PART C — AIRCRAFT AIRWORTHINESS INVESTIGATION

CHAPTER I

AIRWORTHINESS — GENERAL

1. Airworthiness Defined

Airworthiness of an aircraft must be defined before undertaking the investigation of an aircraft accident. *Airworthiness* is a relative concept, depending upon the aircraft in point and how that aircraft is utilized. For example, a particular aircraft may be "safe enough," airworthy for crop dusting, but not for carrying passengers across the Atlantic. The definition depends to some extent on who is defining *airworthiness concepts,* as they may be considered from different viewpoints.

Even though airworthiness may be variously defined, it is understood that the maximum reasonable airworthiness (safety) should be inherent, or designed into the aircraft.

A definition of airworthiness is: *the capability of an aircraft and its systems to operate in all environmental conditions for which they have been designed;* or stated simply, *aircraft fitness for flight.*

If airworthiness exists at a particular time, functional and mechanical integrity of the aircraft and its systems are necessary. Civil aircraft have been developed to a very high degree of airworthiness through careful and conservative **design**, through the use of **redundancy** of many essential elements, through **proper operation,** and through careful **maintenance**. The goal is to reduce the probability of failure of components, and to minimize the threat to airworthiness if a failure occurs. Some of the parameters making up airworthiness are discussed in the following sections.

2. Structures

The integrity of aircraft structural elements is a complex function of material properties, fabrication, design criteria, and environment. Failure is possible in one or a combination of modes. Some failure modes are random, such as single overloads and unique dynamic responses to unknown environments. Other modes such as fatigue, creep, and corrosion, are service induced over a period of time. Structural failure modes result from localized stress phenomena.

Failure may occur from a single overload on a part essentially undamaged by previous service. This means an unanticipated high load (such as a gust of extremely rare magnitude) or a high, but anticipated, load on a part of unanticipated weakness. This type of overload failure has been extremely rare and can be kept rare by careful design, manufacturing, quality control, and proper maintenance of aircraft.

Failure can occur, even under anticipated loads, in a part weakened through previous service life. Strength may decrease in several ways: cumulative fatigue damage, corrosion, stress corrosion, creep (at elevated temperatures), or any combination of these.

The consideration of fatigue is very complex in structural integrity evaluation. It is very difficult to estimate fatigue damage and remaining fatigue lifetime, and there are two quite different approaches toward design to prevent fatigue. One is the "safe life" approach which involves designing the primary structure with a high confidence level of nonfailure for a selected life, and includes plans to retire the structure or critical parts when this lifetime has been reached. This approach demands very exact design with quite accurate predictions of flight conditions, or extremely detailed recording of local strain at all critical spots. In addition, allowance must be made for scatter in material fatigue strengths and for other uncertainties in design and fabrication. To circumvent these difficulties another philosophy recommends a "fail safe" attempt to build the structure in such a way that its strength is not catastrophically reduced by noticeable fatigue cracks, coupled with scheduled inspection for cracks or other evidence of damage.

This approach still requires appropriate design to achieve adequate "safe life", to avoid statistical uncertainties in inspection, and to avoid excessive weight and unreasonable replacement costs. Actually, either approach may be limiting, and some aspect of each is needed.

The total objective in structural airworthiness is to establish and maintain the design integrity of the aircraft structure during its operational life.

3. Propulsion

The powerplant parameters of interest vary with different types of aircraft and engines. The powerplant differs from many other systems in the multiengine aircraft in that to some extent it is *fail safe* without redundance. Failure of an engine, and in some flight conditions two engines of four-engine aircraft, results in a degradation of aircraft performance, but flight is possible. This lack of redundancy, however, makes powerplant performance degradation or failure potentially critical under some operating conditions and emphasizes the importance of powerplant airworthiness investigation.

Aircraft engines have been developed to a high level of operational refinement in a relatively short period of time. The factors limiting the performance, reliability, and service life of these engines are of interest.

Proper performance of a powerplant is dependent on two aspects of the engine: *thermodynamic performance,* and the *mechanical integrity* of the powerplant. These are interrelated since any deterioration of the mechanical integrity of the powerplant ultimately results in degradation of the thermodynamic performance, and operation outside thermodynamic limits is a frequent cause of mechanical failures. However, mechanical difficulties frequently must reach the level of failure before a significant decrease in the thermodynamic performance is noted, and mechanical failures resulting from exceeding thermodynamic limits are frequently delayed.

4. Aircraft Systems

Each system in the aircraft is designed for a specific function necessary to the operation of the aircraft. If the loss of a system's function results in a catastrophic incident, or even in marginal aircraft operation, the system is certainly related to airworthiness.

Functional integrity is the ability of a system or a system component to perform as designed. The probability that it will continue to have functional integrity is evidence of its reliability. Reliability specifications for a given class of components are determined statistically from test data and/or operational data on a number of samples. The actual reliability of a given component may differ considerably from the statistical quotations, however. For this reason, and because the statistical reliability itself may not be deemed satisfactory, the redundant-design concept has become a standardized approach to provide high levels of overall mechanical system reliability in aircraft.

The performance of any system component depends on the behavior of its materials in the environment to which they are subjected. Material changes can occur in the form of corrosion, wear, deformation, and even fracture. (The loads which cause stresses within the material are classed as environment here.) Whether these material changes affect performance or not depends on the functional task of the device in question. For simple load-bearing members, fracture or deformation may be the only serious material change. If the part is an essential piece in rotating machinery, deformation, wear, and fracture could all be significant and might lead to a seizure with complete loss of performance.

The performance or functional integrity of many mechanical devices is sometimes not affected by material changes within the device. If the pushrod operating a critical flight-control surface fractures, or is deformed and seizes, a loss of mechanical integrity occurs. Simultaneously, functional integrity is lost.

This is not true of all mechanical devices. Some, especially those with a number of mechanical parts, can experience material failures of one or more parts and yet maintain part or all of their functional integrity. Thus, a hydraulic pump may experience scored bearings, worn pistons, or other types of failure and continue to perform, possibly at lower performance levels. This effect of material changes on performance is often referred to as *performance degradation.*

5. Manufacturing, Modification, and Maintenance

The importance of proper manufacturing, modification, and maintenance of aircraft in contributing to airworthiness is obvious, but should not be taken for granted. Here lies the responsibility for insuring that the aircraft engines, systems, and equipment conform to the design, specifications, materials, processes, and adequate construction, modification, and maintenance techniques. Continuous quality control helps to insure the airworthiness designed into the aircraft.

6. Flight Characteristics

The *flight parameter* of an aircraft is that parameter most apparent to all concerned in the operation of an aircraft, for it is the parameter which is physically observed and felt.

Stability is the inherent quality of an aircraft to correct for any upsetting condition; it is a primary aircraft design characteristic. It should be noted that there are two distinct types of stability. *Static stability* is the initial tendency that the aircraft displays after being disturbed from equilibrium, and *dynamic stability* is the overall tendency that an aircraft displays after being disturbed from equilibrium. The stability of an aircraft is a very important factor in its flight regime, therefore, based on the type of aircraft and its intended operation, minimum levels of acceptable stability are established.

Another major parameter of flight is *performance.* Performance can be defined as how fast, how high, how quickly, how much distance, and with what payload an aircraft can operate under specific environmental circumstances.

Part C — AIRCRAFT AIRWORTHINESS INVESTIGATION

Chapter II

FLIGHT CHARACTERISTICS

1. Aerodynamics and Performance

In the investigation of aircraft accidents the investigator is confronted with the necessity for some knowledge in a variety of subject areas, such as the human factors, the operations, the structures, the systems, and the various other phases of the investigation. Aerodynamics is a consideration in practically all aircraft accident investigations since it is the medium in which the aircraft operates. An understanding of the principles of aerodynamics provides the investigator an excellent foundation for a clear understanding of what may have happened. If it is necessary to obtain assistance, such understanding will give the investigator a background knowledge so that he can be effective, and specific in his requirements.

The study of aerodynamics requires a good background in mathematics through and above calculus. The theory of aerodynamics found in a text has no value unless we build the airplane and fly it, and thus determine the validity of the theory. The purpose of this text is not to stress mathematics, but to present aerodynamics in layman's language. The objective is to explain some of the fundamentals which have been utilized for a number of years but which have not been discussed freely among pilots and aircraft accident investigators.

1.1. Characteristics of the Atmosphere

Since the aircraft operates in the medium of the atmosphere, it is logical to discuss some of the characteristics or properties of the atmosphere. In various texts on this subject it is noted that the composition of the earth's atmosphere by volume is approximately 78% nitrogen, 21% oxygen, and 1% other gases, which may be water vapor, argon, carbon dioxide, etc. For all practical purposes, the atmosphere maintains a uniform mixture of these gases. Three basic areas to study relative to these properties of an air mass are pressure, temperature, and density. The aircraft operates under the laws of aerodynamics, and when these three properties of the atmosphere change for any reason, an aircraft will have a corresponding response change. It is the lack of understanding (on the part of certain flight personnel) of these various changes that produces an alarming and continuous accident rate.

1.1.1. Pressure

Pressure from a physical aspect is not difficult to visualize. It is the weight of the atmosphere above the level where the pressure is measured; 100% of the pressure is obtained at sea level, approximately 75% of the pressure at 8,000 ft., approximately 50% at 18,000 ft., and 25% of the sea level pressure at 33,000 ft. All of these percentages are approximate, since the percentage varies with the temperature and the location on the earth's surface. The investigator must be conversant with a number of pressures. As an example, dealing with aerodynamics and pressure in general, the term *14.7 lbs. per sq. in.* is used as a standard pressure under standard conditions at sea level; *2,116 lbs. per sq. ft.* is an aerodynamics term of standard pressure at sea level. The first is pressure in lbs. per sq. inch and the other lbs. per sq. foot. In the operations phase of accident investigation the investigator will be dealing with pressure measured in inches of mercury, standard pressure being 29.92 *in. of mercury.*

The term *millibars of pressure* will be utilized with *1013.2 mb.* as a standard pressure. The operation of air carrier aircraft, as well as general aviation high altitude jet aircraft, has made the millibar pressure more significant to the investigator. In dealing with medical personnel, the investigator will be using pressure in the terms of millimeters of mercury, with 760 millimeters of mercury the standard pressure. This is associated with the partial pressures available to the lungs.

1.1.2. Pressure Ratios

The basic pressure of the atmosphere is relatively easy to understand, but as soon as the investigator encounters a problem involving pressure and associated technical data he will encounter the term *pressure-ratio.* This means the comparison of whatever pressure the situation involves with what it would be at sea level under standard temperature and pressure conditions. Pressure ratio is designated by the Greek letter delta (δ), and is equal to the pressure (P) under consideration divided by the pressure (P_o) at sea level under standard temperature pressure conditions (P/P_o). An important point to remember is that the pressure specified as 14.7 lb. per sq. inch, or 29.92 in. Hg., is measured from absolute zero pressure.

1.1.3. Temperature

Temperature is a major factor in the cause of many aircraft accidents since it varies the pressure and the density of the atmosphere. When thinking of temperature, one usually visualizes a thermometer of one type or another, and therefore it is important to discuss the two thermometers in common use to get a picture of change in temperature. Considering the Fahrenheit thermometer first, it indicates that water freezes at 32° and boils at 212° under standard conditions. There is also the 0° mark which is 32 units below the freezing point of water and really has no significance. Study of this thermometer found it to be somewhat unscientific, and for this reason the centigrade thermometer was developed. The centigrade thermometer is based on the freezing and boiling points of water and as the word centigrade connotes, it is divided into 100 units between these limits of the mercury displacement.

To be of use from a scientific standpoint, the temperature must be measured from absolute zero. By this is meant zero molecular motion, since temperature and molecular motion are synonymous.

Problems in aerodynamics, aircraft performance, etc., deal with temperature ratios and these ratios deal with absolute temperature. They are normally noted with the capital T.

