen the skin panels carrying the bending and shear loads applied to the fuselage structure.

In this type of structure the fuselage skin thickness will be varied in different areas of the fuselage in accordance with the load carried in each particular area of the fuselage. The skin is generally riveted to the bulkheads, formers and stringers forming a lightweight, strong and rigid structure.

Both the wings and the fuselage must have openings to provide entry, servicing, fueling, etc. Each opening in a stressed skin must have an appropriate doubler and/or additional structure to carry the loads around the opening such as a cabin door. The additional supporting structure required around an opening is dependent on the size of the opening, load level in the area, and the structural aspects of the closure means.

The various components making up a lightweight aircraft structure are joined together using methods of fastening such as riveting, bolting, screwing, welding, brazing and adhesive bonding. The result is an aircraft structure made up of many individual components such as spars, stringers, longerons, ribs, bulkheads, landing gear, etc., designed to resist the stresses developed by loads applied to the aircraft structure.

Not all components of an aircraft structure have strength as the chief criteria for design such as used in the design of a wing spar. Items such as fairings and cowlings need to exhibit a neat streamlined appearance while being subjected to minimal structural loading.

Before giving a brief description of the analysis of aircraft structure, the investigator must first have a basic grasp of the basics of stress analysis. The external loads applied to an aircraft structure can be categorized into six major areas:

1. Tension
2. Compression
3. Bending
4. Torsion
5. Shear
6. Hoop Tension

2.3.1 Tension

Tension is the stress within a structural member that resists a force (F) which is trying to pull that member apart. The tension stress in a part may be calculated using the equation:

$$S = \frac{F}{A}$$

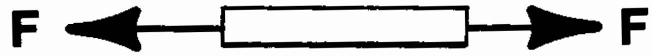
Where: F is in pounds force (lbs.);
A= section area ($in.^2$)
S= stress ($lbs./in.^2$)

The maximum tensile force capability of the section under consideration can be readily obtained by introducing the ultimate tensile stress value for the tensile material into the equation and solving the equation for the tensile force value.

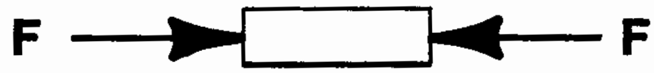
2.3.2 Compression

Compression is the opposite sense of tension, or the application of a load or force that tends to shorten or compress the structural member. The compression stress is calculated using the same formula as that used to determine the tensile stress.

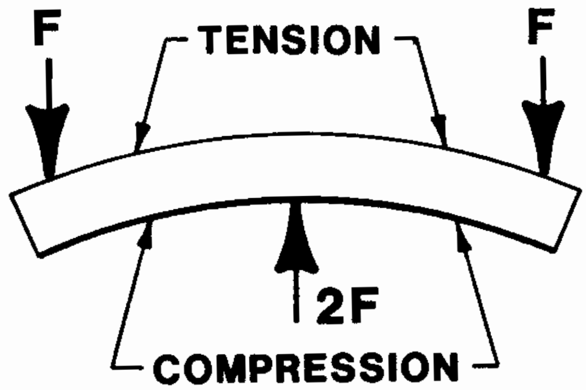
A. TENSION



B. COMPRESSION

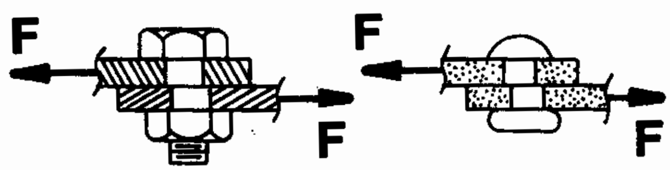


C. BENDING

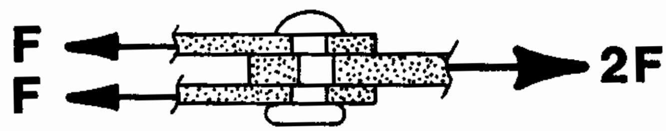


D. TORSION

E. SINGLE SHEAR



F. DOUBLE SHEAR



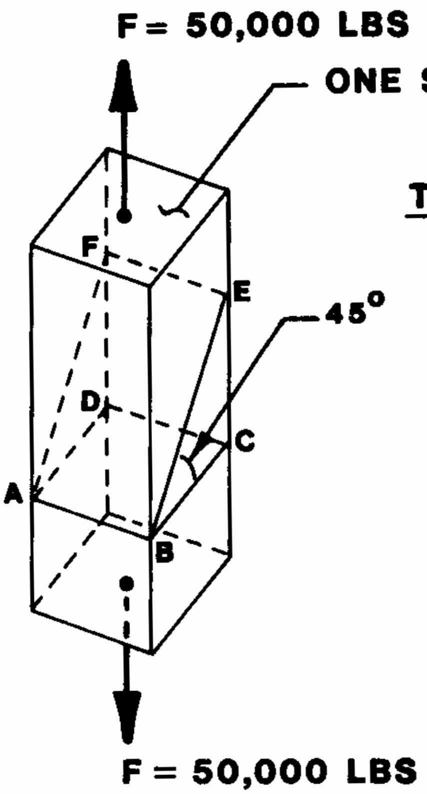
$$\text{STRESS} = \frac{\text{FORCE}}{\text{AREA}} = \frac{F}{A}$$

This Is The Basis Of All Stress. It Is Relatively Simple To Calculate For Items "A","B" And "E", But Under Items "C" And "D" The Magnitude Of The Force "F" As Well As The Area "A" May Both Vary And Therefore Other Formulae Are Required.

Figure 21

Direct stresses.

TENSION INDUCED SHEAR STRESS



TENSION & COMPRESSION INDUCE SHEAR

A DIRECT TENSILE Or COMPRESSIVE Stress Will Induce A Shear Stress Of One Half The Magnitude Of The Direct Stress On A Plane 45° To The Direct Stress.

In This Case The Tensile Stress Is 50,000 PSI On Plane A-B-C-D, And The Induced Shear Stress Is 25,000 PSI On Plane A-B-E-F.

Figure 22

Indirect (secondary) stresses.

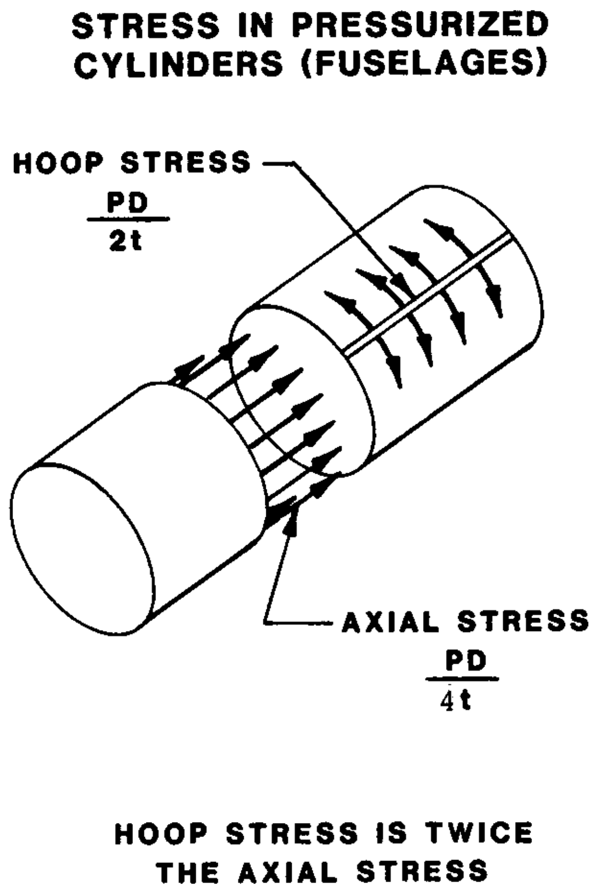


Figure 23

Stresses in a cylinder with internal pressure.

2.3.3 Bending

Bending of a structural member results in a condition which produces both tension and compression stresses in the member. A cantilever beam support at one end with an applied load of F at the free end is an example f this type of conditon. This loading F will produce a tension load on the upper surface and a compressive force in the bottom. No matter what the shape of the section, the highest stress level will occur at the outermost fibers of the structural member. The stress level will uniformly decrease to the center of a symmetrical section and then increase uniformly, but in the opposite direction as shown by Figure 23. The basic flexure formula for stress in bent beams is:

$$S = Mc/I$$

Where: S = stress (psi)
M = bending moment (in-lbs)
c = distance from neutral axis (in)
I = moment of inertia (in^4)

2.3.4 Torsion

Torsion stress is the internal resistance of a structural member to a torque or twisting load. Since torsion is a type of a shear failure, evidence of the direction of rotation will be seen on the fracture surface by observing the scoring marks. The basic formula for torsional stress is:

$$S = Tc/J$$

Where: S = stress (psi)
T = torque (in-lbs)
C = distance from center (in)
J = moment of inertia (in^4)

2.3.5 Shear

Bolted or pinned joints are a special case of shear. Shear stress is the internal resistance of a force tending to cause one layer of a structural member to slide relative to an adjacent layer such as in a fitting as shown by Figure 23 for single shear. This is typical for the connections of many aircraft parts joined by fasteners such as bolts or rivets. The general equation for stress is applicable to this type of joint with the areas representing the fastened area subject to the force such as F. Double shear is developed when a single part carries load by having two areas being subjected to the shearing load.

Tension induces shear on planes inclined to the acting tensile stress (Figure 23).

2.3.6 Hoop Stress

Hoop stress is the term applied to stress conditions created in the shells for pressurized vessels such as the fuselage, hydraulic clynder, or any pressurized line (Figure 23). The tangential stress in the cylinder, usually referred to as the hoop stress is:

$$S_h = \frac{PD}{2t}$$

Where: S_h = hoop stress
P= the operating pressure psi
D= the diameter of the cylinder (in.)
t= the shell material thickness (in.)

The pressure loading also creates a longitudinal stress in the vessel or axial stress which is one-half of the hoop stress:

$$S_A = \frac{PD}{4t}$$

In each of the above stress cases, the member will deform as a result of the applied load. This deformation is measured as strain, as defined in Section 1.3

Ways in Which External Loads are Supported

Basically, the external air loads, landing gear load and any other design loads are resisted internally by tension, compression, shear, torsion and bending forces. In many applications, bending moment can be converted into couple loads of equal and opposite tension and compression forces. The six major loading types: tension, compression, bending, torsion, shear and hoop stress, mainly occur not as separate entities, but in combination as external forces are applied to the aircraft structure. The airloads, as described in the previous section, are external loads applied to the wings and tail surfaces of the aircraft. To a lesser extent, air loads are also applied to the fuselage. As a rule, the more complex forces can be broken down into simpler forces and the effects added together during analysis of a structure.

Many modern aircraft are of cantilever design types, i.e., the wings and tail surfaces are supported and fixed at their root and no external bracing is used. In this type of design, the shear, bending moment and torsion loads are zero at the tip and build up to a maximum value at the root (Figure 24). Deflection (displacement) on the other hand, is greatest at the free end and drops to zero at the attachment point. The designer's job then, is to provide sufficient structural material and stiffness suitably arranged at each station to resist the design loads. The bending moment, which can be visualized as a tension-compression couple, is resisted by the top and bottom covers. These covers may consist, in some designs, of spar caps and stringers; or in other designs, of spanwise corrugations, sandwich construction, or spars alone. In any case, the axial loads from the bending moment are distributed among the various members and stresses are computed. These are compared with allowable stresses for the material being used and margins of safety are computed. A similar method is used for the strut based wing. The deflection of the wing under load is maximum at the tip going to basically zero at the strut attachment. Between the wing attachment and the strut attachment wing deflection is zero at each end with the maximum occurring at mid span.

Stiffness is an important property which limits the amount of deflection. If a structure is too flexible, aeroelastic problems can result. Aeroelastic problems of control system divergence, or flutter can be improved by increasing the inherent stiffness of the affected structure. A stiff structure vibrates at a higher natural frequency and lower amplitude as compared to a softer structure.

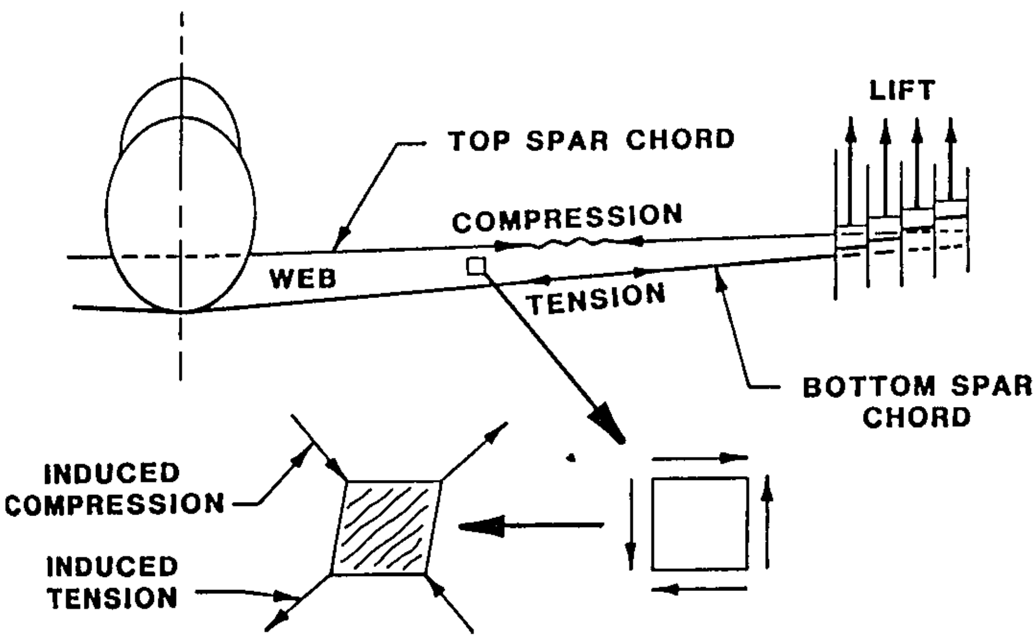


Figure 24

Stresses induced in a cantilever wing
from the lift distribution.

The nature of the deformation and the direction of tension field wrinkles in the shear web allow the investigator to examine wreckage for signs of positive or negative overloads.

In the positive load case illustrated, the bottom spar cap should show stretching; the upper spar cap should be buckled, or crippled; and the shear web should be wrinkled diagonally upward and outboard. Naturally, in the negative load case, all the above signatures would be opposite.

3.0 FRACTURE MODE IDENTIFICATION

3.1 INTRODUCTION

In this section we will be discussing analysis of failures or malfunctions of components or individual members within a system. This discussion will develop the ideas necessary for an infield investigator to readily evaluate suspect parts to identify the primary failure and to interpret the resulting or secondary failures.

A main objective of this section of the text is to provide background information on the various types of failures that can occur. Identification of the mode of failure implies the investigator has properly recognized the particular failure as having occurred by a specific process. Once the mode or type of failure is properly recognized, the cause of that failure can be addressed. This is because each failure mode has only a few key causative factors. More importantly, once the mode and cause of failure are known, appropriate safety recommendations can be developed. On the other hand, if the mode of failure is improperly set, the cause of failure will probably not be correctly identified and potentially dangerous situations will not be corrected. Failure mode identification is a key responsibility of the field investigator. The terminology that goes along with failure mode identification is equally important so that reports convey the proper information to other responsible people in the aviation system.

The cause of failure can be determined once the mode of failure is known and an analysis is conducted of the contributing factors that are significant for the mode of failure being addressed. For instance, in a brittle failure, shock loading and stress concentration effects are important. In this case, if a brittle fracture is observed, the investigator may want to look for evidence of impact loading or improperly sharp corner fillets.

As a general rule, the investigator will need to consider a series of factors before a cause of failure is understood and safety recommendations are offered. The following list summarizes the important contributing factors that are often involved:

1. Mode of failure
2. Fracture origin
3. Fracture direction
4. Material properties
5. Manufacturing factors
6. Service loading factors
7. Maintenance factors
8. Operational (human) factors
9. Consequential (secondary) factors

3.2 SYSTEM OF CLASSIFICATION

3.2.1 Failure Mode Identification Chart

As should now be apparent, there are many different materials, stresses, operational factors and types of failures possible. Fortunately, these interrelated factors can be organized into a system that provides a fairly simple way of handling the problem.

Appendix II is a fold-out that forms the basis for doing a failure mode identification. The remainder of this chapter and a significant amount of classroom time is devoted to a detailed understanding of this chart.

The first thing to be remembered is that all failures are not necessarily fractures. For instance, buckling or distortion may render a particular component failed and unable to perform its intended function. Likewise, while corrosion or wear may initiate fractures in some cases, the corroded or worn part may cause a system failure without actually fracturing. Nonetheless, fracture is a key type of failure and fatigue, in particular, is an all too common failure mode in aircraft structures. The Failure Mode Identification Chart comprises the vast majority of failure modes that the typical investigator will see.

Each investigator should consider keeping a copy of the chart readily available while doing the hands-on field work and analysis and while writing technical reports.

3.2.2 Instantaneous Versus Progressive Failures

All failures can be first classified into one of two broad categories. They are either instantaneous or progressive (see Appendix II). Instantaneous failures are the result of excessive stress causing a separation or dimensional failure. All instantaneous failures are therefore the result of overloading and operation beyond the design intent. Progressive failures, on the other hand, are the result of some "wear and tear" or weakening effect which develops over time. In this case, failure of the weakened structure can occur while operating the aircraft within its design limitations and represents a condition of failure after a critical amount of damage has been done.

3.2.2.1 Instantaneous Failures (Static Failures)

Instantaneous, or what is often called static, failure will occur when loads in excess of the design loads are imposed. This can happen when the aircraft is maneuvered too severely at too high a speed, or when the aircraft is landed too hard. the term "static failure" should not be taken literally because impact loading also causes "static" failures. The damage that results when an aircraft strikes the ground is of the "static" type. Instantaneous failure is therefore, the preferred term for these types of failures. Instantaneous failures are to be contrasted with progressive failures.

All instantaneous failures can be classified as either being ductile or brittle. The difference between ductile and brittle failure modes is the amount of distortion sustained by the part prior to failure. A ductile failure is one that undergoes a significant shape change prior to failure. A brittle failure is one that undergoes little or no apparent shape change prior to failure. Soft aluminum is an example of a ductile material that stretches (elongates) significantly in tension prior to fracture (see Figure 25). Glass is an example of a material that behaves in a brittle manner.

3.2.2.1.1 Instantaneous Ductile Failure, Identifying Characteristics and Contributing Factors

Listed below are the most important identifying characteristics of ductile failures.

1. Shape Change - Since ductility is related to shape change, the more shape change, the greater the ductility. The nature of the shape change will dictate the type of stress applied to the part. For instance, a part that is permanently twisted was subject to excessive torsional stress. In a similar manner, stretching reveals tension, a bent part reveals bending stress, a buckled part reveals compression, bulged containers reveal internal pressure, etc.

2. Necking - Necking is the description of the area very near the fracture where the diameter of a part is locally reduced just prior to breaking. An example of a necked part is shown in Figure 18, which shows a soft aluminum tensile test bar. Necking occurs at the ultimate tensile stress and is the cause of the stress-strain curve maximum (see Figure 4). Greater necking implies greater ductility.

3. Dull, Fibrous Fracture Texture - The fracture surface appears to have a fine textured sandpaper-like surface. The surface is not bright and does not sparkle. The surface appears fresh and is not corroded or worn.

4. <u>Shear Lips</u> - The 45° slanted edges around the fracture surface are called shear lips. These are the last areas of the part to fracture. Therefore, if large shear lips are present, there could not be a surface defect on that area of the part.

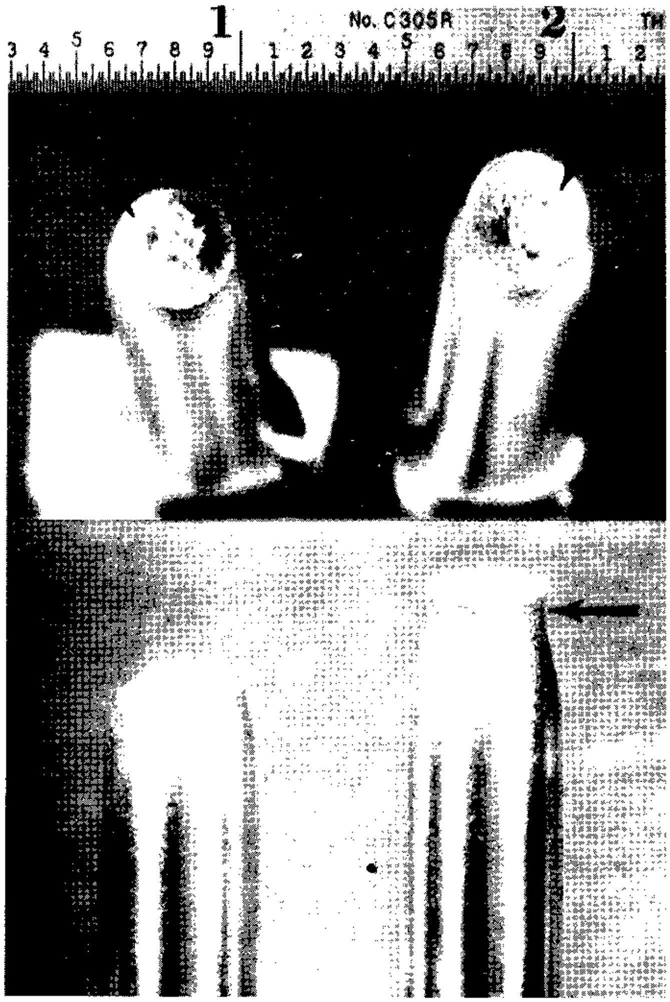


Figure 25

Cup and cone failure resulting from ductile tensile separation after the formation of a necked down area. Note the dull, fibrous fracture surface and the large 45° shear lips around the edge of the fracture.

3.2.2.1.2 Analysis of Ductile Failures

Since ductile failures are the result of excessive load, one can conclude that the applied load exceeded the strength of the part. In this case, it should also be clear that all ductile failures represent operation beyond the operating limitations of the part. This is true except for two very important exceptions. First, the part may have been too weak, such as by missing a heat treatment step in manufacturing, or in maintenance. In such cases, a quick hardness test will tell you whether the part had the correct strength (see page 29). Second, in the event of excessive temperature exposure, such as during an in-flight fire or exhaust gas leak, the strength of the part will be reduced. Therefore, a softened part may fail by ductile overload even though the loading does not exceed the normal structural loading anticipated by the designer. Accordingly, the investigator should look for evidence of excessive temperature before concluding that excessive loading is the only causative factor.

Confirming tests can be performed in the laboratory to further document the failure mode. Metallurgical tests for proper hardness, grain structure and tensile strength may be obtained. In the event of complex structures, it may be worthwhile to assemble a mockup of a system and apply loads to various parts to see if the resulting failures duplicate the findings from the accident aircraft. Likewise, if one part of a system fails by ductile overload, there is often evidence of other parts in the system showing consistent damage or distortion.

3.2.2.2.1 Instantaneous Brittle Failures, Identifying Characteristics and Contributing Factors

Listed below are the most important identifying characteristics of brittle failures.

1. <u>Limited Shape Change</u> - The primary characteristic of brittle failures. On a visual scale, there may be slight change of shape and brittleness is a relative term. Little or no visually apparent shape change is the proper way to judge brittleness.

2. <u>Flat Fracture</u> - Observed when looking at the fracture surface in profile. This is also called a "square break".

3. <u>Fracture Origin</u> - Most brittle fractures have a clearly defined origin. Fracture rays spread out from a specific point, usually at the surface. The marks that radiate away from the origin are also referred to as fan marks. The origin is often located at a surface discontinuity such as a notch or other stress concentration.

4. <u>Chevrons</u> - Marks similar to sergeant's stripes that point like arrow heads back toward the fracture origin. Chevrons often form on fast running brittle fractures and are helpful in isolating the origin area when visually examining large broken structures.

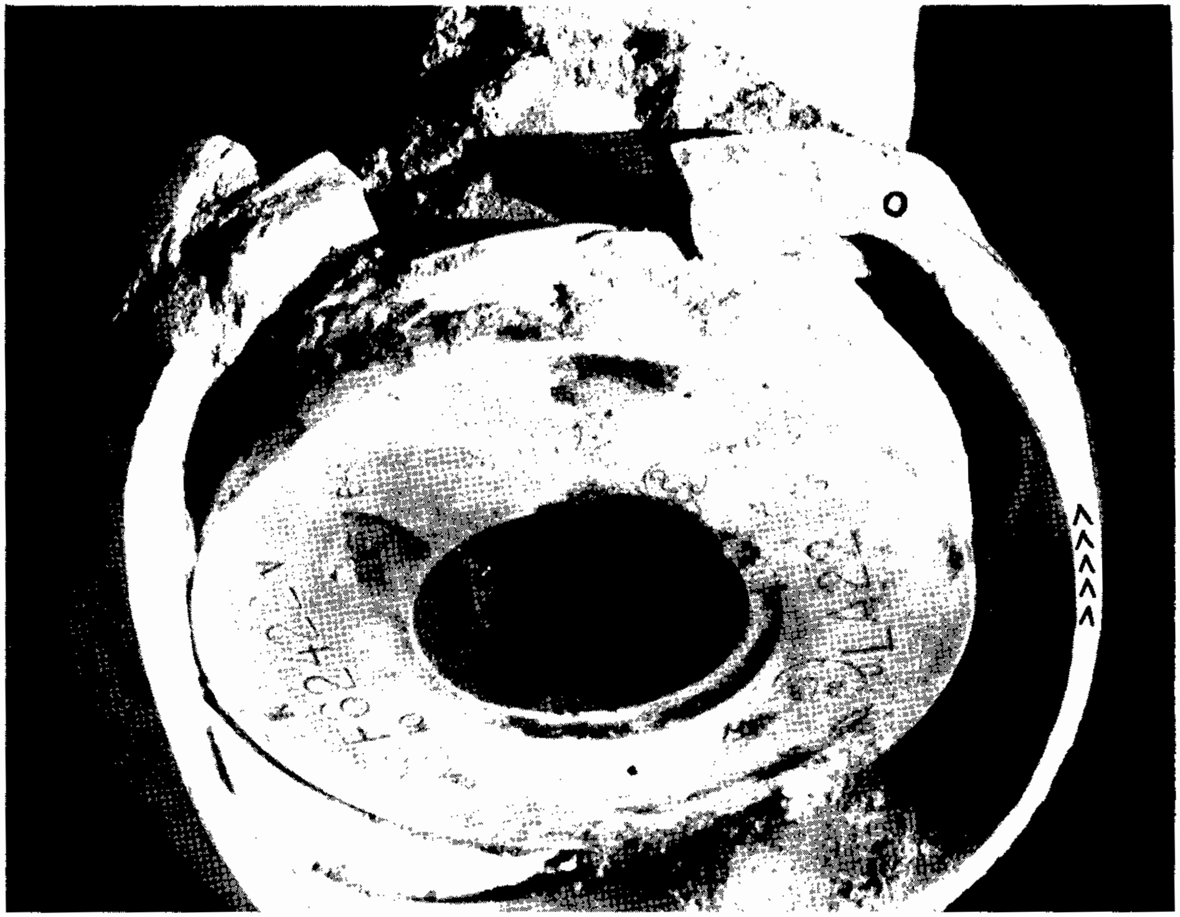


Figure 26

Chevrons appear as "V" shaped marks or arrow heads pointing back to the origin area of a brittle fracture.

5. <u>Coarse, Granular Fracture Texture</u> The fracture will often be bright with shiny facets or granular texture. Castings will usually have the coarsest fracture surfaces.

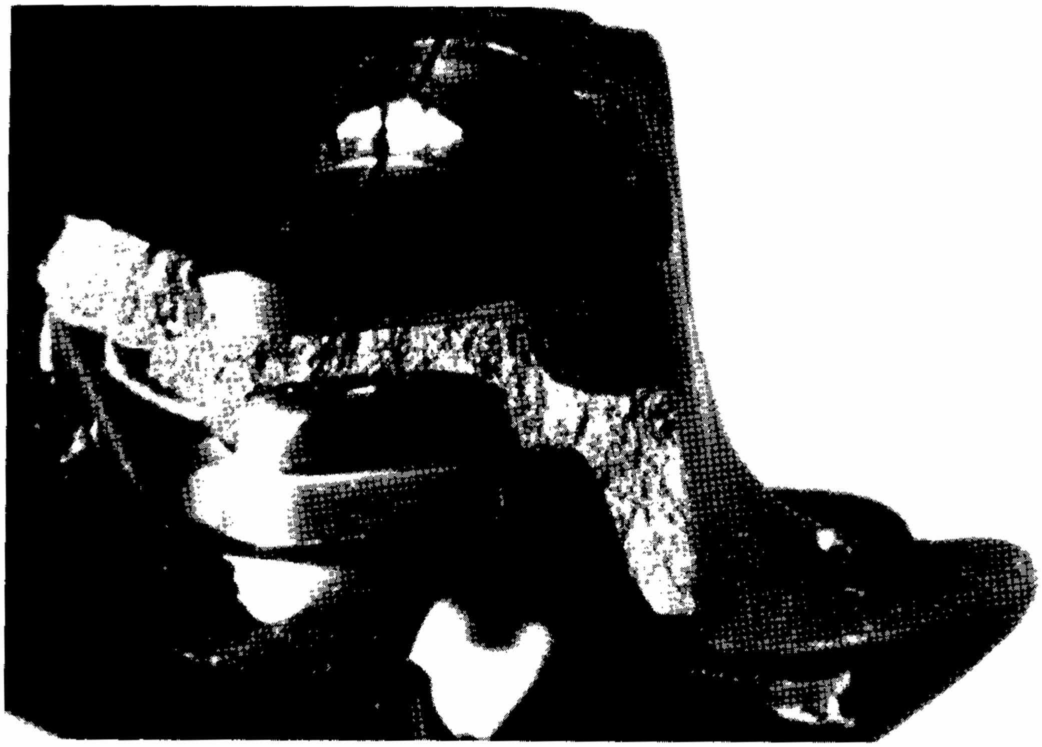


Figure 27

Brittle fracture of an aluminum alloy casting showing
a coarse textured brittle fracture.
caused by impact overloading.

3.2.2.2.2 Analysis of Brittle Failures

Brittle failures occur suddenly, often causing loud noise from the elastic energy stored in the structure. Brittle fractures can be more dangerous than ductile failures because ductile failures stretch before breaking, allowing other parts of the structure to pick up some of the load and provide redundant load paths. Brittle materials are quite susceptible to bending loads which can create very high local stress in brittle materials. Ductile materials have the capacity to yield to bending loads, whereas brittle materials fracture under similar circumstances. Likewise, stress concentrations are not much of a factor for single load events in ductile materials because they can yield to relieve load. Brittle materials experience nearly the full stress concentrating effects of notches and are susceptible to sudden fracture from bending, notches and shock loading, all of which make brittle materials less able to absorb energy during inadvertent overloads or to manage crash energy. Unfortunately, most high strength materials develop strength at the expense of ductility.

At the metallurgical laboratory under high magnification, the fracture may be transgranular (across the metal grains) or intergranular (between the metal grains). Detailed metallurgical analysis in the laboratory may reveal specific causes of brittleness in what should have been a more ductile material. Extreme care should be taken in handling brittle fractures because the initiating defect may be very small and easily damaged by rough handling of the broken part.

3.2.2.3 Recognition of Common Fracture Modes

The amount of distortion, yielding, necking and size of the shear lips in the tensile separation depends on the ductility of the material. Very little, if any, distortion will be found in static fractures in such materials as brittle castings or ultra high strength steels.

Detailed examination of the deformation will disclose indications of the type of loading (i.e. bending, tension, etc.) and the direction of loading. In most cases, the two halves of the fracture will match with one another.

Tensile

Tensile separations are the easiest to identify based on extension deformation prior to fracture. The amount of elongation prior to fracture is governed by the ductility of the material.

Compression

Compression failures occur in two general forms: block compression and buckling. Block compression is generally found in heavy, short sections, whereas buckling is found in long, lighter sections. When buckling occurs locally, it is referred to as crippling (Figure 28). When it occurs in such a way that the whole piece buckles, it is referred to as column buckling. Local buckling and column buckling are easily recognized since the part in all cases is bent from its original shape.

In block compression fractures, the piece separates on oblique panels as in tension, except that there is rubbing of the two halves of the fracture during separation. In some materials, there is a local increase in cross-sectional area (barreling) where the material has yielded.

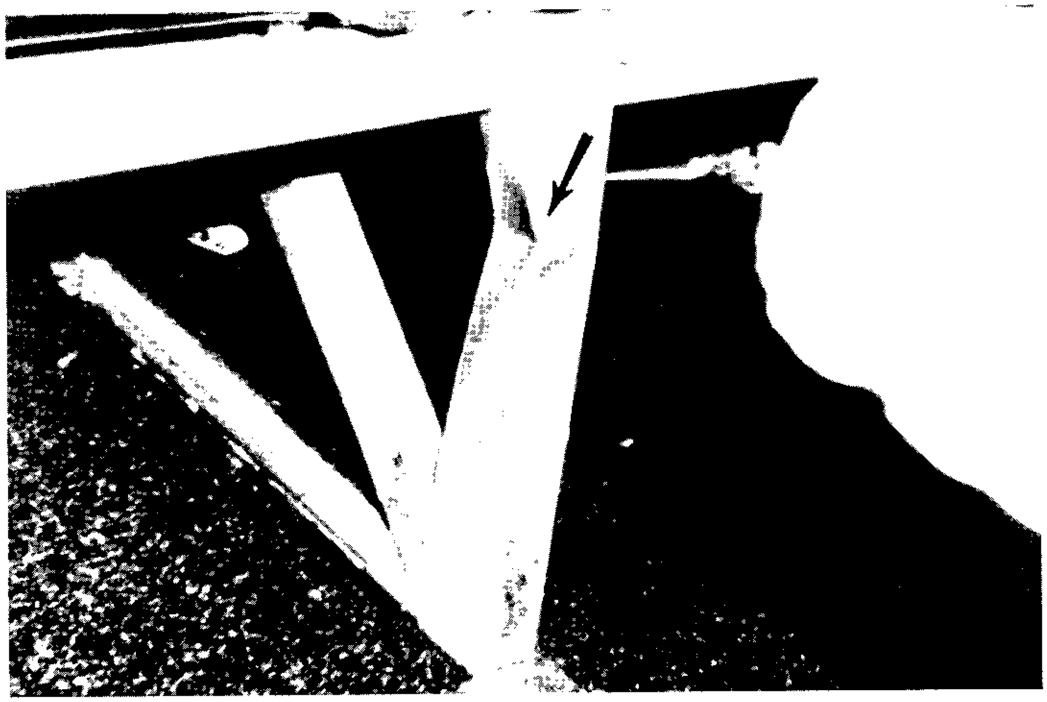


Figure 28

Compressive buckling (crippling) of hollow rectangular forward seat leg structure.

Bending

Bending is resisted by tensile forces on one side of the member and by compression on the opposite side. The appearance of the fracture in the respective areas is as outlined under tension and compression above. The direction of the bending moment causing failure can always be determined from local distortion in the fracture area. As the part finally separates, lipped edges may be found on the inside or compression face of the fracture. This lipping occurs because after the initial tension failure, the final failure on the compression side may be in shear rather than in compression. The shear web between spar caps will show tearing from the tensile failure toward the compressive failure. In many spar failures, the compressive cap fails first by crippling and the wing separates after twisting and deflecting to a large extent. Bending failures are very common because high bending stresses can be developed in most aircraft structures, either by primary or secondary overloading.

Shear

As in compression failures, shear failures can occur in two distinct ways: block shear and shear buckling. In the former type of failure, the two halves of the fracture will slide across each other and the fracture will appear rubbed, polished, or scored. The direction of scoring will give a clue to the direction of the applied shearing force (Figure 29).

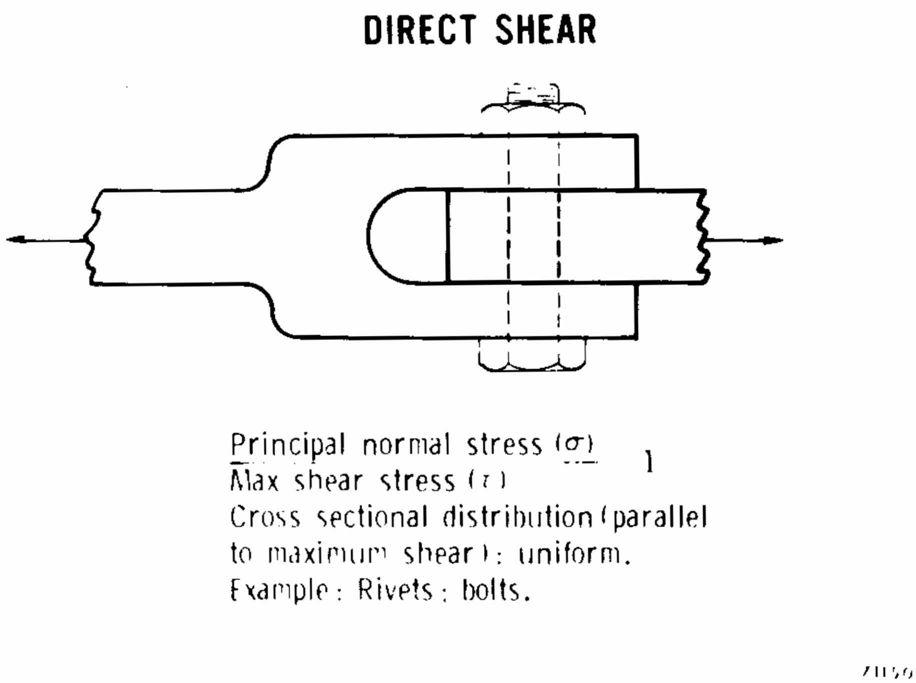


Figure 29

A bolt in a clevis joint is subjected to double shear.

Shear buckling generally occurs in thin sheet metal such as wing skin or spar webs. The shear induces tension at 45° to the applied shear loadings. The sheet will buckle in a diagonal fashion and the direction of force application can be told from the appearance of the buckle (see Figure 30).

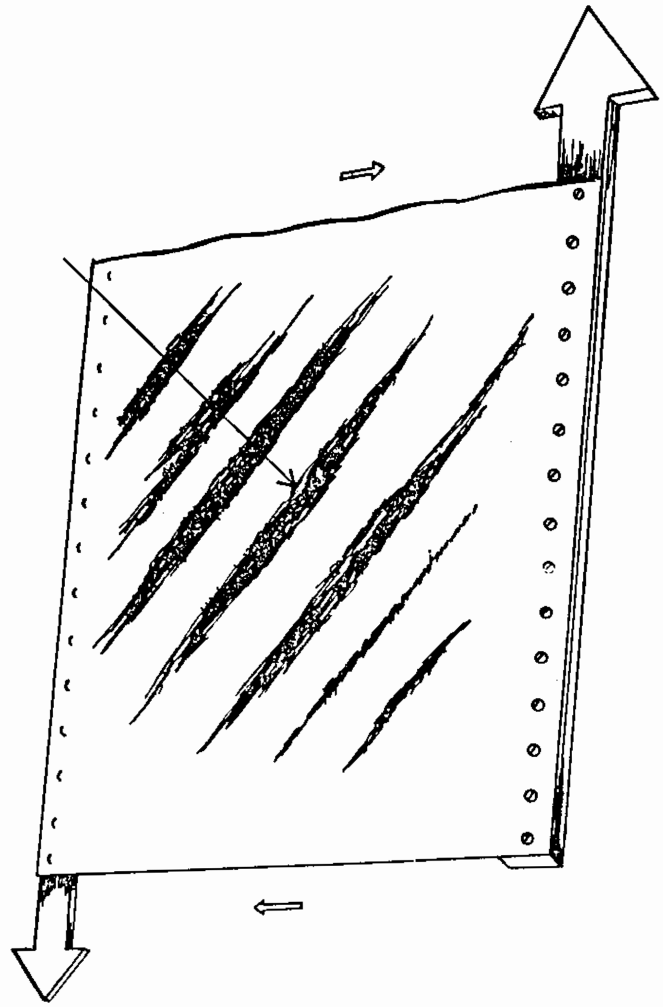


Figure 30

Shear of a panel induces buckling along the 45° diagonal.

Failures of rivets, screws, or bolts in shear are usually accompanied by elongation of the hole and a crescent shape open space will appear behind the rivet (Figure 31). This result can be used to determine the direction of the shearing force. Sheared bolts will show come localized bending. In the electron microscope, elongated dimples will exhibit the direction of shear.

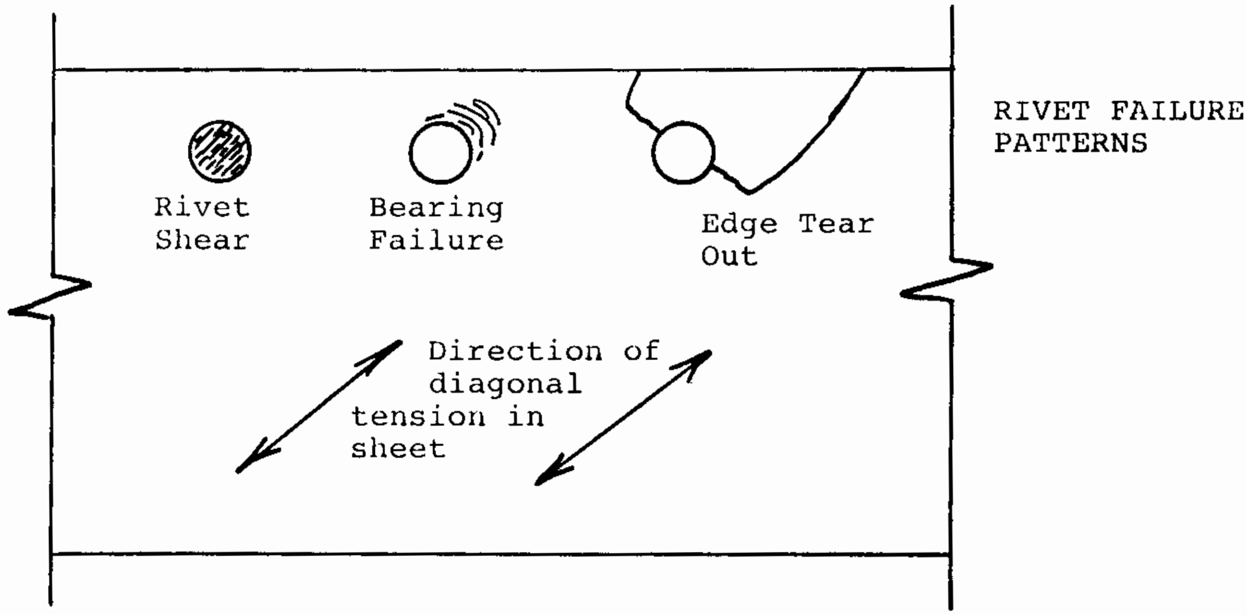


Figure 31

Possible failure modes of riveted skin joints subject to shear loading and diagonal tension.

Torsion

Since torsion is a form of shear, the failure from torsion overload will be somewhat similar to the shear failure. Evidence of the direction of torque can be seen on the fractured surface by observing the scoring marks. Most parts retain a permanent twist and this can be used as an indication. In a tubing member or a large open section like the wing torsion, failures often occur as instability failures in a buckling manner. Again, the direction of the twist can be determined by close examination of the buckle (Figure 32).

Torque tubes and the fuselage are two examples of structural members that act like thin wall tubes.

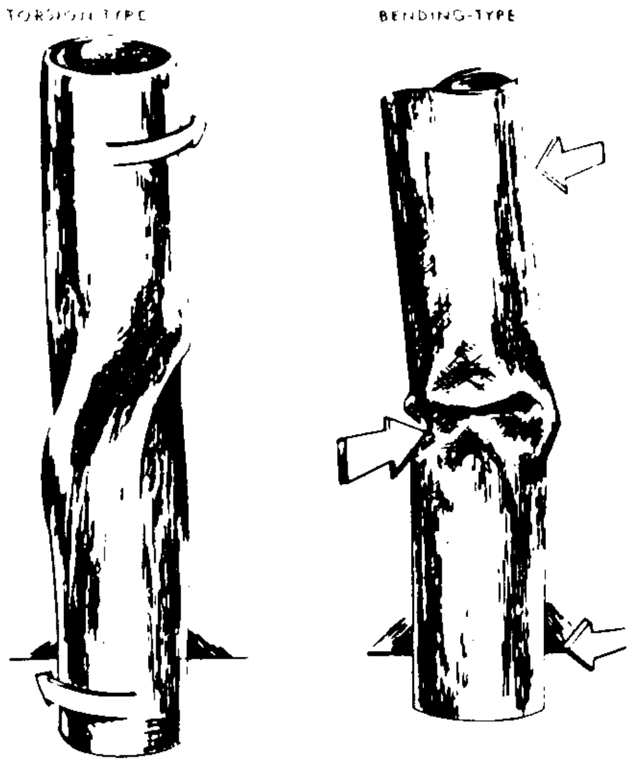


Figure 32

Buckling of this wall member due to
torsion and bending.

Tearing Failure

Tearing failures in sheet metal, or heavier sections for that matter, generally occur in two different forms: shear tearing and tensile tearing.

Shear tearing occurs when the applied forces are acting out of the plane of the sheet. These failures are characterized by a lipping of material on the edges of the sheet and by scoring lines on the fractured surface. The concavity of the scoring can be used to tell the direction of tearing. The direction of tearing is from convex to concave. Sometimes if there is a heavy paint film, the saw-toothed breaking of the paint film can be used to tell the direction of tearing (Figure 33).

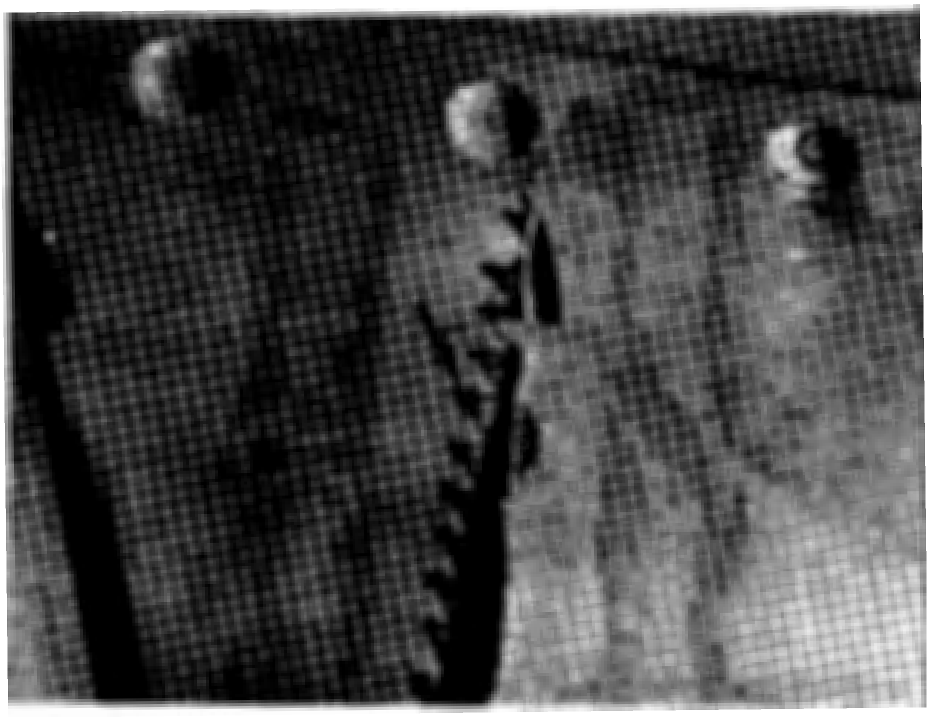


Figure 33

Saw-tooth paint fractures from tearing; the smooth edge torn upward with respect to the saw-toothed edge.

Tensile tearing occurs when the sheet tears under tensile forces in the plane of the sheet or member. This type of fracture is quite common. The edges will usually have 45^{o}F slant fractures. The slant may alternate on either side of the tensile axis. All of the above mentioned types of static, overload failures are described as instantaneous failures.

When examining broken structure to reconstruct the breakup sequence, the skin tear patterns can be used to determine the sequence of failures. For instance, when a skin panel peels away from a surface, the tear patterns will indicate the order that the skin panels tore loose.

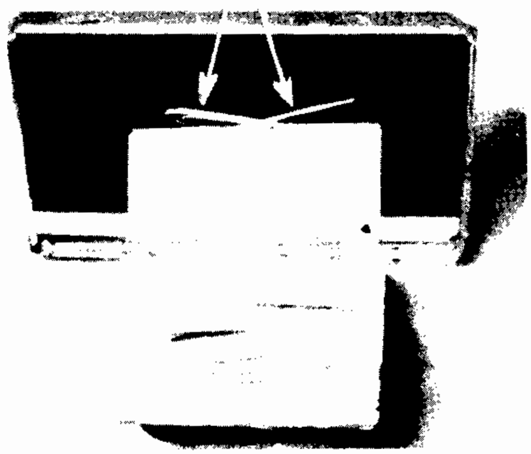
3.2.2.4 Failure Modes of Wood

Wood, like metal, fails in a distinctive manner under different loadings. Generally the type of loading can be determined by examining the fracture. Unlike metal however, wood is not a homogenous mass and has widely different properties in different directions. The type of failure that results depends not only on the type of loading, but the direction of grain in the wood (Figure 34).

Fibres all straight, fracture extends over appreciable length of member.

Tension along grain.

Mirror at 45 degrees to show side view of fracture, surface fibres lifted in different directions.



Tension across grain.

Marks of compression fracture of the fibres on a finest quality spruce spar base plate.

Compression.

Figure 34

Examples of static tension and compression fractures in wood.

Tension Along Grain

Under tension loading, the wood will fail in the individual fibers at different points along the length of the pieces. The resulting appearance then resembles a brush with hairs of different lengths.

Tension Across Grain

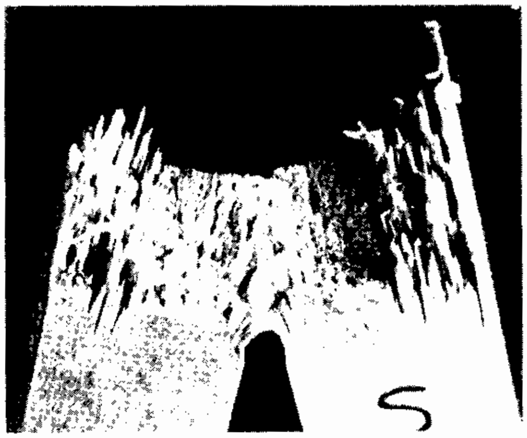
The piece will separate at approximate right angles to the direction of loading and then fail between the fibers. In most cases some fibers, which were at an angle to the fracture plane, will be pulled out at failure in a varied directional manner, unlike shear failures along the grain where the outstanding fibers will generally all point in the direction of the shear load.

Compression

As in metal, compression failures can be in the form of block compression or buckling. Buckling failures will in general, occur in the longer, slender members and the type of failure is easily recognizable. In block compression, the failure results in a collapse of the individual fibers, sometimes along oblique planes with no actual separation occurring in most cases. The compression collapse of the fibers however, reduces the tensile strength and separation generally occurs subsequently under tensile loads. The fracture had a flat carrot-like appearance.

Bending

Bending failures are essentially a combination of tensile and compression failures. There is generally considerable deformation of the fiber at the fracture in the direction of bending (Figures 35 and 36).



Finest quality.

First quality.

Second quality.

Figure 35

Evolution of dynamic bending fracture in samples of spruce of differing qualities. (Note the lower the quality, the less marked the difference between the tension and the compression characteristics.)

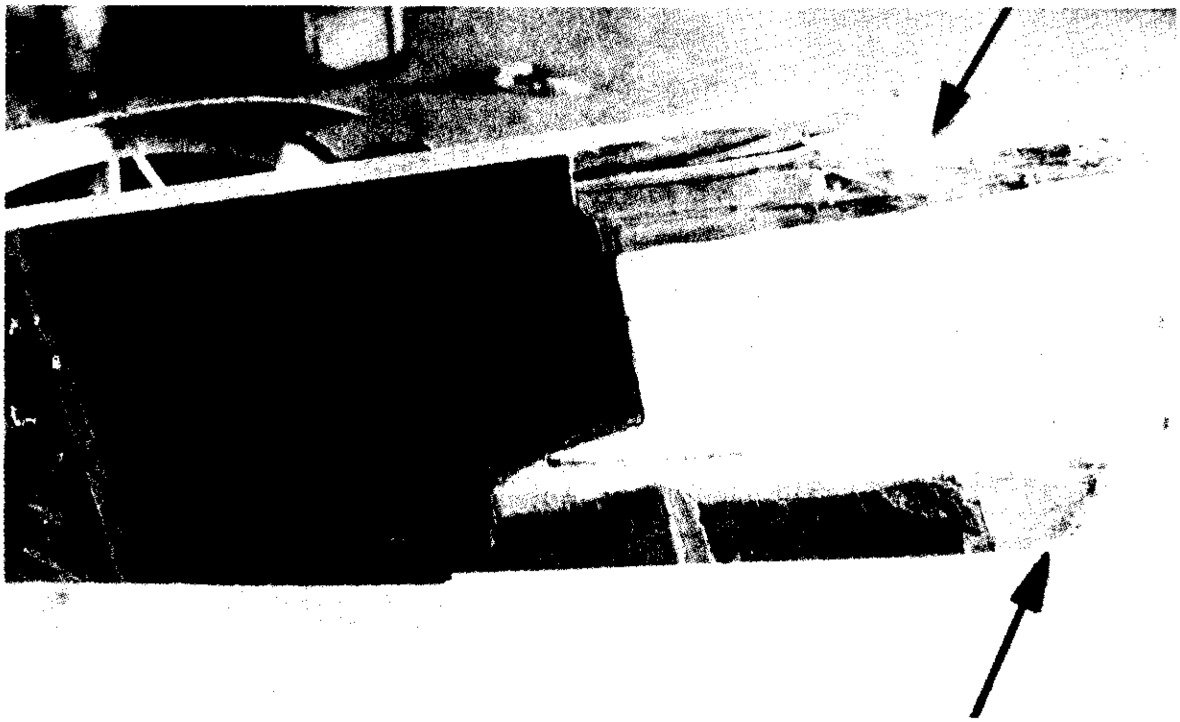


Figure 36

Bending fracture.

Shear

Shear failures usually occur along the grain and some of the fibers which are not in the plane of shear will be deformed in the direction of the force.

Torsion

Under torsion loads, wood will fail along the grain as in tensile failure and the appearance will be similar. The outstanding fibers however, will be distorted in the direction of twist and the part as a whole will generally retain some permanent twist.

Plywood

Since plywood is made up of several layers of wood at varying angles to each other, it is natural to expect that the failure under any particular loading will produce different types of failure in each layer because of the varying grain direction. In examining plywood failures, the failing force on each ply must be compared with the others and the type and direction of loading upon which arrived. The remarks made above for plain wood are applicable to individual ply failures. Plywood sheets often fail by buckling, but this type of failure is easily recognizable.

Glue Joint

Sometimes it is important to be able to determine if the glue joint or bonding two members have failed. When the glue joint has failed, the two halves of the joint are smooth and undamaged. In some cases, there will be evidence of glue adhering to both members, or in other cases, all of the glue will adhere to only one of the members. When the glue bond has held and the failure has occurred in one of the members, some of the wood fiber will stand out on the glue bond which remains on the unfailed member. Generally, in parts where there are different types of wood held together by the glue bond, the failure will occur in the softer wood.

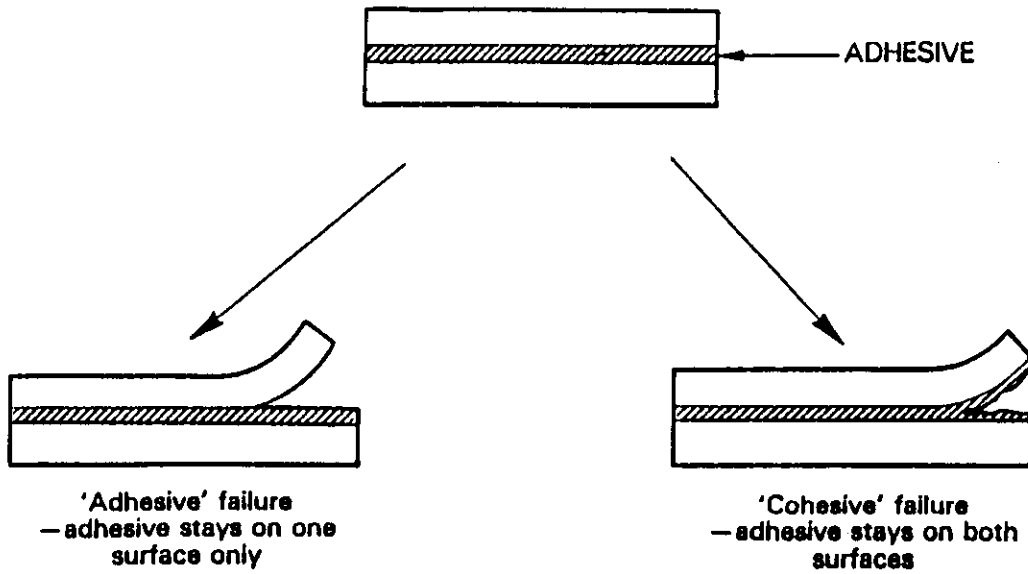
3.2.2.5 Failure Modes of Composites

The analysis of composite material failure modes can be organized into two broad categories. The first category deals with the nature of separation of adhesively bonded joints. The second category deals with various descriptions of failure modes related to filament reinforced matrix material.

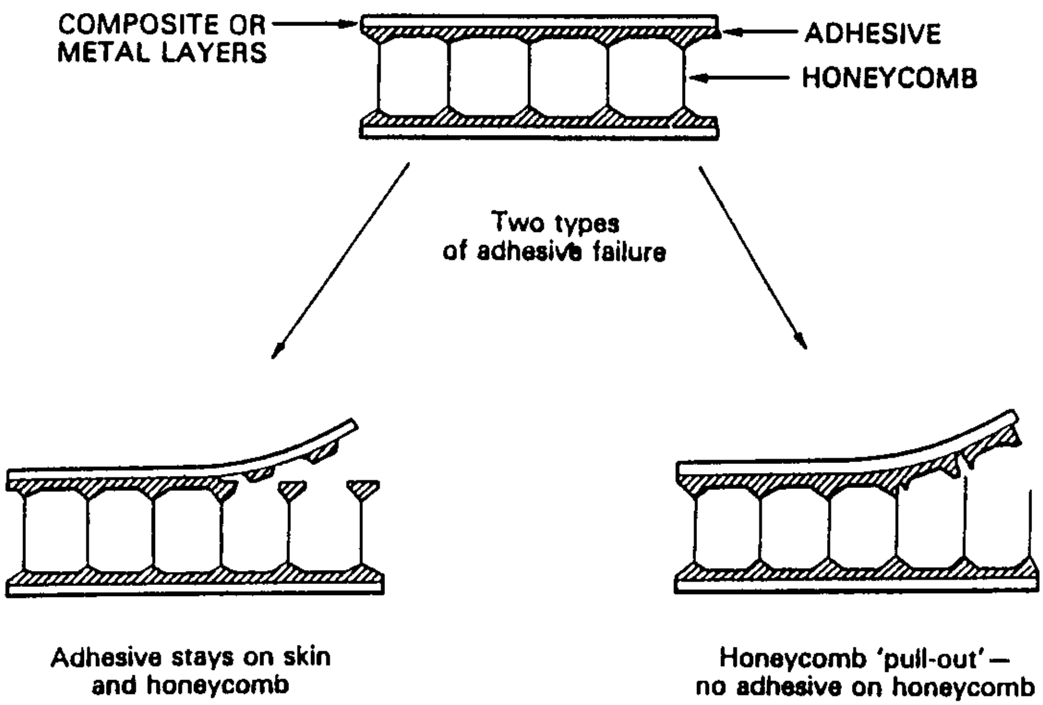
Unbonding

Since in all composite materials we are dealing with at least two materials designed and manufactured to act as an individual structural element, these individual materials are joined or bonded together. This area where the two materials bond is referred to as the interface or bond joint. This bond is both chemical and interlocking in nature.

During manufacture or service, various chemical or loading factors come into play which can weaken or separate the interfacial bonds. "Disbonding" or "unbonding" are terms used to describe a situation of separation at or near the interface. Obviously, if poor technique was used in the original manufacture, the bond strength can be lower than expected, resulting in debonding in service. Figure 37 illustrates different modes of debonding of a honeycomb core sandwich panel. If the debonding occurs completely within the adhesive and not right at the interface, this is termed "cohesive failure". On the other hand, if the debonding occurs at the interface and the adhesive remains with one of the parts, this is termed adhesive failure".



Unbonded



Unbonded

Figure 37

In general, adhesive failures may imply a manufacturing problem or in service attack of the bond joint. Cohesive failures usually result from structural overload and suggest the bond joint was satisfactory to carry the loads applied. If adhesive failures occur, careful laboratory study of the debonded joint may reveal the specific problem giving rise to the weak bond. Care must be taken in handling and preserving the debonded samples to avoid contaminating the surfaces and masking the true cause of failure. Many adhesive failures can be traced to inadequate surface preparation prior to bonding.

A cohesive failure implies that the bond strength exceeded the bulk strength of the adhesive. If the adhesive received the proper cure and the correct adhesive was used, the cohesive failure demonstrates that adequate strength was developed in manufacturing. In some cases, the adhesive joint can exceed the strength of core material and one may find instances where the honeycomb material is fractured and remains bonded to the skin sheet.

Filament/Matrix Failures

In fiber reinforced matrix materials, the fibers are arranged to carry most of the load in tension or compression. The matrix keeps the fibers aligned and carries the shear between fibers or plies. In some cases, fibers are arranged perpendicular to the plies to carry shear as well. Because the matrix and fibers must work together, bonding between the fibers and matrix is critical and on a microscopic scale, adhesive and cohesive failures can control the mode of failure.

In the case of excessive tension loading, the fibers may fail in direct tension loading and, under high magnification, the fiber ends would appear as flat, brittle fractures. The fracture ends will be nearly even with the edge of the matrix fracture surface. This failure mode is referred to as "fiber fracture".

Even though a composite may carry primary loads in tension, shear is developed between each fiber and its surrounding matrix. If the fiber/matrix interface is weakly bonded, fibers may pull out of the matrix resulting in a tension failure with fibers protruding from the fracture face. This failure mode is termed "fiber pullout". Fiber pullout may be caused by improper manufacturing processing of the composite materials.

Under compression loading, the fibers carry loads as if there are numerous parallel columns. As with larger columns, excessive loading can cause buckling or local crippling.

Overall buckling of long, slender members or sheets subjected to edgewise compressive loading may result in elastic instability and a situation of unwanted loss of function or transference of load to secondary structure. Such an elastic failure may or may not result in fracture. Nonetheless, elastic buckling can be a harmful failure mode.

If members under compression loading are short, or are restrained from lateral buckling as a whole, excessive compression loading may cause local areas of the material to form kinks and local zones of disintegration. In areas where the fibers are not straight (such as in woven fabric) composites are more prone to this sort of local failure. Close observation of this failure mode from the edge will usually show where the fibers have crippled and formed a jog.

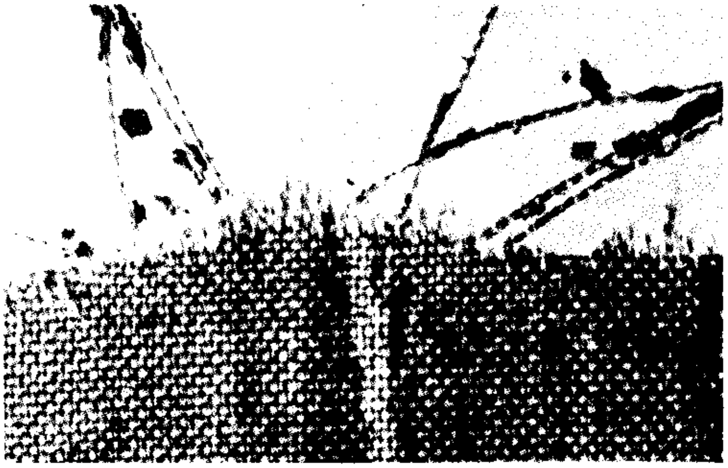
Matrix rich zones in multiple composite structures are susceptible to shear failure under bending stress. When a composite material is flexed, both longitudinal and transverse shear stresses are developed. Because of the high percentage of longitudinal or planar fibers, the transverse shear is not usually critical. However, the longitudinal shear is parallel to the predominate fiber orientation and can result in delamination of the plies. A common quality control test to check for adequate resistance to delamination is subjecting a short beam section of the composite to three point bending.

A short beam comprised mainly of uniaxial fibers may also fail due to fiber fracture on the tension side of the beam or fiber crippling on the compression side of the beam. The fracture appearance in these instances will resemble fractures in wood loaded with the grain oriented lengthwise. The reader is referred to the section on wood fracture modes found elsewhere in this text for typical examples of these failure modes.

3.2.2.6 Failure Modes of Fabric

Tensile

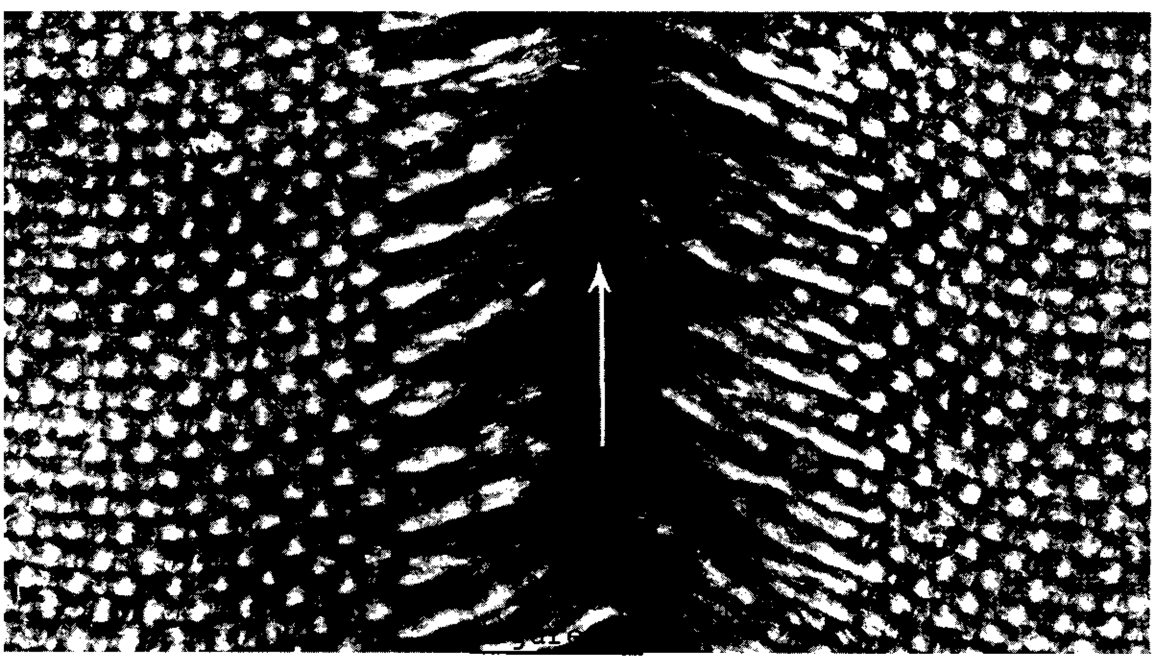
As would be expected, fabric failures result from an overload of the individual threads. If the applied tensile force is parallel to the threads in the cloth, the outstanding thread ends, which have a brush-like appearance, will not be deformed from the line of the load. If the applied tensile force is at an angle to the threads, the threads at the fracture will be deformed in line with the load (Figure 38).



Bushy appearance, fibres straight along threads — *Tension* — **Figure 38** — *Bushy appearance, fibres turned locally at fracture across threads*

Tearing

Under tearing loads, the individual threads fail in tension, but the threads are usually deformed in the direction of the tear. The ends of the threads present the familiar brush-like appearance. The deformation of the threads is much more pronounced than that which is found in tension loading at an angle to the thread line (Figure 39).



Tearing.
Fibers turned into direction of progression of failure.

Teasing

Teasing is the term applied to the appearance of fabric fractures which have been flapping in the airstream after failure. The fabric becomes unraveled, fluffy and sometimes even tied up in knots. Sometimes this can be used as indication of in-flight failure. The condition can, however, be encountered on the ground under high wind conditions and caution must be used in applying this particular characteristic (Figure 40).

Figure 40

Fractured edges "fluffy" due to flapping in airstream.

Some idea of the time of exposure can be determined from the amount of teasing present. Large amounts of teasing might indicate long exposure and/or high airstream velocity.

3.2.2.7 Common Fractures in Plastics and Glass

Failures in plastic windows are difficult to evaluate because in most cases only a small number of fragments are available for examination. The more pieces recovered, the better is the chance of determining the cause. The general procedure used in studying failures in plastics is to piece together the available fragments and then by correlating the individual failure patterns to isolate the initial failure. In the following subsection, information is presented on the appearance of typical tensile, bending and tearing type of fractures. In addition, there are a few general principles which assist in isolating the initial failure. A first path of failure terminates only at an edge of the panel and is generally a smooth curve. Therefore, breaks or fractures which end on other breaks can be dismissed as being secondary failures All breaks should be carefully examined for evidence of bubbles, scratches, nicks or gouges. These will, in general, act as stress raisers and initiate the failure.

Two general types of markings in glass or plastic fractures have been identified and are in general use. These two markings are "rib marks" and "hackle marks" (Figure 41). Rib marks are similar to the familiar fatigue clamshell or beachmarks and are curved lines radiating in the direction of the fracture propagation. The fracture direction approaches a rib mark on the concave side and leaves the convex side. Although rib marks are found on glass and plastic fractures initiated by impact, they can be produced by relatively slow tearing of glass or plastic. Hackle marks are perpendicular to the rib marks and are similar to the fatigue "ratchet marks" which indicate multiple cracks joining with one another. Hackle marks are valuable in identifying the origin of the fracture since they always point in the direction of the initial crack. If the source of the failure is a bubble or the flaw, the hackle marks will very often spread out in ray-like fashion from the flaw.

Tensile

Because of their low ductility, plexiglass and other similar plastics fail in a brittle manner. The failures generally originate at some local weak point in the material or at a scratch or gouge. The initial failure zone is usually flat, smooth and lightly polished. Marks resembling the "herringbone" markings found in metal tearing fractures radiate from the origin of the tensile failure. Moving the piece back and forth to get different lighting on it will sometimes help to make the markings more easily discernable.

Bending

The outer or tensile side of the bend can be generally determined by looking for the flat side of the fracture which is roughly perpendicular to the surface. On the compression side, the failure is usually on an oblique plane and the compression edge is either lipped or rounded off. The rib marks will be asymmetrical with one edge leading. The leading edge corresponds to the side of the piece with the bending induced tension acting (Figure 41).

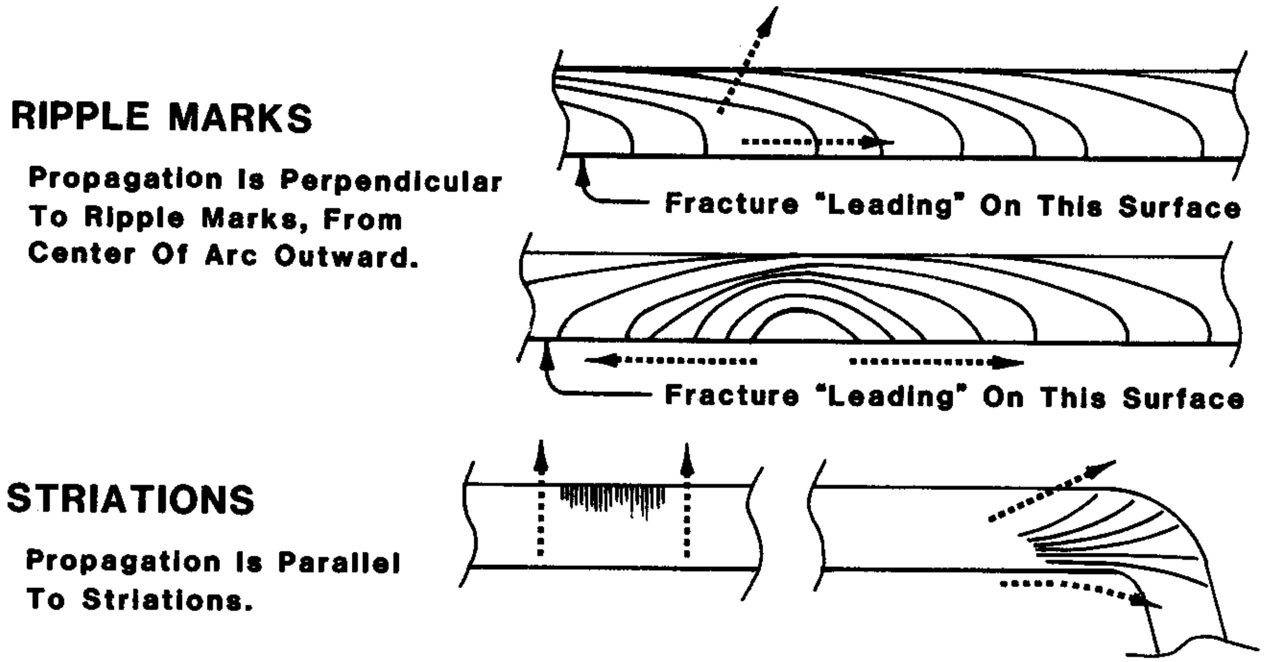


Figure 41

Tearing

Tearing in plastics is essentially a tensile tearing under loads in or nearly in the plane of the surface. Very often bending effects, combined with tension effect, are found in tearing fractures. Curved wave-like lines can be seen on the fracture radiating from the point where the tear started. These curved lines are usually perpendicular to the tension edge of the fracture and curve rapidly until they appear to run tangent to the compression edge. These marks resemble the familiar clamshell or beachmarks found in metal fatigue failures and are generally referred to in plastic fractures as "rib" markings.

3.2.3 Progressive Failures

The most important progressive failure modes are fatigue, corrosion, wear and creep. Progressive failures result from accumulated damage from service related factors such as cyclic stresses, chemical attack, abrasion and thermal effects. Each of these factors can reduce the long-term strength of the part and create the opportunity for structural failure or malfunction while operating the aircraft within the design limitations. Needless to say, progressive failures are potentially disastrous. This section will describe the theory underlying each type of progressive failure mode, discuss the identifying characteristics of each, and summarize the types of further laboratory or engineering analyses that can identify the causative factors.

3.2.3.1 Fatigue, Identifying Characteristics and Contributing Factors

Fatigue is simply the progressive failure of a part under repeated loading. Fatigue fractures start as minute cracks that grow larger under the action of fluctuating stress. Fatigue is not a new problem. In fact, it is a very old problem, dating back to the time when metallic materials were first used for structural components. As early as 1858, Wohler, one of the earliest researchers in the field, concluded after conducting many tests: "Wrought iron and steel will rupture at a stress not only less than the ultimate static strength of the material, but even less than the elastic limit, if the stress is repeated a sufficient number of times". This finding is as true today as it was then.

3.2.3.1.1 Fatigue Fracture Appearance

Fatigue failures will normally have two distinct fracture zones, the progressive zone and the instantaneous overload zone. The progressive zone reveals the size and shape of the fatigue crack just before final separation. When the stress level is low, the fatigue zone is large and vice versa. Stress concentrations affect the general curvature of the beachmarks. In all cases of fatigue under bending loading, the radius of curvature increases as the crack progresses inward. As the stress concentration increases from a value of 1.0 (no stress concentration) up to some high value, the curvature of the arrest marks increases markedly and, at the very high stress concentration, the curvature becomes convex instead of concave. The displacement of the arrest marks shown for the rotary bending case is associated only with this loading and is known as "crack slip". This slip or turning around is against the direction of rotation and this point can be used to determine the rotation direction (Figures 42, 43 and 44).

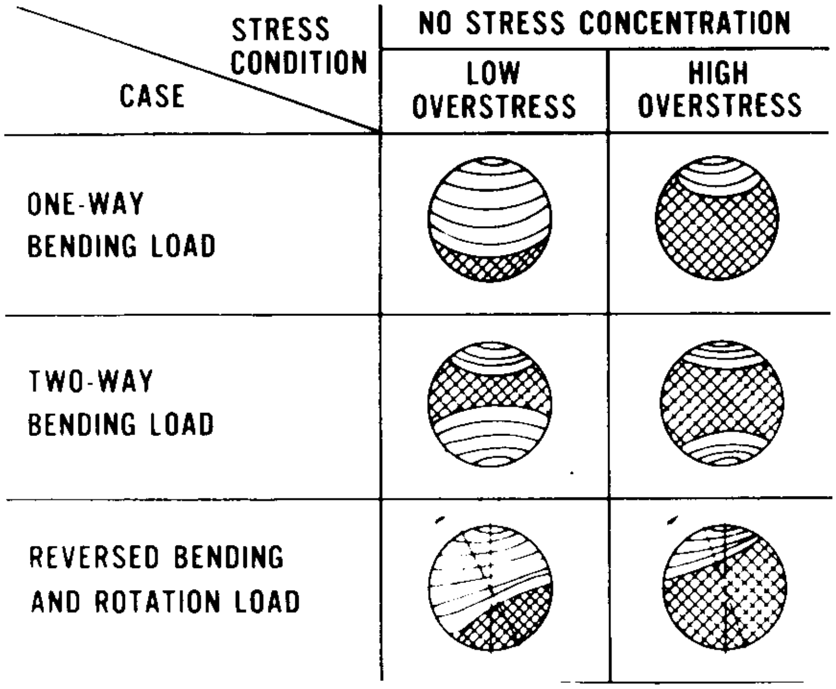


Figure 42

Fracture appearances of bending fatigue failures. Instantaneous zones are shown as cross-hatched areas.