To clearly understand absolute temperature in both the centigrade and Fahrenheit scales, reference **Fig. C II-1.**

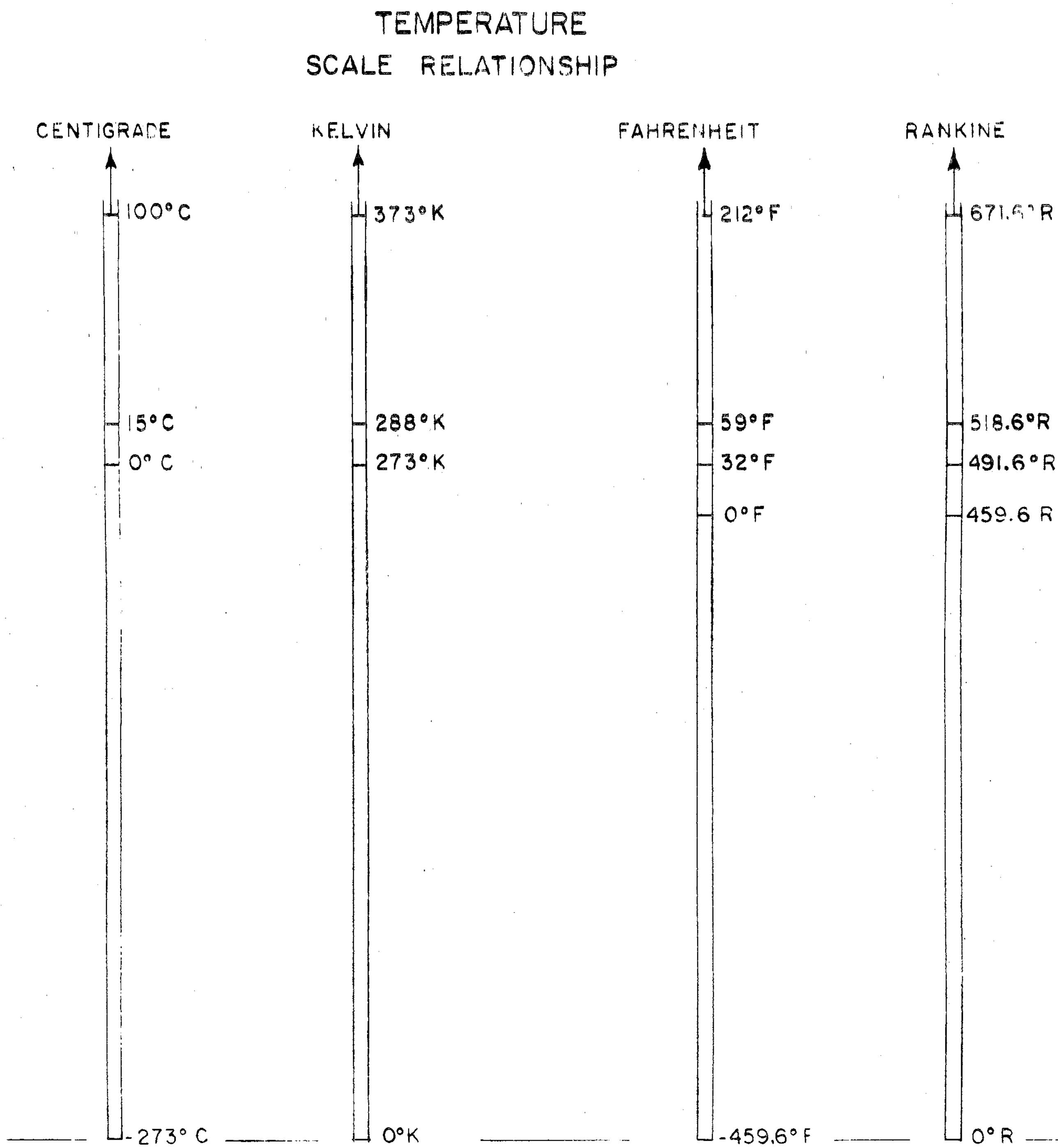
In referencing zero degrees on the centigrade scale, it can be noted that absolute zero is 273°C below this point. In referring to the Kelvin scale, it can be seen that the scale is merely zero referenced at absolute zero. For familiarity purposes, note that 0°C or freezing is 273°Kelvin, a standard day 15°C is 288° Kelvin, and the boiling point of water is 100°C and 373°Kelvin. The Fahrenheit scale (Rankine for absolute temperature) can be analyzed in a similar manner. This subject is discussed since technical reports from government engineering agencies, manufacturers, and operators will refer to temperature in degrees Kelvin and Rankine.

1.1.4. Temperature Ratios

As pressure is expressed in ratios, temperature can be expressed in a similar manner. We are interested not merely in one temperature, but the temperature in comparison to what it would be under standard conditions, therefore, this results in the term *temperature ratio.* Temperature ratio utilizes the Greek letter theta (θ) and is equal to T/T_o. It is very important to understand that this must be absolute temperature. The investigator should understand that if a temperature of 50°C is increased to 100°C this is not a 100% increase in temperature, but rather, a 16% increase, as far as absolute temperature measurement is concerned.

1.1.5. Density

The third factor in the atmosphere is density, basically, the mass of a given body of air. The



Temperature — Scale Relationship

Figure C II-1

aircraft flies under a law of Newton that *Force equals Mass x Acceleration*. The air is accelerated downward by the wing and this action develops a force. It is this work on a given mass M which results in lift. The density of the atmosphere is measured in *slugs per cubic foot*. In the layman's terminology, it is the weight of a cubic foot of air divided by the acceleration of gravity, which is approximately 32.2 *ft./sec./sec.* Standard density at sea level is *0.002378 slugs/cubic foot.*

1.1.6. Density Ratios

Density, as in temperature and pressure, is normally expressed in terms of density ratios; in other words, the density of the atmosphere in any given altitude in comparison to what it is at sea level under standard temperature pressure conditions. The Greek letter sigma (σ) signifies density ratio. Density has been assigned the Greek letter rho (ρ), and therefore **density ratio sigma is equal to rho at altitude divided by rho subzero** (ρ/ρ_o) at sea level under standard temperature pressure conditions. A point worthy of mentioning is that the pressure ratio curve and the density ratio curve are not the same curve when plotted vs. altitude. This means that at an altitude of approximately 22,000 ft. the density is only 50% of what it is at sea level, while at 18,000 ft. the pressure is approximately 50% of what it is at sea level.

In a review of the three characteristics of the atmosphere it becomes quite obvious why a number of accidents occur as a result of certain density altitude conditions. Since an increase in altitude will decrease the pressure and the resultant density, and an increase in temperature will also decrease the density, one can visualize that a high temperature - high altitude takeoff will gravely affect the performance characteristics of an aircraft.

The effect of density can be clearly understood if the above mentioned formula is analyzed for lift; that is, *Force equals Mass x Acceleration.* We must have a given force, which is lift, to fly the aircraft. The mass of air is accelerated downward to generate this lift. If the density of the air is decreased a higher rate of acceleration is required to maintain the same lift. This requirement then means that the engine must put out more horsepower to compensate for the less dense air. Or, in other words, a greater velocity is required to accelerate the lighter mass. The problem becomes more complex at the higher altitude density condition, for the engine capability becomes less at a time when the aircraft is demanding more.

1.2. Altitudes

The investigator will be constantly confronted by problems involving altitudes, such as weather data, witness information, field elevations, cabin pressurization, etc.

1.2.1. Absolute Altitude Measurement

Absolute altitude is the altitude above the terrain or sea. It is the altitude which the investigator needs to know, but finds difficult to actually determine, unless there are scars, such as on an obstruction, or a flight recorder, to give some clue.

1.2.2. Pressure Altitude

The pressure altitude is the altitude as indicated on the altimeter when the altimeter is set to standard conditions of 29.92″ of mercury. This is the altitude used in accident investigation to determine the density altitude.

1.2.3. Density Altitude

Density altitude, which may be considered as the working altitude of the aircraft, is obviously the altitude that must be computed or estimated by the investigator. Density altitude is pressure altitude corrected for temperature, a correction made by the use of a computer or a density altitude chart.

1.3. Airspeeds

There are five basic airspeeds with which the investigator should be somewhat familiar.

1.3.1. Indicated Airspeed

Indicated airspeed is the needle reading on the airspeed indicator. In most cases this is very different from the actual airspeed of the aircraft through the air.

1.3.2. True Indicated Airspeed

True indicated airspeed, **TIAS**, is the indicated airspeed as mentioned previously, corrected for errors within the instrument itself.

1.3.3. Calibrated Airspeed

Calibrated airspeed is true indicated airspeed corrected for position error. By position it is meant the position or location of the pitot static system. This normally requires an airspeed placard in an aircraft, since under great changes of attitude considerable error will be introduced into the instrument reading. It is desirable that the minimum error in this instrument reading occur in the vicinity of the approach speed of the aircraft.

1.3.4. Equivalent Airspeed

Equivalent airspeed is a calibrated airspeed corrected for the compressibility effect of the air. This is a function of highspeed flight at airspeeds in the vicinity of 300 knots and greater. The air becomes compressible, no longer acts as a noncompressible fluid, and therefore the airspeed indicator reads high. The correction for compressibility is always *minus*. In other words, it is always subtracted from the instrument reading, whereas the areas of the instrument error and the pitot static position error may be plus or minus.

1.3.5. True Airspeed

True airspeed reading is what is desired for navigational purposes. It is the equivalent airspeed corrected for density. An airspeed indicator is a pressure instrument which measures the pressure differential between the ambient air pressure (which is referred to as static pressure) and a dynamic pressure which is the pressure of motion. The static pressure is a measure of the air above the instrument, and the dynamic pressure is a function of *the square of the velocity times ½ the density of air (rho)*. This term *rho over 2, V squared,* is referred to as dynamic pressure or "q" in aerodynamics ($q = \rho/2\ V^2$).

1.3.6. Mach Number

In the area of motion or velocity of aircraft through the air we will consider the mach number or the mach meter as mounted on the instrument panel. The definition of mach number is the *ratio of the true air velocity of an aircraft divided by the ambient velocity of sound.* In other words, *M is equal to V over a* ($M = V/a$). The ambient velocity of sound changes with temperature. The value a is equal to **a subzero,** which is the *velocity of sound at sea level times the square root of theta* ($a = a_o\sqrt{\theta}$). Theta is the temperature ratio in absolute temperature and one can see from this formula that the mach number is not a function of pressure or altitude, but a function of temperature only. The point to keep in mind here is that temperature possesses a relationship to pressure and density. The velocity of sound at sea level under standard conditions is 661 knots or 761 mph.

1.3.7. Angle of Attack Indicator

An angle of attack indicator is a device which allows the pilot to read the angle between the longitudinal axis of the aircraft, or the chord line, and the free stream velocity (relative wind). This device has been used successfully on military aircraft, some executive jet aircraft, and on certain air carrier type aircraft.

This instrument is best explained in conjunction with the airspeed indicator, since the airspeed indicator is an indirect angle of attack indicator. The angle of attack indicator is a direct reading instrument and records only one basic function, that of angle. The airspeed indicating system is plagued with mechanical and environmental errors. As an example of an indirect reading instrument, the airspeed indicator will, under a given weight and environmental condition, represent some unknown angle of attack at a given airspeed reading. If the weight, temperature, etc., are changed, the angle of attack could change with no change in airspeed indication.

Another example is the lift over drag ratio of an aircraft and gliding distance. L/D maximum is a shape characteristic of an aircraft and it occurs at some specific angle of attack. L/D max. is independent of weight, and if the aircraft is at maximum gross weight L/D max. occurs at some indicated airspeed. If the same aircraft is very light, L/D max. occurs at some other airspeed. In utilizing an angle of attack indicator it is possible to place the aircraft in a specific flight regime.

There are other uses for the angle of attack indicator besides angle indication. It may be used as the activator of a stall warner and/or a stick shaker. It may also be utilized as a stick pusher sensing device.

1.4. The Wing

The wing area has been separated into individual components since the wing is such an important factor in the design of an aircraft. Basically, the wing descends the aircraft, climbs the aircraft, and turns the aircraft. It is therefore highly desirable for accident investigators to have a thorough understanding of the development of the aircraft wing.

1.4.1. Nomenclature of Parts and Shapes

The *span* of an aircraft wing in this country is measured from tip to tip regardless of the plan form of the wing, whether it be straight wing, elliptical, or sweptwing. This, as we will find later, is an important aerodynamic factor in characteristics and performance.

The *chord* of the wing is a line from the nose of the airfoil section to the aft tip of the section. The letter which represents *wing span* is small b, and the letter representing *chord* is small c. The dihedral angle is by definition an angle between two planes, which in this case would be the right and left-hand wing surfaces of the aircraft. Normally, dihedral is positive, and the dihedral angle is measured from a lateral plane to a specific plane of the wing. Dihedral is primarily used for purposes of lateral stability. Another term is cathedral or antihedral. A number of aircraft designers use this configuration for a specific purpose. Basically opposite to dihedral, the antihedral wing tips are lower than the root sections.

The *angle of incidence* is the angle between the longitudinal axis of the aircraft or the fuselage and the chord line of the wing. This normally is a very small angle, and may be plus or minus, depending on the characteristics of the airfoil sections used in the wing. *Decalage* is an angle involved in the two-wing or biplane type aircraft, and it represents the difference in angle of incidence between the top and lower wing. In the biplane the bottom wing is rather inefficient, and therefore it is mounted on the aircraft at a higher angle of incidence than the top wing. These two angles, *incidence and decalage*, may be of interest in accident work involving newly certificated aircraft, such as aircraft of an experimental category or agricultural aircraft. The determination of the balancing airfoil or mean aerodynamic chord of a biplane is somewhat more complicated than that of a monoplane.

A design characteristic of a wing is called *wash-out*, and this in turn will be followed by the term *wash-in*. The term *wash-out*, which is very common in aircraft design, is in actuality an aerodynamic twist of the wing. In layman's language, the wing is twisted to give a different angle of attack at the tip of the wing than that at the root section. The way to differentiate beween wash-out and wash-in is to associate wash-in with an increase in angle of attack towards the tip. Very few aircraft have ever used this system of wash-in at the tip, but it is very common to have the tip washed-out, where the angle of attack will be less at the tip than it is at the root section. The purpose of this is to achieve a better stall characteristic of the wing, where some area inboard on the wing stalls before the tip section to prevent adverse rolling in the advanced stall state.

Another shape characteristic is called *taper ratio* and involves the ratio of the tip chord to the root chord. We can say that a wing has a higher taper ratio when the tip chord is small in comparison to the root chord. Generally speaking, wings with a high taper ratio have rather undesirable stall characteristics because the tip section tends to stall ahead of the root section. There is a definite reason for this which will be discussed later when the subject of Reynolds number is explained.

Another item relative to wing design and shape is *aspect ratio.* This is an extremely important characteristic of the wing which should be thoroughly understood by the student of aerodynamics and the air safety investigator. In the subject of aspect ratio refer to the first two items under discussion of the wing, the wing span and the wing chord. The basic formula for aspect ratio is the *span small b divided by the chord small c.* Generally speaking, wings are of an aspect ratio of approximately 6 to 1. One of the reasons for this is that most of the wind tunnel research by the National Advisory Committee on Aeronautics (NACA) was with model wings of an aspect ratio of 6. It is found from actual flight experience that a wing with an aspect ratio of 6 displays rather good characteristics. The formula for aspect ratio, *AR equals small b divided by c,* is the physical relationship between the span and the chord that must be visualized to understand what aspect ratio actually is. The general term in aerodynamics is this basic formula of *b over c multiplied by b over b* which, of course, has a value of one. And in studying this equation we have in the numerator *b x b* which would be *b* squared, and in the denominator we have the span *b x the chord c,* which is actually the wing area S. So, in multiplying *b over c x b over b* we arrive at the equation of *span squared*

$$\mathrm{AR} = \frac{b}{c} \times \frac{b}{b} = \frac{b^2}{S}$$

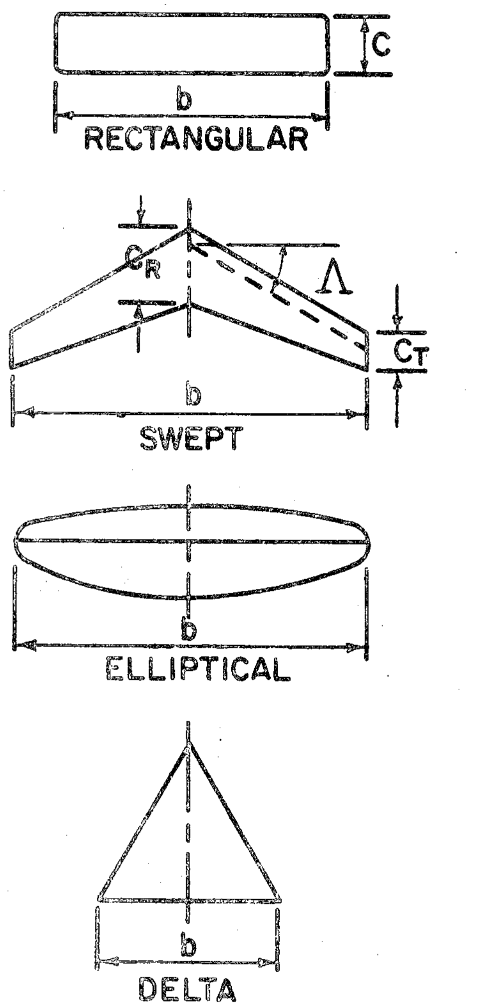
over area. This is the overall formula for aspect ratio, regardless of wing or airfoil shape.

Sweepback angle is a characteristic of an aircraft in which the wing is swept back so that the tip sections are aft of the root sections. In the study of the mean aerodynamic chord, we will find the reason why sweepback is measured in a certain manner in this country. The angle of sweepback is measured from the lateral axis aft and around to the quarter chord point of the wing, and not to the leading edge of the wing. There is a reason for this, since the design engineers are always concerned with the quarter chord wing line. This is a point where the aerodynamic characteristics of the wing are known and sweepback is measured from the lateral axis back to this quarter chord line of the wing. In line with the above, the subject of the mean aerodynamic chord is one of major importance. It is a subject not too well defined except in aerodynamic texts. If the aerodynamic characteristic of airfoils in association with the mean aerodynamic chord is understood, the implications of weight and balance, and stability and control become obvious.

1.4.2. The Mean Aerodynamic Chord

By definition, *mean aerodynamic chord* is an airfoil with a certain chord length which represents the aerodynamic characteristics of the complete wing as if it were a rectangular wing. In actual aerodynamic work it is a laborious task to determine the mean aerodynamic chord. It is determined by use of all the airfoil characteristics throughout the span of the wing, and it involves a location of this theoretical airfoil in a spanwise direction, in a vertical direction, and a fore and aft direction in the wing. Quite often it actually is not an exact location within the wing. It is also found that a centroid or center of gravity of a given wing area will give an airfoil section an actual chord, which is known as the **mean geometric chord.**

The mean geometric chord of a tapered wing can be determined in the following manner: A line is drawn from the midchord point of the root section to the midchord point of the tip section. This is the average chord or median line. The tip chord length is then added to the root chord length on either end. The root chord length is then added to the tip chord length, but it is added to the opposite end to which it had been added to the root chord. In other words, one segment to the trailing edge, and one segment to the leading edge. The external points of these two lines are connected by a diagonal line. The mean geometric chord (mean aerodynamic chord) is located where this diagonal line crosses the previously drawn median line. **Figure C II-2.**



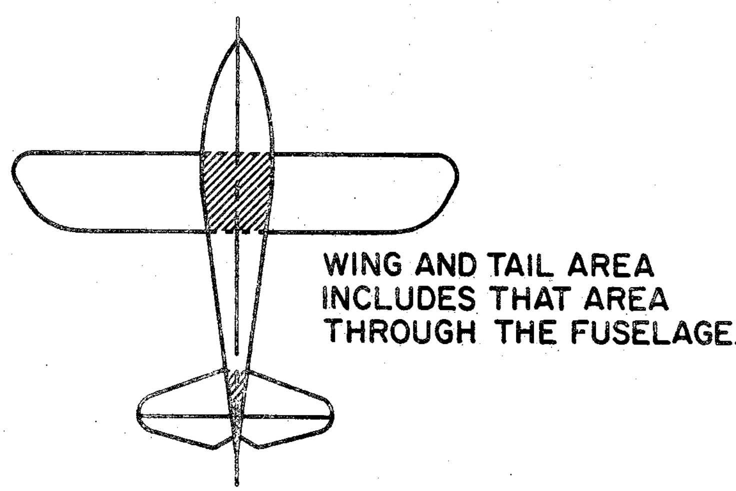
ASPECT RATIO

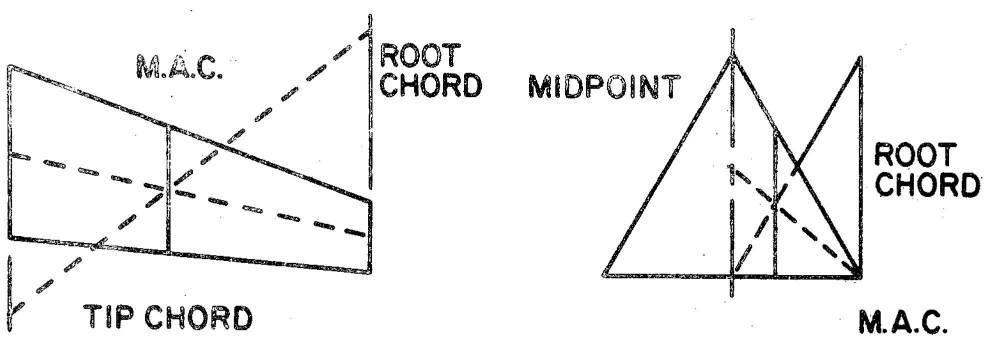
$$AR = \frac{b}{c} = \frac{b \times b}{c \times b} = \frac{b^2}{S}$$

S = SURFACE (AREA) = c x b

Λ = ANGLE OF SWEEP IN DEGREES.
IT IS MEASURED FROM THE LATERAL AXIS TO THE QUARTER CHORD LINE OF THE WING.

λ = TAPER RATIO = C_T/C_R





GEOMETRY OF DETERMINING THE MEAN GEOMETRIC CHORD WHICH IS BASICALLY EQUAL TO THE MEAN AERODYNAMIC CHORD.

Figure C II-2

1.5. The Airfoil

The airfoil, or the basic shape of the wing, is an extremely important factor in the design of an aircraft. Considerable time is spent in design, deciding which airfoil to use, or which series or family of airfoils, and how to arrange these airfoils to make a workable wing.

1.5.1. Shape, Development, and Nomenclature

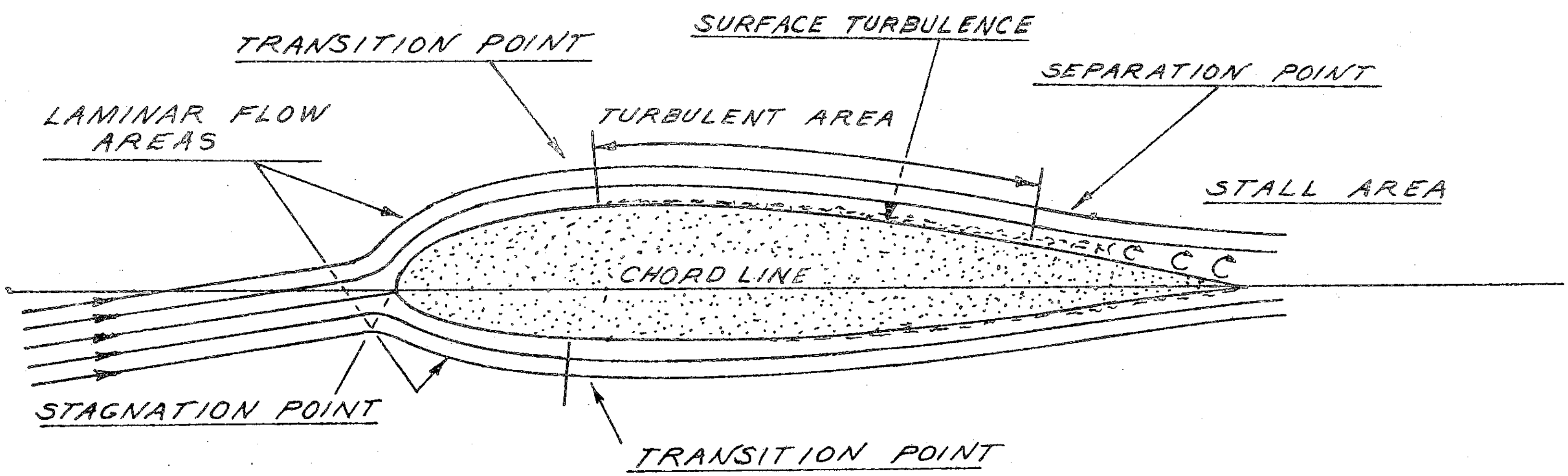
A clear picture of the working characteristics of an airfoil section is shown in Fig. C II-3.

1.5.2. Nomenclature (Aerodynamic)

A series of points on an airfoil affect the general characteristics of the shape. The knowledge of these various points on the airfoil and what they represent can be directly associated with accident causal factors. The first to consider is the stagnation point. Actually, there are two stagnation points on an airfoil, one on the leading edge, and one on the trailing edge. We shall discuss only the leading edge stagnation point. By definition, this is a point on the leading edge of an airfoil where the air flow velocity (free stream velocity, or relative wind) is brought to a velocity equal to zero. As we can see, referring to Fig. C II-3, the stream lines separate on either side of this point and flow below and above the wing. As the angle of attack of this airfoil is increased, the stagnation point moves downward. It is this movement of the stagnation point which prompted Dr. Leonard M. Greene to develop the stall warning indicator in such common use on general aviation aircraft. This is the small vane which protrudes from the leading edge of the wing. It is held down by a light spring pressure and by dynamic pressure. When the stagnation point reaches the area of the vane or below, it will move to an "up" position, and either activate a light, blow a horn, or both, in the cockpit to indicate an approach to a stall. One manufacturer utilizes a hole in the leading edge of the wing, so positioned that it will produce an audible whistle or vibration as the stagnation point approaches this particular area. This type of stall warner has an advantage over the other type because no electrical equipment is required. It must be pointed out here that the stall warning devices used on general aviation aircraft do not *prevent* stalls, they merely warn of an approaching stall.

The next point to consider on an airfoil is shown in Fig. C II-3. This is the *transition point*. The flow of air preceding the leading edge of the wing, and that which passes up over the top and bottom leading edge curvature of the wing, is *laminar*. By this it is meant that each stream line is parallel to the other, and remains so down to the surface of the wing. It is desirable to keep this flow laminar as far aft on the airfoil as possible. In examining the metal aircraft wing, note that the surface from the leading edge to a point approximately one-fourth to one-third aft of the chord is flush riveted, whereas the rear portion of the wing surfaces on top and bottom may be riveted with brazier head rivets which protrude above the skin of the wing. In the transition area, or transition point, the airflow changes from laminar flow to turbulent flow next to the surface. This is a very thin layer of turbulence which increases in thickness as it approaches the wing section trailing edge. One might say at this point that if a person desired all possible speed from an aircraft he would wax the leading edge portion of the wing. Beyond this point, additional waxing and polishing would be useless, since the surface is working in a thin layer of turbulence.

Proceeding aft on the upper wing surface this turbulent area gets thicker and thicker until finally another point known as the separation point is reached. This is the point where the airflow breaks away from the surface of the wing and thus initiates the stall. Note in examining Fig. C II-3 that as the stall progresses a reverse in flow occurs, since the air has been slowed along the surface. Aerodynamicists refer to this as deenergization of the boundary layer. In layman's language, it means that the surface of the skin has decreased the airflow velocity next to this surface. Visualize that if the surface of the wing is rough it will have the effect of rapidly deenergizing the air velocity next to the surface, and in turn, it will cause the separation point as well as the transition point to move forward.



Airfoil Airflow Characteristics
Figure C II-3.

This makes it easy to visualize why frost is so critical on an aircraft wing and horizontal tail. Frost on the surface has a very effective de-energizing characteristic on the airflow, and as a result, the transition and separation points move forward to produce a premature stall.

In referring to the leading edge area of the airfoil section it can be seen that it is desirable to maintain laminar flow as far aft as possible. As stated previously, this is the reason manufacturers normally use flush rivets in this particular area. Laminar flow airfoils are airfoils that maintain laminar flow over a large area. They were developed just prior to and during WW II. The P51 aircraft is an example of one of the early aircraft employing a laminar-flow type wing. The laminar-flow wing is also utilized on a number of the modern general aviation aircraft.