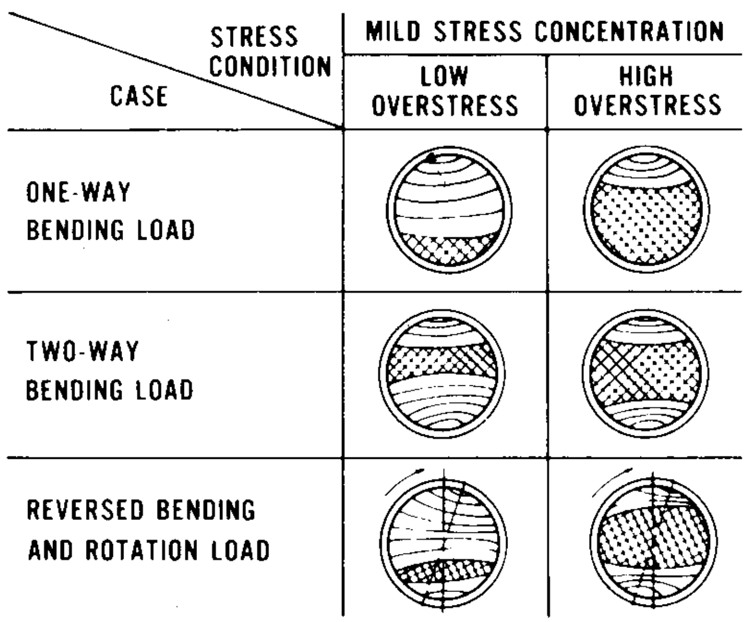


Figure 43

Fracture appearances of bending fatigue failures Instantaneous zones are shown as cross-hatched areas.

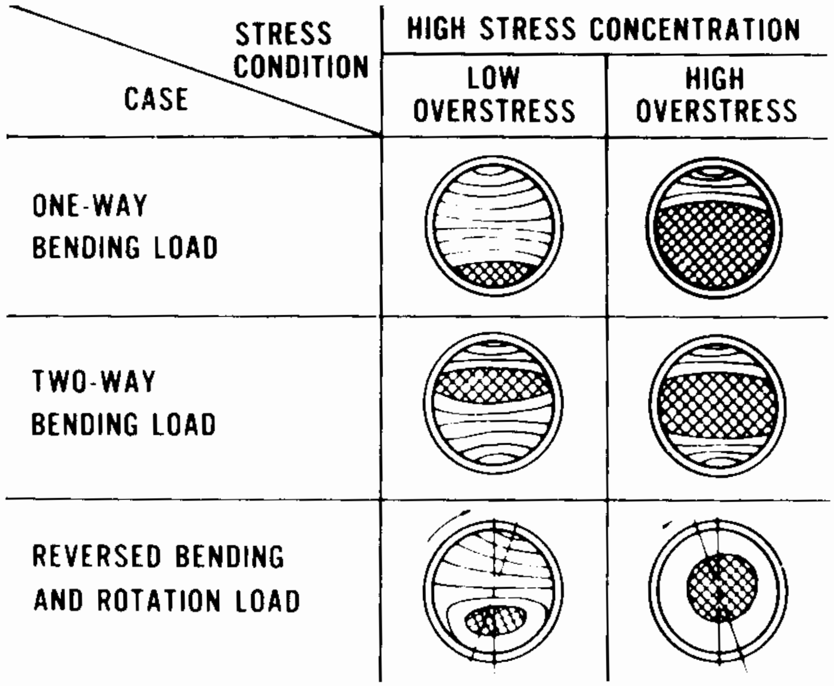


Figure 44

Fracture appearances of bending fatigue failures. Instantaneous zones are shown as cross-hatched areas.

These general features can be used to determine the type of bending loading applied, and, qualitatively, the stress level and presence or absence of stress concentrations.

3.2.3.1.2 Tension Fatigue Failures

Because of initial eccentricities in a part, or because of eccentric loading, pure tension loading rarely occurs in service. Usually some amount of bending accompanies tension on axial loading. As in bending fatigue failures, the relative size of the fatigue zone and the instantaneous zone can be used as a measure of the stress level which produced the failure. The origin will often be associated with a stress concentration. The fatigue zones will have a similar appearance to those described in the section on bending fatigue.

3.2.3.1.3 Torsion Fatigue Failures

Torsional fatigue cracks usually progress along a helical path. The direction of grain flow may also influence the direction of the initial crack in some materials. Fatigue arrest markings may not always be found on the fracture and secondary means, such as absence of ductility and observing the angle of the failure plane, and must often be used to identify failures of this type. Torsional fatigue fractures are usually very smooth from the rubbing of the two fatigue halves of the fracture before separation of the initial crack will start in one plane and then slip off into another. In searching out torsion fatigue failures, the investigator is usually aided by the knowledge that cyclic torsion loading is present in the service application. In this regard, torsional fatigue should be suspected when examining failures of crankshafts, flap drive torque tubes, coil springs, splined shaft members, etc.

3.2.3.1.4 Basic Theory

Although fatigue failures have been occurring for a hundred years or more, the actual mechanism of fatigue is not totally understood.

The best explanation of why parts fail in fatigue under repeated loading is given by the "slip theory".

The "slip theory" works as follows: A metal is composed of a large number of crystals whose axes have a random orientation. When tensile stress is applied to a piece of the metal, the actual internal shear stress affects some of the crystals more than others. Consequently, stresses at the boundaries between crystals will vary in magnitude. When a small stress is applied to the piece, slip planes will develop within a few of the crystals, i.e., plastic yielding has taken place. Although the piece has not been stressed beyond the elastic limit, a few grains within the piece have undergone permanent deformation. Work hardening of the crystals accompanies slipping. As additional slipping takes place, bands of persistent slip develop and microscopic notches are formed. Layers or "blocks" of the grains slide back and forth, but the accumulated damage in these bands of intense activity prevent the metal from returning to its original configuration when the stress reverses. In the early stages of crack formation, this sliding (shearing) dominates. The crack may stop at the grain boundary if the neighboring grains are able to supply stress relief in the form of plastic flow. If they are unable to do this, the crack will spread from grain to grain by way of transgranular cracking (across the grains). The crack may grow at different rates at different times due to the variability of loads experienced. Ultimately, once the crack has grown large enough, the remaining metal is unable to withstand the next stress cycle and the part fractures completely. The final overload part of the fracture can be in tension, bending or torsion, depending upon the kind of load applied.

FATIGUE FAILURES

Identifying Characteristics:

Fatigue failures may be identified by a two zone fracture surface. The fatigue zone will exhibit a smoother appearance and arrest marks (beachmarks) showing the progression of the crack as it enlarged. The remainder of the fracture will be more coarse, consistent with the instantaneous fracture mode operating at final separation.

The fatigue zone origin is found by tracing the beachmarks back to their point of focus. The fatigue crack may have started at a notch, flaw or other feature which acted as a stress concentration. Fatigue can also start on a smooth surface if the applied cyclic stresses are high enough.

3.2.3.1.5 Flutter

Flutter is an instability type of phenomenon involving self-excited oscillations of control surfaces, wings, or propellers. Its occurrence is dependent upon the interrelationship of the aerodynamics forces, inertia forces and elastic forces of the system. When divergent flutter does occur, the amplitude of the oscillation builds up and extremely high loads are developed, resulting generally in structural failure of the aircraft or one of its components. For this reason, flutter can be considered as a special variation of excessive cyclic loading and can be handled by the investigator in the same general manner as for that category of failure.

The possibility of flutter is considered during the design and testing of modern aircraft and all aircraft certificated by the Federal Aviation Administration have been demonstrated to be free from flutter when operated within their design limitations. Flutter can occur in service, if the original mass balance or component stiffness is altered by repair or modification work, or if excessive free play is permitted through poor maintenance. In two recent accidents, the repainting of the tail surfaces (without removing the old paint) changed the static balance and incipient flutter developed. Aileron flutter has occurred due to the build up of ice inside the ailerons. Elevator trim tab flutter has occurred when tab push rods disconnected or tab control cables corroded and broke (see Figure 45). Free tabs will normally flutter and may drive the elevators to destructive oscillations. In this example, the forward end of the tab push rod showed evidence of hammering inside the elevator, cyclic scrapes were found on the tab push rod, and the elevator hinges over-travelled in both directions.

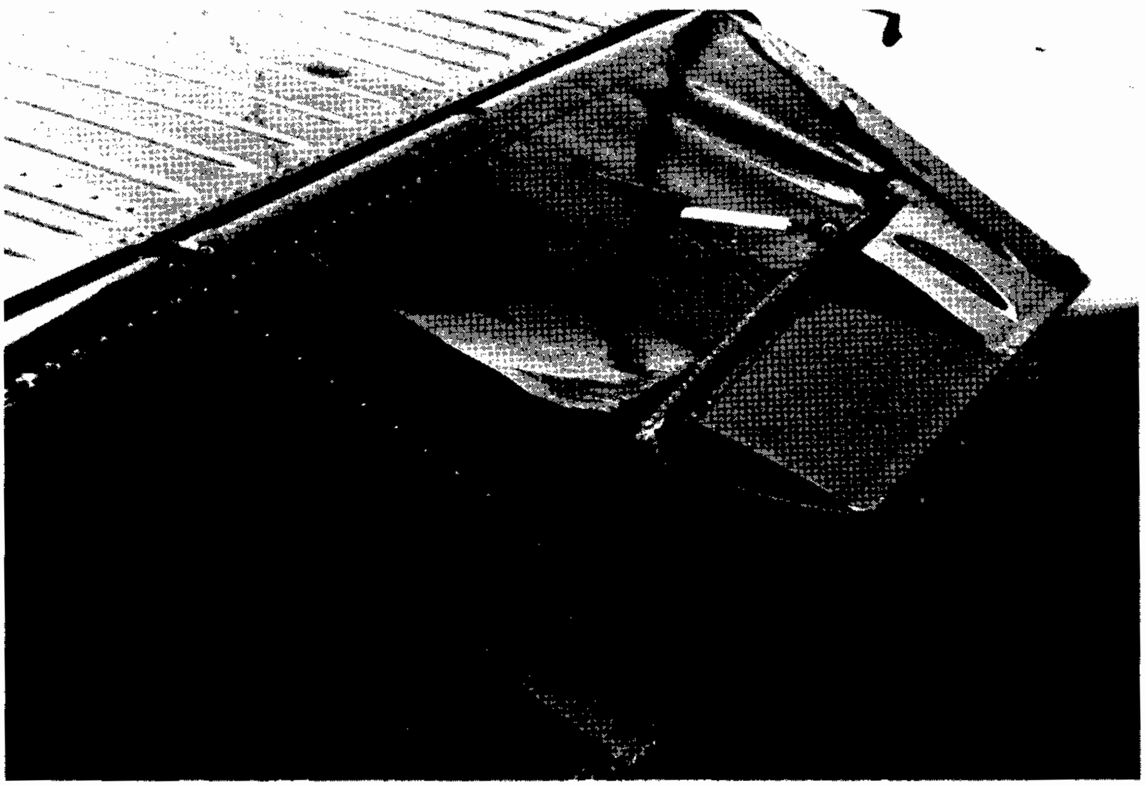


Figure 45

Beech Baron elevator with trim tab flutter when push rod disconnected.

In general, flutter is a remote possibility. When in-flight structural failure occurs, many parts will be subjected to unnatural loads and the secondary overloads can be quite complex. The investigator must be careful not to confuse secondary damage due to overload with flutter effects. Flutter will create various signs of repetitive cyclic overloads. These signatures are commonly found on over traveled hinges, push rods, bell cranks and control surface stops. Cyclic twisting of wings may introduce 45° creases in both directions on upper and lower skins or working of rivets and other fasteners. Pilots that survive a flutter incident will report violent shaking of the airframe and instruments, and rapid motions of the control yoke and/or rudder pedals.

3.2.3.2 Corrosion

Corrosion is the natural effect of the environment chemically attacking the metal. Most structural alloys must be protected from the environment to avoid corrosion. Most parts are coated to achieve a barrier between the metal and the corrosive environment. In some cases, proper design, incorporating drains or ventilation, will prevent the environment from becoming too aggressive. In general, corrosion damage implies either the coating has been damaged or deteriorated, or the structure has undesirable characteristics which trap corrodents. As described in the fatigue section, one major problem caused by corrosion is corrosion induced fatigue failure. This section will describe basic corrosion theory and discuss the major forms of corrosion that are applicable in aircraft structures.

3.2.3.2.1 Corrosion Theory

Metals are produced from ores which are typically metal oxides. Left unprotected, the metal will have a natural tendency to reoxidize to its more stable form. Dry air is not however, very corrosive and the presence of moisture is usually required.

Water with dissolved gases or salts can be very corrosive to bare metals or coated metals which have damaged coatings. The water acts as an electrolyte which assists in the process of corrosion. At the most basic level, corrosion involves metal atoms becoming ionized (electrically charged atoms) to form oxides, hydroxides, sulfides, etc. This implies that electrical charge must be involved in the corrosion reactions. The electrons freed from the atoms are also involved by reacting with the oxygen or with hydrogen ions in the water. Corrosion then involves the release of electrons from the metal atom which can then dissolve into the water (corrosion) and the reaction of the electrons with a substance that can "pick up" the extra electrons. Because these reactions involve the transfer of electrons at isolated spots on the surface of the metal, corrosion is said to be electrochemical in nature.

3.2.3.2.2 Galvanic Corrosion

As we have just seen, corrosion involves charge transfer or current flow. If two different metals are electrically connected and both are exposed in the same corrosive environment, galvanic corrosion can occur. Galvanic corrosion is the result of the active metal corroding faster than usual with the passive metal acting as the place where the electrons are discharged. This is the principle of operation of a battery. In a battery, the anode (positive terminal) is the corroding metal and a voltage is developed because of the electrons being attracted to the cathode (negative terminal).

Galvanic corrosion may occur in aircraft structures where two metals are fastened together and the area of the interface collects water. Uncoated steel in contact with aluminum, or copper in contact with aluminum are two potential galvanic couples. Galvanic corrosion is identified by the localized corrosion of the active metal in the pair, near the interfacial surface.

Galvanic corrosion is prevented by coating either or both metals (isolate the metals from the environment) or insulate between the metals (break the electrical contact). Of course, a design change may be warranted to better drain the area, if the coatings are subject to excessive failures.

3.2.3.2.3 Intergranular Corrosion

Intergranular corrosion is the result of galvanic corrosion effects occurring on a microscopic level. Inside metal alloys, small particles of different chemical composition exist between metal grains. If one type of particle or area in the microscopic structure is attacked more readily than the remainder of the alloy, galvanic effects are at work. Exfoliation of aluminum alloys and sensitization of stainless steels are described in this section.

Exfoliation

In the heat treatment of aluminum alloys, particularly those alloyed with copper such as alloy #2014 or #2024, the grains develop a nearly pure aluminum surface with particles of copper between the grains. As a result, corrosion can rapidly penetrate between the aluminum grains leaving most of the grain itself nearly unaffected. This is a form of intergranular corrosion on a microscopic level.

In extruded structural shapes made of aluminum alloys, the grain structure is highly directional. The grains are much longer in the direction of the extrusion axis. If a corrodent can attack the end grain exposed at the end of the part, or at the sides of drilled holes, corrosion penetration can be severe.

The part suffering intergranular corrosion appears to be splitting open or swelling. The corrosion products wedge the grains apart forming a severely corroded and deformed appearance. This form of intergranular corrosion is referred to a exfoliation.

> **EXFOLIATION**
>
> **Identifying Characteristics:**
>
> **Exfoliation is identified by the lengthwise fragmentation of extruded aluminum alloy shapes such as spar caps. Corrosion of the elongated grain boundary forms light, gray colored aluminum oxide corrosion product which causes the affected area to appear to swell up. The exfoliated area may also seem to be made of multi-layered aluminum sheets separated by powdery corrosion product.**

Sensitization

Stainless steels such as Types 304 and 321 are also subject to intergranular corrosion under some circumstances. Stainless steel is corrosion resistant because the high chromium content allows a thin, transparent oxide film to form on the surface that protects the metal from atmospheric corrosion.

These stainless steels also contain approximately 0.08% carbon which, if heat treated properly, causes no problem. If Type 304 stainless steel is heated to between 850^{o}F and 1450^{o}F, the carbon combines with the chromium to form chromium carbides inside the metal. Unfortunately, these carbides form in the grain boundaries and deplete the grain boundary areas of chromium. As a result, the grain boundaries are not "stainless" and intergranular corrosion occurs in water that is even slightly acidic. This phenomenon is given the name "sensitization". A sensitized stainless steel is one that will experience intergranular corrosion due to grain boundary carbide precipitation. Sensitization frequently occurs in welded stainless steels because the heat affected area near the weld is in the critical temperature range long enough for the chrome carbides to form. (See Figure 46).

Figure 46

Metallurgical cross-section showing chromium carbides precipitated in the grain boundaries of a stainless steel exhaust system causing sensitization, magnification 200X.

Type 321 stainless steel is specially formulated to be resistant to sensitization in the as-welded condition. Titanium is added (approximately 1/2%) to the alloy to preferentially react with the small amount of carbon present and to avoid the chromium being depleted. While welding 321 stainless steel chromium carbides do not form. However, there is a very narrow band near the weld bead where neither titanium nor chromium carbides form. So long as the alloy is not subsequently reheated to the sensitizing range this causes no problem. If reheated in the critical range, chrome carbides form and titanium carbides do not; the area immediately beside the weld bead can experience intergranular corrosion. Mufflers and exhaust systems are a particular problem in this regard. Higher alloy materials such as Inconel are now being used to avoid this and related problems.

Intergranular corrosion of mufflers and exhaust systems of general aviation aircraft can pose a carbon monoxide hazard and/or a fire hazard. The corrosion normally proceeds from the inside and is not readily detected until the wall is perforated. Obviously, an external visual inspection is no sure way to avoid in-flight problems between inspections. The intergranular failure will have the appearance of rock candy around the edges of the crack or the hole. If hot exhaust gases escape for an extended period of time, the fracture surfaces may erode, making the failure mode identification more difficult. Old looking rough edges are a good sign of intergranular corrosion. A more detailed laboratory examination may be able to offer more supporting evidence and to help date the age of the problem.

3.2.3.2.4 Stress Corrosion Cracking

Stress corrosion cracking is the result of the simultaneous effects of relatively mild corrosion and tensile stress. Either acting alone would not be harmful. For some alloys, the synergistic effect results in formation of cracks which enlarge with time. Once the stress corrosion crack becomes large enough, it may initiate fatigue cracking (due to a notch effect) or an instantaneous failure may occur, depending upon the nature of the service loads. Higher strength alloys, such as hardened steels, high strength aluminum alloys, and titanium are susceptible to stress corrosion cracking under certain conditions of stress in specific corrosive environments.

In wrought alloys, stress corrosion cracking is most prevalent when the material is stressed in a direction applying tension across the grain pattern. Either residual or applied stress may cause cracking. Locations where grain flow emerges from the sides of forgings are particularly susceptible to this problem.

Structural aluminum forgings are subjected to quality control steps to make certain the grain flow in the forging is proper. If the grain flow perpendicular to a highly stressed surface, stress corrosion or fatigue is much more likely. The grain flow pattern in a forging is sometimes unpredictable and extreme care in forging practice is a necessity. Destructive metallurgical laboratory tests will usually be required to determine the grain flow pattern and its possible relationship to the failure.

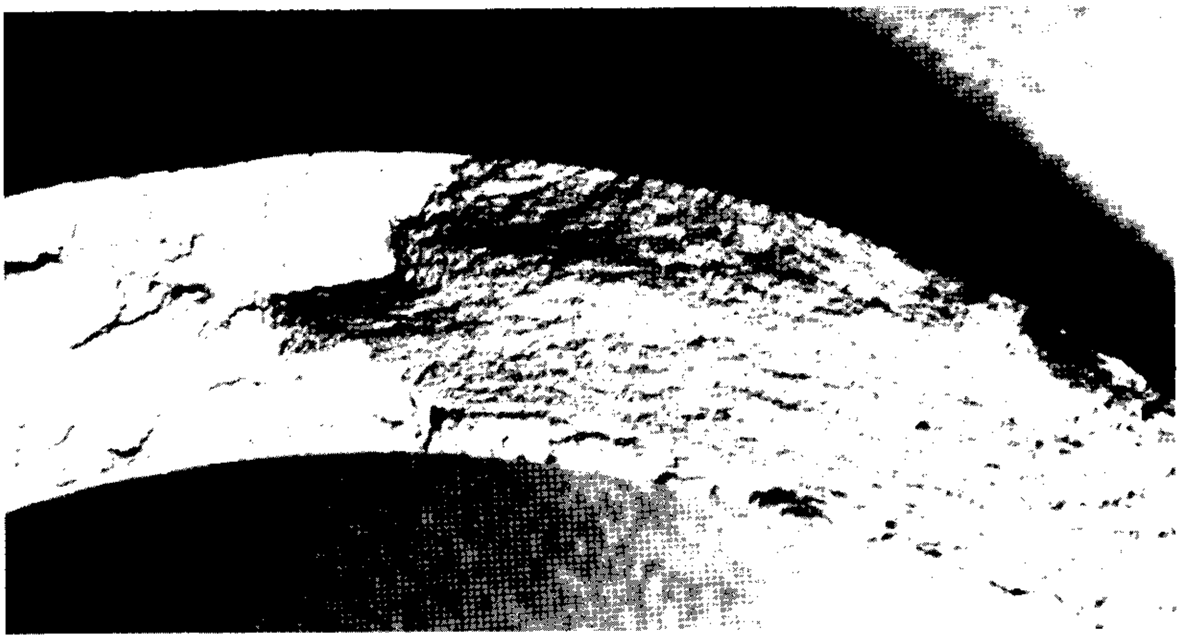


Figure 47

Fracture surface of a nose wheel fork forging
that has split along the forging parting line.
Note the two zone fracture, the flatter
area (top) is stress-corrosion cracking;
the fibrous area (bottom) is the final overload zone.

STRESS CORROSION CRACKING

Identifying Characteristics:

Stress corrosion cracking may be identified by the presence of multiple, brittle appearing cracks in high strength alloys. Under magnification, the crack path is often found to have followed the grain boundaries and to be branched (secondary cracks spreading sideways from the main crack). The cracks will form perpendicular to the major tensile stress axis which may be from either applied or residual stresses in the part. A great amount of external corrosion damage is not a necessary condition for stress corrosion cracking to form.

3.2.3.2.5 Hydrogen Embrittlement

High strength steels are susceptible to cracking when hydrogen enters the metal. The hydrogen may come from the corrosion reaction with water or may be electrochemically introduced during electroplating. Many steel aircraft parts are cadmium plated. If high strength parts are electroplated, they must be heated to approximately 400°F for several hours to bake out the hydrogen. If not baked, once the part is loaded, the hydrogen moves through the metal and allows cracks to form after a period of time. (Figure 48)

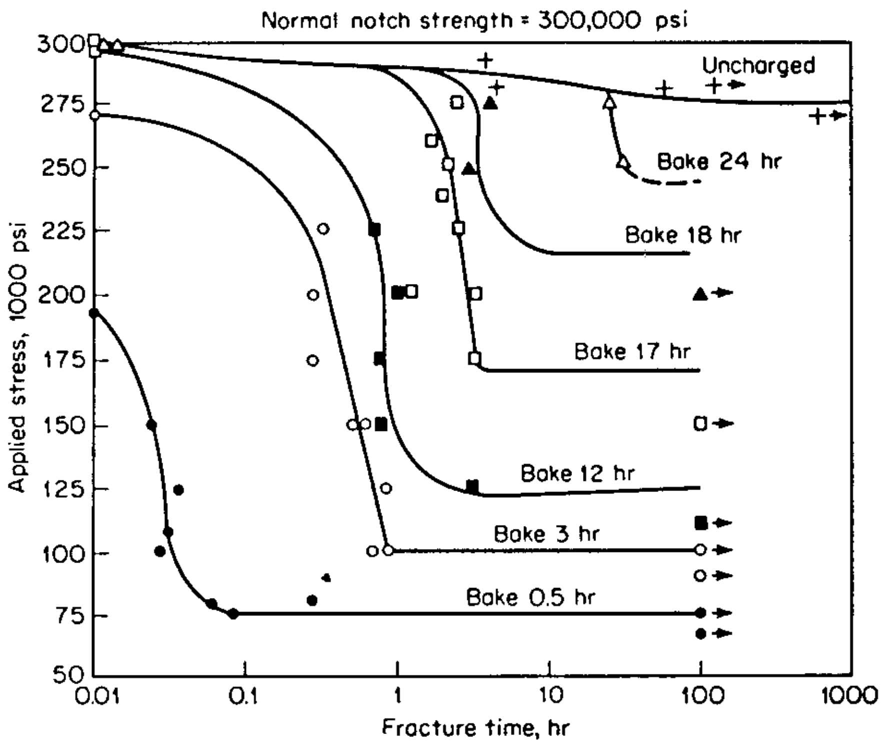


Figure 48

Hydrogen baking chart illustrating the effect of baking time at 400°F needed to restore the strength and ductility of high strength steel.

If the hydrogen is absorbed during corrosion in service, it could be considered a special case of stress corrosion cracking. Obviously, the source of the hydrogen must be determined if future cases are to be avoided. If the hydrogen is because of a missed baking operation, an entire lot of parts are at risk. A laboratory metallurgical examination should be conducted for such parts.

Hydrogen embrittlement cracks will normally be intergranular (between the grains) and will be highly branched. Once formed, hydrogen embrittlement cracks may serve to initiate fatigue or instantaneous fractures, depending upon the circumstances of the loads applied.

> **HYDROGEN EMBRITTLEMENT**
>
> **Identifying Characteristics:**
>
> **Hydrogen embrittlement results in brittle half moon shaped cracks in high strength steels. The surface of the progressive part of the crack is often tarnished or rusted. Under magnification, the cracks are usually intergranular and highly branched. A detailed metallurgical study is usually needed to confirm this failure mode.**

3.2.3.3 Wear

Wear is the result of long-term mechanical effects slowly removing material from the surface of parts. There are two broad classifications of wear failure modes as explained below.

3.2.3.3.1 Abrasive Wear

Abrasive wear is the term given to the process of small particles abrading the surface of a part. The abrasives can usually be thought of as being contaminants. As such, proper filters and cleanliness are primary means of defense. If abrasive corrosion is suspected, numerous laboratory tests are available to analyze the oil to identify the potential sources of abrasives and to determine which materials are being adversely affected by the wear.

3.2.3.3.2 Adhesive Wear

Adhesive wear is the result of microscopic projections from two sliding surfaces projecting through the oil film and making physical contact. During this contact, pressure welding of microscopic areas occur which are then torn apart as the sliding continues.

During the break-in phase of operation, a controlled amount of adhesive wear is desirable. After the two surfaces are broken in, the wear rate is extremely low. If, however, there is a momentary loss of lubrication, or excessive pressure, velocity, or temperature, adhesive wear can become destructive. Listed below are terms given to progressively more damaging stages of adhesive wear:

> Break-in - short-term wear rate followed by negligible wear in service.
>
> Scoring - the mildest form of damaging adhesive wear. The surface has the appearance of fairly smooth shallow grooves.
>
> Galling - a seriously damaging form of wear characterized by a very rough surface with pieces of metal adhering to the surface which have torn loose from the mating surface.
>
> Seizure - catastrophic failure due to friction welding of the two surfaces together. If the part is turning under high power, the part may rapidly heat to the point of melting. Seized bearings in a turbine engine would be an example of this failure mode.

Fretting

Fretting results from heavy pressure, low amplitude rubbing between surfaces. Fretting is often responsible for surface damage that can initiate fatigue cracks.

FRETTING

Identifying Characteristics:

Fretting is identified by the appearance of surface discoloration and roughening due to the small, relative motion between two surfaces clamped together. In steels, the surface discoloration is red-brown. Aluminum fretting debris is generally a finely divided black powder. Fretting is often identified with the origin of a fatigue crack.

3.2.3.4 Creep (Stress Rupture)

Creep and stress rupture are the result of the metal deforming slowly due to the mutual effects of stress and high temperature. Turbine disks and exhaust valves are two examples of aircraft components that may fail by long-term creep and stress rupture.

When the amount of overheating becomes severe, structural materials may stretch significantly in a few minutes, creating harmful distortion and possible in-flight separation.

3.2.3.4.1 Long-Term Creep

Long-term creep failures are also called stress rupture failures. When metal parts are held under load at elevated temperatures, a slow elongation takes place. After a period of time, ranging from hundreds to thousands of hours, at temperature and while under stress, internal microscopic defects begin to form and grow. In the later stages of creep damage, pores begin to form on the grain boundaries and fissures open up at places where several grains intersect. These internal defects grow and join up to produce small cracks in the part. By this time, the remaining life is short and the part is said to be in third stage creep. In the last stage, the part will deform at a much more rapid rate and when enough cracks develop, the part will suddenly rupture.

Stress rupture is the expected failure mode of parts that must operate at high temperatures. Exhaust valves, turbine disks and burner cans are examples of parts that are subject to long-term creep damage. Periodic inspection is required of parts subject to creep damage. A typical method of inspection of turbine disks relies on measurement and recording of the diameter of the disk. After the centrifugal force and heat exposure have created a certain amount of growth, the wheel is scrapped. Creep damage cannot be safely repaired.

LONG-TERM CREEP/STRESS RUPTURE
Identifying Characteristics:
Long-term creep failures are identified by a network of brittle looking fissures, roughly perpendicular to the tensile stress axis. The fissures are oxidized and appear burned. The sudden overload of the remaining structure will appear relatively fresh and will be brittle or ductile, depending upon the material involved.

3.2.3.4.2 Short-Term Creep

Short-term creep represents a severe temperature and stress problem with extensive distortion over a period of time as short as a few minutes or so. For short-term creep to occur, an operational upset condition is likely. Short-term creep will be identified by extraordinary temperature signatures and gross distortions and stretching. For instance, an overheated spar cap may have the appearance of pulled taffy. Another example would be a broken rod cap bolt where the matching fractures both look like bullet points. Such rod cap failures are a good indication of oil starvation in the engine followed by extreme frictional heating, leading to a rapid failure.

A short-term creep failure may be caused by in-flight fire. The structural aluminum alloys and aluminum castings melt at between 1100°F and 1200°F, which is lower than the temperature of most flames. Therefore, uncontained in-flight fire is a severe safety hazard. Structural aluminum alloys will lose much of their strength when heated to temperatures even several hundred degrees below their melting point. Some alloys melt over a wide temperature range creating a "mushy state" before they completely melt. This heat softening or partial melting will allow normal flight loads or the air-stream to distort and/or separate the parts which can result in in-flight breakup or loss of control.

Puddled metal or resolidified molten droplets are usually associated with ground fire. Air blasted metal spray and metal pulled apart like taffy are usually signs of in-flight fire. Metal which is partially melted and then subjected to the shock of ground impact frequently forms a characteristic appearance called "broom straw" (Figure 49).

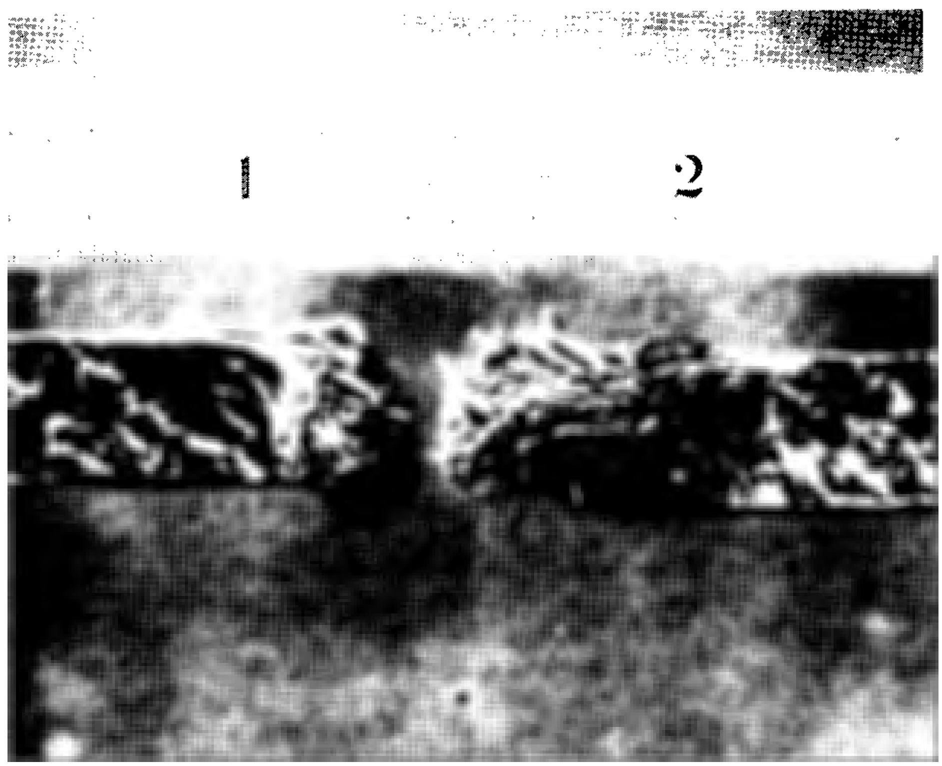


Figure 49

Broomstraw effect from shock loading a partially melted aluminum alloy extrusion.

4.0 ON-SITE TECHNIQUES AND PROCEDURES

4.1 GENERAL APPROACH

The task confronting an investigator upon arrival at the scene of an accident often seems bewildering. However, experience has shown that if the investigator follows an orderly procedure of investigation, the cause of the accident can in almost all instances be found within a reasonable amount of time. First things must come first and in general, there is no short cut to a successful investigation. The investigation must be a planned one with logical courses of action and each particular line of investigation must be followed in a systematic manner. It is especially important that the investigator refrain from arriving at hasty conclusions since this often results in the culmination of an investigation before the true causes are uncovered.

4.1.1 Elimination Technique

When an aircraft crashes, there may be any one of a thousand and one reasons why it did so. The overall task confronting the investigator is one of initiating a program aimed specifically at eliminating those possibilities which could not conceivably have been involved under the particular circumstances. Thus, if the weather is clear, it may be possible to eliminate weather as a factor without further investigation. Similarly, if the accident occurred during landing approach, those possibilities associated with takeoff configurations or circumstances can obviously be eliminated. If all of the major structural components are found at the scene of the accident, it may sometimes be possible to state that no structural failure of the wings, tail surfaces, fuselage, etc., occurred in the air since if failure had occurred, it would be reasonable to expect that the parts would be some distance from the main wreckage. Although no possibility can be completely eliminated until all of the pertinent facts are developed, the more unlikely ones should be set aside in favor of the more likely ones. Very often certain possibilities suggest themselves. Others are eliminated soon after it is established just how the aircraft contacted the ground. The general approach is to gradually eliminate the more unlikely possibilities until a relatively small number remains. Then, by careful, painstaking investigation, the true probable cause and contributing factors can usually be uncovered.

4.1.2 Types of Structural Failure

Categorizing of aircraft accidents is extremely difficult and can often be misleading since in almost every accident sufficient variation of detail occurs to make each accident distinct and slightly different. If the discussion is limited to accidents involving structural failure or malfunctioning, two broad classifications are noted: major component failure and partial failure or malfunctioning.

4.1.2.1 Major Component Failure

As the title indicates, this category is associated with in-flight failure or separation of some major component such as the wing, tail surface, aileron, control system, or fuselage. The relative incidence of their occurrence is approximately in the order listed, with major failures of the fuselage or control system occurring very infrequently. In general, major component failures result from:

1) Excessive loads imposed upon the components
2) Deterioration of static strength through fatigue
3) Inadequate design strength.

Since all civil aircraft are designed and tested to at least the minimum standards of the Federal Aviation Regulations, failures directly attributable to inadequate design strength are remote if the aircraft is operated within its design limitation. Sometimes however, especially when the aircraft is first introduced, different loadings are experienced than those anticipated and static failures occur within the operating limitations. This occurs so infrequently as to be of no particular concern to the accident investigator, but a certain amount of suspicion should always be directed to failures involving new designs. Most of the component failures attributable to inadequate strength are usually associated with deficient repair or modification work, or with an improperly manufactured part or component. Since the manufacturer's standards and procedures are supervised by Federal Aviation Administration (FAA) manufacturing inspectors, major manufacturing errors are kept to a minimum. Faulty repair or modification work is responsible for a large number of failures in this grouping. Improper rivet size or spacing, deficient fabric repairs and poor workmanship are some of the major causes for failure.

Excessive loads are developed when an aircraft is operating outside its limitations of load factor and/or speed. Very often these large loads are imposed inadvertently as when control is lost in severe turbulence. Often the pilot deliberately performs severe maneuvers for which the aircraft was not designed. In either case, the loading on the wing, tail, fuselage, etc. builds up to a value in excess of the design limit and static failure results. The circumstance immediately preceding the failure as developed from witness statements is most helpful in establishing excessive loads as the direct cause. Static failures and a consistent fracture sequence will be found upon reconstruction.

Fatigue failures continue to be one of the major causes of structural failures of aircraft parts and components. This basic cause should always be suspected until other facts or circumstances are developed to disprove it as being a factor. As indicated in the section on "Fatigue", this type of failure can result from a number of causes. In general, fatigue failures are due to: 1) poor maintenance; 2) defective manufacturing; or 3) inadequate design. Since fatigue is usually associated with large numbers of cycles of repetitive loading, this type of failure is rarely found in new aircraft with low service time.

4.1.2.1 Partial Failure or Malfunctioning

Accidents in this general category are by far more difficult to investigate, since there is usually no obvious evidence available with which to make a rapid determination, such as a wing being found two miles from the main wreckage scene. Partial failure or malfunctioning of a major component generally results in altered flight characteristics and these in turn are responsible for the accident. Some of the general causes of accidents in this category are jammed controls, improper distribution of load on board, control surface improperly rigged, incorrect installation of parts, hard-over signals from autopilots, etc. Since accidents of this type are frequently associated with recent repair or alteration work, the investigator can often discover valuable clues by studying the aircraft's history as reflected by log book entries or by other sources. In one recent twin engine transport fatal accident, a cotter pin had not been installed through the bolt which connected the elevator cable to the bell crank at the tail after a maintenance overhaul with the result that the bolt vibrated out, elevator control was lost, porpoising developed during the landing approach and the aircraft struck the ground in a near vertical attitude.

Figures 50a and 50b show an example of a fatal accident that occurred on the first flight after repainting the aircraft. The paint partially jammed the elevator cable resulting in a loss of control.

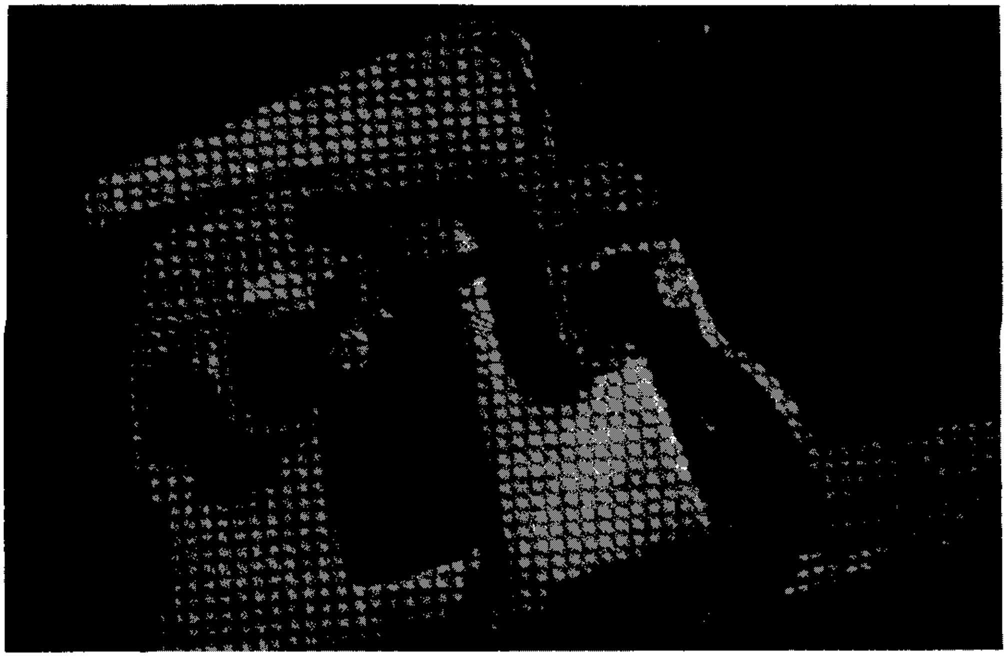


Figure 50a

Repainted elevator cable clevis which makes contact with a bulkhead cutout.

Figure 50b

Matching mark showing clevis interference point. This resulted in loss of pitch control freedom and a fatal accident.

The general procedures used for accidents in this category are to follow routine investigatory practices, systematically following up on various leads and clues until the cause is determined. Certain techniques are available to reduce the amount of work required to complete the investigation. Of these, the elimination technique explained previously is one of the most useful. In most accidents, an experienced investigator can quickly eliminate unlikely possibilities and can isolate the general area in which the initial difficulty is located.

When a malfunction results in an accident, only a certain number of potential causes are consistent with the accident circumstances. The investigator is faced with finding the critical cause and effect. A detailed knowledge of the aircraft systems and deductive logic are the prime tools to use.

4.2 PROCEDURES AND TECHNIQUES

In the preceding paragraphs of this section, the basic causes and contributing factors which are associated with in-flight structural failures of major components have been briefly noted. These points should be of assistance to the accident investigator in evaluation of the failure after it is found. Initially however, his concern is directed toward determining what failed first. Fortunately, failures in this particular category (major component failure) are relatively easy to search out after an accident. This is true because in almost every case the component separates from the aircraft after failure. Since separation generally results, the failed component is found some distance from the main wreckage. When the component or components separate at a low altitude, the parts are strewn along the flight path in the approximate order of their separation. When the component or components separate at a high altitude, the interrelationship of component mass, aerodynamic shape, speed at separation and winds aloft all affect the trajectory of the part and careful study of these factors is required to determine the order of separation from the ground wreckage trail. Methods are available to approximate the trajectories of wreckage parts. Some investigators have had considerable success in evaluating the significance of wreckage trails in accidents of this type.

4.2.1 Initial Wreckage Examination at Site

In the following paragraphs of this section, only those items directly connected with the wreckage examination are covered. It should be pointed out that prior to the actual detailed wreckage examination, the investigator should have completed a general survey of the surrounding area, noting specifically the types of terrain, whether contact was made with trees, buildings, etc., by the airplane before hitting the ground and whether any appreciable burning occurred at or near the wreckage site.

4.2.1.1 General Examination

When the investigator arrives at the scene of a suspected structural failure accident, one of his first objectives should be to conduct a "walk-around inspection" in order to obtain an overall impression of the accident scene from which a general plan of investigation can be formulated. This initial examination should be confined to observation of obvious indications, such as the absence of some component, general failure patterns of the wing, fuselage, tail, etc., fire damage or collision markings. At this stage, an effort should be made to determine the attitude and relative speed of the aircraft before impact. The amount of telescoping of the structure and the size and number of the pieces of wreckage are generally used in the speed-at-impact determination. The extent of damage of one wing panel versus the other, the damage areas on the fuselage, tail, etc., together with the ground markings are used in the attitude-at-impact determination. During this phase of the examination, the wreckage should be disturbed as little as possible.

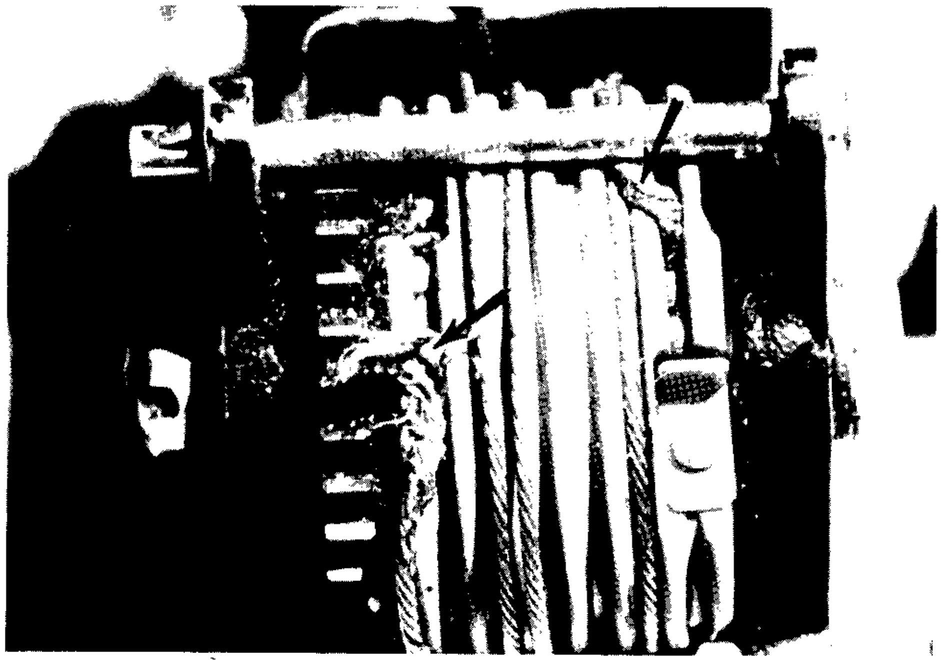


Figure 51

Partially broken and frayed trim tab cable due to investigator "exercising" the system to check out partial binding and continuity.

After the walk-around inspection has been completed, the investigator can proceed to carry out his overall plan of investigation. It is usually advisable to begin this phase by obtaining the necessary data for use in preparing the wreckage distribution chart. When this work has been completed, the detailed examination of individual pieces of wreckage can be initiated. At this stage, particular attention should be directed toward significant smears, scores, indentations, the extent and type of damage, the surrounding ground and the position of the piece relative to the ground evidence. If, during this preliminary examination, the investigator observes any unusual smears, consideration should be made to its significance and whether or not a laboratory examination is necessary. Smear samples must be taken as early as possible since movement of the wreckage will in many instances destroy their value. This is particularly true when fire is involved and samples of ash may be desired. Clear, detailed notes and suitable clarifying sketches should be made of all significant points learned during this general examination.

In many investigations, the investigator will have isolated the cause of the structural failure during the procedure outlined above and additional examination of the wreckage may not be necessary. In some investigations, the cause cannot be found at this point and other techniques must be resorted to in order to determine the part or component which initially failed.

It should be noted that the general plan of investigation as outlined above is only a suggested one and that the circumstances surrounding a particular accident, or the investigator's own working habits may dictate substantial deviations. In general, the suggested plan is a practical one and its use should insure the development of all significant facts within a reasonable length of time. Apparent short cuts often lead to additional work. In this connection, it has been observed that investigators, who immediately after arriving on the accident scene, begin to turn over and rearrange the wreckage without first making adequate notes, frequently are required to spend considerable time puzzling over markings made during the moving process (Figure 52). Deviations from orderly procedures should only be tolerated as a last resort when unusual circumstances dictate such a course.

4.2.1.2 Wreckage Distribution

The wreckage distribution chart is one of the most useful tools the investigator can utilize and often takes the place of or supplements a detailed write-up of that portion of the accident report. This is especially true in the case of accidents involving structural failures. Frequently, failure patterns and failure sequences suggest themselves when the completed distribution chart is carefully studied. In those instances where the wreckage is removed to another site for study, the wreckage distribution serves as the only record of how the various pieces were located at the accident scene. The significance of later findings often depends upon reference to the original wreckage distribution chart. If one was not prepared, the investigation could be seriously hampered.

In determining the type and amount of information to be included on the chart in any specific case, the investigator must be guided by the conditions and circumstances surrounding the particular accident. In every instance, all major components, parts and accessories should be listed and suitable identifiable symbols or titles for each noted. Photographs should be taken of all significant parts before they are moved and so noted. The initial ground contact markings and other ground markings (made by propellers, fuselage, nose, wing tips, etc.) should also be indicated on the chart. When terrain features appear to have a bearing on the accident or on the type or extent of structural damage, they should be noted on the distribution chart. Pertinent dimensions, descriptive notes and locations from which the photographs were taken are additional items which contribute to the completeness of the chart.

In addition to the wreckage distribution chart, other sketches are often desirable and sometimes necessary. Main spar chord failures, skin damage and control surface or system failures are some of the details which can often be handled with more clarity by means of sketches which show station lines, dimensions of breaks, tears, etc. In general, photographic coverage will be adequate. However, when close-up photographs of important failures cannot be made because of lack of equipment, poor lighting, etc., sketches should be made for inclusion in the accident report.

4.2.1.3 Wreckage Mockups

The "reconstruction" technique is one of the most useful procedures available to the investigator for the isolation of the cause of a structural failure. "Reconstruction" means the assembling of the various pieces of wreckage in their relative position before failure. Generally, this technique is employed only for specific components such as a wing panel, tail surface, or control system, although in rare instances it has been found necessary to reconstruct almost all of the major components. The reconstruction procedure is a twofold proposition. First, the various pieces are identified and arranged in their relative positions. Second, a detailed examination is made of the damage to each piece and the relationship of this damage to the damage on other adjacent or associated pieces. This latter work is the chief purpose behind the reconstruction.

The chief difficulty in reconstructing a component such as a wing lies in the identification of the various wreckage pieces. If the wing is broken into a relatively few large pieces, the task is much simplified. If it is broken into a large number of small pieces, the reconstruction job may be extremely difficult. The most positive means of identification is through part numbers which are stamped on most aircraft parts. Part numbers of structural members are frequently not listed in parts catalogs, but can be found in the engineering drawings for the aircraft. When part numbers are unreadable or not found, indirect methods must be resorted to for identification. The coloring (either paint or primer), the type of material and construction, external markings, rivet or screw size and spacing, all can be used to assist in the identification of different parts. For large sections such as spar chords, it is often possible to match the two halves of the fracture. The identification process is sometimes puzzling, since pieces which are normally flat are often found curved, and pieces which are normally curved are often found flat. The investigator soon learns not to be confused by the torn, twisted, buckled condition of a piece of wreckage and to search out the identifiable features pointed out above. Care must be taken to avoid damaging fracture surfaces which may require detailed examination later.

The chief purpose of reconstructing the aircraft or one of the major components is to permit a detailed examination of the various wreckage pieces. When the various parts are placed in their correct relative positions, it is possible to study the continuity or lack of continuity of damage on associated pieces. If wrinkles in one skin panel section are continuous across a tear or break into another panel, then it generally can be stated that the forces causing wrinkling were applied before the forces causing the fracture. This kind of determination is most useful in differentiating between in-flight damage and impact damage, or between primary and secondary failures. The continuity of smears and scores across breaks is an additional point to note during the detailed examination. In-flight fire versus ground fire can be distinguished in this same general manner. Overall failure patterns, including directional indications of the forces involved, can in almost all cases be determined by relating the damage of individual pieces. The manner and direction in which rivets, screws and bolts are sheared is a useful indication in this work. Good notes and sketches should be made throughout this detailed examination. When it will add to the clarity of the accident report, photographs of the reconstruction, including close-ups of significant details, should be made.

At Accident Scene

The reconstruction technique is most frequently employed at the accident scene. This is especially true if the accident has occurred in a relatively open area and the weather is not unusually inclement. Before the reconstruction work begins, a specific procedure should be followed, i.e., overall photographs made, wreckage distribution chart completed, a walk-around inspection conducted and adequate notes made on the manner in which the various pieces were first found. Parts from the suspected area should be collected, identified and arranged on the ground in their relative position. Major components such as the wing, tail and fuselage are laid out separate from one another for ease of later examination. If the suspected area is at the junction of the major components, these areas are reconstructed separately. Individual cable runs with their associated bell cranks, idlers and quadrants are laid out separately, again for ease of examination. If significant markings are found on any of these latter items, corresponding markings can be sought out in the relative position in the wing, fuselage, etc. Reconstruction work at the accident site is fairly straightforward and no great difficulty presents itself unless the accident has been very severe and there are a large number of small pieces of wreckage. In this case, identification is difficult and time consuming, but the results of employing the reconstruction technique are, in most cases, extremely worthwhile.

Away from Accident Scene

Very often the location of the accident or the prevailing weather conditions preclude the reconstruction of suspected components at the accident scene. In this case the investigator must decide whether or not it is warranted or necessary to transport the wreckage or portions thereof to another location for further examination. This decision should be based on a consideration of the type of accident, the facts developed as of that time and the type of information that could be developed from the reconstruction procedure.

Since additional damage will undoubtedly be done to the various wreckage pieces during the transportation process, the investigator should make doubly sure that he has a complete set of notes on all significant smears, scores, tears, etc. All major pieces should be suitably tagged, identified and keyed to the wreckage distribution chart. Minimum disassembling should be done. If it is found necessary to disconnect bolted assemblies, a record should be made of the sequence of the various washers, spacers, nuts, etc. In many cases, control cables will have to be cut to separate portions of the wreckage. When this is done, care should be taken to identify and tag all cuts. Unless these simple precautions are followed, valuable evidence may be lost and the investigator's task may be considerably magnified. The cause of jammed or binding components should be found prior to forcing relative motion.

When the reconstruction is performed away from the accident site, in a hangar, for example, it is usually possible to do a more complete job of reconstruction. Parts can be hung on wooden mockups or frameworks, or suspended from above to achieve a three-dimensional arrangement which resembles more closely the unfailed aircraft. If the parts are arranged on frameworks off the floor, it is possible to examine the upper and lower sides without additional rearranging. Aside from the possible use of mockups, framework, etc., reconstruction away from the accident scene is the same as the reconstruction at the accident site. In all of this work, the goal is to permit a more detailed examination and analysis of the various pieces of wreckage.

The assistance of technical specialists from the airframe manufacturer or component part supplier is often very helpful. Their detailed knowledge of the hardware or ready access to technical information is needed during the reconstruction.

4.2.1.4 Examination for Smears, Scoring, etc.

In the preliminary examination at the accident scene, the investigator's immediate concern is to determine if a structural failure occurred before impact. Toward this end, the initial chief interest is in separating ground impact damage from in-flight failure. In this regard, much valuable information can be gathered from a careful study of the various smears and scores found on different parts of the wreckage. When possible, this study should be made before the wreckage is disturbed, since movement of the wreckage may destroy valuable clues or create misleading ones. The study and analysis of wreckage smears and scores is an extremely valuable aid in the investigation of collision accidents (Figure 52).

A smear can be defined as a deposit of paint, primer, rubber or oil film transferred from one part to another part during the process of the two sliding or rubbing across each other. This sliding or rubbing action frequently occurs after an in-flight structural failure. For example, a failed wing panel often makes such a contact with the rear portion of the fuselage or tail section. If the wing panel had been painted with a distinctive color, it would be common to find colored smears on the fuselage or tail components. These paint smears usually pile up against protuberances such as rivet heads or skin laps. The direction of the smearing force can generally be determined from the fact that the pile up of paint will be found on the side of the protuberance away from the direction applied force. Smear deposits are sometimes found in the recessed slots of screws. In some cases, excess deposits are pushed out from the ends of the slots and deflected over in the direction of the smearing force.

Laboratory examinations can usually reveal the nature of the smear substance and can usually pinpoint the direction of the smearing force. In a recent collision accident, this procedure was used to good advantage to scientifically verify which components had been in contact. This find had an important bearing on the final flight path determination.

Figure 52

Rubber smear from the nose wheel of one aircraft due to a mid-air collision. Reconstruction showed the impact angle to be within 5° of head-on. (Note rubber smear shows the tire serial number backward.)

Score marks are produced when one part slides or scrapes across another. The score marks result when some sharp edge on one of the pieces gouges the other piece. Sometimes only the paint film is gouged, while more frequently actual metal is gouged and an indentation or trough is formed. Close examination of the score marking under a magnifying glass or microscope will reveal directional markings and metal residue which is deformed in the direction of the scoring force. When a skin panel containing a protruding head rivet seam strikes a glancing blow in a painted skin panel, a series of parallel score markings in the painted film is usually produced. If corresponding smear deposits can be found on a particular row of rivets and if the rivet pitch is known, the relative position of the two bodies during contact can usually be established.

Score marks can be used to establish that the damage occurred prior to impact and not afterward. If score marks are found on several related pieces of wreckage, the consistency and continuity of the scores across the pieces after they are placed in their relative positions will show that the scoring was made before the pieces were torn apart. This type of evidence can often be used to establish that the scored component struck or was struck by another component, thus leading to a logical sequence of in-flight breakup.

Many other distinctive markings may be found on pieces of wreckage and a careful study of such markings will very often provide many valuable clues. When a rotating propeller cuts through metal, it leaves a very distinctive saw-toothed pattern. The jagged "teeth" are deformed in the direction of the cutting force and curled over in an easily distinguishable manner. The amount of curling, the extent of the jaggedness and the length and width of the cut all provide indications of the propeller rpm and forward speed during the cutting interval. An aircraft control cable is another item which produces a distinctive marking when it strikes or is dragged across a skin panel. The general indication is a series of tiny parallel lines. The exact shape and size of these cable markings can often be used to determine the direction in which the cable was moving when the markings were made. Peculiar shaped indentations on parts or on skin panels can sometimes be matched with the piece which made the marking and thereby provide a clue to the sequence of failure.

It is possible to be misled by cutting marks produced by an ax or hacksaw used in the salvage operation. The investigator should learn to be familiar with this type of marking and be capable of distinguishing this type from the others described.

4.2.1.5 Care and Handling of Fractures

All fractures should be visually examined for any evidence of progressive or precrash failure modes. The parts will be handled during the recovery and reconstruction phases of the investigation. Due care must be exercised to avoid damage to fracture surfaces.

Inadvertent contact between fractures will permanently destroy fracture details which may be important in determining the sequence of failure. Rubbing or touching the fracture surfaces may add or subtract chemical impurities which may also be important.

Fracture surfaces should only be cleaned when the investigator is certain that the material on the surface was introduced in the crash and leaving the potential corrodent on the fracture would cause certain harm before it could be examined in the laboratory. Fresh water rinsing followed by drying with anhydrous alcohol or acetone should be enough. Steel parts which may be susceptible to rust should be sprayed with clean oil. Place soft padding over the fractures (cotton, clean cloth, paper towels) and securely tape the padding in place. Tag the parts and include information as to how the fractures have been treated. The investigator should also consider sending the lab a separate sample of any suspected corrodent so the lab can analyze it and determine whether it may have masked part of the evidence. Figure 54 is a cartoon which illustrates a number of ways fractures may be seriously damaged. Be careful!

H A N D L E   F R A C T U R E S   W I T H   C A R E

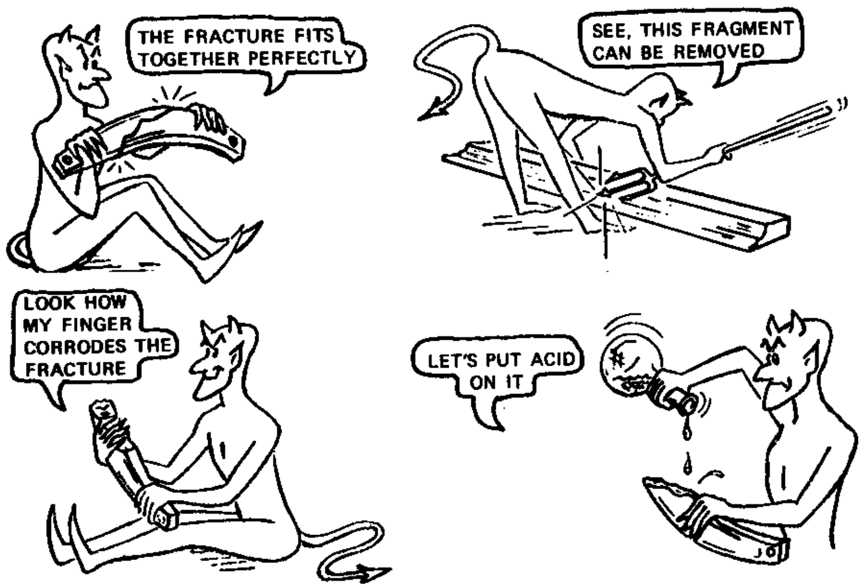


Figure 53

Examples of things not to do.

4.2.1.6 Information Forwarded with Parts

As indicated, laboratory testing is a valuable tool which can be employed to good advantage in many accident investigations. To take full advantage of the technique, it is required that the investigator forward complete information relative to the circumstances surrounding the failure. Unless this is done, a positive determination of the cause may not be possible. The investigator should include instructions relating to exactly what is suspected and what is hoped to be established by testing. The forwarding of parts with the innocuous instruction "for testing" does not provide the technician with the information necessary for the arrangement of a test program and is to be discouraged. It is not expected that the investigator will know exactly what tests should be conducted in every instance, but he should at least have some reasons for suspecting that the forwarded part was involved in the initial failure. These suspicions are what the technician needs to know in order to set up an intelligent test program.

In addition to adequate instructions, as complete a history on the part as can be developed should be forwarded with the failed part. This history should include information such as:

a) When the part was installed in the aircraft
b) Total number of hours on the parts
c) Time since overhaul or inspection
d) Whether any previous difficulty had been reported
e) Other pertinent data which might shed light on how the part failed and why the part failed.

This type of information is extremely important to the technician.

Without this type of information, it is very often impossible to evaluate the significance of failure due to fatigue, corrosion, poor maintenance, etc. The investigator should strive to develop all pertinent facts relating to the failure. In searching for the cause of a particular failure, it is impossible to have too much information at hand for study and evaluation. This is especially true when the technician attempts to project the specific failure to similar type aircraft and decide upon corrective action. In other words, it is not sufficient to establish that a part failed due to fatigue. The purpose of the investigation must be extended to determine why the part failed from fatigue, so that the danger can be avoided on other aircraft. Detailed information is required for this work and the field investigator is often the only person who is in a position to develop the pertinent facts.

5.0 IN-FLIGHT FIRE STUDIES

Each material has its own characteristics with respect to how it will fail when subjected to fire (elevated temperatures). An example: A ductile material has different characteristics than a brittle material and the failure pattern will be different. However, it must be remembered that a brittle material may become ductile at elevated temperatures. Because of these different characteristics of metallic materials, it is important to identify the type of material being investigated. It is imperative that the fire investigator work with and support the materials and/or structures engineer. If there are doubts of the kind of material, laboratory support must be requested to identify the material and its characteristics. The following will provide some information for field observations.

5.1 COLOR OF METALS

Check the color of the metal. Is the color silvery, like polished aluminum; yellow, like brass or gold; gray, like zinc or lead? The color may provide information for identification.

5.2 MAGNETIC TESTING

Magnetic testing consists of determining whether the specimen is attracted by a magnet. Usually, a metal attracted by a magnet is iron, steel, or an iron base alloy containing nickel, cobalt or chromium. However, there are exceptions to this rule since some nickel and cobalt alloys may be either magnetic or nonmagnetic. Never use this test a a final basis for identification. The strongly attracted alloys could be pure iron, pure nickel, cobalt or iron-nickel-cobalt alloys. The lightly attracted alloys could be cold worked stainless steel, or monel. The nonmagnetic alloys could be aluminum, magnesium, silver, copper base alloy, or an annealed 300 type stainless steel.

5.3 SPARK TESTING

Some metals can be easily identified when the specimen is held at an even pressure against a high speed emery wheel. The more iron in a specimen, the lighter the spark will be. As the percentage of iron decreases, the percentage of nickel increases and the spark will darken in color.

Some examples are:

Iron/steel - straw white

300 Series Stainless Steel- red-orange turning white

Monel - course red marks

Cobalt - very short red sparks

Titanium - long brilliant white sparks

The following metals will not spark:

Aluminum, Brass, Cadmium, Magnesium, Silver, Gold, Copper, Lead

There are nitric and hydrochloric acid tests which will provide a spot color test. The base laboratory or the corrosion control shop can provide assistance.

ACID TESTS TO IDENTIFY DIFFERENT METALS

NOTE: Clean surface of metal with file or grinder before applying acid.

1. Concentrated Nitric Acid: Apply one drop

REACTION:

Aluminum-no reaction	Magnesium-effervescent (boils)
Brass-blue/green	Monel-blue/green
Cadmium-yellow	Nickel-pale green
Cobalt-red	Silver-surface becomes gray/ white
Copper-blue/green	Tin-surface becomes white
Cu-Nickel-blue/green	Zinc-effervescent (boils)

2. Nitric acid diluted 50-50 with water: Apply one drop

REACTION:

Hastelloy "B"-deep blue	Magnesium Steel-brown
Iron-brown or black	Pure Moly-brown

3. Nitric and Hydrochloric: Apply <u>one</u> drop of each acid.

REACTION:

Cobalt base-green/blue   Nickel Base-green

1. <u>Fire/Overtemp Effects on Aluminum</u>:

   a. Soften/anneal
   b. Harden/solution heat treat and age
   c. Partial melting (1100^{o} - $1200)^{o}$F
   d. Complete melting
   e. Conductivity change
   f. Tensile strength change
   g. Ductility change

2. <u>Fire/Overtemp Effects on Magnesium</u>:

   a. Same as for aluminum
   b. Plus corrosion
   c. May also burn and leave white ash

3. <u>Fire/Overtemp Effects on Steel</u>:

   a. Soften/tempered/normalized
   b. Harden/heat treat/martensite
   c. Grain boundary carbides
   d. Melting (2600^{o}F)
   e. Decarburized layer

4. <u>Fire/Overtemp Effects on Titanium</u>:

   a. Tensile strength
   b. Fire
   c. No effective method of NDI

TIME TO FAILURE
FOR COMPONENTS EXPOSED TO 2000^{o} F FIRE

Thin skin aluminum panel	Fails less than 30 seconds
Empty aluminum lines	Fails less than 60 seconds
Pressurized fluid lines no flow	Fails less than 5 minutes
Pressurized fluid lines flow	Fails greater than 5 minutes
Electrical wiring harnesses (800^{o}F Pyrolysis insulation)	Fails less than 15 seconds

5.4 GENERALIZED FIRE EVIDENCE TO OBSERVE

Fire resistance is not a property of a particular material, but is a characteristic of a particular system environment. Such distinction in material form is often overlooked in interpreting fire data.

The melting point of sheet aluminum is generally around 1180°F, but a few alloys may melt as low as 980°F. Most aluminum forgings or casting will melt around 1100°- 1200°F.

When aluminum alloys are heated to the melting range, they will wrinkle and pull apart leaving bright cracks and fissures. If heated sufficiently, they will form stalactites and appear like wrinkled bags.

Titanium discolors from tan to light blue, to dark blue, to gray as temperature increases. At approximately 1300°F, an oxide scale is formed which flakes off easily. At approximately 3100°F, titanium will melt. Molten titanium absorbs oxygen from the atmosphere which will continue (self-sustaining) to burn or boil up to 5600°F.

Stainless steel (300 series) discolors starting at 800°-900°F from tan to light blue, to bright blue, to black. When examining this metal, the investigator should check both sides; the side which has the lighter blue was the side opposite the heat source.

Zinc chromate paint primers are tan at 450°F, brown at 500°F, dark brown at 600°F, and black at 700°F.

Cadmium plating begins to discolor at 500°F and bead up at 550°F. Glass cloth fuses at 1200°F. Silicone rubber blisters at 700°F. Neoprene rubber blisters at 500°F. Wire insulation is a good guide to lower temperature ranges, if the material is known, i.e., nylon spaghetti melts at 250°F-250°F.

MELTING POINTS OF COMMON AEROSPACE MATERIALS

Material	Use	Temperature
Rubber	Fuel hose, linings,etc.	400°-500°F
Plastic	Insulators, liners, etc.	450°-F & over
Glass Cloth	Insulators, liners, etc.	Fuses at 1200°F
Neoprene Rubber	Seals,clamp liners, etc.	Blisters at 500°F
Silicone Rubber	Seals, gasket, etc.	Blisters at 700°F
Stainless Steels/High Chrome Alloys	I.G.V.s,cases, supports	2550°-2789°F
Udimet 500	Turbine blades	2350°-2600°F
High Temp Alloy	Turbine wheels or disc	2500°-2600°F
Medium Carbon Steels	Compressor rotor disc	2690°-2790°F
Low Carbon Steels	Turbine casings, etc.	2690°-2790°F
Titanium	Compressor blades, cases supports	3100°-3272°F
Copper	Bearing cages, wires, etc.	approx. 2000°F
Brass	Bearings, bushings, etc.	1600°-2000°F
Aluminum Alloy	Compressor cases, seals, fittings, etc.	1000°-1250°F
Magnesium Alloy	Bearing supports and frames, housings, etc.	1200°-1250°
Paints and Corrosive Coatings	Used on low and medium carbon steels	600°F

Cadmium Plating	Hardware, etc.	Begins to discolor at 500°F, melts and beads up at 550°-600°F

AUTOIGNITION TEMPERATURE OF COMMON AEROSPACE MATERIALS (°F)

Canvas (nontreated)	204
Denatured Alcohol	750
Hydraulic Hose (Buna-N Rubber)	950
Leather	850
Nylon Covered Wire	1000
Glass Matts	950
Lube Oil (Mil-I-7808)	790
Hydraulic Oil (Mil-H-5606)	468
Plywood	790
Rubber Covered Wire	900
Vinyl Covered Wire	900
Rubber Asbestos Material	900
Styrene	914
Teflon	1050

The above values are not absolute; they will vary according to manufacturer.

TEMPERATURE LIMITS FOR OTHER MATERIALS ($^{\circ}$F)

Glaze or Electrical Porcelain	2250
Enamel Flakes or Blisters	1200-1400
Glass Softens	1400-1600
Paraffin Wax Melts	129
Zinc Melts	786
Silver Solder Melts	1165-1450
Silicone Rubber (Considerable Softening at Sustained Service)	425
Neoprene Rubber Blisters	500
Cellulose (Filled Malamine, Heat Distortion)	400
Nylon (Polyamide, Heat Distortion)	300-360
Nylon Spaghetti Melts	250-350
Paper/Phenolic Delamination and Distortion	250
Syrene Elastomer, Distortion at Sustained Service	220
Polystyrene Distortion	210
Plastic PVC, Heat Distortion	185
Melamine/Formaldehyde Distorts	266-400

5.5 LABORATORY EVALUATION

During the course of a particular investigation, the investigator may decide that additional study or testing of a specific part or item may be necessary or desirable. A wide range of laboratory facilities is available.

At the present time, three government agencies are available to perform test work on failed aircraft parts. Wood parts are tested by the Forest Products Laboratory. Metallic parts are tested by the NTSB Metallurgical Laboratory or National Bureau of Standards (NBS). Most of the chemical analyses are performed by the Federal Bureau of Investigation Laboratory, although the NBS also does some of this type of testing. On certain occasions, tests are conducted at the manufacturer's plant or independent laboratory under the investigator's supervision and control. All of the test work performed by other government agencies is paid for by a transfer of funds to the particular agency. For this reason, the investigator should evaluate the importance of the information to be gained from testing and the relationship of this information to the determination of the probable cause of the failure.

The various types of tests that can be conducted are too numerous to be listed in detail. Some of the more frequently conducted tests are:

a) Tests on metallic parts for evidence of fatigue cracking, poor welding, substandard material properties, poor heat treatment, stress corrosion cracking, inadequate dimensional properties, etc.;

b) Tests on wooden parts for evidence of inadequate glue bond, substandard material properties, moisture content, improper grain slope in splice connectors, etc.;

c) Tests on smears, scores, cuts, etc. to determine the nature of the substance and direction of applied forces, etc.;

d) Tests on fuel and oil to detect presence of foreign substance or nonconformity with standard specifications.

6.0 IN-FLIGHT BREAKUP (IN-FLIGHT SEPARATION)

6.1 SEQUENCE OF FAILURE

When a structural part or component fails in-flight, generally a chain of events is started during which other parts or components fail. In a high speed maneuver, the wing load becomes high positive (up bending) and the balancing tail load becomes high negative (down bending). If the loading is excessive, either the wing will break off in an upward direction, or the tail will break off in a downward direction. These circumstances will cause secondary failures and characteristic responses of the aircraft.

6.1.1 Wing First Sequence

When the wing fails first due to upward static overload, the separated wing will bend up and back over the fuselage. At the same time, the unbalanced lift from the opposite wing causes a rapid roll acceleration with the side of the aircraft missing its wing rotating downward. In some instances, the roll rate is rapid enough to cause a torsional failure in the empennage. The separated wing often impacts the tail surfaces causing matching impact marks and smears between the broken off wing and the leading edges of the tail. The impact with the tail may be severe enough to cause secondary failures in the tail structures (Figure 54a, 54b, and 54c).

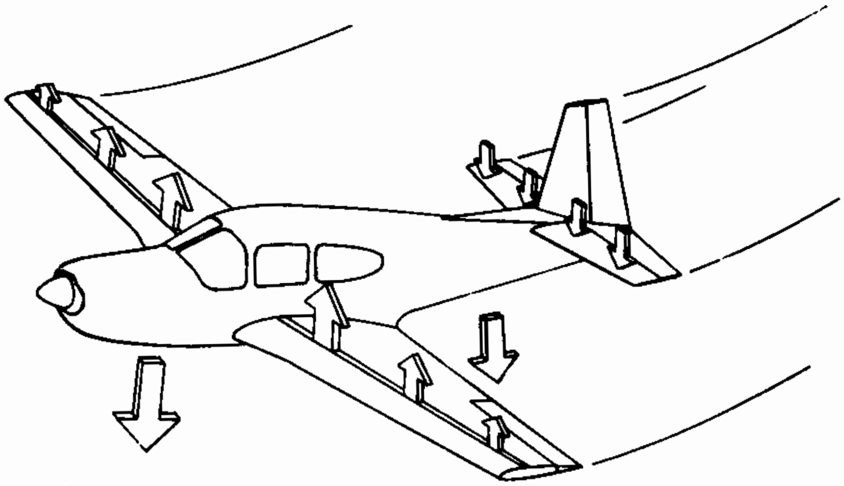


Figure 54a

In a high speed maneuver, the wings are loaded to high positive G and the tail to high negative G.

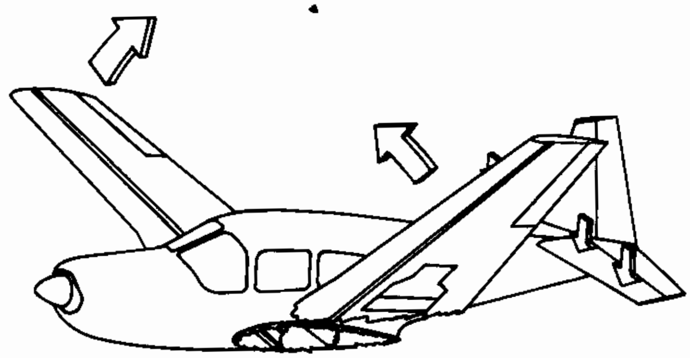


Figure 54b

The wing fails due to excessive upward loading, the aircraft rolls abruptly.

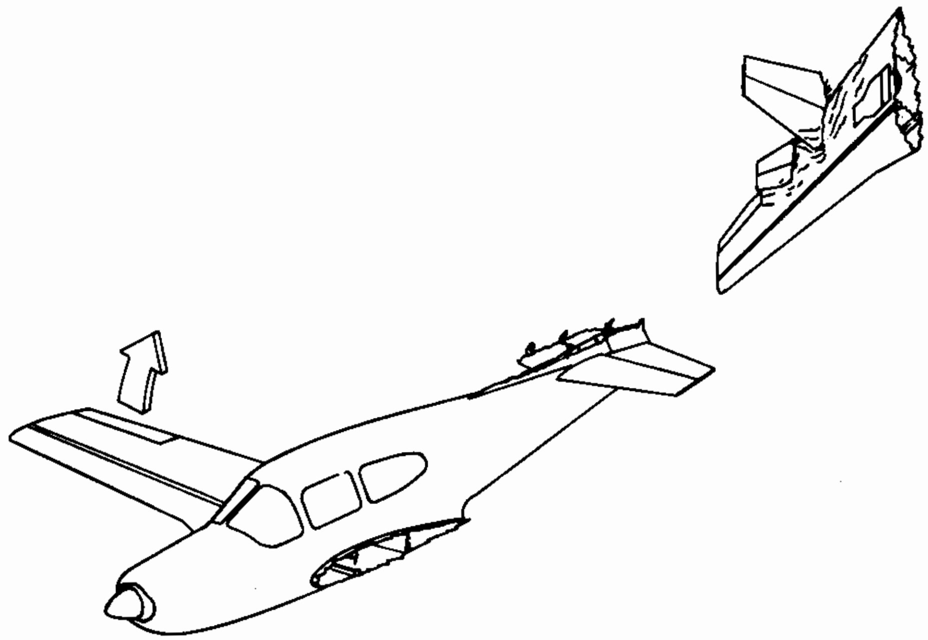


Figure 54c

The separated wing often strikes the tail causing secondary failures and/or smears.

Wing First Characteristics:

If the wing fails first in-flight, it will most often bend upward break off with characteristic instantaneous overload signatures. The rapid rolling may then cause the broken off wing piece to strike the tail leaving mutual indentations and smears. The tail may also then fail as a secondary effect.

6.1.2 Tail First Sequence

In the tail first sequence, the loss of one or both horizontal tail surfaces causes a sudden loss of downward balancing load. The aircraft pitches downward abruptly resulting in a negative angle of attack. The down loading on the wing at high speed results in secondary, negative overloading of the wings (Figure 55a and 55b).

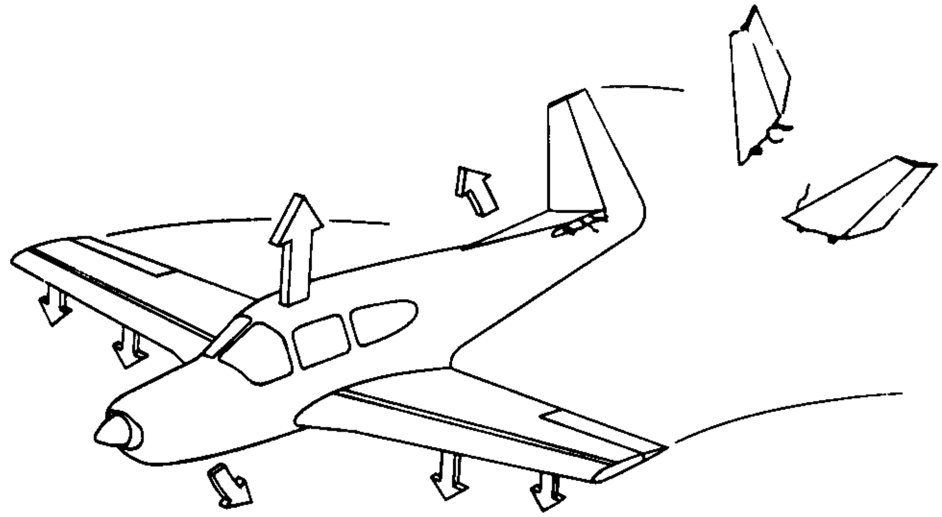


Figure 55a

The high down loading on the tail causes downward tail failure. The loss of tail balancing load causes a rapid pitch-over and down loading on the wing.