Procedure for development and testing of airfoils is worthy of a few comments, since rather an elaborate process is employed to develop the technical information on a given shape. Various shapes will act as airfoils; however, each shape requires accurate wind tunnel testing to determine the actual physical characteristic for reporting in a written, graphical form. Almost any airfoil shape will generate lift if enough power is available to obtain the required velocity. This is the origin of the oldtime statement of the 1930's, "I can fly a barn door if I have enough power." The modern airfoil, however, is considerably more sophisticated than the barn door.

The first item to cover in the airfoil is the *chord line*. This is the line drawn from a leading edge to the trailing edge, a reference line. This line is actually used to plot the airfoil, and it is also utilized to attach the wing to the aircraft. The angle between this line and the longitudinal axis of the aircraft, as previously stated, is the *angle of incidence*.

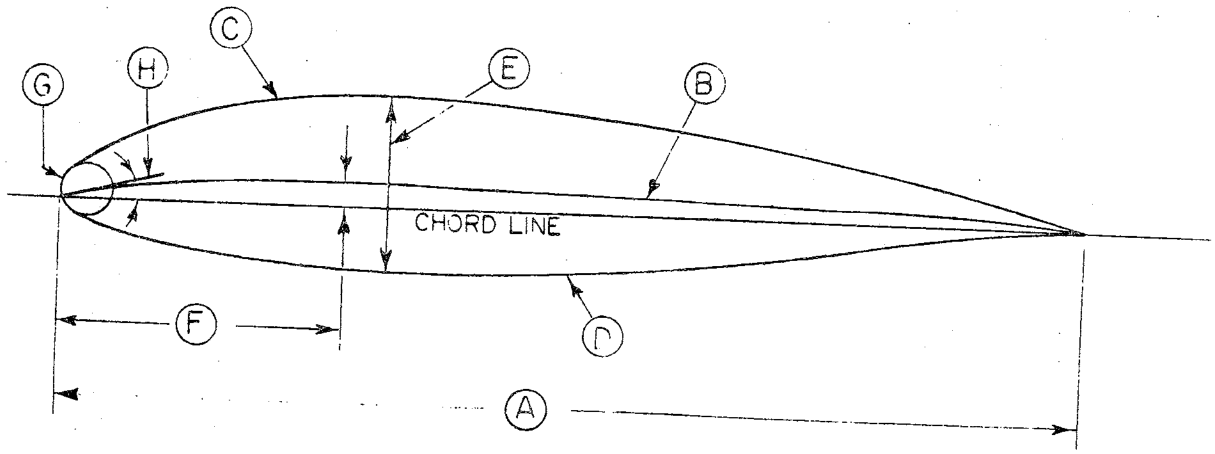
In the study of airfoil shapes we find a second line known as the *mean camber line*. This line is defined as the line which results from a plot of all points half way between the upper surface and the lower surface of the airfoil. This is the line used to mathematically develop an airfoil section. By this it is meant that equations can be written for the curves involved. It is necessary to refer to Fig. C II-4 to illustrate the chord line and a mean camber line.

In referring to this figure, note that in this particular illustration the mean camber line is above the chord line, and therefore this airfoil section has positive camber. Any time the mean camber line goes below the reference chord line, we have negative camber. We might also make the statement that a symmetrical airfoil would have the mean camber line and the chord line as the same line, and therefore the symmetrical airfoil has no camber. Since airfoils are developed from this mean camber line, airfoils series, or families of airfoils, can be explained. The numbering system for the five-digit airfoil section series is as follows. The first integer indicates the amount of camber in terms of the relative magnitude of the design lift coefficient — the design lift coefficient is thus three-halves of the first integer. The second and third integers together indicate the distance from the leading edge to the location of maximum camber. The distance in per cent of the chord is one-half the number represented by these two integers. The 23012 airfoil therefore has a design lift coefficient of 0.3 ($3/2 \times 0.2 = 0.3$), has its maximum camber at 15 per cent of the chord ($30/2 = 15$), and has a thickness of 12 per cent of the chord.

In the four-digit series airfoil section, the numbering system is based on the section geometry. The first integer indicates the maximum displacement of the mean camber line in per cent of the chord. The second integer indicates the location of the point of maximum camber in per cent of the chord. The 2412 airfoil section thus would have a 2 per cent camber located 40 per cent of the chord aft of the leading edge, and it is 12 per cent thick.

It is desirable in the construction of a cantilever wing to have it thicker at the root section than at the tip. The center airfoil, therefore, might be a 23015 or 23018 airfoil, meaning that it was 15 per cent or 18 per cent thick. This could taper out to middle areas where it might be a 23012 airfoil, and possibly go out to a 23009 at the tip section. There are many wing designs which have a different family of airfoils within the wing section itself, in other

AIRFOIL SECTION TERMINOLOGY



(A) CHORD
(B) MEAN CAMBER LINE
(C) TOP SURFACE
(D) BOTTOM SURFACE
(E) MAXIMUM THICKNESS
(F) POSITION OF MAXIMUM CAMBER
(G) NOSE RADIUS
(H) NOSE RADIUS LOCATION

THE SYMMETRICAL AND LAMINAR FLOW AIRFOILS ARE SHOWN IN OTHER FIGURES OF THIS CHAPTER.

Airfoil Section Terminology
Figure C II-4.

words, a different section at the tip than at the root. As a matter of fact, some of our jet transport aircraft have wing sections of different types in three individual areas. These are from the fuselage to the first outboard engine, a section between the engines, and the third section from the last engine outboard to the tip. The inboard section may actually be inverted to obtain a desired characteristic of the wing. A number of series of airfoils exist throughout the world; this discussion will be confined to NACA type airfoils.

Another series of interest is the NACA laminar-flow type airfoil such as the 65412. Each of these digits represents some technical information, as in the 23012 series, but this series is in no way related to the 23012 series. The digit 6 designates the fact that this is a laminar-flow airfoil. The second digit, 5, signifies the chordwise position of minimum pressure in tenths of the chord from the leading edge. The third digit, 4, represents the lift coefficient of 0.4 which is at a point of minimum drag of this section. In other words, this would be a design cruise lift coefficient and the last two numbers are the same as in the older 23012 series. This means that the airfoil is 12 per cent thick airfoil. To be specific, on a 100-inch chord the airfoil would be 12 inches thick. There are hundreds of airfoil shapes, some with a scientifically developed background, others are airfoils which carry the name of the person who developed the shape.

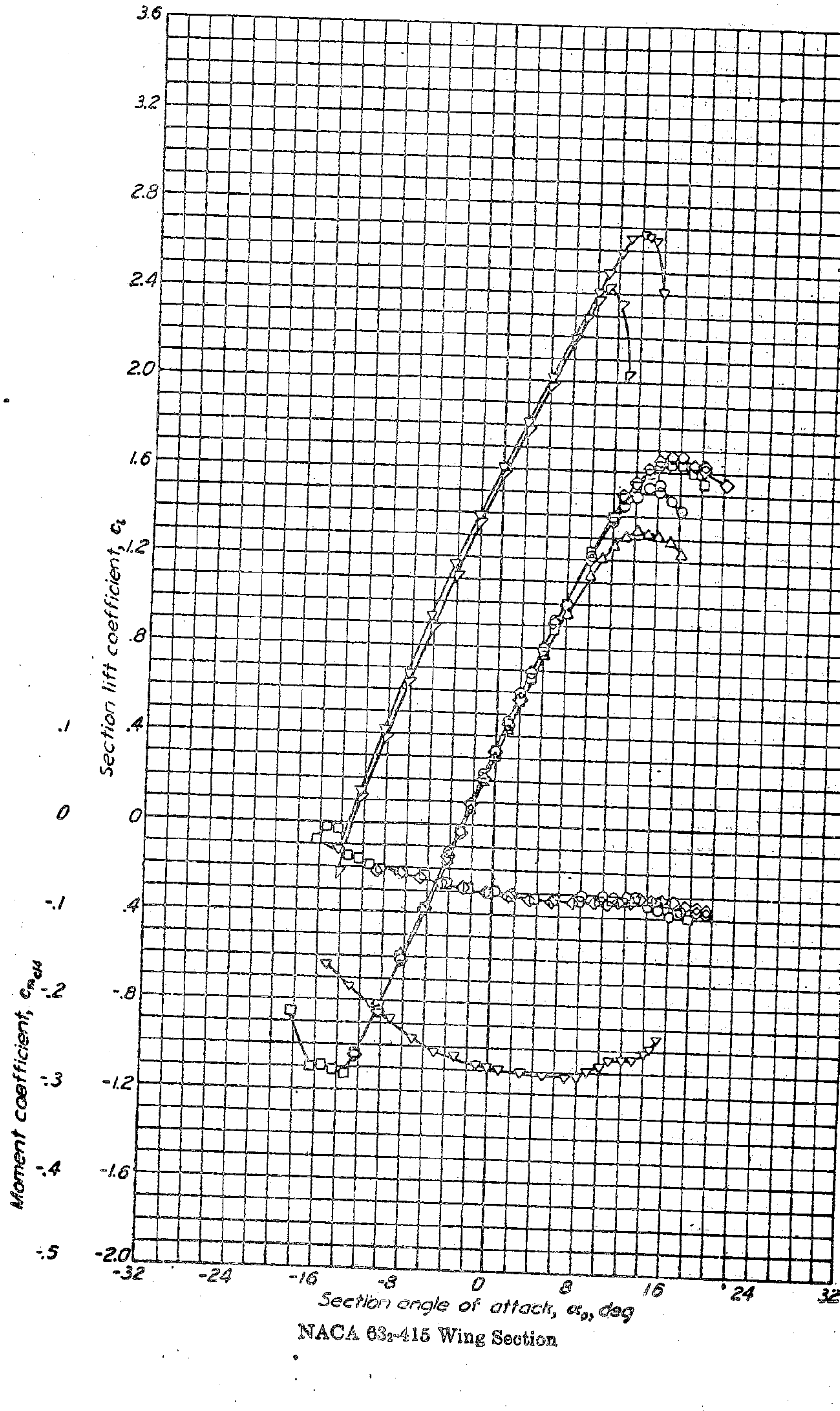
One of the most popular of these is the Clark Y series. Other types of airfoils used in the 1930's were the Gottingen, Eiffel, and RAF series. A more modern airfoil type is the laminar-flow series which came into common use during WW II. Note Fig. C II-5; these airfoils possess drag curves with what is known as a drag bucket. There is a small range of angle of attack or lift coefficient where the drag decreases appreciably. In order to utilize properly this laminar-flow airfoil, it is necessary to have adequate power to bring the airplane up into a designed speed range where we can utilize the lower lift coefficient, and operate in the drag bucket or low shape-drag range.

The *aerodynamic center* of the airfoil is a common term found in all aerodynamic books but one omitted from most pilot-type aerodynamic books as seemingly unimportant. Actually, it is an extremely important area on an airfoil, as will be noted in the discussion of weight and balance, stability and control, and inflight breakup under a high-speed flight condition.

If a cambered airfoil is tested in a wind tunnel, it will display a pitchdown or nosedown pitch characteristic through the range of maximum lift coefficient (which is a stall) to a low angle of attack, or zero-lift condition. This is referred to as a pitching moment, a characteristic of practically all airfoils except symmetrical, and some highly modified airfoils. This pitching moment characteristic was one reason for the development of the 23000 series discussed previously, the idea being to develop an airfoil with a minimum pitching moment.

A symmetrical airfoil has a zero pitching moment, and for this reason it is used almost universally in helicopter main rotor blades. (**Figure C II-6**) The aerodynamic center is an airfoil property which should be clearly understood. It is defined as the point on an airfoil about which the pitching moment coefficient is constant. It is a point on an airfoil which is approximately one quarter of the way back from the leading edge. One must also visualize that the center of pressure under a varying angle of attack moves fore and aft on the airfoil. It is well forward at a high angle of attack, and it is well aft at a low angle of attack. It is then a matter of taking moments around some point on the airfoil to determine the location of a point where the pitching moment is constant. In layman's language, this means that at the lower angles of attack the center of pressure is well aft on the airfoil, and the lift is of a relatively small value. Therefore, the moment arm from the aerodynamic center (which in this case is approximately the quarter chord point)to the relatively small lift force will be some fixed moment. If there is an increase in the angle of attack, the lift force will increase, and at the same time it will have moved forward on to the airfoil.

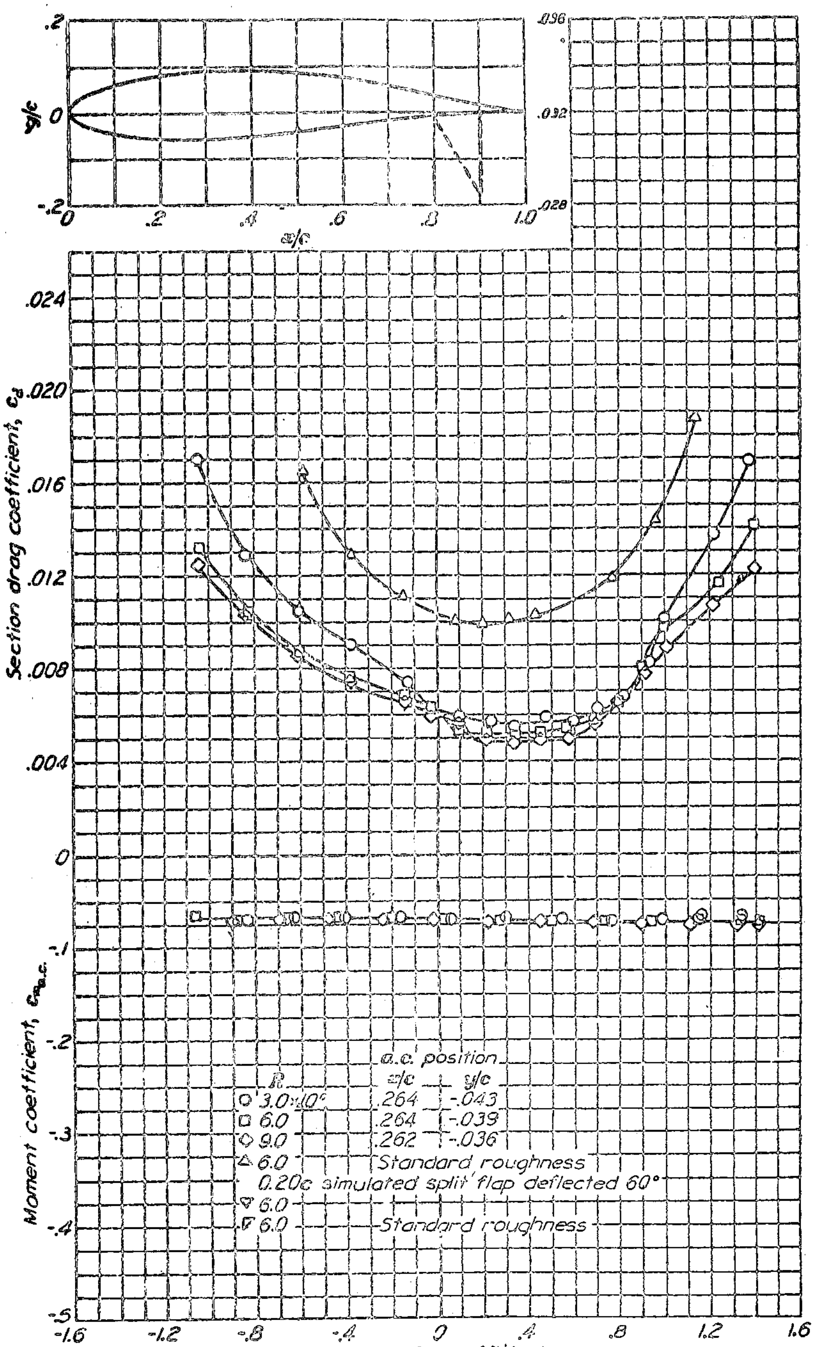
In generalities, if the force is decreased, the moment arm is longer, and if the force is increased, the moment arm is shorter. This is a



NACA 63_2-415 Wing Section

Airfoils

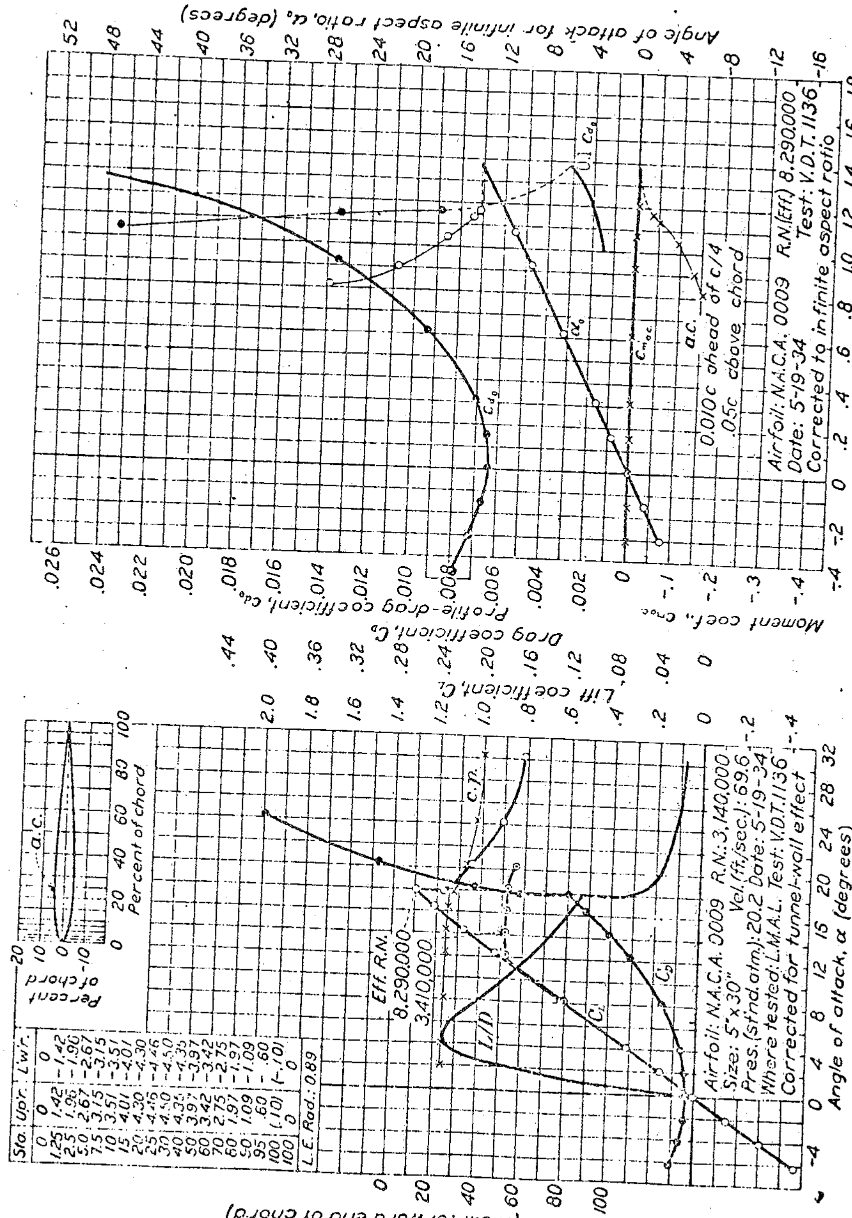
Figure C II-5a.



NACA 63₂-415 Wing Section (*Continued*)

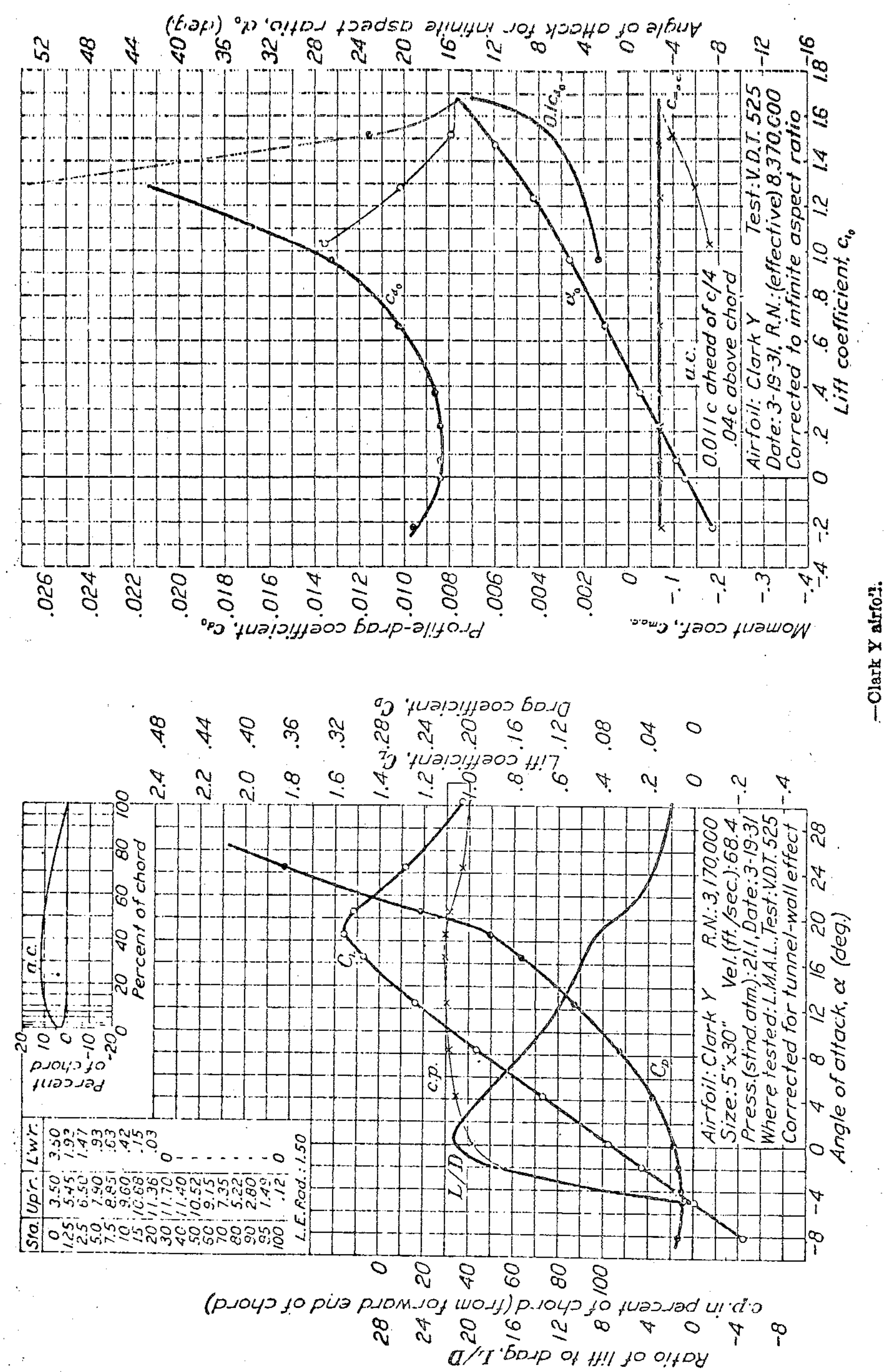
Airfoils

Figure C II-5b.



—N. A. C. A. 0009 airfoil.

Angle of Attack
Figure C II-6.



—Clark Y airfoil.

Airfoil Characteristics
Lift, Drag, and Moment

Figure C II-7.

physical presentation of the aerodynamic center. The aerodynamic center is a fixed point on an airfoil determined by wind tunnel tests, and for all practical purposes it is at the quarter chord point.

If one were to examine the data on a series of cambered airfoils, it would be found that the position is approximately 0.249 to 0.269 per cent of the chord. The above explanation should be visualized as a wind tunnel operation where the velocity is constant, and only the angle of attack is changed. It must be remembered that the pitching moment increases as the square of the velocity.

Since wing pitching moments are a part of stability and control as well as a factor in inflight breakup problems, it is necessary to understand why the wing pitching moment increases as the square of the velocity.

In the wind tunnel, the velocity normally remains constant, and the angle of attack and resultant lift are varied. In the aircraft, the velocity and angle of attack are varied, and the lift remains constant for any given instant of study.

In studying a level flight condition at slow speed, the lift equals the weight, the angle of attack is high, and the center of pressure is close to the aerodynamic center. The moment arm between the aerodynamic center and the lift force is short, therefore, the pitching moment would be at some low value.

In the high-speed flight regime, the weight, and therefore the lift force, is still the same. The angle of attack, however, is low, and the center of pressure is well aft on the airfoil. The force of lift times its moment arm from the aerodynamic center will therefore be a higher value than in the slow-speed regime.

As an example, if an aircraft with a cambered airfoil has a given pitchdown moment at 100 knots, it will have four times this pitchdown moment at 200 knots, and of course, nine times this moment at 300 knots. This fact makes it easy for the investigator to understand the breakup sequence in a loss-of-control, weather-type accident.

In understanding the mean aerodynamic chord as an airfoil which represents the characteristic of the wing as if it were a rectangular wing, and the fact that it, in turn, possesses an aerodynamic center which is located for all practical purposes at the quarter chord point, it becomes easy to visualize why the wing is attached to the fuselage in a particular position for weight and balance purposes. See **Fig. C II-8.** The horizontal tail, therefore, can be visualized as the balancing surface for the wing, fuselage, and powerplant combination. For example, if one were to take a series of airplanes and note the center of gravity limits of each, it would be found that they fall in the ball park area of approximately 10 to 15 per cent of the M.A.C., fore and aft of the aerodynamic center. This will vary with different aircraft configurations, but as an investigator, one should visualize this as a general area of the C.G. limits beyond which the horizontal tail surfaces cannot generate enough power to control the aircraft (forward C.G.), or conversely, the aircraft becomes extremely sensitive or unstable (with an aft C.G.). It is this pitching moment of the wing, plus the pitching moment characteristics of the engine thrust, the fuselage shape, etc., which must be counterbalanced by the normally negative load on the horizontal tail.

In a stable conventional aircraft all moments add up to a condition that requires a download on the horizontal tail to balance the aircraft. It must be visualized that with an abnormally forward C.G. in the aircraft it is possible to exceed the download aerodynamic carrying capability of the tail. In the slower-flight regimes it is possible to have an uncontrollable pitching down condition of the aircraft. This is one of the areas in which FAA engineering personnel will flight check the aircraft during the certification tests.

The other regime will be the maximum aft C.G. limit in which the download on the tail will disappear, or become positive, because of the excessive aft C.G. condition. Under this condition, the aircraft will become highly unstable, or very sensitive on the controls, to the point where over controlling could result.