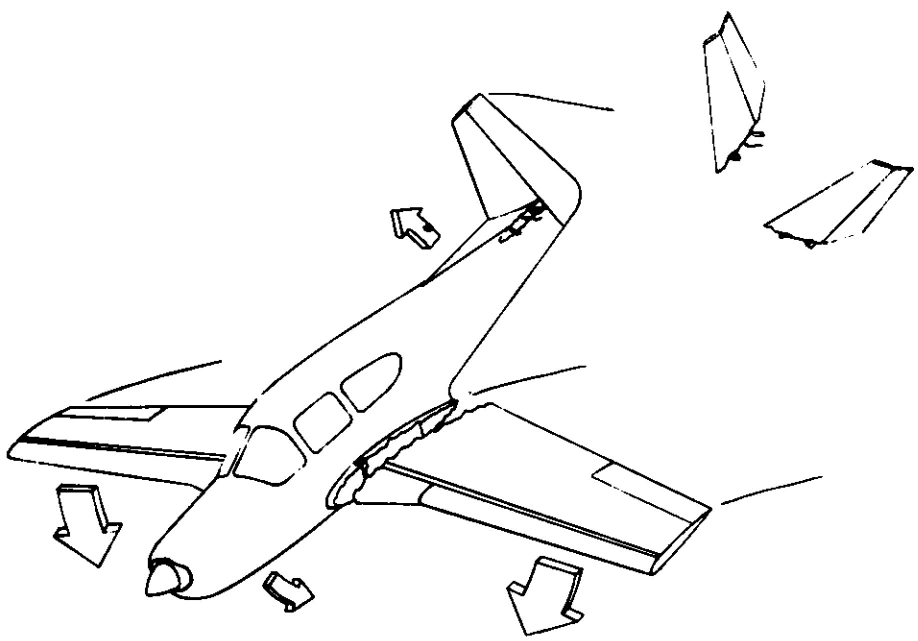


Figure 55b

The excessive down load on the wings causes one (or both) wings to fail by down bending.

BREAKUP SEQUENCE

Tail First Characteristics:

Once aerodynamic forces exceed the strength of the tail, it will usually separate in a downward bending fashion. The spars of the horizontal stabilizers will show permanent evidence characteristic of bending failure. The skin may also develop diagonal buckling if the leading edge rotates during separation. Once the tail is lost, rapid pitch down of the aircraft often results in downward overloading of the wings as a secondary failure. However, the wings may still show effects of prior excessive positive loadings.

In either case, before the first failure occurred, excessive loads were imposed on the airframe. Residual effects of these overload forces can often be found, even after secondary overload separation in the opposite direction. For instance, in the first sequence the wing was first exposed to high positive forces until the tail failed causing a negative overload failure of the wing. Such load "reversals" may be mistakenly identified as flutter. The investigator must also carefully evaluate the cause of load "reversals" found in other secondary pieces that flap in the airstream during the sequential breakup. Such load reversals are not the result of true flutter.

Figures 56a through 56d are illustrations of an in-flight breakup sequence of a King Air resulting from a loss of control and subsequent excessive pilot effort in the attempted recovery. The positive overload first bent the wings upward until the left outer wing panel separated. The unbalanced lift from the remaining right wing created rapid pitch, roll and yaw accelerations which caused the left engine to separate upward and the right engine downward. Wreckage signatures matched the outboard end of the left wing panel to an impact mark near the door. Once the engines were lost, the aft center of gravity caused the empennage to drop and the horizontal tail surfaces to be overloaded and separate upward.

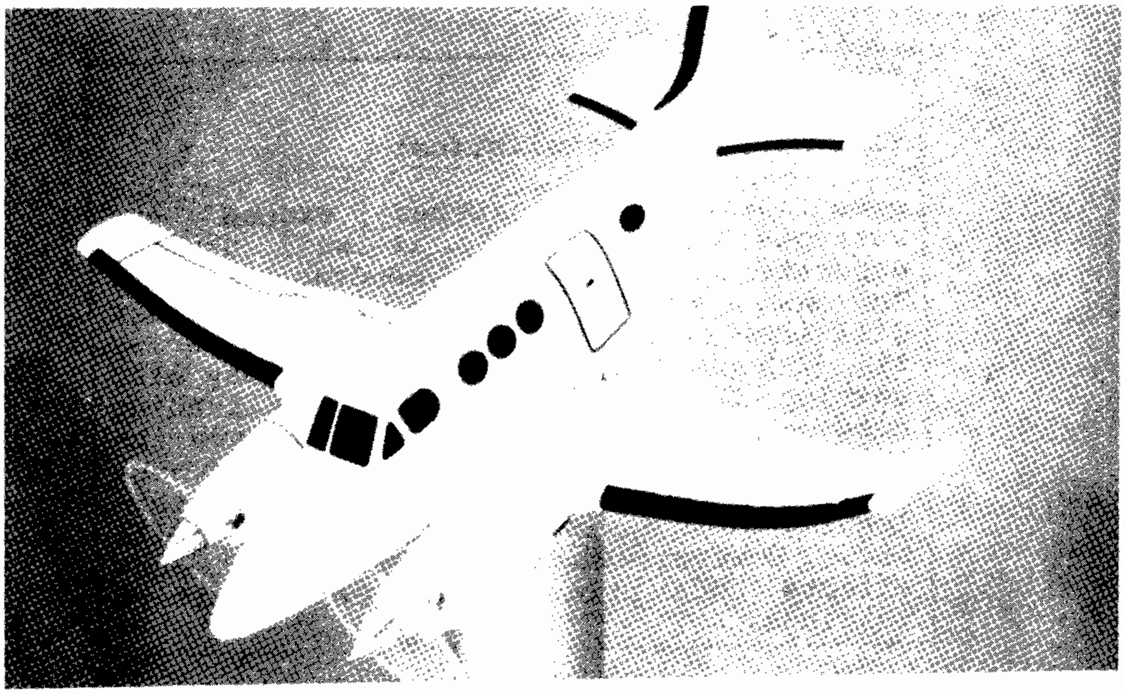


Figure 56a

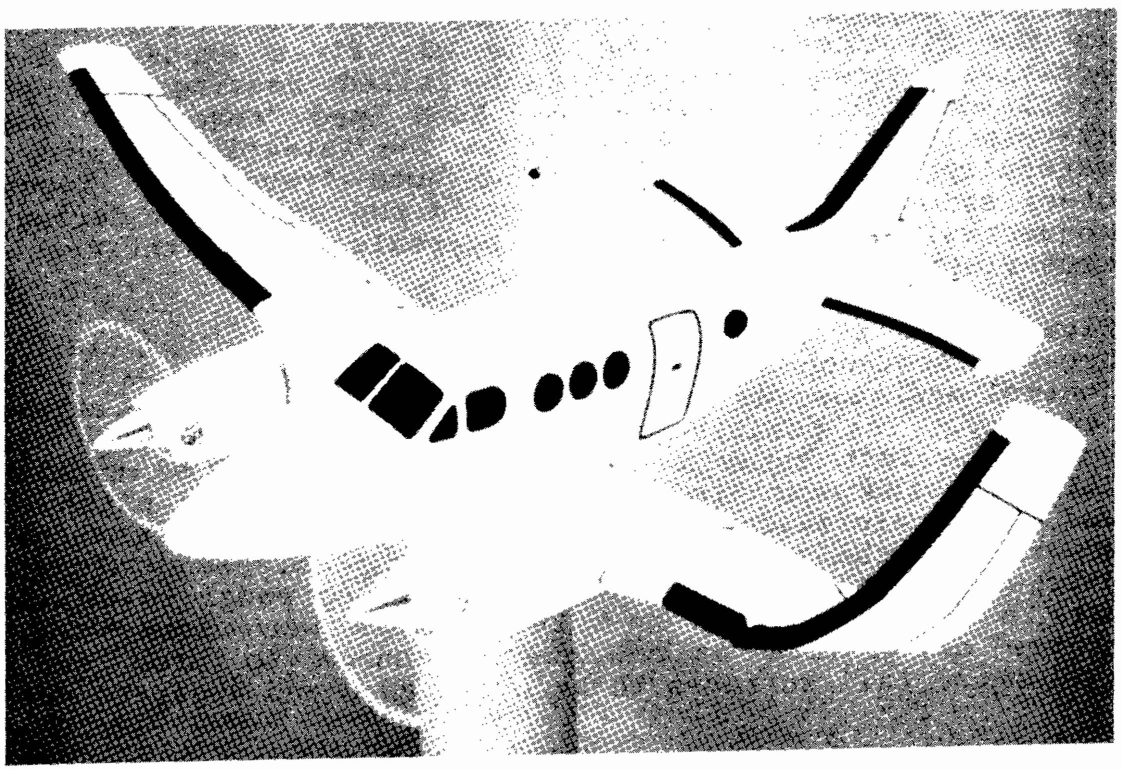


Figure 56b

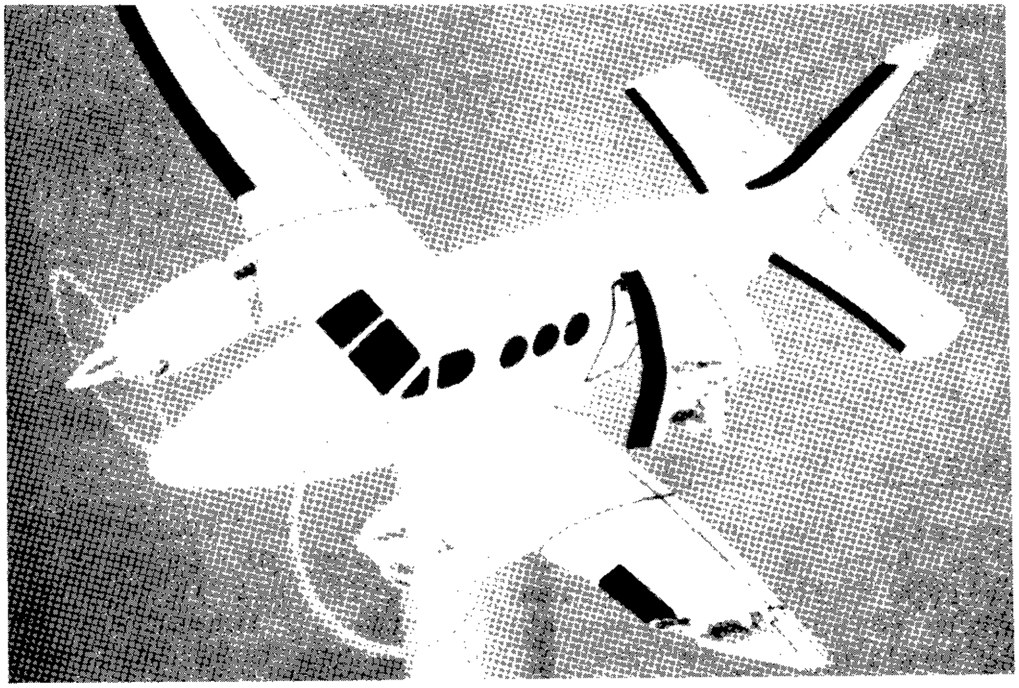


Figure 56c

Figure 56d

6.1.3 Relating to Aircraft Attitude Just Before Accident

If the investigator had followed the procedures outlined it would be possible for example, to determine that the left wing panel had failed in-flight. However, it still remains to be determined why the wing panel failed and if the failure was consistent with the flight attitude at the instant of failure. This kind of determination is necessary in order to rule out the possibility of a design deficiency or to establish the imposition of excessive loads. The investigator must compare the failure loading with known loadings for various flight attitudes to arrive at some indication of the speed of the aircraft and the maneuver being performed at the time of breakup.

6.2 PRIMARY AND SECONDARY FAILURES

In determining the sequence of failure, it is extremely helpful to have a thorough understanding of primary and secondary-type failures. A primary-type failure is one which occurs while adjacent or associated parts are intact and when a loading similar to the design loading has been applied to the failed piece.

When the aircraft or its separated components strike the ground, substantial impact damage usually results. The investigator's task is to first separate the in-flight damage from the ground impact damage. Next, he must search out among the in-flight failures the initial failure. Finally, he must isolate the exact cause for this initial failure.

As the various points are developed, the investigator should constantly integrate the new evidence. If the investigation has been proceeding systematically and if the detailed examination has been performed with thoroughness, definite modes of failure will become evident. It will be found that certain failures must have preceded others for the observed damage to have been made. As the work progresses further, a definite sequence of failure will be established.

A primary-type failure of one of the wing main spars would involve the compression failure of one spar chord and/or buckling of the spar web and/or the tension failure of the other spar chord. A secondary-type of failure is one which occurs when the integrity of adjacent parts has been destroyed by previous failures. In general, the loading producing failures differs from the design loading in type. Thus, if both spar chords of a wing spar are found failed by twisting or bending forces, the failures would be secondary. Some knowledge of the design functions of the various aircraft structural parts is necessary to make determinations of this type. Most secondary failures will be the result of ground impact.

In examining the structure after accident, it is desirable to be able to distinguish between in-flight damage and damage by impact with the ground. A good understanding of the above basic points is extremely useful for making this determination. In examining the various failures, the mode of failure should be compared with the type of failure that would normally be expected for the particular piece. For example, a spar cap in a wing is designed for tension and compression forces.

The basic tool in all of this work, however, is a thorough knowledge of how the structure carries the loads.

6.3 ROTORCRAFT

This section is intended to be general and in no way describes a particular model aircraft. It is suggested that where specific facts relating to a helicopter are in question, the manufacturer's publications and specifications should be consulted.

The design concepts of the modern rotary wing aircraft set it apart from fixed wing aircraft by virtue of the method by which lift and thrust for propulsion are produced. In many models a single main rotor accomplishes both tasks by simultaneously increasing the angle of attack of all main rotor airfoils (thrust) or cyclically changing them to tilt the "Disc", which is formed while the system is rotating at nearly constant RPM. This cyclic action provides directional control in the hover state, roll control and longitudinal pitch control in flight. To add to the complexity of the system, torque effect, induced upon and otherwise untreated by the fuselage must be counteracted by another rotating tail rotor blade assembly while the aircraft is hovering or passing through the translational phase of flight. This translational effect occurs at approximately 16 to 24 knots considering the wind speed and actual velocity of the aircraft are combined. The main rotor system becomes more efficient thereafter. Thus, the term "Rotary" can be extended to encompass a design which incorporates many rotating assemblies. These include systems which produce power, transmit it to a main transmission, drive accessories, such as generators and hydraulic pumps (these rotate also), and turn the main tail rotors. Intermediate to the tail rotor assembly will likely be gearboxes providing gear reduction and/or a change of angle along the drive system.

The greatest common denominator describing component failures in rotorcraft is fatigue. Cyclic stresses which are generated in by each rotation of a system cause fatigue "damage" in the material. As the process continues, the fatigue damage accumulates due to repeated cyclic stressing. Some of the high strength steels used are highly "notch" sensitive. This notch sensitivity can be described by an example. A mechanic inadvertently creates a very small tool mark (stress raiser) on a component while accomplishing a maintenance task. The tool mark becomes a notch at which a fatigue crack initiates. This crack continues to propagate under the application of cyclic stresses. The fatigue crack grows cyclically until either the crack reaches the critical crack size for the material, or the remaining ligament is subjected to the ultimate tensile strength of the material. Tensile residual stresses which are not relieved during manufacturing can further reduce fatigue lives.

If we consider the mass rotating as represented by the main rotor system, it may be easy to understand extreme stresses which are reacted by the root section of each airfoil and its attachment assembly. To lend credance to this concept, consider the tip cap of one main rotor system attached to the outermost end of the blade and weighing a few ounces at static, but can be calculated to exceed 84 pounds at operating RPM. One can deduce from this there is a greater likelihood of the separation of most or an entire blade from the system as opposed to an outboard section unless impact with a fixed object should play a role.

Loss of a blade or section thereof very often results in total collapse of a rotor system due to imbalance, although history has shown aircraft were successfully landed with part of a blade separated. Where the mode of rotor failure is questionable to the investigator, it is suggested a materials engineer should be consulted to help determine the exact cause of the separation. This manual provides ample reference by which the investigator can conduct the investigation.

A long moment arm for anti-torque reaction and a reduction of component size and weight are facilitated by mounting the tail rotor gearbox and assembly as close to the aft end of the tailboom as possible. This, however, causes a rather complex, lightweight drive system to be present. Each section of the drive shaft, the couplings between each gearbox and certain tail rotor components undergo tremendous torsional stress.

Over stress can be seen as buckling and permanent deformation to the shaft sections. Not surprising is a tail rotor strike that will often create some of the same signatures. The bearings on the shafting support systems can become worn, begin to deteriorate, and ultimately fail. Gearbox designs usually allow for the rotation of the gears to sling oil for lubrication system flow. Lack of lubricating oil may be a suspect item when a gearbox has suffered case failure. Again, it would be advisable to consult a materials engineer where a failure is questionable.

Gas turbine engines are used in many rotorcraft. They have been proven as a viable power source in helicopters by thousands of hours of reliable operation. Yet, there is an inherent design requirement which causes great temperature change within the hot section of the engine. Extreme temperature change is another method by which materials can be fatigued. For instance, turbine wheels are generally designed with high strength cast alloy steels. One material microstructure is very stable below temperatures up to approximately 1700°F. Thereafter, the process of resolution of the grains comprising the steel may begin. Resolution weakens the microstructure and can cause what is commonly referred to as stress rupture failure. One reason for establishing a gas temperature limit and acceptable transitory periods (in terms of seconds) above the limit is to preclude changes in the microstructure of the materials forming combustors, turbine blades and nozzles.

Flight control components in rotorcraft are composed of push-pull tubing, clevises swaged or riveted to their ends, bellcranks and in many design, hydraulic servos to reduce pilot effort while flying the machine. Aircraft quality braided cable is also used. These components undergo extensive compressive/tension cycles as the operator selects a myriad of control positions. Some experiencing shear loads, bolts attaching clevises, for instance. To compound design difficulty, at some point in the main and tail rotor control groups, push-pull motion must incorporate a device which follows rotary motion; yet "tilts" to allow the airfoils to cyclically increase and decrease angle of attack or simultaneously accomplish blade angle pitch change. These are called swash plates or gimbal rings. This suggests a complex system and complex forces acting upon each component. To give an example, the pitch change links attaching to the main rotor swash plate and each blade experience loads from compression/tension and centrifugal forces. One can readily deduce analyzing a component failure here could be extensive. The subject most perplexing to accurate investigation of control failures is relating them to impact or distinguishing them from preimpact separations.

The final subject area which will be discussed regarding rotorcraft is the fuselage of the aircraft. In many respects, monocouque designs may be the most susceptible to cyclic stresses induced by vibration. The fuselage has the greatest capability for flexure, but hardened areas will react to cyclic stress from dynamic vibration by fatiguing. Sometimes this occurs at an alarming rate. Most materials used to form fuselage structures are aluminum alloys, although an increasing element of fuselage design technology centers on research and development of composite materials. Other sections of this manual explain in more depth the applications and characteristics of composites. At present, the most common composite structures on rotorcraft are main and tail rotor airfoils and cabin structure.

We have briefly discussed some of the potential detrimental effects of fatigue as the subject relates to rotorcraft. This is by no means the only consideration when approaching a component failure, nor is the subject completely unique to helicopters. The other sections of this manual present valuable information to the investigator in recognizing failure modes applicable to rotorcraft and should be an aid in the solution of failure analyses.

CLOSING REMARKS

The investigator is encouraged to review this material occasionally. It is a continuing challenge to sharpen investigative skills. The more familiar the investigator becomes with the principles of metallurgy, design, loads, fracture and investigative techniques, the better prepared he or she will be to properly conduct complex investigations.

The basic purpose of the classroom lecture and this text is to help investigators and inspectors perform better investigations and author better reports. The technical language used in reports and safety recommendations is very important and must be precise to avoid confusion and improper actions of others. This text also goes beyond a simple "cookbook" approach. Information has been organized and presented to assist the student in seeing the interrelationships between materials, design, maintenance and flight safety. A basic understanding of these technologies is expected to make qualitative improvements to aviation safety.

This text only covers the highlights of each topic and while the information is intended to be generally useful, not all circumstances can be covered. In-depth expertise, developed with study and experience, will also teach the investigator the important exceptions.

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The following references contain helpful general technical information that can be obtained through many technical libraries such as local colleges.

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INDEX OF TERMS

APPENDIX I

Periodic Table of The Elements

In the periodic table the elements are arranged in order of increasing atomic number. Vertical columns headed by Arabic numerals are called *Groups*. A horizontal sequence of elements is called a *Period*. The most active elements are at the top right and bottom left of the table. The staggered line (Groups 13-17) roughly separates metallic from non-metallic elements.

Groups—Elements within a group have similar properties and contain the same number of electrons in their outside energy shell.

—The first group (1) includes hydrogen and the alkali metals.

—The last (18) contains the *inert gases*.

—Group 17 includes the *halogens*.

—The elements intervening between groups 2 and 13 are called *transition elements*.

—Short vertical columns without Arabic numeral headings are called subgroups.

Periods—in a given period the properties of the elements gradually pass from a strong metallic to a strong non-metallic nature, with the last number of a period being an inert gas.

LIGHT METALS (Groups 1–2) — NON METALS (Groups 13–17)

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
Hydrogen 1.0080 H 1																	Helium 4.003 He 2
Lithium 6.939 Li 3	Beryllium 9.012 Be 4											Boron 10.811 B 5	Carbon 12.01115 C 6	Nitrogen 14.007 N 7	Oxygen 15.999 O 8	Fluorine 18.998 F 9	Neon 20.183 Ne 10
Sodium 22.990 Na 11	Magnesium 24.312 Mg 12											Aluminum 26.981 Al 13	Silicon 28.086 Si 14	Phosphorus 30.974 P 15	Sulfur 32.064 S 16	Chlorine 35.453 Cl 17	Argon 39.948 Ar 18
Potassium 39.102 K 19	Calcium 40.08 Ca 20	Scandium 44.956 Sc 21	Titanium 47.90 Ti 22	Vanadium 50.942 V 23	Chromium 51.996 Cr 24	Manganese 54.938 Mn 25	Iron 55.847 Fe 26	Cobalt 58.933 Co 27	Nickel 58.71 Ni 28	Copper 63.54 Cu 29	Zinc 65.37 Zn 30	Gallium 69.72 Ga 31	Germanium 72.59 Ge 32	Arsenic 74.922 As 33	Selenium 78.96 Se 34	Bromine 79.909 Br 35	Krypton 83.80 Kr 36
Rubidium 85.47 Rb 37	Strontium 87.62 Sr 38	Yttrium 88.905 Y 39	Zirconium 91.22 Zr 40	Niobium 92.906 Nb 41	Molybdenum 95.94 Mo 42	Technetium (99) Tc 43	Ruthenium 101.07 Ru 44	Rhodium 102.91 Rh 45	Palladium 106.4 Pd 46	Silver 107.87 Ag 47	Cadmium 112.40 Cd 48	Indium 114.82 In 49	Tin 118.69 Sn 50	Antimony 121.75 Sb 51	Tellurium 127.60 Te 52	Iodine 126.90 I 53	Xenon 131.30 Xe 54
Cesium 132.90 Cs 55	Barium 137.34 Ba 56	Lanthanum 138.91 La 57	Hafnium 178.49 Hf 72	Tantalum 180.95 Ta 73	Tungsten 183.85 W 74	Rhenium 186.21 Re 75	Osmium 190.2 Os 76	Iridium 192.2 Ir 77	Platinum 195.09 Pt 78	Gold 196.97 Au 79	Mercury 200.59 Hg 80	Thallium 204.37 Tl 81	Lead 207.19 Pb 82	Bismuth 208.98 Bi 83	Polonium (210) Po 84	Astatine (210) At 85	Radon (222) Rn 86
Francium 223 Fr 87	Radium (226) Ra 88	Actinium 227 Ac 89	Unnilquadium 261 Unq 104	Unnilpentium 262 Unp 105	Unnilhexium 263 Unh 106	Uns 107	Uno 108	Une 109	110	111	112	113	114	115	116	117	118

Cerium 140.12 Ce 58	Praseodymium 140.91 Pr 59	Neodymium 144.24 Nd 60	Promethium (147) Pm 61	Samarium 150.35 Sm 62	Europium 151.96 Eu 63	Gadolinium 157.25 Gd 64	Terbium 158.92 Tb 65	Dysprosium 162.50 Dy 66	Holmium 164.93 Ho 67	Erbium 167.26 Er 68	Thulium 168.93 Tm 69	Ytterbium 173.04 Yb 70	Lutetium 174.97 Lu 71
Thorium 232.04 Th 90	Protactinium (231) Pa 91	Uranium 238.03 U 92	Neptunium (237) Np 93	Plutonium (242) Pu 94	Americium (243) Am 95	Curium (247) Cm 96	Berkelium (249) Bk 97	Californium (251) Cf 98	Einsteinium (254) Es 99	Fermium (253) Fm 100	Mendelevium (256) Md 101	Nobelium (254) No 102	Lawrencium (260) Lr 103

Key

Information	Color
Name of Element	Red
Atomic Weight	Yellow
Atomic Symbol	Black
Atomic Number	Green

chemistry

NAVY

Mechanical Equivalent of Heat is the work required to produce a unit quantity of heat.

0.427 kilogram-meter (kg-m) = 1 calorie

$$\frac{\text{Work}}{\text{mechanical equivalent of heat}} = \text{heat}$$

VALENCES SHOWN BY COMMON RADICALS

Name	Symbol	Valence
Acetate	$C_2H_3O_2$	—1
Bicarbonate	HCO_3	—1
Bisulfate	HSO_4	—1
Carbonate	CO_3	—2
Chlorate	ClO_3	—1
Chromate	CrO_4	—2
Ferricyanide	$Fe(CN)_6$	—3
Ferrocyanide	$Fe(CN)_6$	—4
Hypochlorite	ClO	—1
Nitrate	NO_3	—1
Nitrite	NO_2	—1
Permanganate	MnO_4	—1
Phosphate	PO_4	—3
Sulphate	SO_4	—2
Sulphite	SO_3	—2

EXPONENTIAL NUMBERS

Multiplication by a positive power of 10 corresponds to moving the decimal point to the right; multiplication by a negative power of 10 corresponds to moving the decimal point to the left.

1.33×10^4 is 13,300

1.33×10^{-4} is 0.000133

Numbers expressed with powers of 10 cannot be added or subtracted directly unless the powers of 10 are the same.

$1.23 \times 10^4 + 1.23 \times 10^5 =$
$1.23 \times 10^4 + 12.3 \times 10^4 = 13.5 \times 10^4$

$1.23 \times 10^{-4} - 1.23 \times 10^{-5} =$
$1.23 \times 10^{-4} - 0.123 \times 10^{-4} = 1.11 \times 10^{-4}$

When the powers of 10 are multiplied, exponents are added; when divided, exponents are subtracted.

$(1.23 \times 10^4) \times (1.23 \times 10^5) =$
$(1.23 \times 1.23) \times (10^4 \times 10^5) = 1.51 \times 10^9$

$$\frac{1.23 \times 10^{-4}}{1.23 \times 10^{-5}} = \frac{1.23}{1.23} \times \frac{10^{-4}}{10^{-5}} = 1.00 \times 10$$

SOME ACID—BASE INDICATORS

Indicator	Color Change Acid	Basic
Alizarin Yellow	Yellow	Red
Bromocresol Green	Yellow	Blue
Litmus	Red	Blue
Methyl Orange	Red	Yellow
Methyl Red	Red	Yellow
Phenolphthalein	Colorless	Red
Thymol Blue	Red	Yellow

PRESSURES AND DENSITIES

$$\textbf{Pressure} = \frac{\text{Force}}{\text{Area}}$$

1 column of water 1 foot deep = 62.4 pounds per square foot, or 0.433 pounds per square inch. 1 column of water 1 centimeter deep = 1 gram per square centimeter.

Specific Gravity = number of times a substance is as heavy as an equal body of water, or Specific gravity (liquid) =

$$\frac{\text{Weight of Liquid}}{\text{weight of equal volume of water}}$$

$$\textbf{Density} = \frac{\text{Weight}}{\text{Volume}}$$

Pressure = depth × density, or force per unit area. An increase in pressure is transmitted equally through the liquid.

Specific Gravity (Solid) =

$$\frac{\text{Weight of Body}}{\text{loss of weight in water}}$$

OR Specific Gravity (Solid)

$$\frac{\text{Weight of Body}}{\text{weight of equal volume of water}}$$

One cubic yard of air weighs about 2 pounds. Atmospheric pressure at sea level = about 15 pounds per square inch.

DEFINITIONS: CONCENTRATION OF SOLUTIONS

1) Mole Fraction: The number of moles of solute per unit total moles of solution.

2) Molarity: The number of moles of solute per liter of solution.

3) Molality: The number of moles of solute per 1,000 g. of solvent.

4) Formality: The number of formula weights of solute per liter of solution.

5) Normality: The number of equivalents of solute per liter of solution.

BASIC LAWS AND TABLES

UNITS

Kilo—means one thousand
Centi—means one-hundredth
Milli—means one-thousandth
Micro—means one-millionth

1 Kilometer (km) = 1,000 meters = 0.621 mile

1 Meter (m) = 100 centimeters = 39.4 inches

1 Centimeter (cm) = 10 millimeters (mm) = 0.394 inches

1 Kilogram (kg) = 1,000 grams = 2.20 pounds

1 Gram (g) = 1,000 milligrams (mg) = 0.0353 ounce.

1 Liter (l) = 1,000 milliliters = 1.06 quarts

1 Milliliter (ml) = 1.000027 cubic centimeters (cc)

1 Atomic Mass Unit = 1.66×10^{-24}g

Avogadro Number = 6.0235×10^{23}

TEMPERATURE MEASUREMENTS

In scientific work, the Centigrade or Celsius (°C) and Kelvin (K) scales are most commonly used. The Kelvin scale is an absolute temperature scale, in which zero degrees ideally represents the lowest attainable temperature.

Comparison of Various Temperature Scales

(°F) Fahrenheit	(K) Kelvin	(°C) Centigrade
212°	373°	100°
32°	273°	0°
−459°	0°	−273°

Fahrenheit—Centigrade: Conversions

$C = 5/9\,(F - 32)$
$F = 9/5C + 32$

Boyle's Law:

$p_1 \times V_1 = p_2 \times V_2$ at constant temperature. Zero degrees Kelvin is the lowest possible temp.

In Kelvin absolute temperature scale: water boils at 373K, freezes at 273K.

Charles' Law: $\frac{V_1}{V_2} = \frac{T_1}{T_2}$ at constant pressure.

Combination of Charles' and Boyle's Laws:

$$\frac{V_1p_1}{T_1} = \frac{V_2p_2}{T_2}$$

When heated through one degree Centigrade, any gas expands 1/273 of its volume at 0 degrees Centigrade if the pressure remains constant.

One BTU is the heat required to raise the temperature of 1 pound of water through 1 degree Fahrenheit.

One Calorie is the heat required to raise the temperature of 1 gram of water through 1 degree Centigrade.

Specific Heat: heat required to raise the temperature of a unit mass of that substance through 1 degree. If q is total heat and m is mass, $q = m \times s \times (t_2 - t_1)$.

Heat of melting, or heat of fusion, L, is the quantity of heat needed to melt one unit weight of a substance without changing its temperature, or $q = m \times L$

80 calories of heat is required to melt 1 gram of ice without raising its temperature.

Boiling Point of Liquid: that temperature at which the vapor pressure is equal to the pressure above the liquid.

efixes and symbols to form decimal multiples and/or submultiples.

Power of Ten	E Notation	Decimal Equivalent	Prefix	Phonic	Symbol
10^{12}	E+12	1 000 000 000 000	tera	ter'a	T
10^{9}	E+09	1 000 000 000	giga	ji'ga	G
10^{6}	E+06	1 000 000	mega	meg'a	M
10^{3}	E+03	1 000	kilo	kil'o	k
10^{2}	E+02	100	hecto	hek'to	h
10	E+01	10	deka	dek'a	da
10^{-1}	E−01	0.1	deci	des'i	d
10^{-2}	E−02	0.01	centi	sen'ti	c
10^{-3}	E−03	0.001	milli	mil'i	m
10^{-6}	E−06	0.000 001	micro	mi'kro	μ
10^{-9}	E−09	0.000 000 001	nano	nan'o	n
10^{-12}	E−12	0.000 000 000 001	pico	pe'ko	p
10^{-15}	E−15	0.000 000 000 000 001	femto	fem'to	f
10^{-18}	E−18	0.000 000 000 000 000 001	atto	at'to	a

Coefficient of Friction: $\mu = \frac{F}{N}$
μ = coefficient of friction
F = force of friction N = force normal to surface

Velocity: $v_{av} = \frac{d}{t}$
v_{av} = average velocity
d = distance traveled t = elapsed time

Acceleration: $a_{av} = \frac{v_f - v_i}{t}$
a_{av} = average acceleration v_i = initial velocity
v_f = final velocity t = elapsed time

ewton's 2nd Law of Motion: $F = m \cdot a$
= force m = mass a = acceleration

w of Universal Gravitation: $F = G\frac{m_1 \cdot m_2}{d^2}$
F = force of attraction $m_1 \cdot m_2$ = product of masses
G = gravitational constant d = distance between their centers

Centripetal Force: $F = \frac{m \cdot v^2}{r}$
F = centripetal force
m = mass v = velocity r = radius of path

Pendulum: $T = 2\pi\sqrt{\frac{l}{g}}$
T = period l = length g = acceleration of gravity

Work: $W = F \cdot d$
W = work F = force d = distance

Ideal Mechanical Advantage: IMA = d_A/d_L
d_A = distance through which applied force Facts
d_L = distance load moves against force W without friction
Actual Mechanical Advantage: AMA = W/F

Mechanical Equivalent of Heat: $W = J \cdot Q$
W = work J = mechanical equivalent of heat Q = heat

Kinetic Energy: $K = \frac{1}{2} m \cdot v^2$
K = kinetic energy m = mass v = velocity

Potential Energy: $V = m \cdot g \cdot h$
V = potential energy g = acceleration of gravity
m = mass h = vertical distance (height)

Relationship between Mass and Energy: $E = m \cdot c^2$
E = energy m = mass c = velocity of light

Wave Formula: $v = f \cdot \lambda$
v = wave speed f = frequency λ = wave length

Uniformly Illuminated Surface: $E = \frac{\Phi}{A}$
E = illumination
Φ = luminous flux A = uniformly illuminated area

Images in Mirrors and Lenses: $\frac{S_o}{S_i} = \frac{D_o}{D_i}$
S_o = object size D_o = object distance
S_i = image size D_i = image distance

Focal Length of Mirrors and Lenses: $\frac{1}{f} = \frac{1}{D_o} + \frac{1}{D_i}$
f = focal length
D_o = object distance D_i = image distance

Snell's Law: $n_i \sin \Theta_i = n_j \sin \Theta_j$
n_i = refractive index of ith material
Θ_i = angle between ray and normal to surface

Math/Physics

Electric Current: $I = \frac{q}{t}$
I = current q = quantity of charge t = time

Coulomb's Law of Electrostatics: $F = k\frac{q_1 \cdot q_2}{d^2}$
F = force between two charges
k = proportionality constant
$q_1 \cdot q_2$ = product of charges
d = distance separating charges

Capacitance of a Capacitor: $C = \frac{q}{V}$
C = capacitance of a capacitor
V = potential difference between plates
q = charge on either plate

Ohm's Law of Resistance: $E = I \cdot R$
E = emf of source I = current in the circuit
R = resistance of the circuit

Joule's Law: $Q = I^2 \cdot R \cdot t$
Q = heat energy I = current R = resistance t = time

Faraday's Law of Electrolysis: $m = z \cdot I \cdot t$
m = mass z = electrochemical equivalent
I = current t = time

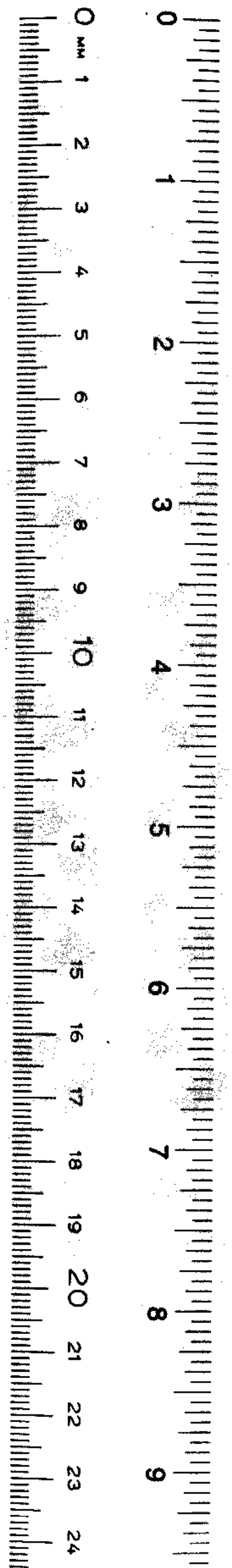
Induced emf: Coil in a Magnetic Field: $E = -N\frac{\Delta\Phi}{\Delta t}$
E = induced emf N = number of turns
$\Delta\Phi/\Delta t$ = the change in flux linkage in a given interval of time

Induced emf: Conductor in a Magnetic Field: $E = B \cdot l \cdot v$
E = induced emf B = flux density of the magnetic field
l = length of conductor
v = velocity of conductor across magnetic field

Instantaneous Voltage: $e = E_{max} \sin \Theta$
e = instantaneous voltage
E_{max} = maximum voltage
Θ = angle between the plane of the conducting loop and the perpendicular to the magnetic flux (displacement angle)

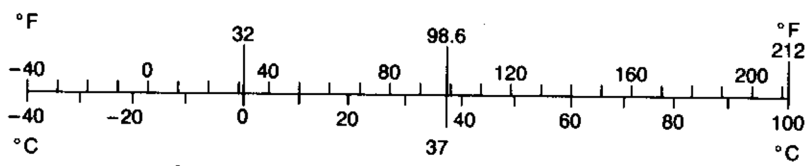
Instantaneous Current: $i = I_{max} \sin \Theta$
i = instantaneous current
I_{max} = maximum current
Θ = displacement angle

METRIC CONVERSION FACTORS (Approximate)