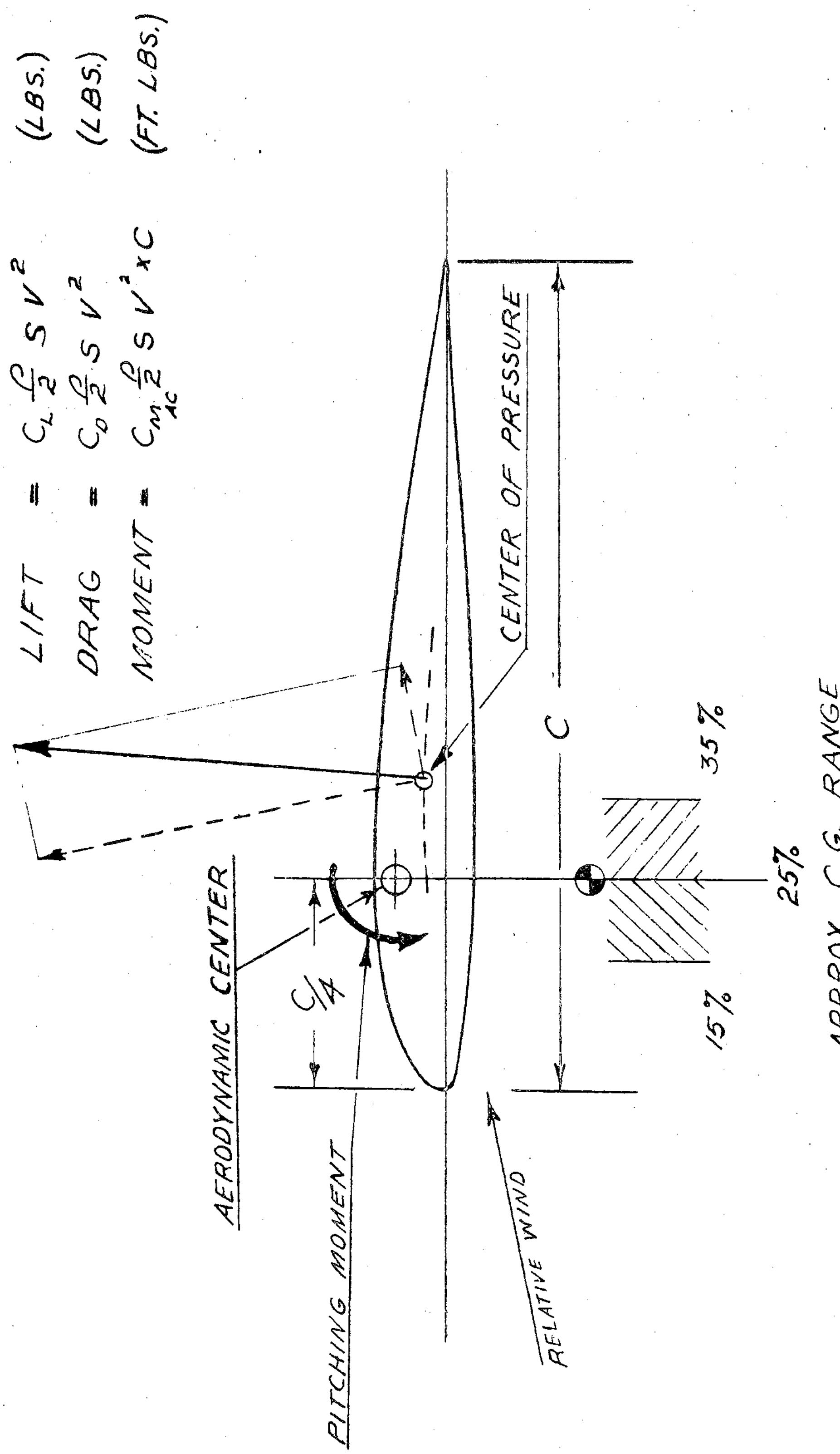


Illustration of lift, drag, and moment coefficient equations
Figure C II-8.

1.5.3. Related Equations

To illustrate the relationship of lift, drag, and moment of the airfoil or wing, reference is made to Fig. C II-7 which shows the lift characteristic curve, the drag characteristic curve, and the pitching moment curve of an airfoil section. These curves are plotted versus the angle of attack across the horizontal or X-axis. The three characteristics of lift coefficient, drag coefficient, and moment coefficient are plotted on the vertical or Y-axis.

Figure C II-8 illustrates the three equations associated with these coefficients. A lift equation is equal to the lift coefficient times rho over 2 SV squared, where rho is the density of the air, S the wing area, and V the velocity in feet per second, of the aircraft ($Lift = C_L \; \rho/2 \; SV^2$). The drag equation is the same basic equation of C sub D, or drag coefficient, times rho over 2 SV squared ($Drag = C_D \; \rho/2 \; SV^2$). These equations result in pounds of lift and pounds of drag, respectively. The third equation, which is the moment equation, is equal to C sub M, sub AC, which is the moment coefficient around the aerodynamic center, times rho/2 SV squared ($Moment = C_{M_{AC}} \; \rho/2 \; SV^2c$). This quantity will result in a force times a distance c, the last letter in the equation, which is the mean aerodynamic chord. This distance (c) when multiplied by the wing lift force will give the total pitching moment in the wing.

1.5.4. The Stall

The stall has been an operational problem of an aircraft (heavier than air) since man began to fly. It is a simple thing in itself, just a matter of exceeding the critical angle of attack of a given airfoil. It can be a lethal thing, for there have been numerous stall, or stall-and-spin accidents every year since the flying machine came into being. The stalled attitude of an aircraft, in the hands of a skilled pilot, is a beautiful thing to watch at an air show. The snaproll, the hammerhead, the spin, the falling leaf, the vertical reverse, and the lomcevak are all maneuvers involving the stalled condition of an aircraft.

The stall is highly misnamed. There are the power-on, power-off, acceleration, the secondary, the deep, the stable, and the advanced stalls. There is only *one* stall – that which exceeds the critical angle of attack. The named stalls are merely flight regimes.

Since we must live with the stall, there are some aircraft which possess such bad rolling characteristics in a stalled condition that stall strips are placed on the leading edge to initiate an early stall in a strategic area. Some aircraft, because of undesirable stall characteristics, are required by the FAA to be equipped with a stall warner. This stall warning device is standard equipment on many aircraft as a safety device.

Stall-proof aircraft have been developed in an effort to preclude the spin. This type aircraft is prevented from stalling merely by restricting elevator travel so that the aircraft cannot be held in an attitude to obtain the stall. There have been fatal spin accidents in these spin-proof aircraft. Once this type aircraft is stalled and spun the recovery may be difficult because of the small or restricted controls. It is possible to get this type aircraft in such an attitude that it falls into a high angle of attack and subsequent spin condition.

It must be understood by the investigator that a complete aircraft under certain conditions can fall. The statement that an aircraft cannot fall is an old wives' tale taught by some instructors to instill confidence in the student. If this were so, it would be safe to do stalls 50 feet above the ground. Scientifically, the theory is explained: Any time the weight exceeds the lift, the aircraft will accelerate towards the earth. The acceleration rate is in proportion to the differential of lift and weight. One merely has to ride through a tail slide or lomcevak maneuver to experience the full effect of this scientific explanation. Fortunately, as the aircraft falls, it again begins to fly, provided there is sufficient altitude, or, it does not remain in a high angle of attack, stalled condition.

As an investigator, keep in mind that an aircraft must be stalled in a spin. If it is not stalled, the aircraft is merely in a steep spiral.

1.6. The Momentum Theory of Lift

The *momentum theory of lift* is based upon the equation *Force equals Mass times Acceleration ($F = Ma$).* This means that the wing will act upon a mass of air and deflect it downward through some angle. It is this downwash or angular acceleration of the air that requires an equal opposite reaction, which is lift. This theory appears to be a rather elementary approach to lift generation. However, it can be utilized to develop all of the lift and drag equations developed from other theories of lift. In studying the momentum theory in which a mass of air is deflected, one might ask what happened to Bernouli's principle in which a change in air velocity over the wing changes the pressure? There is no change in the principle as put forth by Bernouli. Obviously, one wants to use an efficient wing to accelerate a mass of air. A barn door wing could be used to lift an aircraft, but this is rather an inefficient deflector of air. By utilizing Bernouli's principle, and an efficient airfoil shape, the deflector can be greatly improved over a barn door.

To understand the momentum theory, visualize the wing as accepting a cylinder of air in which the wing span is the diameter of this cylinder. If the plan form of the wing is elliptical, such as that used on the Spitfire aircraft of WW II, it will actually affect a nearly perfect cylinder of air. If we consider a tapered cantilever wing, the total area the wing accepts is not perfectly cylindrical, but becomes slightly elliptical, and a correction factor will be introduced. A straight rectangular wing will also deflect an elliptical section of air. It accepts this tube of air, and then deflects it downward through some angle. In viewing Fig. C II-9 note that we can have a cross-sectional area of a wing span in diameter and one foot in thickness to form a volume. It can be seen that the velocity, density of the air, and mass or volume of air will eventually result in the lift equation, if we desire to derive it by this method.

1.6.1. The Downwash Angle

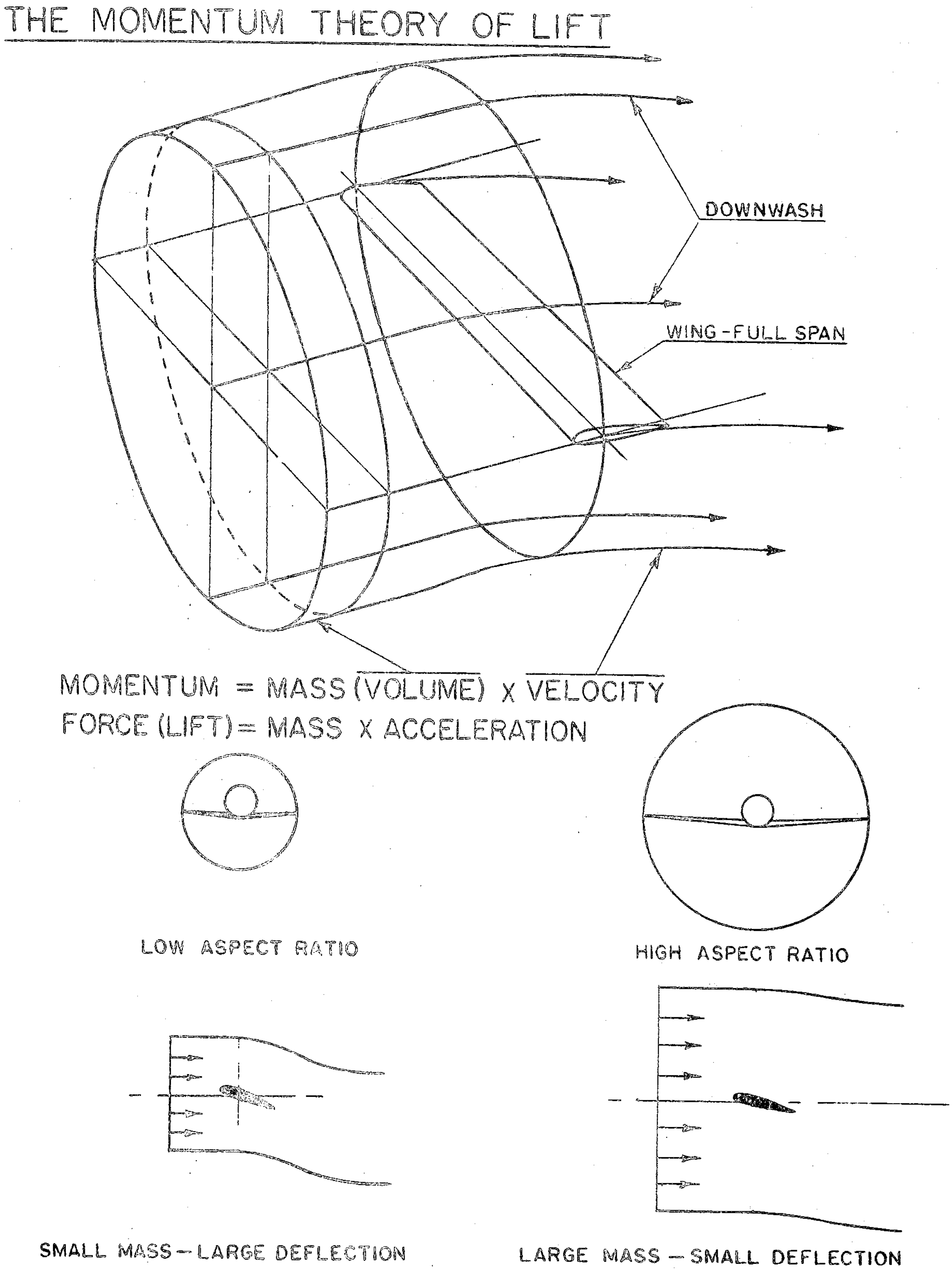
In the momentum theory of lift, actual flight is easily visualized, since this is what is taking place: The cylinder of air strikes the wing, and the wing deflects the mass of air downward at some downwash angle. This normally results in a downwash on the horizontal stabilizer which assists in creating the download required for stable flight conditions. Referring to the shape of the wing, a study of aspect ratio in conjunction with the momentum theory of lift will show that aspect ratio plays an important part in aircraft design for a specific mission.

In examining the sailplane, the B52 bomber, and the U2, it is obvious that these aircraft possess very high aspect ratios. These aircraft with large wing spans and relatively short chords will work upon a large mass of air, and therefore must deflect it through a relatively small angle to generate lift.

On the other hand, in an examination of an aircraft of delta wing design, which has a rather low aspect ratio and short span, it is noted that in order to sustain the weight of the aircraft, the wing must necessarily take this small tube of air and deflect it through a relatively large angle. In this case, what is lacking in the mass (volume) of the air must be made up in acceleration of the air. In observing the takeoff and approach attitudes of aircraft such as the F102, F106, and B58, it is noted that the attitude or angle of attack is extremely high compared to the conventional wing aircraft. This configuration of the wing plane has the characteristic of developing very high induced drag which will be explained later. Although the momentum theory of lift is relatively easy to visualize as it takes place in a conventional aircraft, other complications are involved in this basic aerodynamic phenomenon.