Conversions TO Metric Measures

	Symbol	When You Know	Multiply By	To Find	Symbol
LENGTH	in	inches	2.54	centimeters	cm
	ft	feet	30.48	centimeters	cm
	yd	yards	0.9	meters	m
	mi	miles	1.6	kilometers	km
AREA	in^2	square inches	6.5	square centimeters	cm^2
	ft^2	square feet	0.09	square meters	m^2
	yd^2	square yards	0.8	square meters	m^2
	mi^2	square miles	2.6	square kilometers	km^2
		acres	0.4	hectares	ha
MASS (Weight)	oz	ounces	28	grams	g
	lb	pounds	0.45	kilograms	kg
		short tons (2000 lb)	0.9	tonnes	t
VOLUME	tsp	teaspoons	5	milliliters	ml
	Tbsp	tablespoons	15	milliliters	ml
	fl oz	fluid ounces	30	milliliters	ml
	c	cups	0.24	liters	l
	pt	pints	0.47	liters	l
	qt	quarts	0.95	liters	l
	gal	gallons	3.8	liters	l
	ft^3	cubic feet	0.03	cubic meters	m^3
	yd^3	cubic yards	0.76	cubic meters	m^3
TEMP. $°C=\frac{5}{9}(°F-32)$	°F	Fahrenheit temperature	5/9 (after subtracting 32)	Celsius temperature	°C



Conversions FROM Metric Measures

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

711-0642 **(170)**

FAILURE MODE

INSTANTANEOUS (OVERLOAD)

EXAM \ MODE	DUCTILE	BRITTLE
VISUAL UP TO 10X	• DISTORTION PRIOR TO SEPARATION • NECKING • DULL, FIBROUS FRACTURE • SHEAR LIPS	• VERY LITTLE DEFORMATION • FLAT FRACTURE • SMALL SHEAR LIPS • ORIGIN, FAN MARK • CHEVRONS • NOTCH EFFECT • IMPACT EFFECT
MICROSCOPIC UP TO 1000X	• GRAINS STRETCHED ALONG FRACTURE PLANE • GRAINS LOCALLY WORK HARDENED • FRACTURE ACROSS THE GRAINS	• GRAINS NOT DEFORMED • FRACTURE INTERGRANULAR OR CLEAVAGE
SCANNING ELECTRON MICROSCOPE UP TO 20,000X	• FRACTURE SURFACE USUALLY SHOWS "DIMPLES" LESS THAN 1/10,000 INCH ACROSS	• INTERGRANULAR FRACTURE APPEARS LIKE ROCK CANDY • CLEAVAGE FRACTURE APPEARS LIKE NUMEROUS FLAT TILTED PLANES
CONFIRMING TESTS	• TENSILE TEST OR HARDNESS TEST FOR PROPER HEAT TREATMENT AND STRENGTH	• CHARPY IMPACT OR FRACTURE TOUGHNESS TEST FOR EXTREME BRITTLENESS

E IDENTIFICATION CHART

APPE

PROGRESSIVE (DEFECT OR WEAKNESS PRIOR TO F

FATIGUE	CORROSION	WEAR
• TWO ZONE FRACTURE • BEACHMARKS • ORIGIN • RATCHET MARKS • HIGH STRESS/LOW CYCLE • LOW STRESS/HIGH CYCLE	• GENERAL SURFACE DAMAGE • INTERGRANULAR - EXFOLIATION OF Al ALLOYS - SENSITIZATION OF STAINLESS STEEL • STRESS CORROSION - INTERGRANULAR CRACKING OF Al ALLOYS, GRAIN EFFECT - CHLORIDE CRACKING OF STAINLESS STEEL - HYDROGEN EMBRITTLEMENT OF HIGH STRENGTH STEEL	• ABRASIVE WEAR - SCRATCHING - GOUGING • ADHESIVE WEAR - BREAK-IN - SCORING - GALLING - SEIZURE
• FATIGUE PART OF FRACTURE IS STRAIGHT ACROSS THE GRAINS • POSSIBLE "DEFECT" AT ORIGIN	• INTERGRANULAR VERSUS TRANSGRANULAR PATH • EFFECT OF SURFACE ATTACK, COATINGS, ETC.	• NATURE OF EMBEDDED PARTICLES • AMOUNT OF METAL TRANSFER • AMOUNT OF METAL WORN AWAY
• FATIGUE STRIATIONS OFTEN SEEN AT 1000X, APPEAR AS PARALLEL WAVY LINES • NATURE OF ORIGIN CAN BE EXAMINED IN DETAIL	• INTERGRANULAR VERSUS TRANSGRANULAR PATH • NATURE OF ATTACK TO SURFACE • CHEMICAL ANALYSIS BY EDS	• NATURE OF EMBEDDED PARTICLES • SURFACE APPEARANCE AT HIGH MAGNIFICATION • CHEMICAL ANALYSIS BY EDS
• CYCLIC LOAD TESTS • CHECK PART GEOMETRY FOR PROPER SURFACE CONDITION • INVESTIGATE SERVICE HISTORY	• ENVIRONMENTAL EXPOSURE TESTS WITH OR WITHOUT STRESS • CHEMICAL ANALYSIS OF FILMS OR DEPOSITS • CHECK COATINGS AND ENVIRONMENTAL EXPOSURE AND DRAINS	• NUMEROUS LAB WEAR TESTS ARE POSSIBLE • CHEMICAL ANALYSIS OF OIL, WEAR DEBRIS, OR ABRASIVES • POSSIBLE LUBE FAILURE OR OIL STARVATION, FILTERS

ON CHART

APPENDIX II

(DEFECT OR WEAKNESS PRIOR TO FAILURE)

CORROSION	WEAR	CREEP
GENERAL SURFACE DAMAGE INTERGRANULAR - EXFOLIATION OF Al ALLOYS - SENSITIZATION OF STAINLESS STEEL STRESS CORROSION - INTERGRANULAR CRACKING OF Al ALLOYS, GRAIN EFFECT - CHLORIDE CRACKING OF STAINLESS STEEL - HYDROGEN EMBRITTLEMENT OF HIGH STRENGTH STEEL	• ABRASIVE WEAR - SCRATCHING - GOUGING • ADHESIVE WEAR - BREAK-IN - SCORING - GALLING - SEIZURE	• SHORT-TERM - GROSS OVER HEATING - "TAFFY PULL" • LONG-TERM - STRESS RUPTURE - OXIDIZED FISSURES - BRITTLE CRACKS - OVERLOAD ZONE NORMALLY DUCTILE - ENLARGED SIZE
• INTERGRANULAR VERSUS TRANSGRANULAR PATH • EFFECT OF SURFACE ATTACK, COATINGS, ETC.	• NATURE OF EMBEDDED PARTICLES • AMOUNT OF METAL TRANSFER • AMOUNT OF METAL WORN AWAY	• CREEP FISSURES INTERGRANULAR • GRAIN BOUNDARY BUBBLES AND CRACKS
• INTERGRANULAR VERSUS TRANSGRANULAR PATH • NATURE OF ATTACK TO SURFACE • CHEMICAL ANALYSIS BY EDS	• NATURE OF EMBEDDED PARTICLES • SURFACE APPEARANCE AT HIGH MAGNIFICATION • CHEMICAL ANALYSIS BY EDS	• SURFACE DETAILS AT HIGH MAGNIFICATION • NATURE OF FILMS ON CREEP FISSURES
• ENVIRONMENTAL EXPOSURE TESTS WITH OR WITHOUT STRESS • CHEMICAL ANALYSIS OF FILMS OR DEPOSITS • CHECK COATINGS AND ENVIRONMENTAL EXPOSURE AND DRAINS	• NUMEROUS LAB WEAR TESTS ARE POSSIBLE • CHEMICAL ANALYSIS OF OIL, WEAR DEBRIS, OR ABRASIVES • POSSIBLE LUBE FAILURE OR OIL STARVATION, FILTERS	• CREEP TESTS • NDT FOR INTERNAL CRACKS • INVESTIGATE SERVICE HISTORY

APPENDIX III

U.S. DEPARTMENT OF TRANSPORTATION

HANDBOOK OF DESCRIPTIVE TECHNICAL TERMS

TRANSPORTATION SAFETY INSTITUTE

6500 SOUTH MACARTHUR BOULEVARD
OKLAHOMA CITY, OKLAHOMA 73125

FOREWORD

We gratefully acknowledge the following organizations who participated in the original compilation and subsequent reprintings of this glossary:

USAF Inspection & Safety Center
USAF Air Training Command
Oklahoma Air National Guard (137 TAW)
The General Electric Company
General Motors Corporation
Pratt & Whitney Aircraft Group

Editors Note: Two experienced investigators will often use a different descriptive term when referring to the same damage. We see this in correspondence, published reports, safety inspections, accident/incident reports, and in normal conversation. This problem is not peculiar to any group of people and applies industry wide. This handbook is republished in an attempt to standardize the technical language used in aircraft accident and incident reports.

First TSI Reprint - February 1983

INDEX

PART CONDITION CODE

(Main Terms)

Condition	Definition	Asso. Terms	Illus.
Arced	Visible effects (burn spots, fused metal) of an undesired electrical discharge between two electrical connections.	Flashed Over	
Battered	Damaged by repeated blows or impacts (not humanly inflicted).		21
Bent	Sharp deviation from original line or plane usually caused by lateral force. Examples are: kinked pipe, creased or folded sheet metal.	Creased Folded Kinked	22
Binding	Restricted movement such as tightened or sticking condition resulting from high or low temperature, foreign object jammed in mechanism, etc.	Sticking Tight	
Bowed	Curved or gradual deviation from original line or plane usually caused by lateral force and/or heat.		23
Brinnelled	Circular surface indentions on bearing races usually caused by repeated shock loading of the bearing: i.e., ball or roller indention.		24
Broken	Separated by force into two or more pieces.	Fractured	25

Condition	Definition	Asso. Terms	Illus.
Bulged	Localized outward or inward swelling usually caused by excessive local heating and/or differential pressure.	Ballooned Swelling	26
Burned	Destructive oxidation usually caused by higher temperature than the parent material can withstand.		27
Burrs	A rough edge or a sharp projection on the edge or surface of the parent material.		
Carboned	Accumulation of carbon deposits.	Carbon covered Carbon tracked Coked	28
Chafed	Frictional wear damage usually caused by two parts rubbing together with limited motion.		
Checked	Surface cracks usually caused by heat.		
Chipped	A breaking away of the edge corner or surface of the parent material usually caused by heavy impact (not flaking).		
Circuit - Grounded	Undesired current path to ground (common).		
Circuit - Open	Incomplete electrical circuit due to separation at or between electrical connection points.		
Circuit - Shorted	Undesired current path between leads or circuits that normally are at a different potential.		

Condition	Definition	Asso. Terms	Illus.
Collapsed	Inward deformation of the original contour of a part usually due to high pressure differentials such as a collapsed bellows.	Crushed	
Corroded	Gradual destruction of the parent material by chemical action. Often evidenced by oxide build-up on the surface of the parent material.	Rusted Oxidation	
Cracked	Visible (not requiring special fluorescent or magnetic penetrants) partial separation of material which may progress to a complete break.		30
Crossed	Material damage to parts (as in the case of crossed threads) or part rendered inoperative (as in the case of crossed wires) as a result of improper assembly.		
Curled	A condition where the tip(s) of compressor blades or turbine buckets have been curled over due to rubbing against the engine casings.		31
Dented	A surface indention with rounded bottom usually caused by impact of a foreign object. Parent material is displaced, seldom separated.	Peened	32
Deposits	A build-up of material on a part either from foreign material or from another part not in direct contact.	Metalizing (Undesirable)	33

Condition	Definition	Asso. Terms	Illus.

NOTE

Dimensions under minimum and over maximum are condition codes involving the physical dimensions of a part, such as diameter, width, length, etc. where the discrepancy is not caused by normal wear and tear.

Condition	Definition	Asso. Terms	Illus.
Dimension under min.	Under blueprint dimension or other dimensions published in an authoritative publication (not caused by wear).		
Dimension over max.	Over blueprint dimension or other dimension published in an authoritative publication (not caused by wear).		
Disintegrated	Separated or decomposed into fragments. Excessive degree of fracturing (breaking) as with disintegrated bearings. Complete loss of original form.	Shattered	34
Distorted	Extensive deformation of the original contour of a part usually due to impact of a foreign object, structural stresses, excessive localized heating or any combination of these.	Buckled Depressed Twisted Warped	35
Eccentric	Part(s) wherein the intended common center is displaced significantly.	Non-concentric	
Emission-Low	Applies to the output of electric tubes, indicates an unsatisfactory tube.		
Eroded	Carry away of material by flow of fluids or gases, accelerated by heat or grit.		

Condition	Definition	Asso. Terms	Illus.
Flattened Out	Permanent deformation beyond tolerance limits usually caused by compression.		
Frayed	Worn into shreds by rubbing action.		
Fused	Joining together of two materials usually caused by heat, friction, or current flow.		
Galled	Chafing or severe fretting caused by slight relative movement of two surfaces under high contact pressure.		36
Glazed	Undesirable development of a hard, glossy surface due to rubbing action, heat or varnish.		
Gouged	Scooping out of material usually caused by a foreign object.	Furrowed	37
Grooved	Smooth, rounded furrow or furrows of wear, usually wider than scoring, with rounded corners and smooth on the groove bottom. Example: a ball bearing wearing into a race would cause a grooved condition.		38
Hot-Spot	Subjected to excessive temperature usually evidenced by change in color and appearance of part.	Heat discolored Overheated Heated excessively	39
Indications	Cracks, inclusions, fractures etc. not visible without fluorescent or magnetic penetrants.		

Condition	Definition	Asso. Terms	Illus.
Magnetism-Low	Low or lost magnetism of permanetly magnetized parts such as motor field cores, etc.		
Melted	Deformation from the original configuration due to heat, friction or pressure as with melted bearings or insulation.		
Mis-matched	Improper association of two or more parts.		
Mis-positioned	Improper installation of a part resulting in damage to the installed part or to associated parts.	Mis-aligned Reversed	
Nicked	A sharp surface indention caused by impact of a foreign object. Parent material is displaced, seldom separated.		40
Out-of-round	Diameters of part not constant.		
Out-of-square	Deformation of right angle relationship of part surfaces.		
Peeled	A breaking away of surface finishes such as coating, platings, etc; peeling would be flaking of very large pieces; a blistered condition usually precedes or accompanies flaking.	Blistered Flaked	41
Pick-Up	Transfer of metal from one surface to another. Usual cause is the rubbing of two surfaces without sufficient lubrication.		42

Condition	Definition	Asso. Terms	Illus.
Pitted	Small irregular shaped cavities in the surface of the parent material usually caused by corrosion, chipping, or heavy electrical discharge.		43
Plugged	Pipe, hoses, tubing, channeling, internal passage, etc. which are totally or partially blocked.	Clogged Obstructed Restricted passage	
Porous	Voids located internally, in the surface or completely through a material. Usually applied to cast material or to welds.	Pock-marked Perforation-weld	44
Resistance-high	High electrical resistance in an electrical circuit, causing improper component or circuit operation.		
Rolled-over	Lipping or rounding of a metal edge.	Lipped Turned Metal	45
Rough	Usually applies to operation as opposed to surface finish; i.e., a condition of bearings (which cannot be disassembled further) where during the spin test the rotation is rough.		
Rubbed	To move with pressure or friction against another part - such as compressor rub.		
Ruptured	Extensive breaking apart of material usually caused by high stresses, differential pressure, locally applied force or any combination of these. Examples: burst bellows, blown casing, etc.	Blown Burst Split	46
Scored	Deep scratch or scratches made during part operation by sharp edges of foreign particles.		47

Condition	Definition	Asso. Terms	Illus.
Scratched	Light narrow, shallow mark or marks caused by movement of a sharp object or particle across a surface. Material is displaced, not removed.		48
Seized	Parts bound together because of expansion or contraction due to high or low temperature, foreign object jammed in mechanism, etc.	Frozen Jammed Stuck	
Sheared	Dividing a body by cutting action, i.e., division of a body so as to cause its parts to slide relative to each other in a direction parallel to their plane of contact.	Cut	49
Spalled	Sharply roughened area characterized by progressive chipping-away of surface material. (Not to be confused with flaking.) Usual causes are surface cracks, inclusions or any similar surface injury causing a progressive breaking away of the surface under load.		50
Stretched	Enlargement of a part as a result of exposure to operating conditions.	Growth	
Stripped	A condition usually associated with threads or insulation. Involves removal of material (threads) by force.		
Torn	Separation by pulling apart.		51
Worn excessively	Material of part consumed as a result of exposure to operation or usage.		

PART CONDITION CODE

(Associated Terms)

Condition	Reference
Arced	See Arced
Ballooned	See and code Bulged
Battered	See Battered
Bent	See Bent
Binding	See Binding
Blistered	See and code per peeled
Blown	See and code per Ruptured
Bowed	See Bowed
Brinelled	See Brinelled
Broken	See Broken
Buckled	See and code per Distorted
Bulged	See Bulged
Burned	See Burned
Burst	See and code per Ruptured
Burrs	See Burrs
Carboned	See Carboned
Carbon Covered	See and code Carboned
Carbon Tracked	See and code Carboned
Chafed	See Chafed
Charred	See and code per Burned
Checked	See Checked

Condition	Reference
Chipped	See Chipped
Circuit-Grounded	See Circuit-Grounded
Circuit-Open	See Circuit-Open
Circuit-Shorted	See Circuit-Shorted
Clogged	See and code per Plugged
Cocked	See and code Mispositioned
Coked	See and code Carboned
Collapsed	See Collapsed
Corroded	See Corroded
Cracked	See Cracked
Creased	See and code per Bent
Crossed	See Crossed
Crushed	See and code per Collapsed
Curled	See Curled
Cut	See and code per Sheared
Dented	See Dented
Deposits	See Deposits
Depressed	See and code per Distorted
Dimension under min.	See Dimension under min.
Dimension over max.	See Dimension over max.
Disintegrated	See Disintegrated
Distorted	See Distorted

Condition	Reference
Eccentric	See Eccentric
Emission-Low	See Emission-Low
Eroded	See Eroded
Flattened Out	See Flattened Out
Flaked	See and code per Peeled
Flashed Over	See and code Arced
Folded	See and code per Bent
Fractured	See and code per Broken
Frayed	See Frayed
Frozen	See and code per Seized
Furrowed	See and code per Grooved
Fused	See Fused
Galled	See Galled
Glazed	See Glazed
Gouged	See Gouged
Grooved	See Grooved
Growth	See and code per Stretched
Indications	See Indications
Heat Discolored	See and code per Hot-Spot
Heated Excessively	See and code per Hot-Spot
Hot-Spot	See Hot-Spot
Jammed	See and code per Seized
Kinked	See and code per Bent

Condition	Reference
Lipped	See and code per Rolled-over
Magnetism-Low	See magnetism-Low
Melted	See Melted
Metalizing (Undesirable)	See and code per Mis-positioned
Mis-matched	See Mis-matched
Mis-positioned	See Mis-positioned
Nicked	See Nicked
Non-concentric	See and code per Eccentric
Obstructed	See and code per Plugged
Out-of-round	See Out-of-round
Out-of-square	See Out-of-square
Overheated	See and code per Hot-Spot
Oxidation	See and code per Corroded
Peeled	See Peeled
Peened	See and code per Dented
Perforation-weld	See and code per porous
Pick-up	See Pick-up
Pitted	See Pitted
Plugged	See Plugged
Pock-marked	See and code per Porous
Porous	See Porous
Resistance-high	See Resistance-high

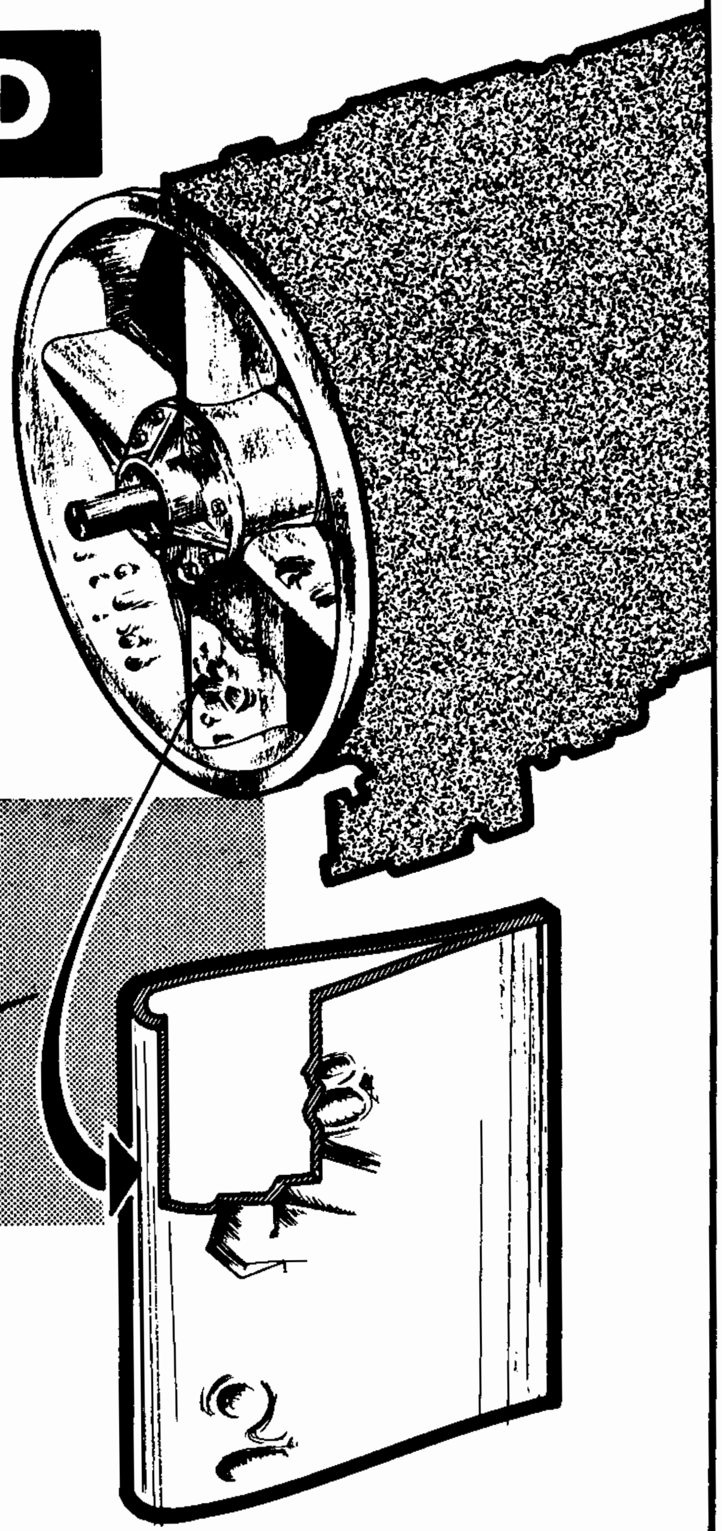
Condition	Reference
Resistance-low	See Resistance-low
Restricted Passage	See and code per Plugged
Reversed	See and code per Mis-positioned
Rolled-over	See Rolled-over
Rough	See Rough
Rubbed	See Rubbed
Ruptured	See Ruptured
Rusted	See and code per Corroded
Scored	See Scored
Scraped	See and code per Chafed
Scratched	See Scratched
Scuffed	See and code per Chafed
Seized	See Seized
Shattered	See and code per Disintegrated
Sheared	See Sheared
Spalled	See Spalled
Split	See and code per Ruptured
Sticking	See and code per Binding
Stretched	See Stretched
Stripped	See Stripped
Stuck	See and code per Seized
Swelling	See and code per Bulged

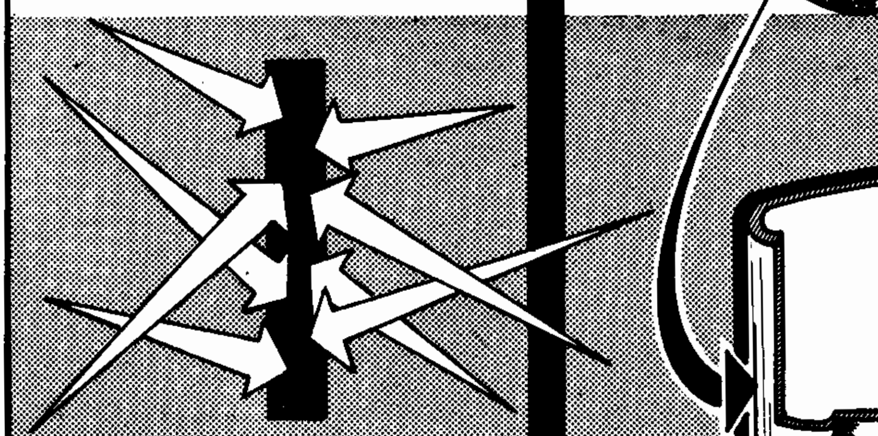
Condition	Reference
Torn	See Torn
Turned-metal	See and code per Rolled-over
Twisted	See and code per Distorted
Tight	See and code per Binding
Undesirable Metalizing	See and code Deposits
Warped	See and code per Distorted
Worn excessively	See Worn excessively

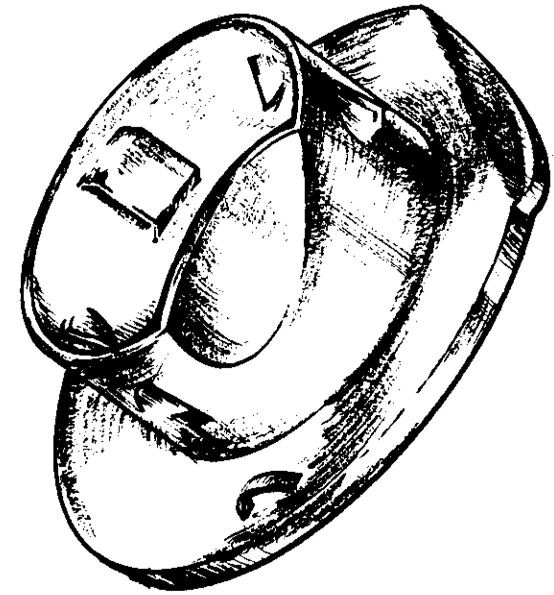
TABLE OF CONTENTS

BATTERED

DAMAGED BY REPEATED BLOWS OR IMPACTS (NOT HUMANLY INFLICTED)

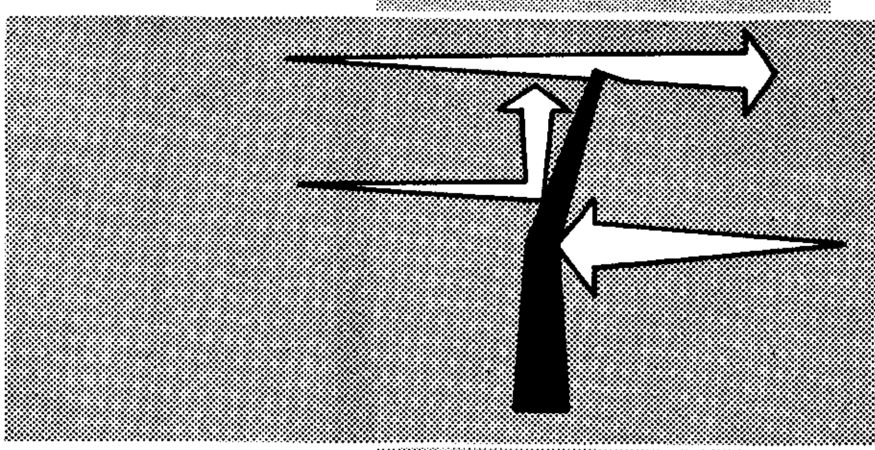


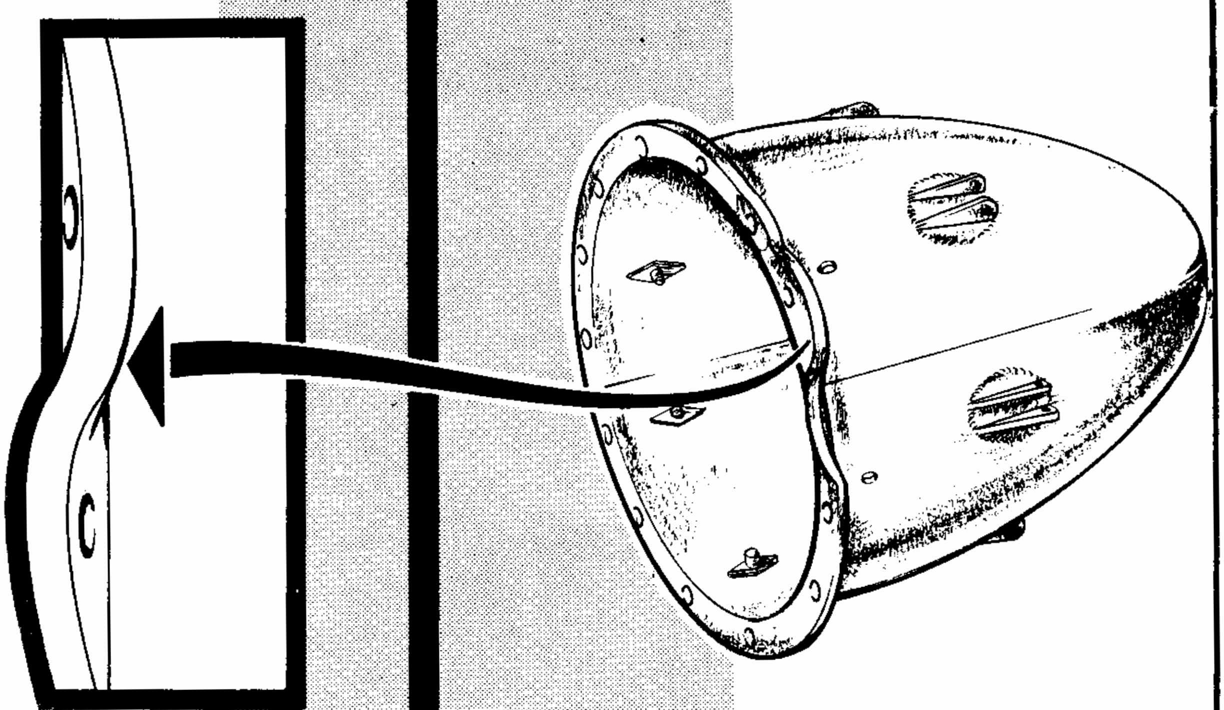




BENT

SHARP DEVIATION FROM ORIGINAL LINE OR PLANE USUALLY CAUSED BY LATERAL FORCE. EXAMPLES ARE: KINKED PIPE, CREASED OR FOLDED SHEET METAL





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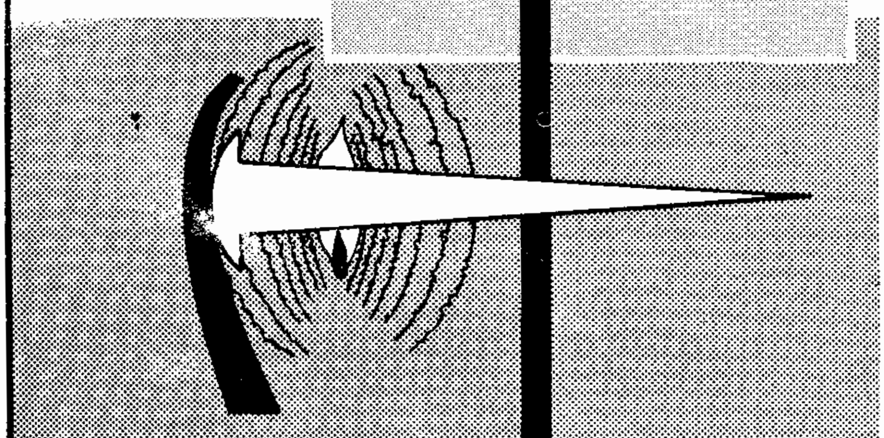
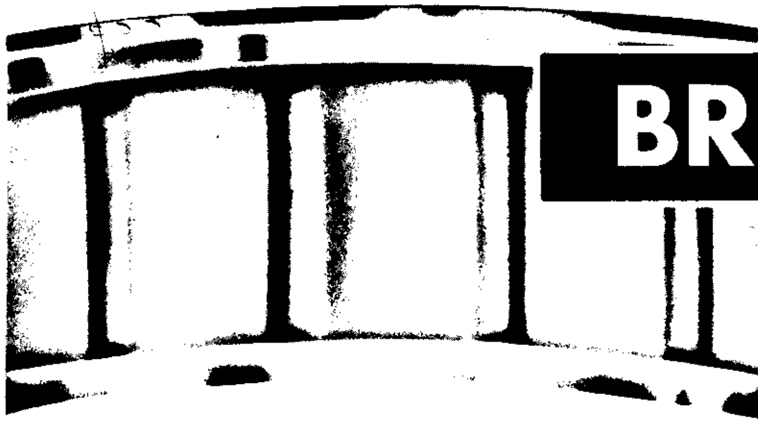
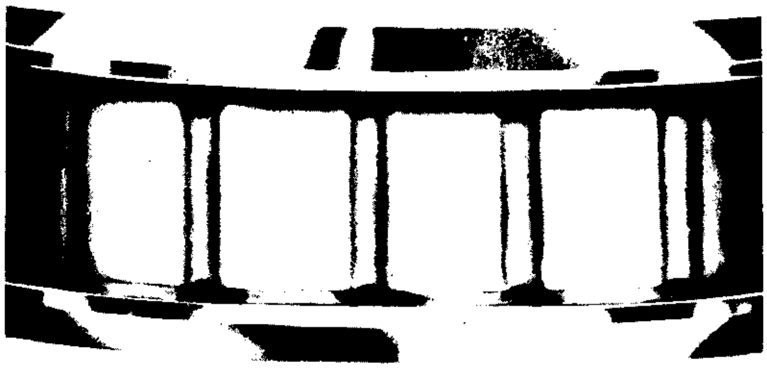


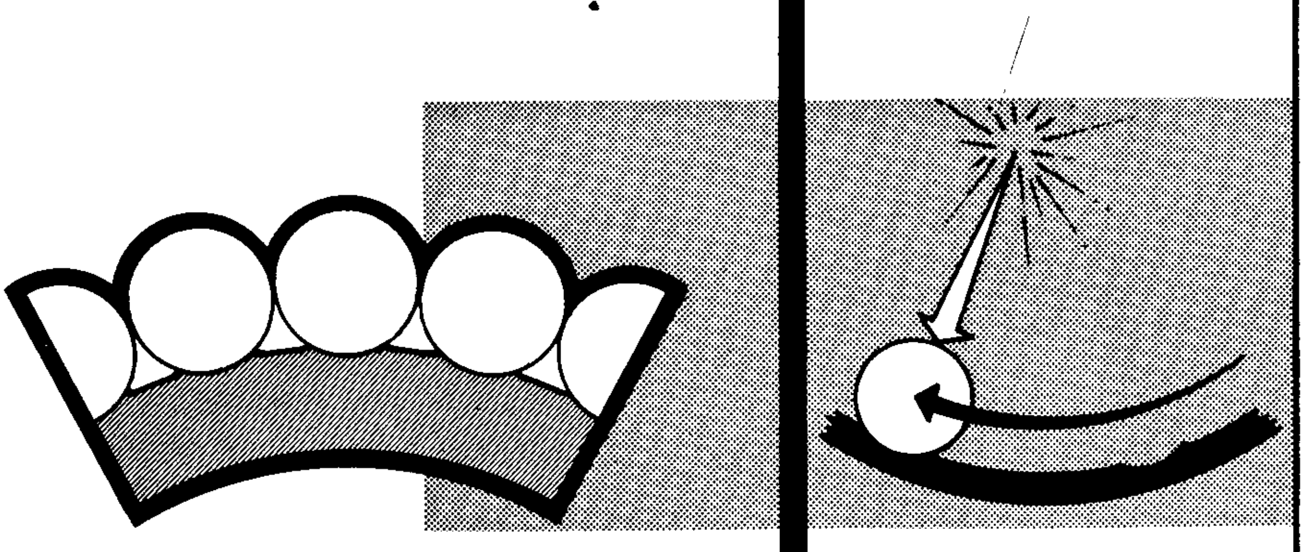
CURVED OR
GRADUAL DEVIATION
FROM ORIGINAL LINE
OR PLANE USUALLY
CAUSED BY LATERAL
FORCE AND/OR HEAT

BRINELLED

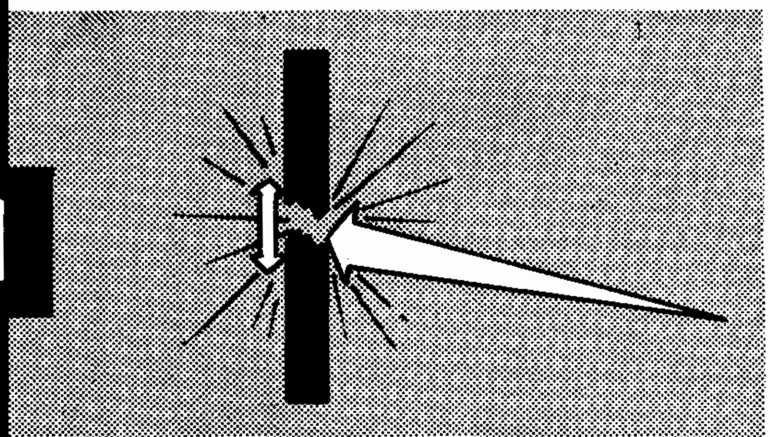




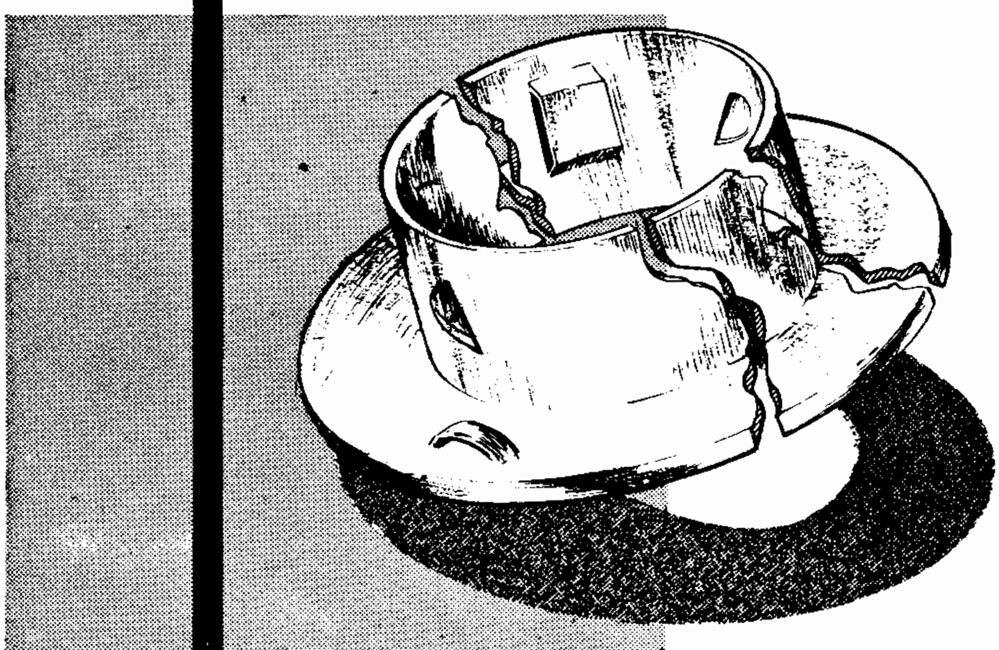
CIRCULAR SURFACE INDENTIONS ON BEARING RACES USUALLY CAUSED BY REPEATED SHOCK LOADING OF THE BEARING: i.e., BALL OR ROLLER INDENTION



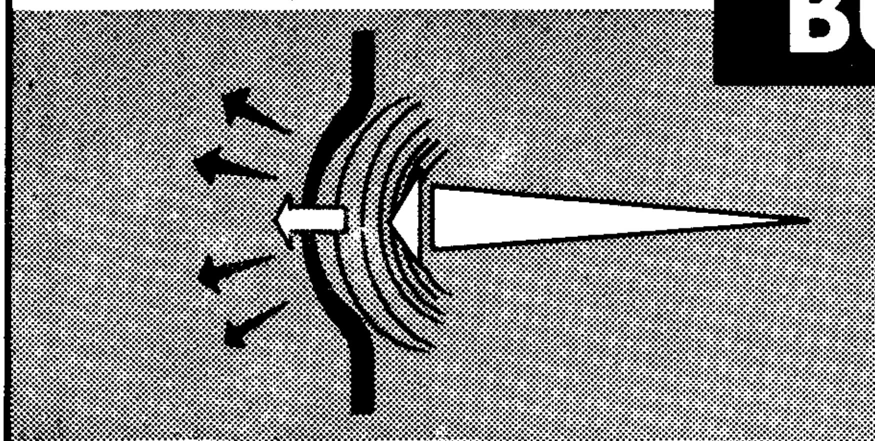
BROKEN



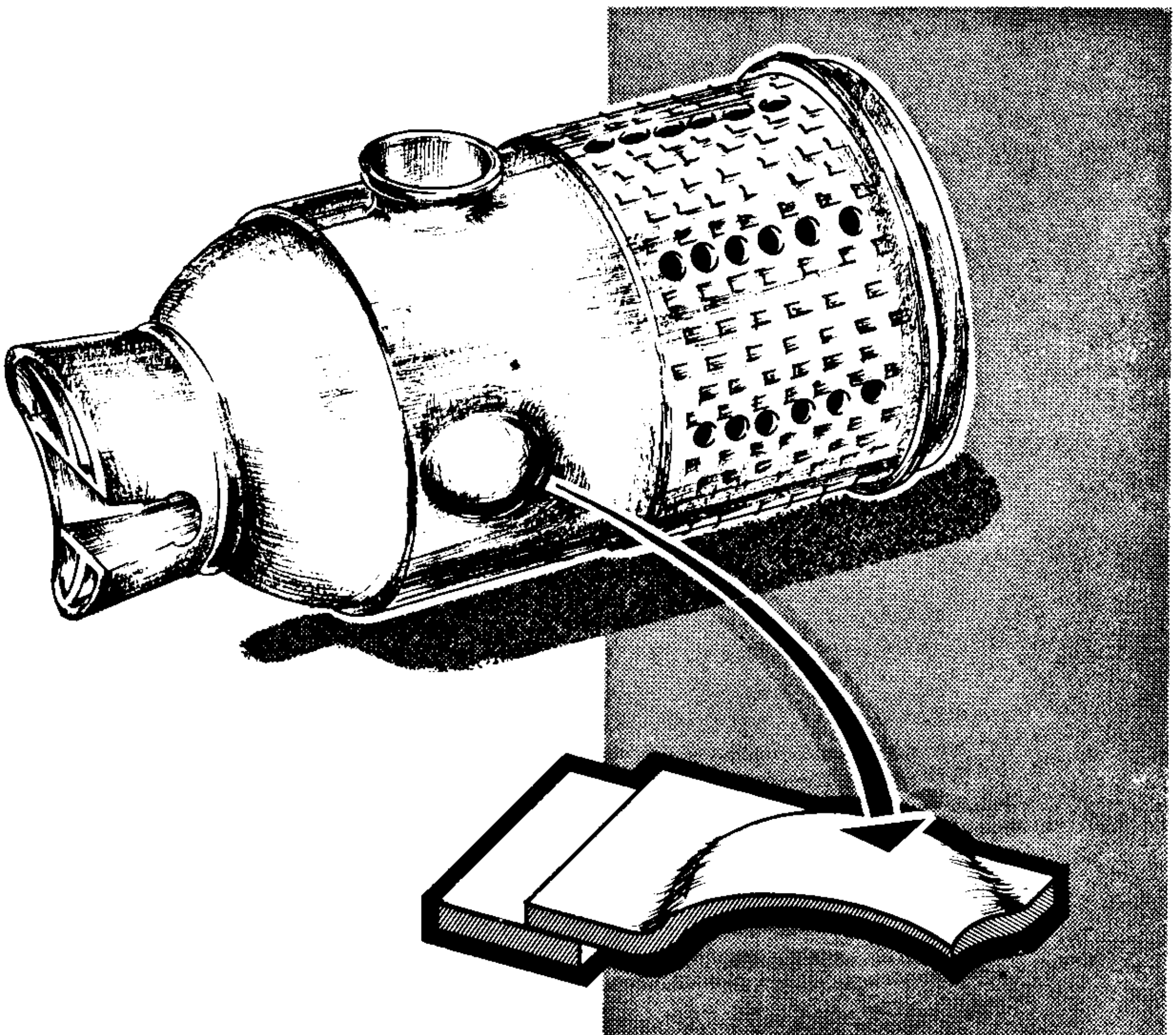
SEPARATED BY FORCE INTO TWO OR MORE PIECES



BULGED

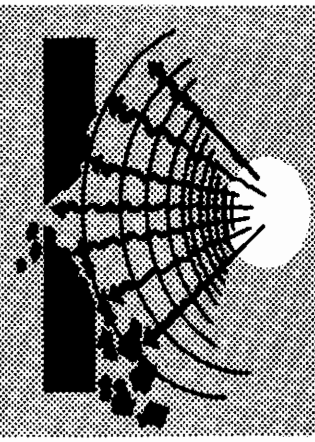


LOCALIZED OUTWARD OR INWARD SWELLING USUALLY CAUSED BY EXCESSIVE LOCAL HEATING AND/OR DIFFERENTIAL PRESSURE.

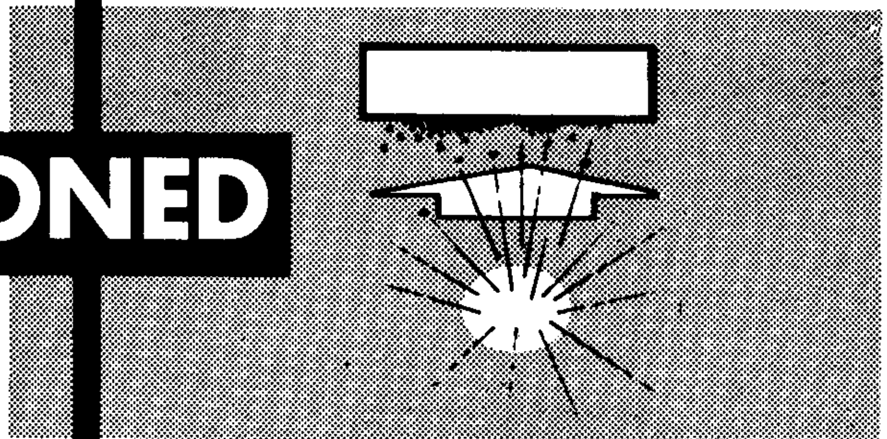


BURNED

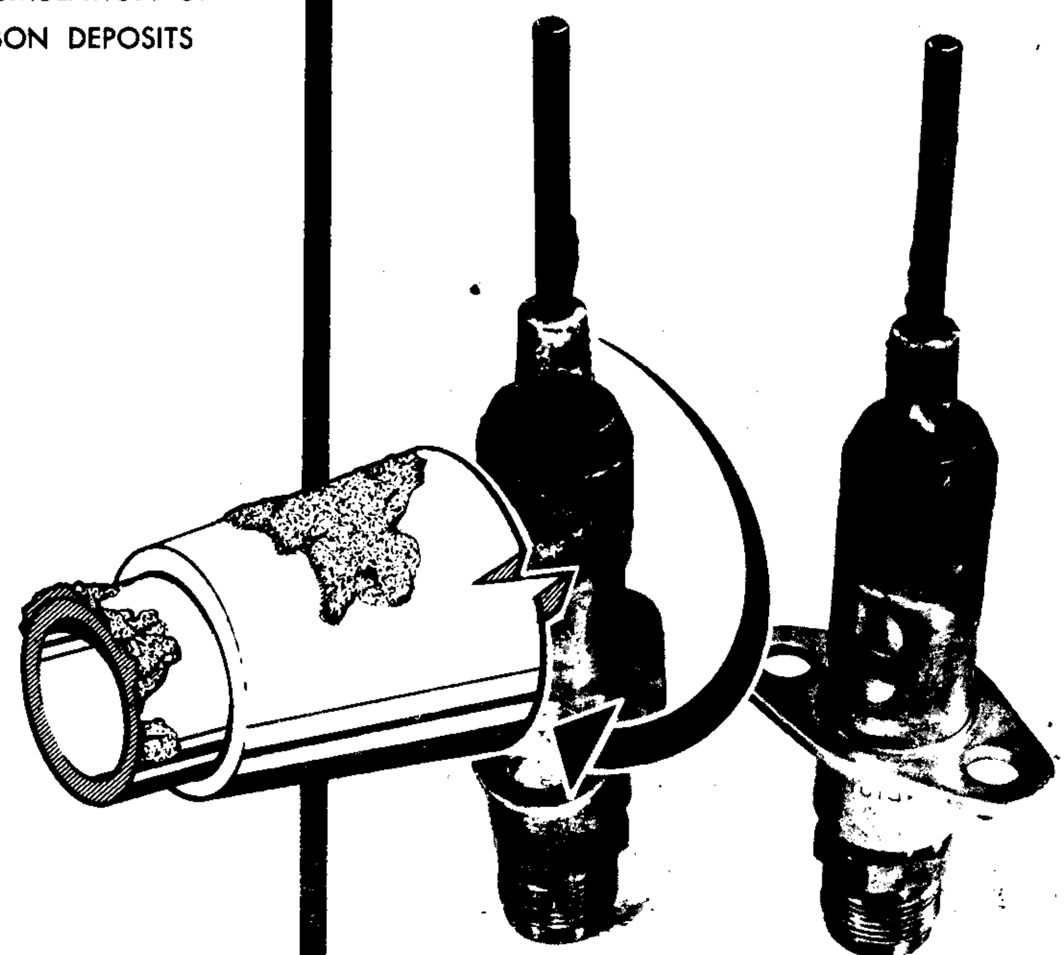
DESTRUCTIVE OXIDATION USUALLY CAUSED BY HIGHER TEMPERATURES THAN THE PARENT MATERIAL CAN WITHSTAND



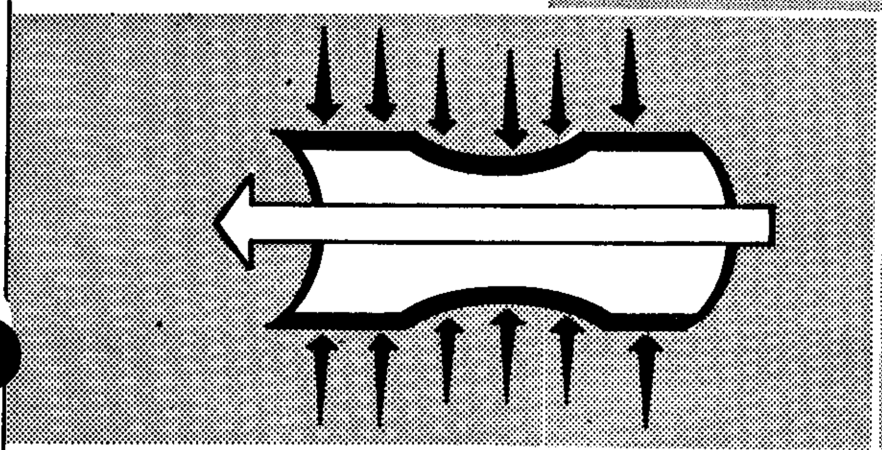
CARBONED



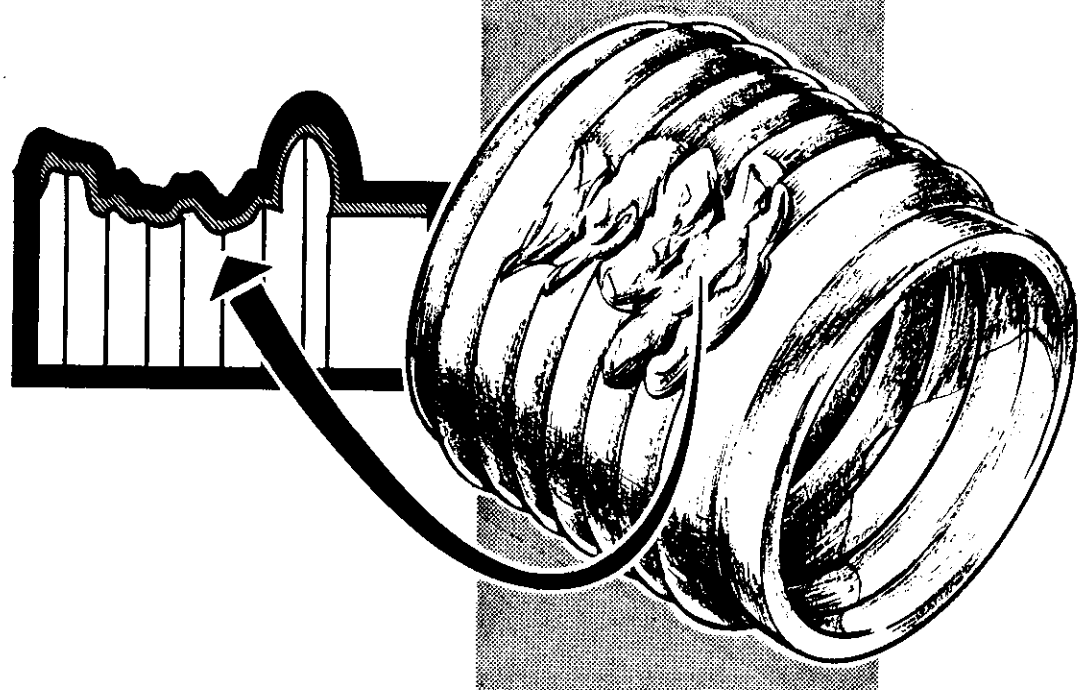
ACCUMULATION OF
CARBON DEPOSITS



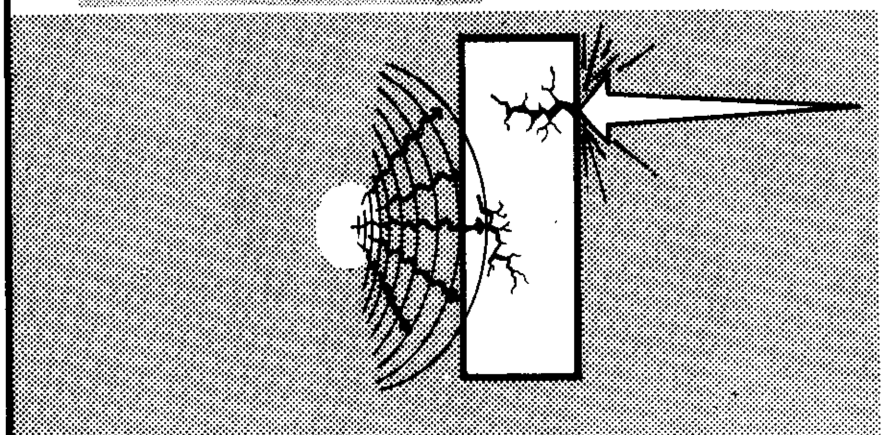
COLLAPSED



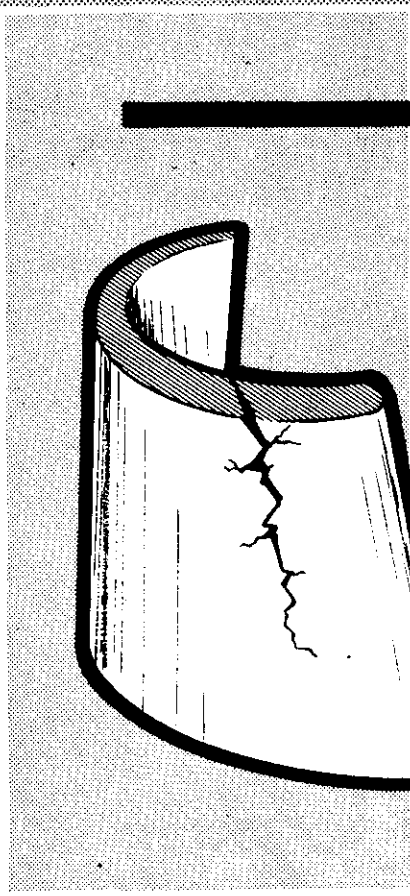
INWARD DEFORMATION OF THE ORIGINAL CONTOUR OF A PART USUALLY DUE TO HIGH PRESSURE DIFFERENTIALS SUCH AS A COLLAPSED BELLOWS



CRACKED



...VISIBLE (NOT REQUIRING SPECIAL FLUORESCENT OR MAGNETIC PENETRANTS) PARTIAL SEPARATION OF MATERIAL WHICH MAY PROGRESS TO A COMPLETE BREAK



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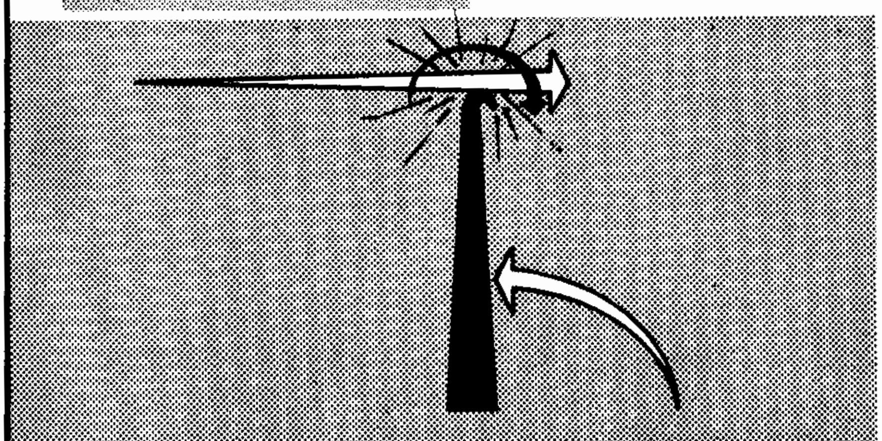
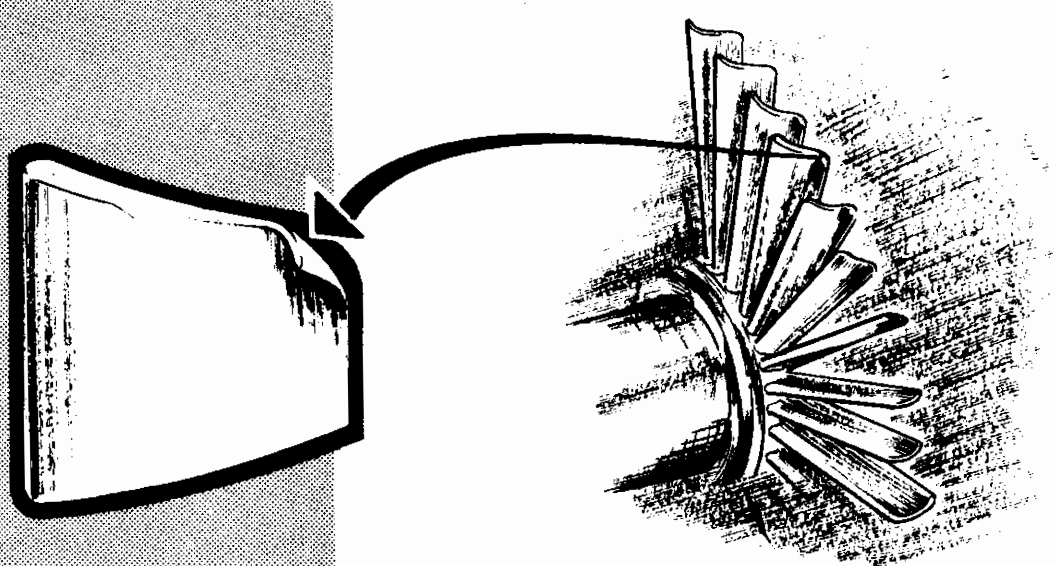
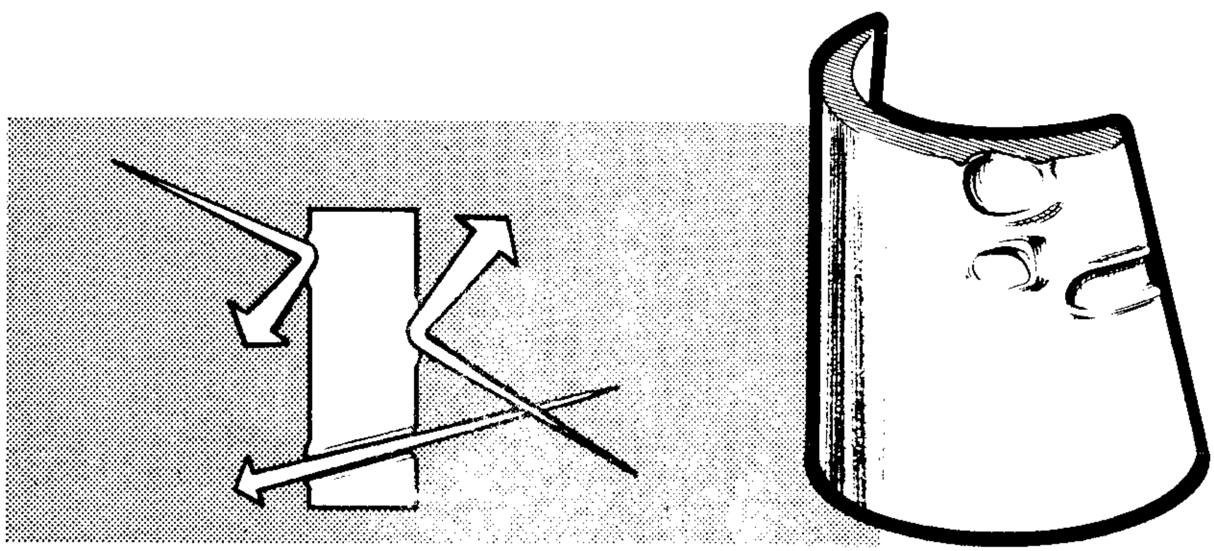


A CONDITION WHERE THE TIP(S) OF COMPRESSOR BLADES OR TURBINE BUCKETS HAVE BEEN CURLED OVER DUE TO RUBBING AGAINST THE ENGINE CASINGS

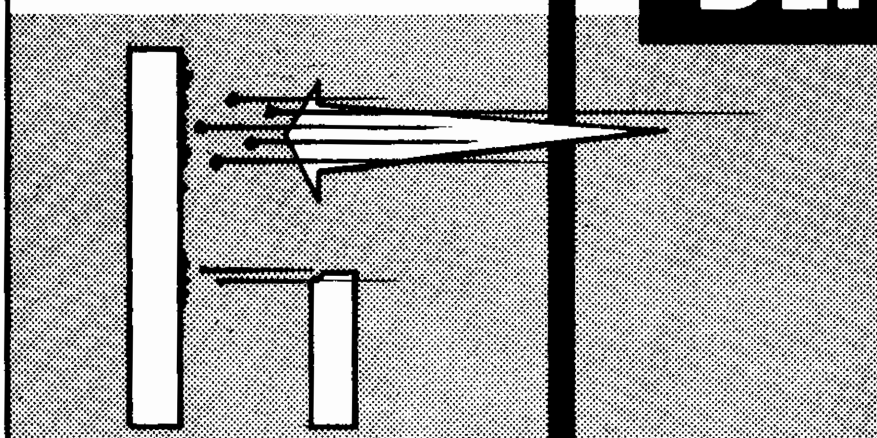


DENTED

A SURFACE INDENTION WITH ROUNDED BOTTOM USUALLY CAUSED BY IMPACT OF A FOREIGN OBJECT. PARENT MATERIAL IS DISPLACED, SELDOM SEPARATED

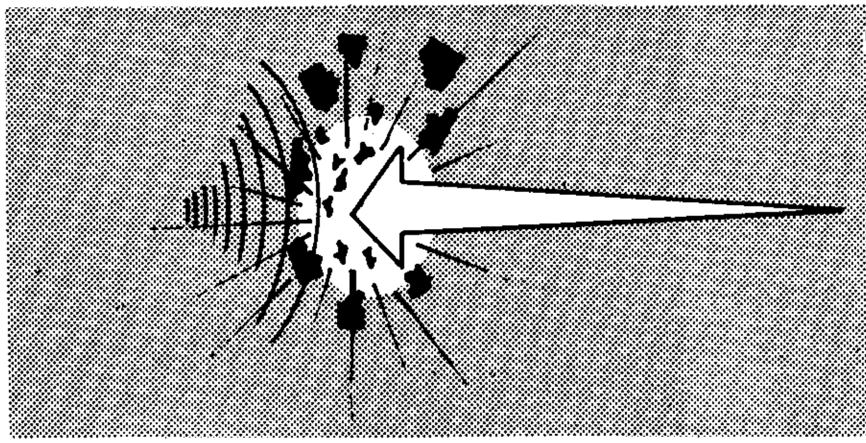


DEPOSITS·



A BUILD-UP OF MATERIAL ON A PART EITHER FROM FOREIGN MATERIAL OR FROM ANOTHER PART NOT IN DIRECT CONTACT

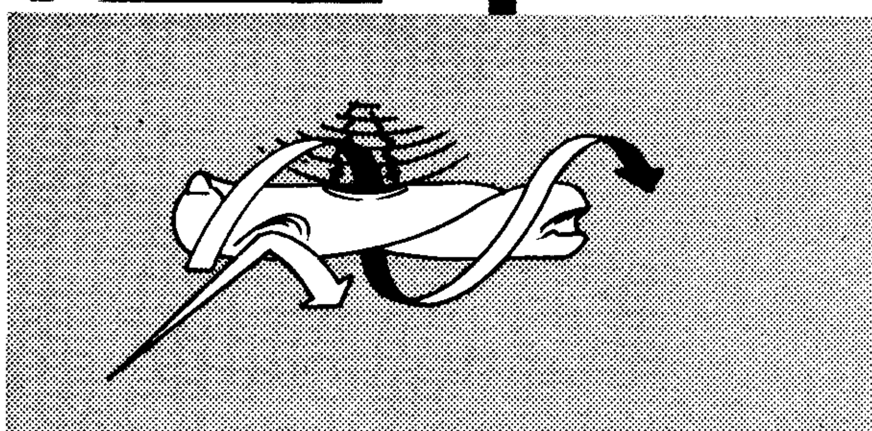
SEPARATED OR DECOMPOSED INTO FRAGMENTS. EXCESSIVE DEGREE OF FRACTURING (BREAKING) AS WITH DISINTEGRATED BEARINGS. COMPLETE LOSS OF ORIGINAL FORM.

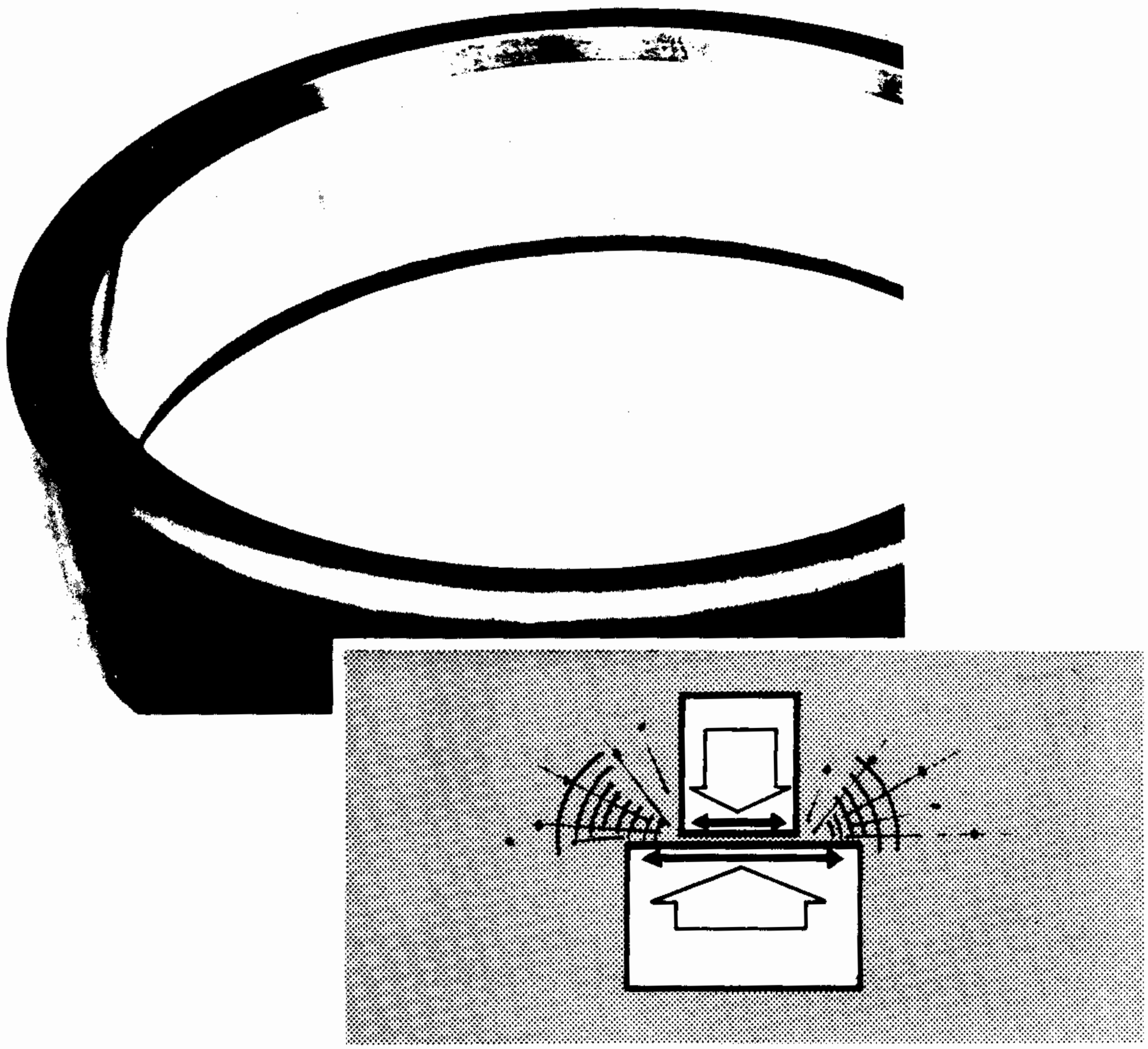


DISINTEGRATED

DISTORTED

EXTENSIVE DEFORMATION OF THE ORIGINAL CONTOUR OF A PART USUALLY DUE TO IMPACT OF A FOREIGN OBJECT, STRUCTURAL STRESSES, EXCESSIVE LOCALIZED HEATING OR ANY COMBINATION OF THESE



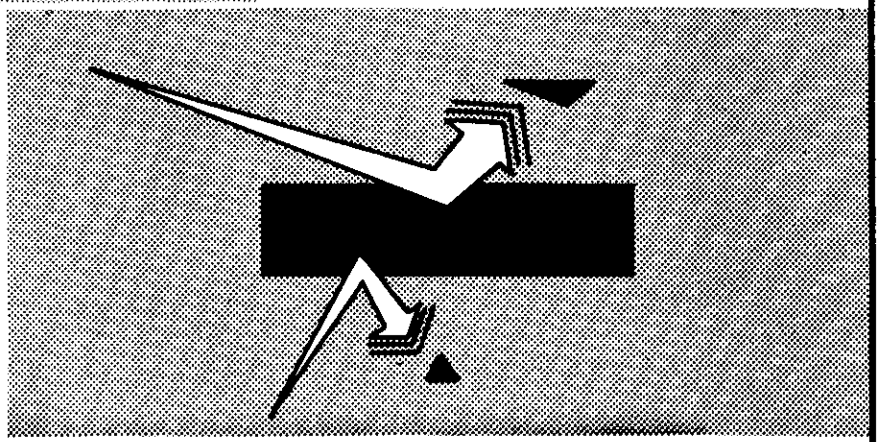


GALLED

CHAFING OR SEVERE FRETTING CAUSED BY SLIGHT RELATIVE MOVEMENT OF TWO SURFACES UNDER HIGH CONTACT PRESSURE

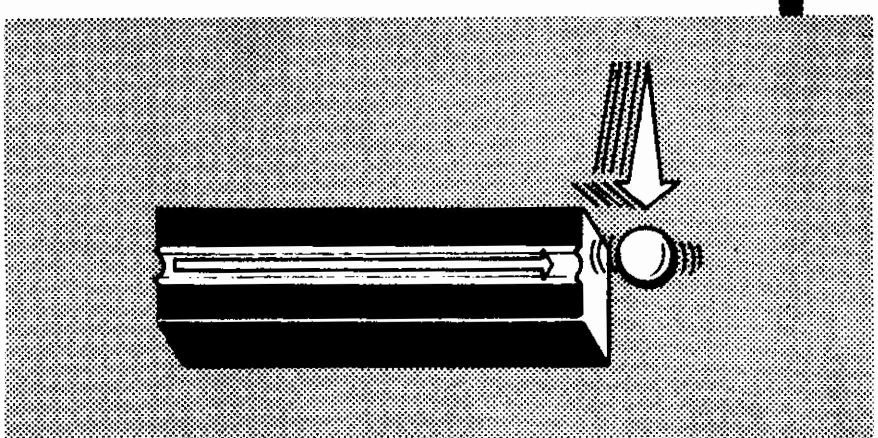
GOUGED

SCOOPING OUT OF MATERIAL USUALLY CAUSED BY A FOREIGN OBJECT



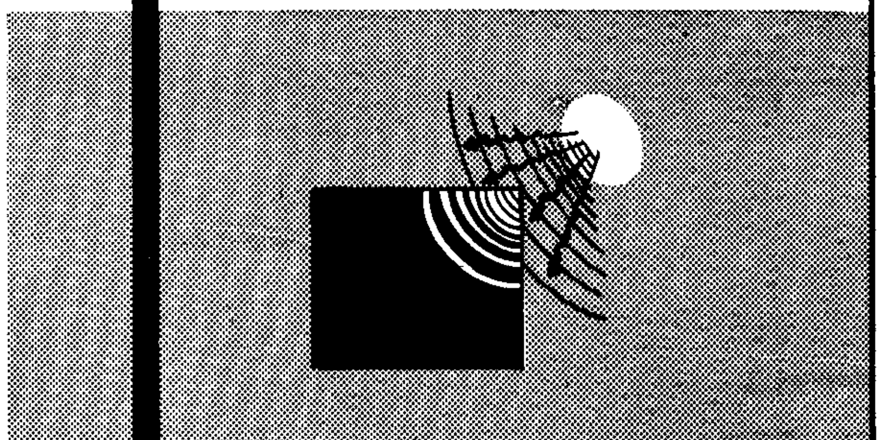
SMOOTH, ROUNDED FURROW OR FURROWS OF WEAR, USUALLY WIDER THAN SCORING, WITH ROUNDED CORNERS AND SMOOTH ON THE GROOVE BOTTOM. EXAMPLE: A BALL BEARING WEARING INTO A RACE WOULD CAUSE A GROOVED CONDITION

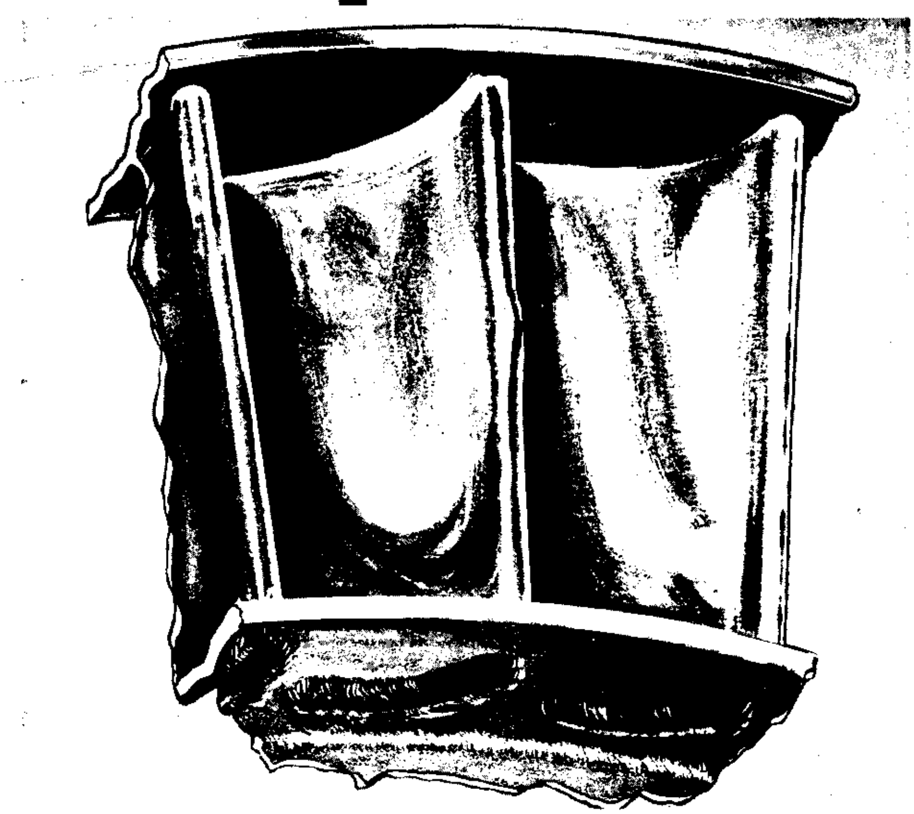
GROOVED



HOT-SPOT

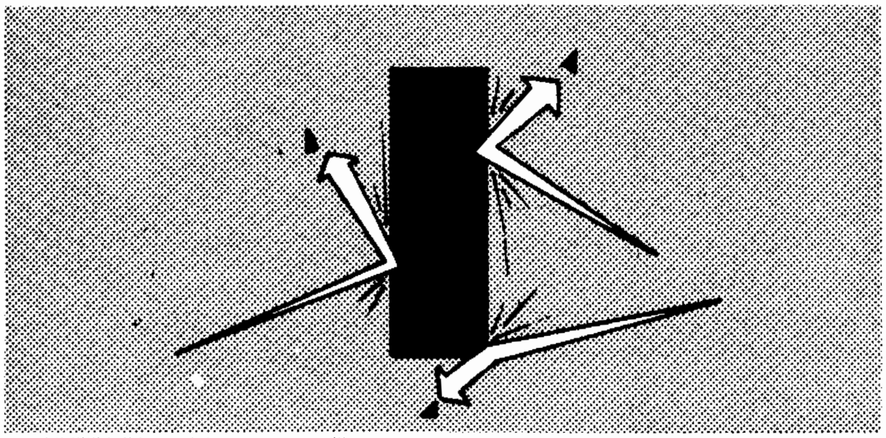
SUBJECTED TO
EXCESSIVE
TEMPERATURE
USUALLY EVIDENCED
BY CHANGE IN
COLOR AND
APPEARANCE OF PART