1.6.2. Three Dimensional Flow

One facet of aerodynamic flow is known as three-dimensional flow. Airflow can be visualized as traveling in one dimension in a direction from the leading edge to the trailing edge. As it passes over an airfoil, the air will pass above and below in a second or vertical dimension. The third dimension is in a spanwise direction, or spanwise flow.



Momentum Theory of Lift

Figure C II-9.

1.6.3. Vortex Generation

Figure C II-10 illustrates a conventional rectangular wing. Note that the airflow on the top of the wing is straight fore and aft in the center of the wing, but in progressing toward the wingtips on the top surface the airflow tends to flow inward from both tips at a slight angle. The dotted line represents the airflow on the bottom surfaces of this wing. It can be noted that the air flows in straight lines from leading edge to the trailing edge in the center of the wing. Progressing outward toward the tips, the airflow assumes an outward flow toward the tips because of the difference in pressure between the bottom and top surfaces. The high pressure air on the bottom moves toward the tip to enter the low pressure area on top the wing. Examine part B of **Fig. C II-10a**, which is a trailing edge of the wing; note that the spanwise flow creates a shearing action across the trailing edge of the wing, which in turn causes rotation or small vortices to form. The size of these vortices increases as they move out toward the tip, as evidenced in the plan view in **Fig. C II-10a.** The spanwise flow increases in magnitude inversely as a function of the distance from the wingtip.

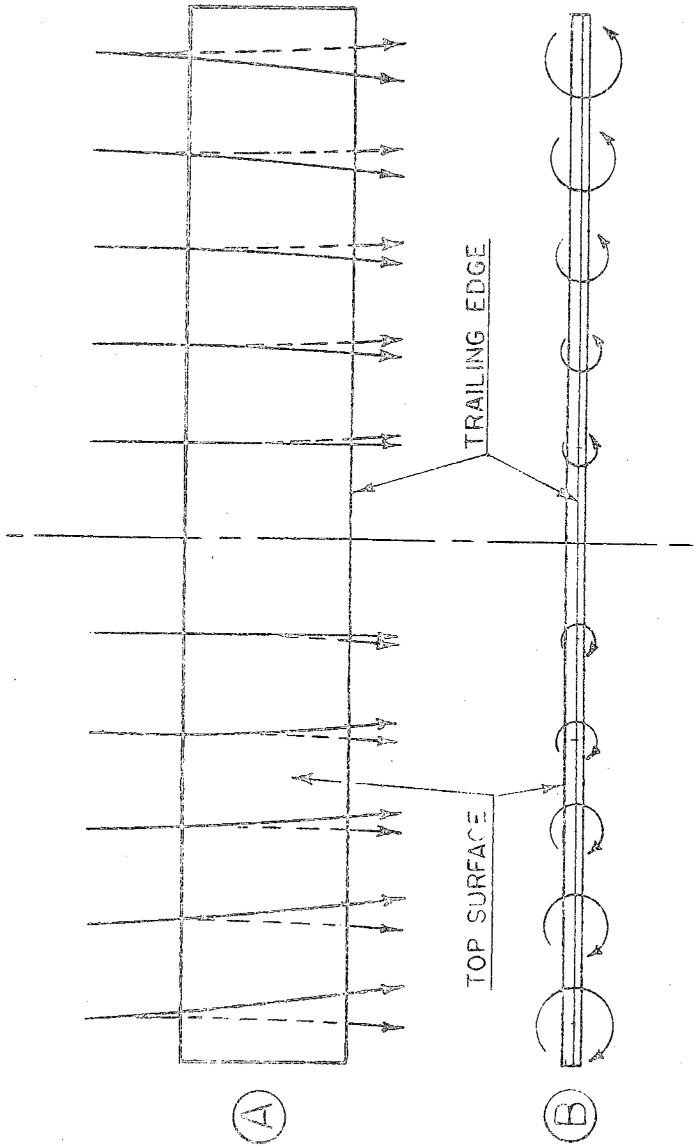
In the high aspect ratio wing, a large span and relatively small chord, it is obvious that it is more difficult for the air to spill over the tips, primarily because of the distance involved. Conversely, the low aspect ratio wing, a short span and large chord, it is relatively easy for the high pressure air on the lower portion of the wing to spill into the low pressure area on top of the wing. Aspect ratio then will be one factor in the process of vortex generation which in turn will have an effect on induced drag. In **Fig. C II-10b**, a cross section of an airfoil, note that vortex action creates a downwash of the air as it passes over the wing. The greater the vortex action, the greater will be the angle of downwash. The airflow in the area of the aerodynamic center is parallel to a line equal to one-half the downwash angle. ϵ(*Epsilon*) The total vortex action is the sum of the bound vortex and the vortex of downwash.

1.6.4. Induced Angle of Attack

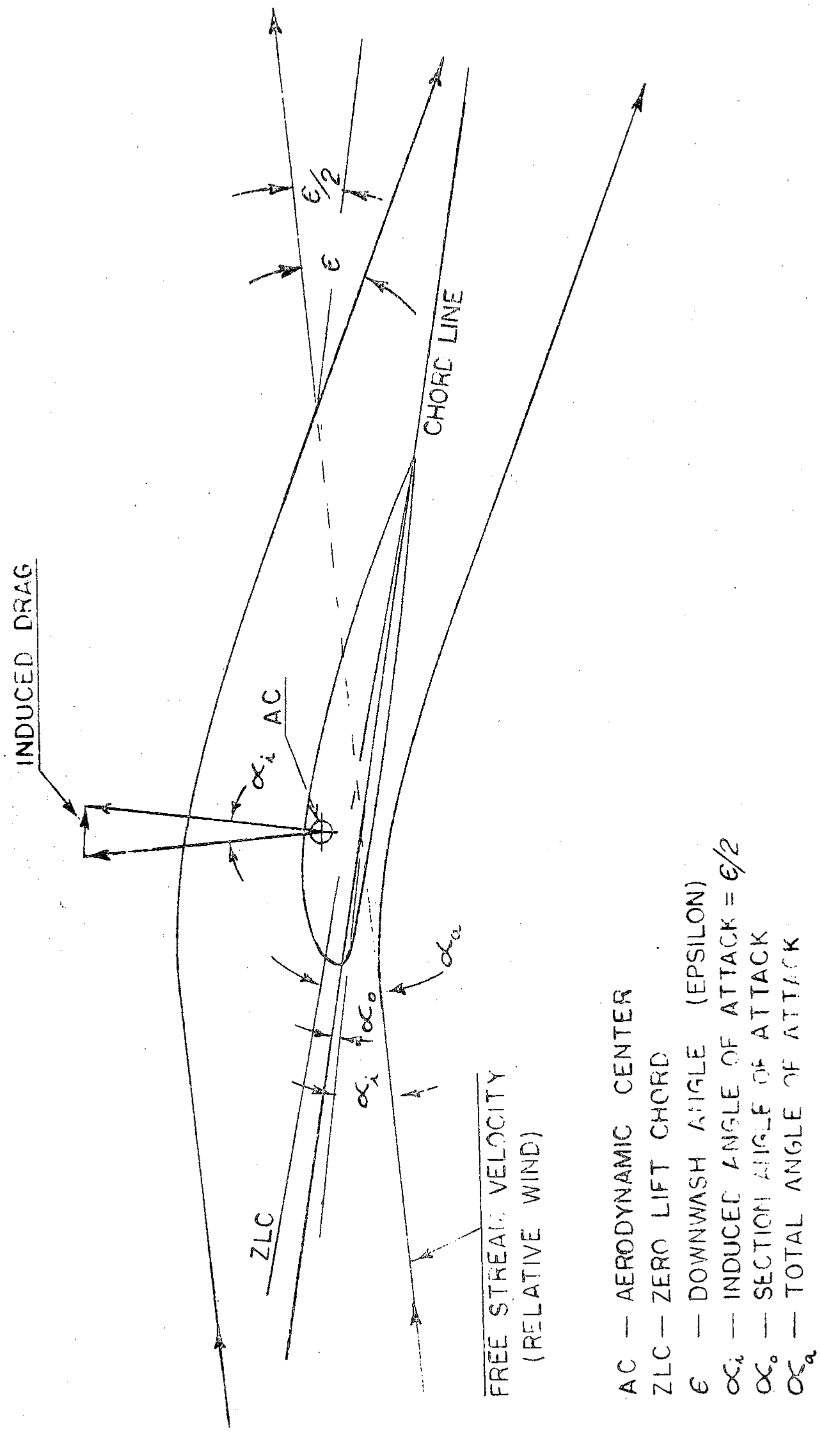
Wind tunnel tests proved that the airflow in the vicinity of the aerodynamic center or the quarter chord point (the effective relative wind) is parallel to a line which is equal to one-half the downwash angle. This downwash angle in turn induces a higher angle of attack to the airfoil referred to as alpha sub I or alpha induced. The total angle of attack (in referring to **Fig. C II-10b**) will be noted to be alpha, which is the original angle between the chord line and the free stream velocity. The total angle of attack then is equal to alpha sub zero, the airfoil section angle of attack, plus alpha sub i, the additional wing characteristic angle of attack ($\alpha = \alpha_0 + \alpha_i$). The spanwise flow increases toward the tips and therefore the vortex action increases toward the tip of the wing. For this reason, the airfoil will assume a higher angle of attack toward the wingtip, and this in turn will cause the tip airfoil to stall earlier than the root section. In a great percentage of aircraft, the wing is twisted around the aerodynamic center of a conventional metal wing or rigged in a strut-braced wing, thus reducing the angle of attack in the area of the wingtip. Other methods have been used to improve stall characteristics. One method is the use of leading edge wing slots, and another is the use of a highly cambered airfoil section to give better tip stall characteristics. End plates and tip tanks also improve this situation to a certain degree.

1.6.5. Induced Drag

Figure C II-10b illustrates the reason for induced drag. Note that the line which is labeled as free stream velocity, or relative wind, describes an angle between the chord line and this line. This is the *true angle of attack.* Progressing to the quarter chord point, there is a change in the angle of attack which is the *induced angle of attack.* The original lift, normally drawn perpendicular to the relative wind, now inclines aft since the relative wind in the vicinity of the aerodynamic center is at a different angle. This causes the lift vector to incline aft.



Vortex Generation
Figure C II-10a.



Induced Drag
Figure C II-10b.

The small component shown on the top triangle is the induced drag. It can be noted that as the aircraft is slowed a higher angle of attack results, and the rearward inclination of the lift vector progresses in angular displacement aft. This is an important fact to consider as a potential causal factor in aircraft accidents. The problem arises when the aircraft is on the final approach in a slowed flight regime which necessitates a higher angle of attack. This higher angle of attack results in greater vortex action, the lift vector inclines aft, and the induced drag increases. If the aircraft is allowed to establish an abnormally high sink rate, this means that the angle of attack goes to a higher value, the lift vector inclines aft at a greater angle, and the induced drag increases.

Although the wing may be well below the stall, a high degree of power is required to overcome such a flight regime. This is commonly referred to as "getting behind the power curve," or "in the region of reverse command." This is not a new concept, for it has been known for a number of years.

The problem of induced drag began to manifest itself with the advent of higher performance aircraft. High performance aircraft obey the same laws of aerodynamics and physics as the low performance aircraft, however, increased weight, increased thrust, and increased velocity can result in an accident of a greater release of energy and resultant destruction. It is therefore necessary to re-examine the induced drag equation. This is the formula for the coefficient of induced drag which is a direct indicator of induced drag. The coefficient of induced drag is equal to *C sub L squared over pi AR.*

$$C_{D_i} = \frac{C_L^2}{\pi AR}$$

It can be seen that the aspect ratio is a factor in induced drag. An aircraft can encounter problems with induced drag on the takeoff and final approach merely because of the configuration of the aircraft. As an example, in an aircraft with a low aspect ratio and a high wing loading, or span loading, the problem manifests itself, since in the above equation it can be noted that the lift coefficient is squared.

In the jet transport utilized today, the aspect ratio is fairly high, in the regime of 7 to 1 to 8 to 1. However, these aircraft have abundant power available, and therefore are allowed to carry very high wing and/or span loadings. As a result, a high lift coefficient is required. At the time of liftoff the aircraft is near maximum weight, therefore, the lift coefficient has a very high value. As a result, the induced drag has a very high value which is synonymous with a maximum development of wingtip vortex action.

The same problem can be expected on the final approach in which the aircraft is slowed to a high angle of attack and high lift coefficient for the lower velocity. This in turn generates the high vortex action which is again synonymous with induced drag.

With some of our modern transports problems have occurred because they are so efficient, as far as lift generating capabilities are concerned. They can be flown quite slowly on the final approach, which in turn generates enormous values of induced drag. Failure to keep such an aircraft in a state of equilibrium can establish high sink rates which compound the problem. The point to remember is that induced drag is associated with high angle of attack and lower airspeeds. As the airspeeds increase up to the cruise regime, the induced drag, which is a result of rearward inclination of the lift vector, virtually disappears. The drag of the wing then is primarily parasite or shape drag, also referred to as form drag.

Numerous devices have been installed on the wing to prevent spanwise airflow and the resulting vortex action and induced drag at slow speed. One common device is the boundary fence, a fore and aft fence or wall placed upon the wing surface to prevent spanwise airflow. Another is the installation of wingtip tanks. The shape of the tank acts as a boundary fence at the wingtip retarding spanwise airflow, and in effect, increasing the aspect ratio of the wing. There are three basic advantages in the installation of tip tanks. One is the improved aerodynamic characteristic; the second is the reduction of bending moments within the wing in flight; the third is the extended range of the aircraft. Some disadvantages to a tip tank installation will be covered in another subject area.

2. Stability and Control

An aircraft should display satisfactory handling characteristics throughout its flight regime from V_{so}, which is full-flap landing configuration stalling speed, up to maximum operational speed, V_{no}. The aircraft should provide sufficient stability to return to a normal flight regime when disturbed by some outside influence. Also, to achieve the desired performance, the aircraft should display a proper control response.