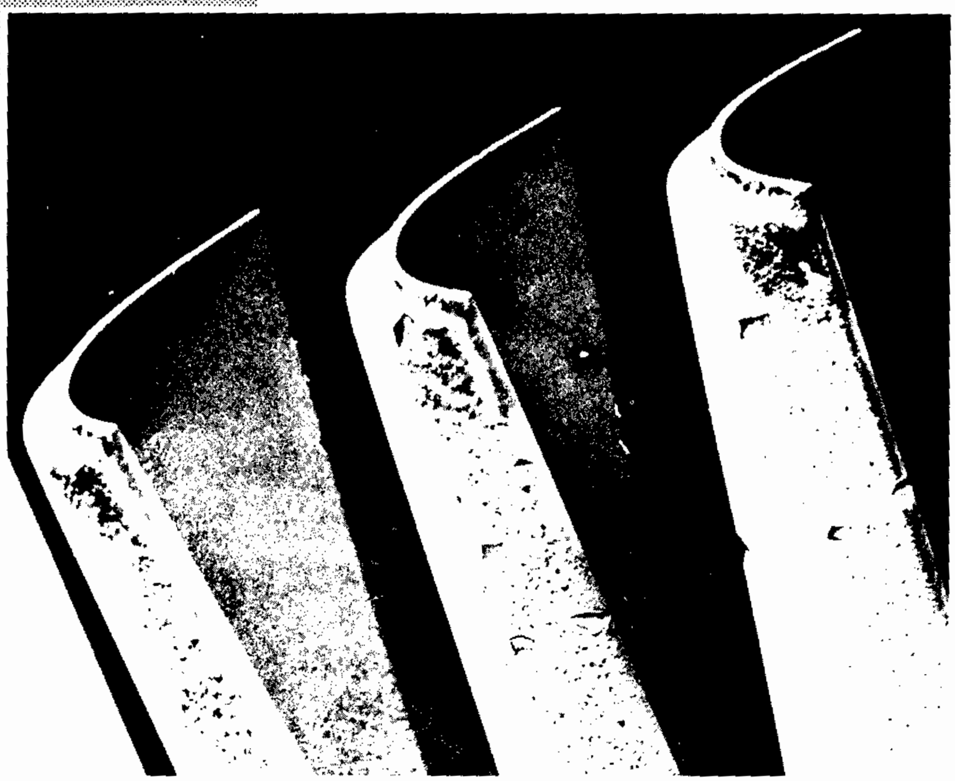


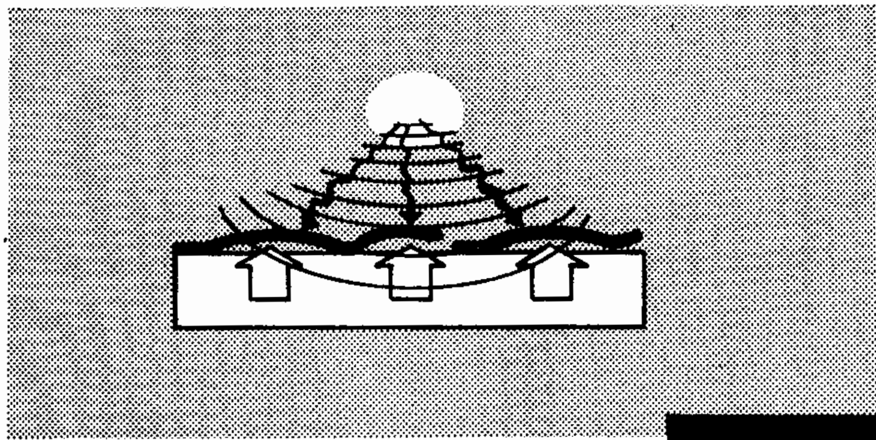


NICKED

A SHARP SURFACE INDENTION CAUSED BY IMPACT OF A FOREIGN OBJECT. PARENT MATERIAL IS DISPLACED, SELDOM SEPARATED

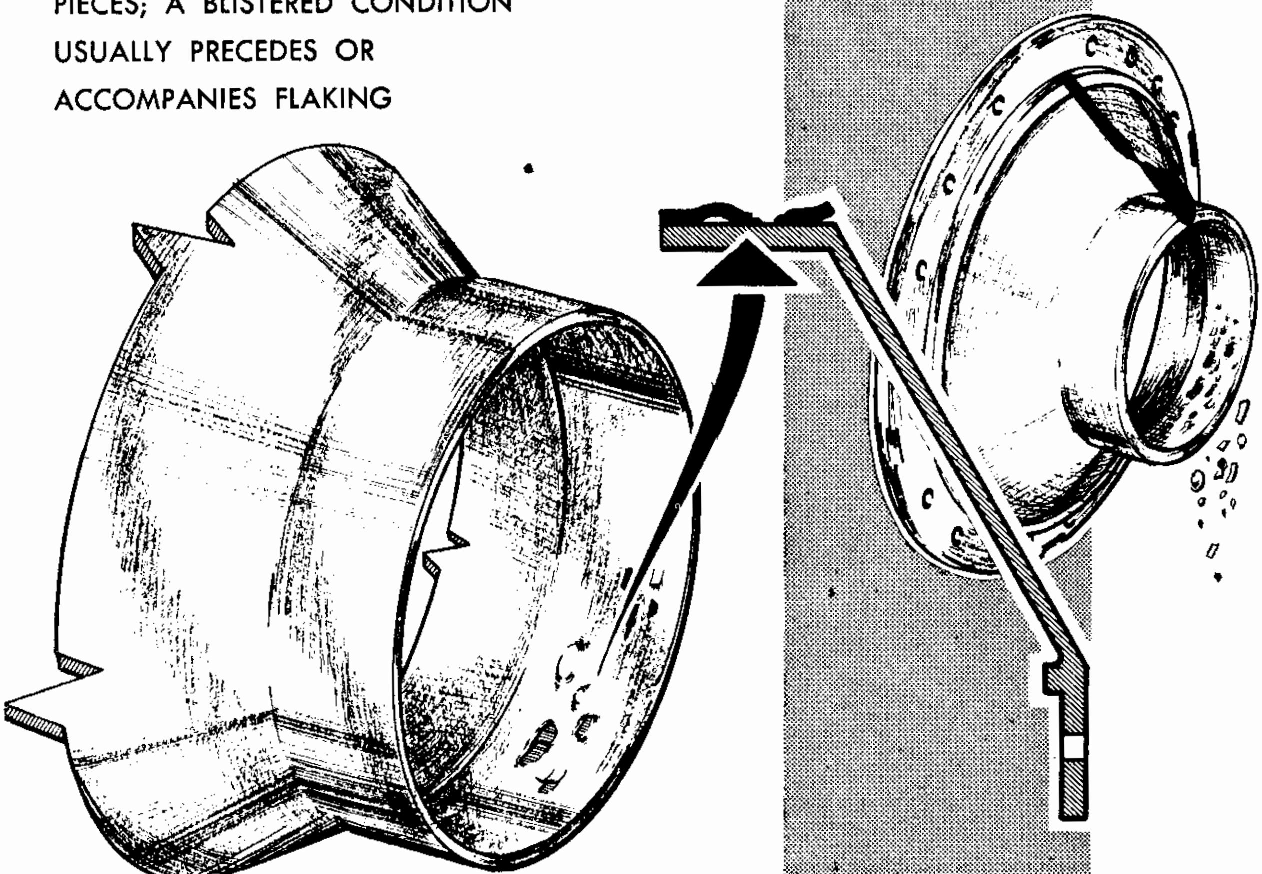




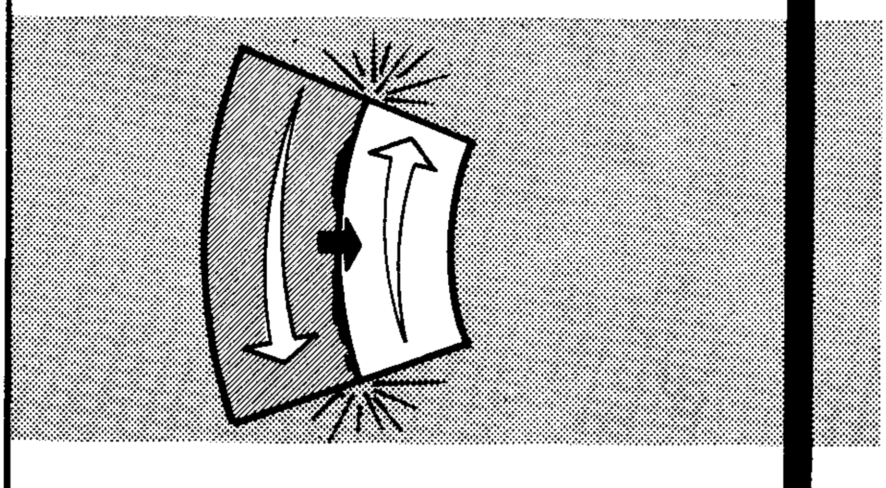


PEELED

A BREAKING AWAY OF SURFACE FINISHES SUCH AS COATING, PLATINGS, ETC; PEELING WOULD BE FLAKING OF VERY LARGE PIECES; A BLISTERED CONDITION USUALLY PRECEDES OR ACCOMPANIES FLAKING

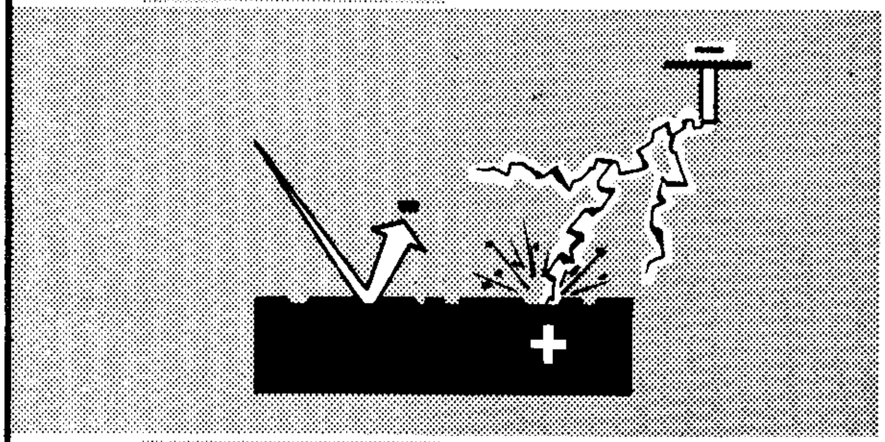


PICK-UP

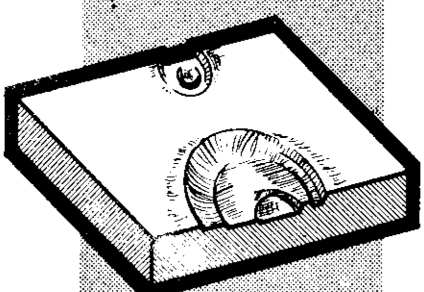


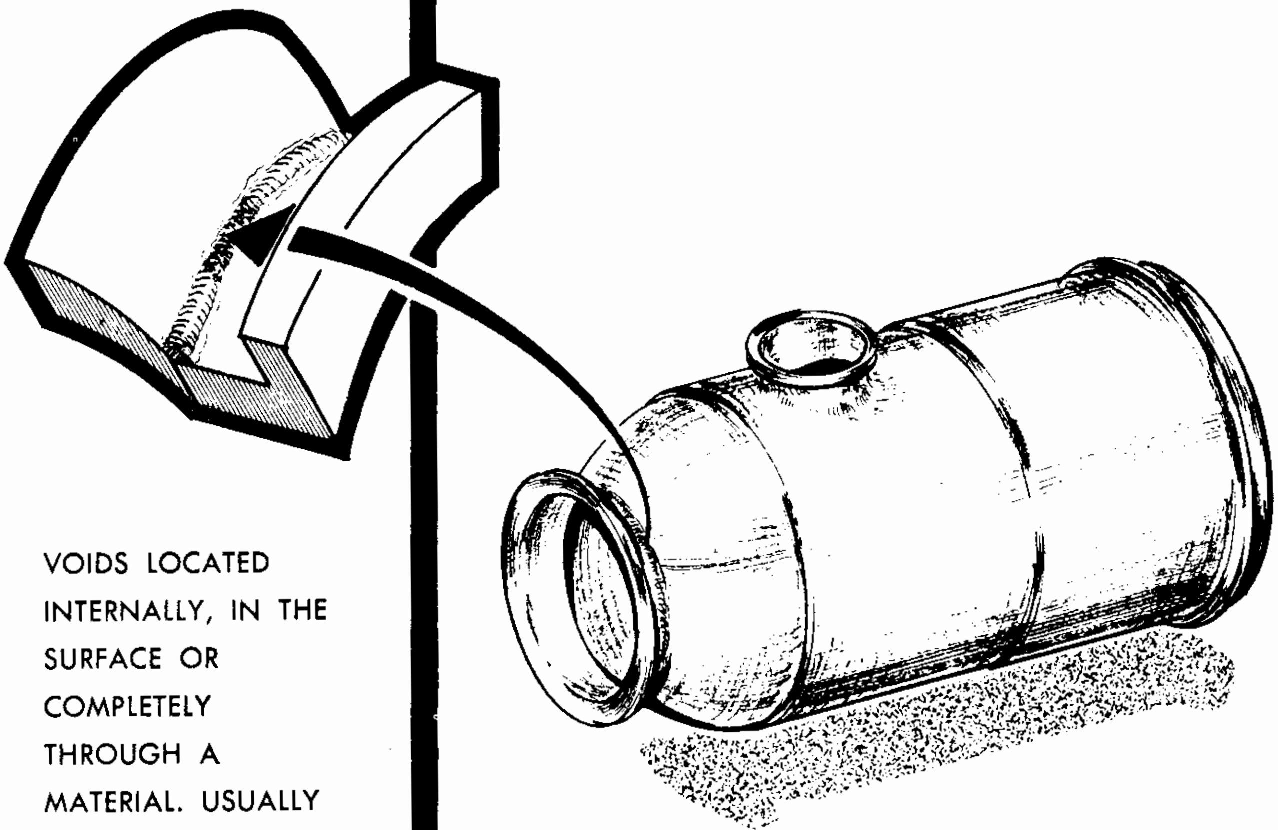
TRANSFER OF METAL FROM ONE SURFACE TO ANOTHER. USUAL CAUSE IS THE RUBBING OF TWO SURFACES WITHOUT SUFFICIENT LUBRICATION

PITTED



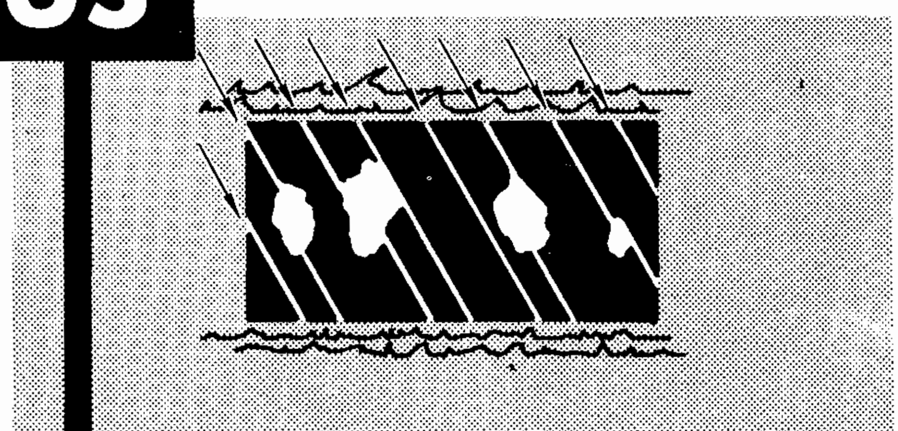
SMALL IRREGULAR SHAPED CAVITIES IN THE SURFACE OF THE PARENT MATERIAL USUALLY CAUSED BY CORROSION, CHIPPING, OR HEAVY ELECTRICAL DISCHARGE





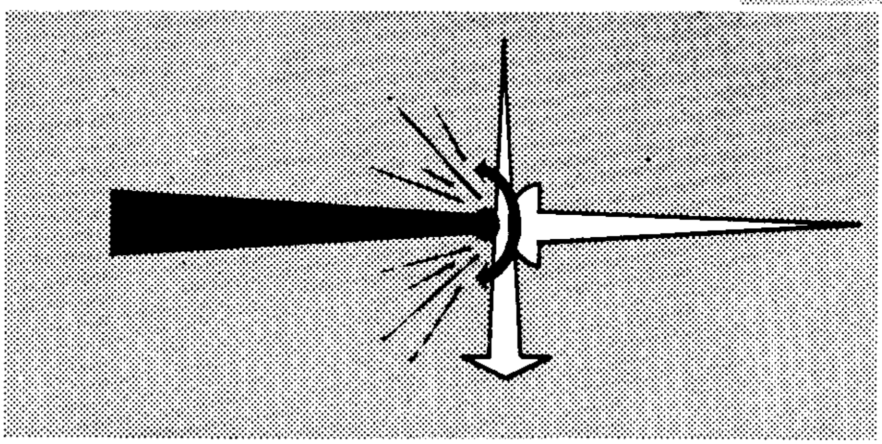
VOIDS LOCATED INTERNALLY, IN THE SURFACE OR COMPLETELY THROUGH A MATERIAL. USUALLY APPLIED TO CAST MATERIAL OR TO WELDS

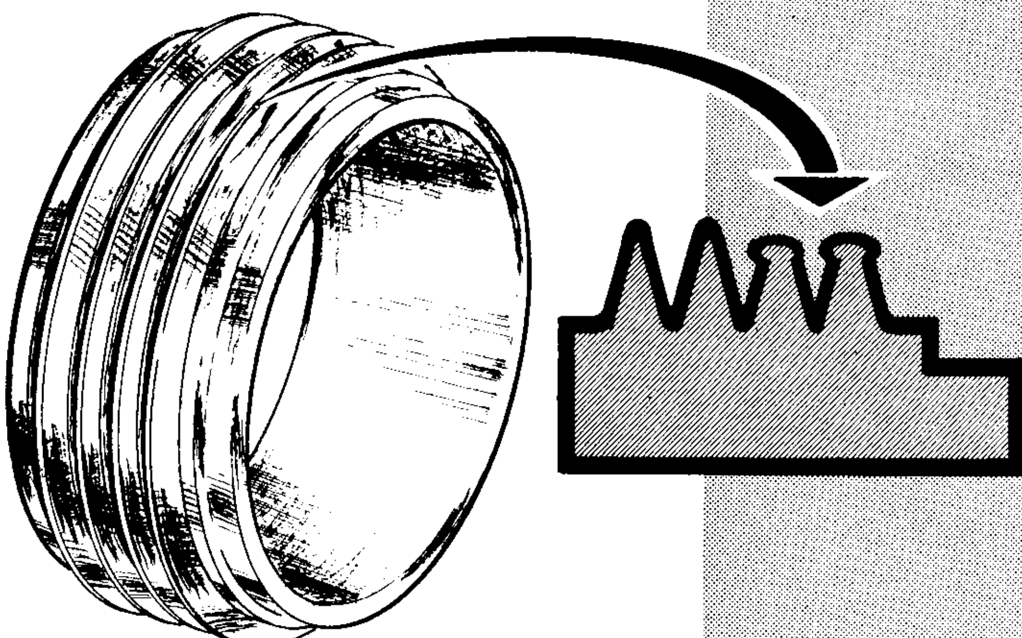
POROUS



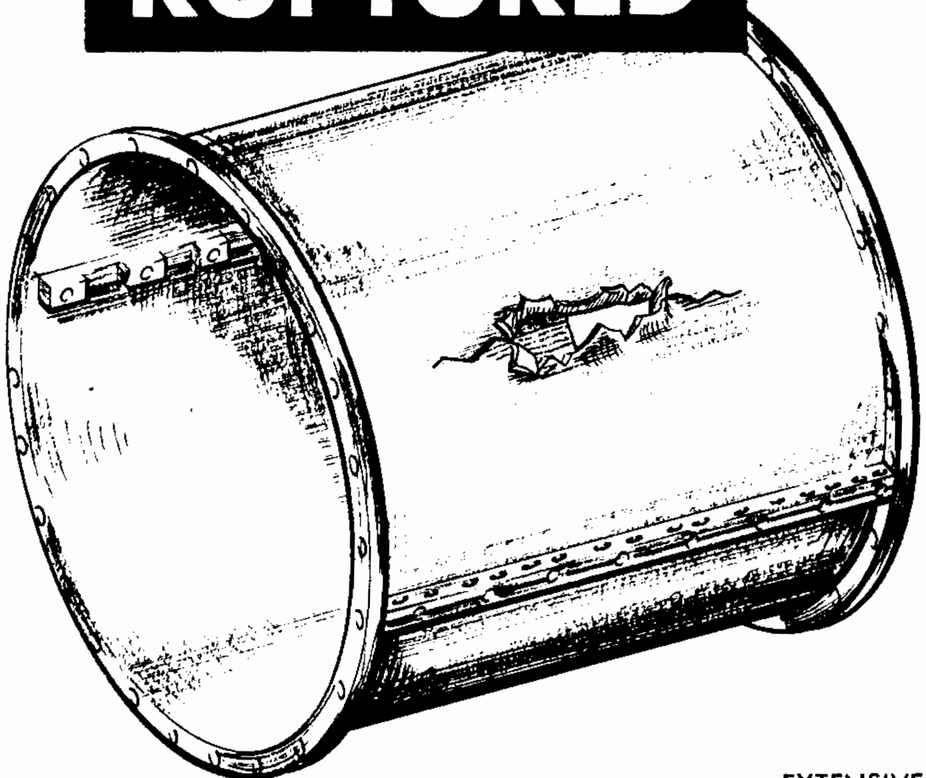
ROLLED-OVER

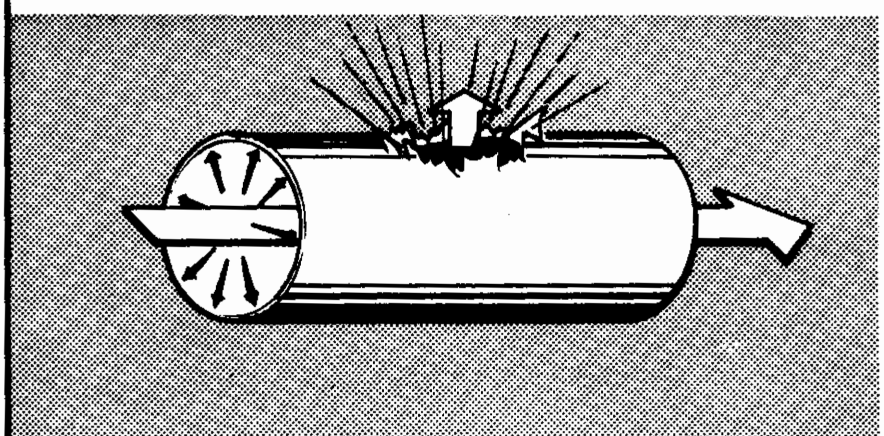
LIPPING OR ROUNDING OF A METAL EDGE



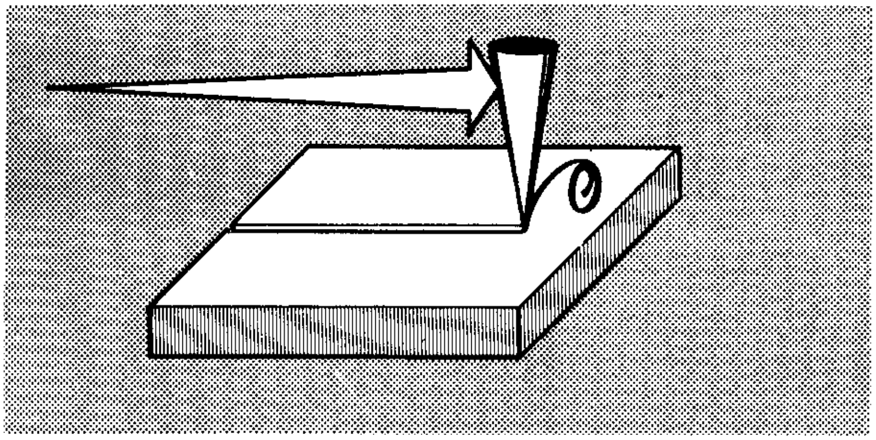


RUPTURED



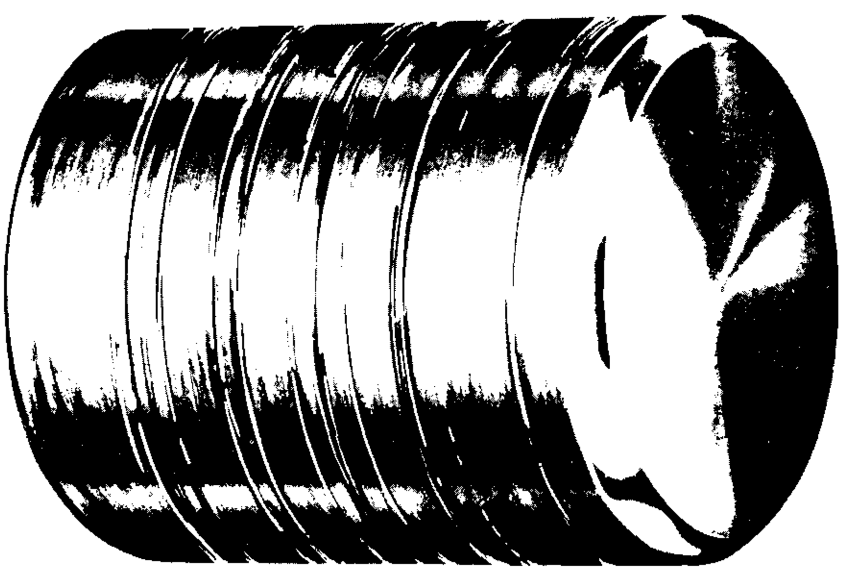


EXTENSIVE BREAKING APART OF MATERIAL USUALLY CAUSED BY HIGH STRESSES, DIFFERENTIAL PRESSURE, LOCALLY APPLIED FORCE OR ANY COMBINATION OF THESE. EXAMPLES: BURST BELLOWS, BLOWN CASING, ETC.

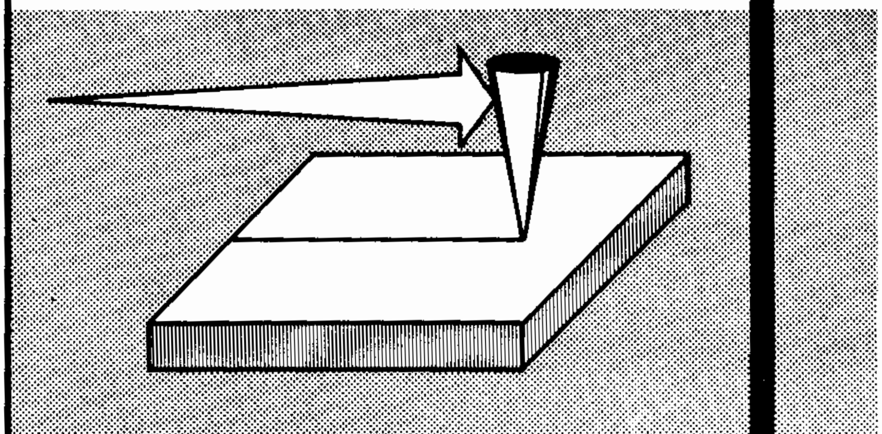


DEEP SCRATCH OR SCRATCHES MADE DURING PART OPERATION BY SHARP EDGES OF FOREIGN PARTICLES

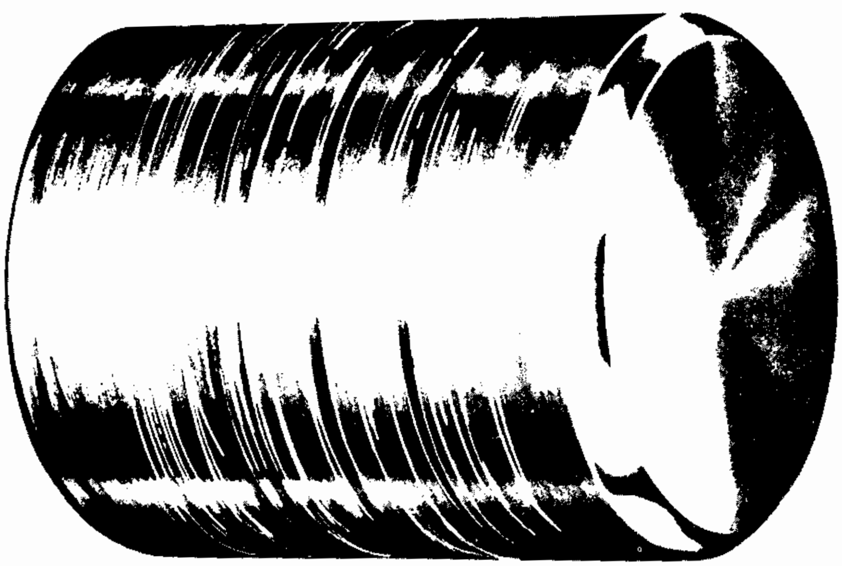
SCORED



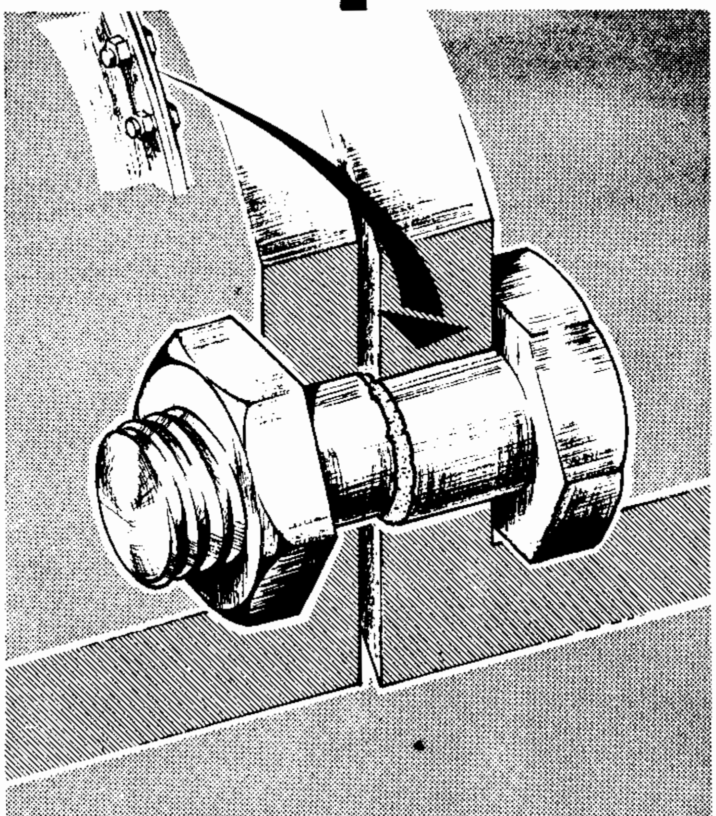
SCRATCHED



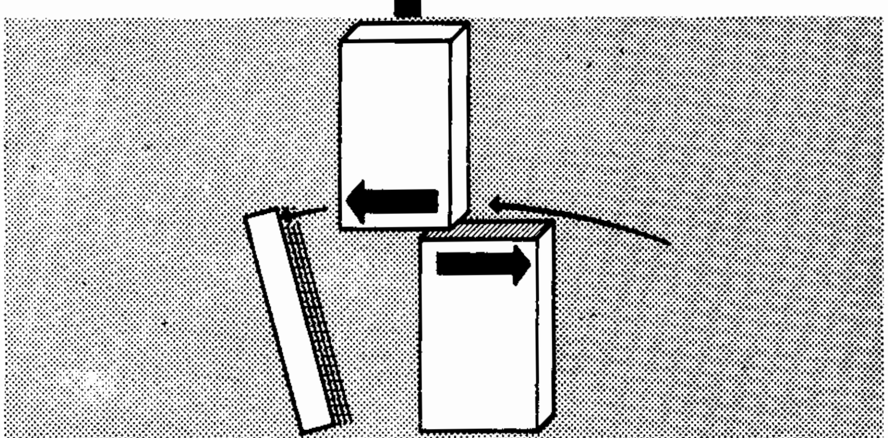
LIGHT NARROW, SHALLOW MARK OR MARKS CAUSED BY MOVEMENT OF A SHARP OBJECT OR PARTICLE ACROSS A SURFACE. MATERIAL IS DISPLACED, NOT REMOVED

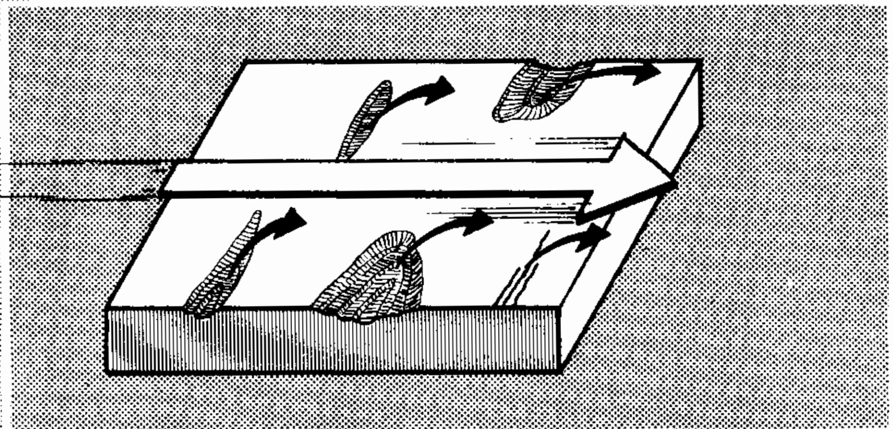


SHEARED

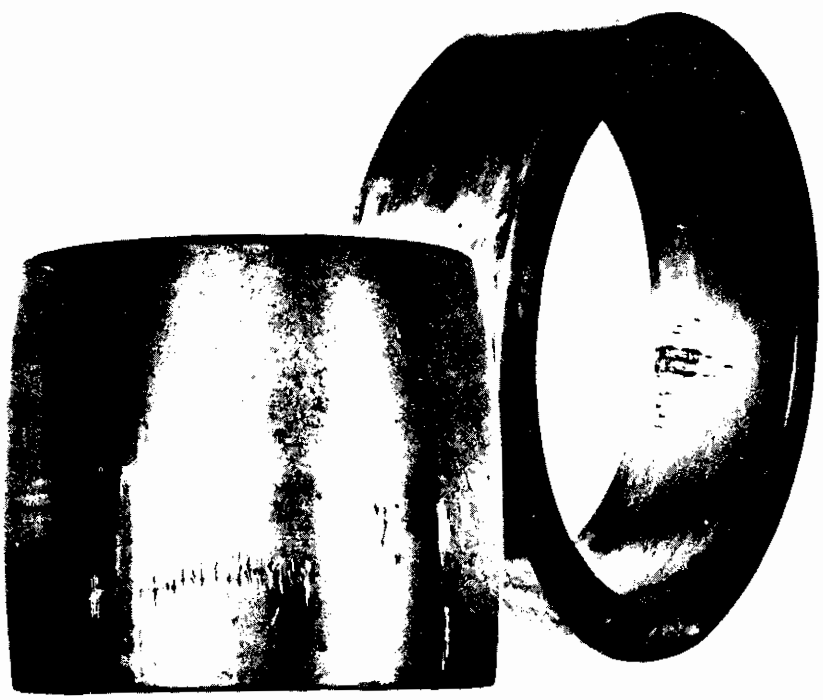


DIVIDING A BODY BY CUTTING ACTION, i.e., DIVISION OF A BODY SO AS TO CAUSE ITS PARTS TO SLIDE RELATIVE TO EACH OTHER IN A DIRECTION PARALLEL TO THEIR PLANE OF CONTACT



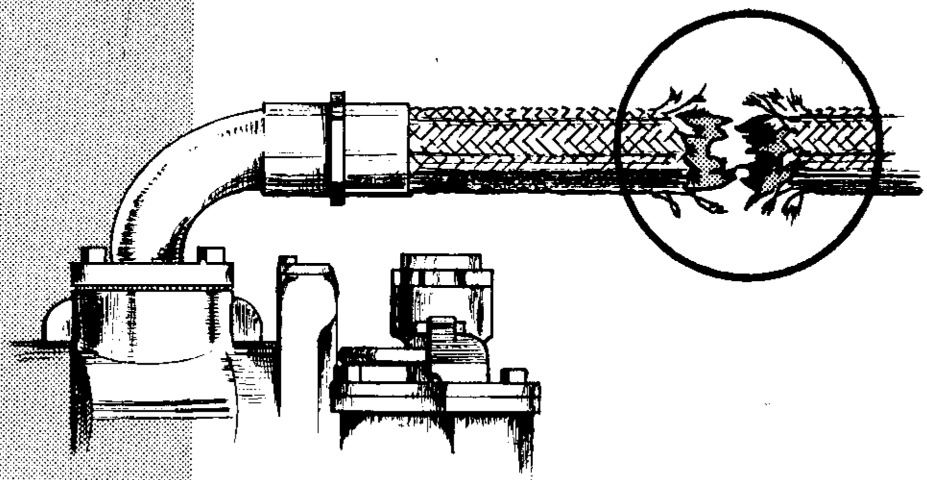


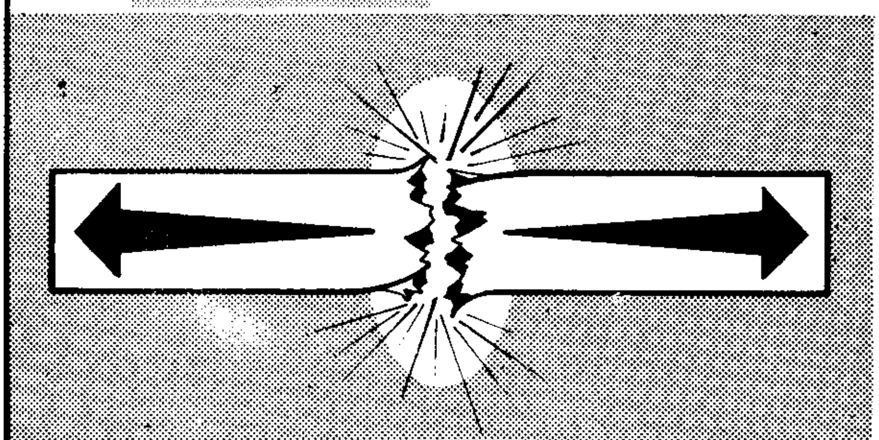
SPALLED



SHARPLY ROUGHENED AREA CHARACTERIZED BY PROGRESSIVE CHIPPING-AWAY OF SURFACE MATERIAL. (NOT TO BE CONFUSED WITH FLAKING). USUAL CAUSES ARE SURFACE CRACKS, INCLUSIONS OR ANY SIMILAR SURFACE INJURY CAUSING A PROGRESSIVE BREAKING AWAY OF THE SURFACE UNDER LOAD

TORN





SEPARATION
BY
PULLING
APART