2.1. Stability — Controllability — Maneuverability

In this area of stability and control three subject areas should be defined. *Stability* is the inherent flight characteristic of an aircraft after being disturbed by an unbalanced force or moment. This definition of stability of the aircraft covers the characteristics, stable or unstable. One might say that an aircraft possesses stability with the connotation that it is good. When the statement is made to a test pilot that an aircraft possesses stability, a question immediately arises in his mind: "Sure, it possesses stability, but is it good, bad, or marginal stability?"

Controllability is the ability of an aircraft to respond to control surface displacement, and to achieve the desired condition of flight.

Maneuverability is defined as an aircraft design characteristic of motion around the three axes governed by weight, inertia, control surface size and location, structural strength, and powerplants.

A term in that branch of physics known as mechanics should be thoroughly understood prior to any study of stability and control. This is *equilibrium,* defined as the state of an object (in this case an aircraft) when the summation of all forces and moments equals zero. This means that all vertical forces must equal zero. In other words, the total lift of an aircraft must equal the total weight plus aerodynamic downloads on the aircraft. In the horizontal direction this means that the sum of all side loads and fore and aft loads must equal zero. In the area of moments, the aircraft must have zero moments around the three rotational axes of the aircraft in order to be in a state of equilibrium.

2.1.2. The Six Degrees of Freedom

An aircraft possesses six degrees of freedom which differentiate the aircraft from earth-bound vehicles.

There are three degrees of translational freedom, meaning that the aircraft can move in a fore and aft direction, move or translate sideways (slip to the right and to the left), and it can move up and down.

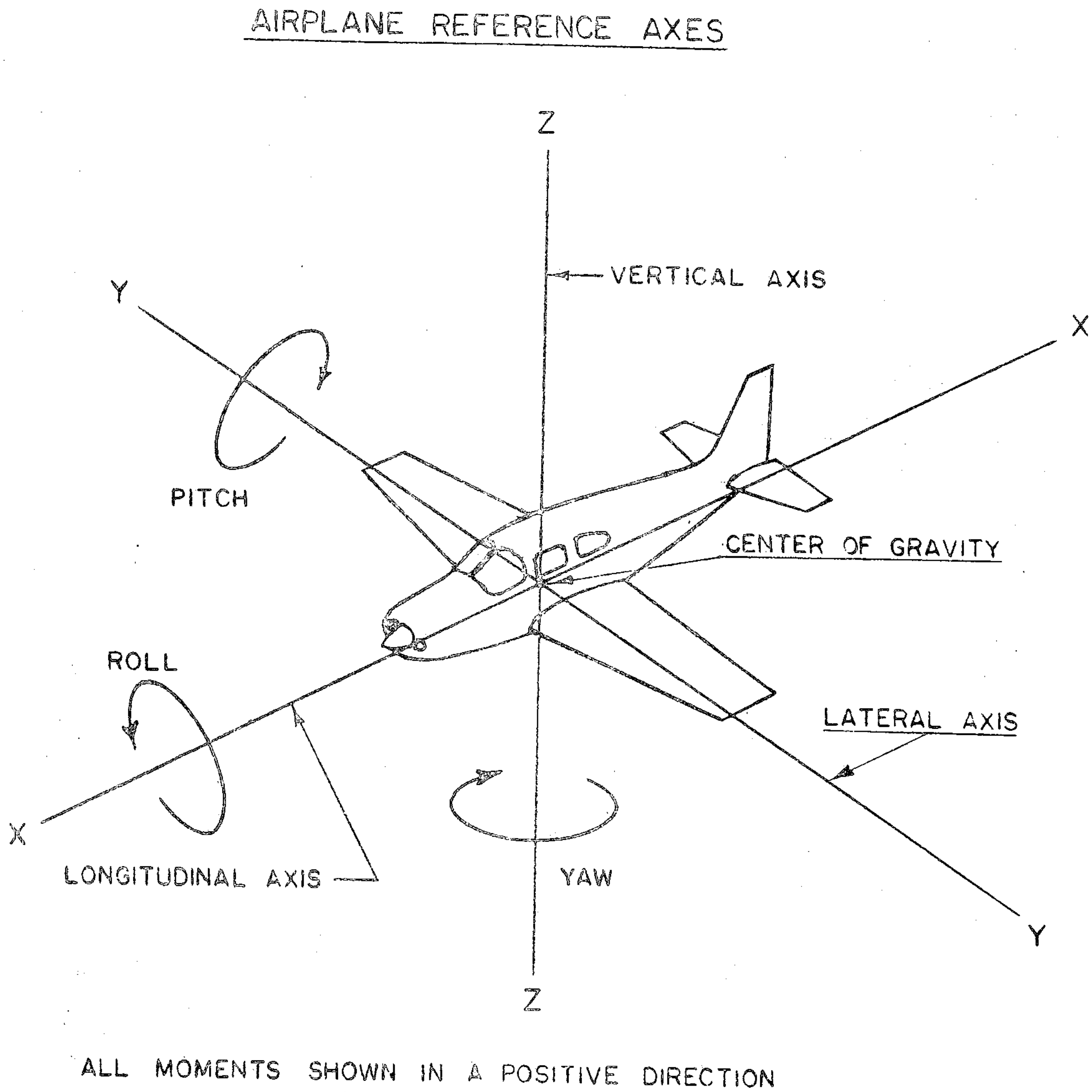
Translational means that the aircraft can move on a line in either of two planes, that is, vertical and horizontal. Horizontal includes fore and aft, and left and right directions.

The aircraft also possesses three degrees of rotational freedom in that as a body it can rotate around the X, Y, and Z axes of the aircraft. Note in Fig. C II-11 that X axis is the longitudinal axis; the aircraft will roll around this axis. The Y axis is the lateral axis, and the aircraft will pitch up and down around this axis. The Z axis is the vertical axis, and the aircraft will yaw left and right around the vertical axis. It is also noted that *the three axes intersect at the center of gravity of the aircraft,* an important point to remember in studying aircraft motion.

All rotation takes place around the center of gravity of the aircraft while the aircraft is in flight. A plus and minus sign convention has been established relative to motion around these three axes. On the longitudinal X axis a roll to the right is a positive rolling moment; on the lateral Y axis a pitchup is a positive pitching moment; and on the vertical Z axis a yaw to the right is positive.

2.2. Static Stability

For an aircraft to possess safe and desirable control qualities, it must possess certain stability characteristics. Stability is classified in two areas, static and dynamic. Static stability is the initial tendency of a system, in this case



Airplane Reference Axes
Figure C II-11.

an aircraft, to return to equilibrium conditions after some disturbance. It is further subdivided into three areas known as positive, neutral, and negative stability. The important point to remember is that static stability is only the *tendency* to return to equilibrium conditions. Reference **Fig. C II-12.**

2.2.1. Positive Static Stability

As a practical example of positive static stability, visualize an aircraft trimmed for cruise flight conditions. Back pressure is applied to the control system until the aircraft has assumed some nose-high pitch attitude. The control pressure is then released, and the characteristic of the aircraft is studied. If the nose of the aircraft tends to pitch down or return to a level flight condition, the aircraft has displayed a positive static stability condition. This is also verified from the nosedown pitch condition.

2.2.2. Negative Static Stability

If the nose of the aircraft pitches to a higher attitude when the controls are released, it is displaying a negative static stability.

2.2.3. Neutral Static Stability

If the nose of the aircraft remains in the same nose-high attitude when the controls are released, it is displaying neutral static stability.

2.3. Dynamic Stability

The second phase of stability is known as dynamic stability, defined as the resulting motion with time. Dynamic stability also is classified as positive, neutral, and negative. Ref. **Fig. C II-12.**

An important relationship between static and dynamic stability is that positive static stability is a requirement for dynamic stability, but static stability does not necessarily imply dynamic stability. To visualize the difference between static and dynamic stability, take a practical application of an aircraft trimmed for cruise flight conditions. If back pressure is applied so that the nose is placed at some angle above level flight conditions, and the control column is released, one of the following conditions may result: positive, negative, or neutral dynamic stability.

2.3.1. Positive Dynamic Stability

If, after displacement from a level trim condition, the nose of the aircraft continues to rise and descend through a level flight condition, and during this oscillation the amplitude is less than the one previously displayed (in both the up and down condition from level flight) the aircraft is displaying positive dynamic stability. This means that the aircraft will eventually damp itself out and return to the level flight regime for which it was trimmed.

2.3.2. Negative Dynamic Stability

When the aircraft is displaced from a level trim flight attitude, either up or down, and the resulting motion increases the amplitude each time, the aircraft is displaying negative dynamic stability.

2.3.3. Neutral Dynamic Stability

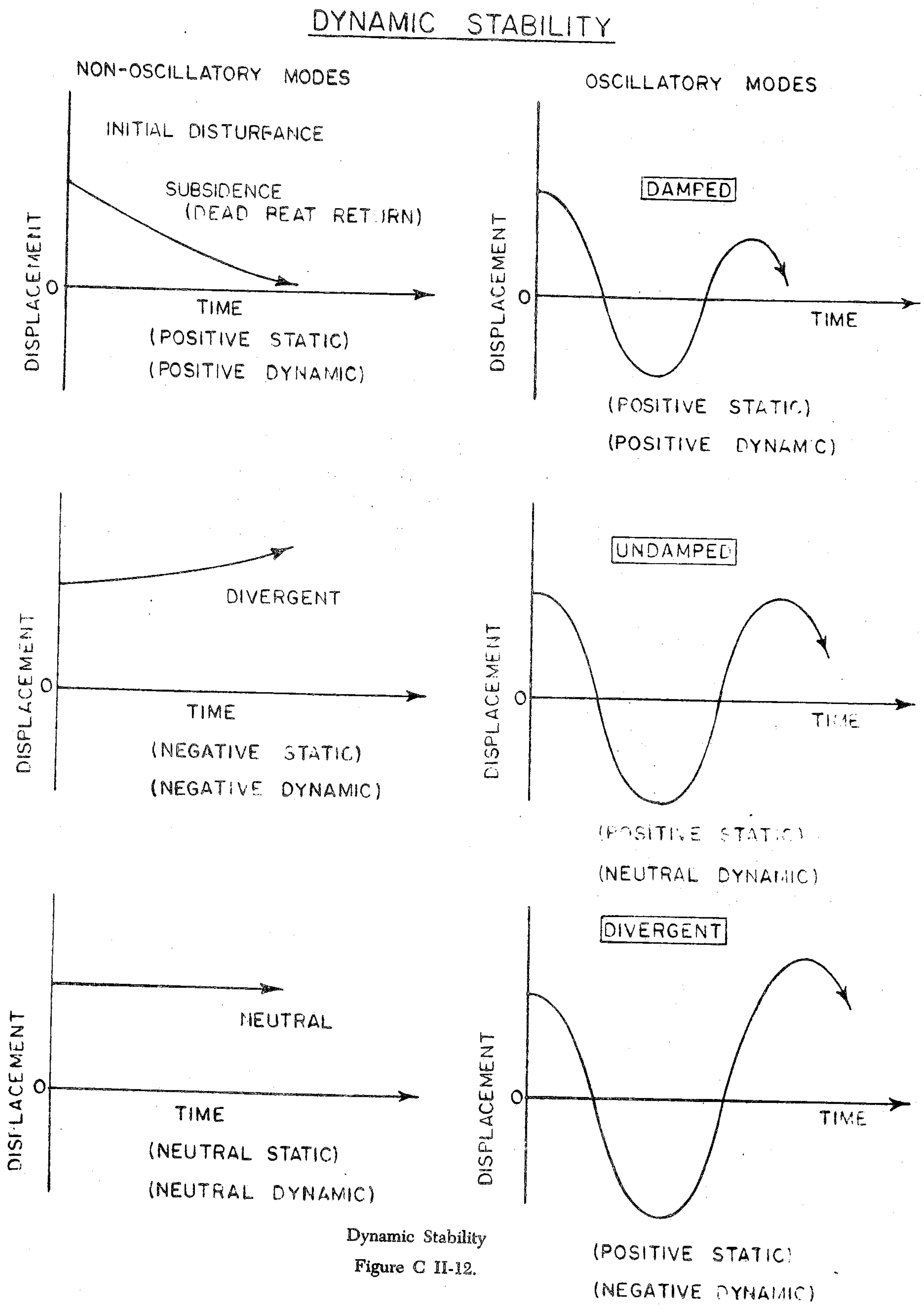
If the aircraft is displaced as previously, the displacement of the aircraft as it oscillates above and below level flight attitude is the same each time and continues at the same amplitude; the aircraft is then displaying a condition of neutral dynamic stability.

2.3.4. Longitudinal Stability and Control

General aviation aircraft which display negative dynamic stability characteristics are certificated within this country. This in no way makes the aircraft dangerous or undesirable, and they fully meet the FAA Part 23 and 25 requirements for stability. The FAA requirement for longitudinal stability reads as follows:

> Any short period longitudinal oscillation between the stalling speed and maximum dive speed must be heavily damped with the primary controls (a) free, (b) fixed.

In the discussion of longitudinal stability, which is rotational around the lateral axis of



Dynamic Stability
Figure C II-12.

the aircraft, it is found that numerous items affect the pitching characteristics of the aircraft. In order to remain consistent in the signs for moments and directions, longitudinal stability should be visualized from the left side of the aircraft, that is, with the nose to the left and the tail to the right. In dealing with longitudinal stability we therefore have pitching occurring in the aircraft around the center of gravity. If the nose pitches up, the aircraft is displaying a positive pitching moment characteristic, and if it pitches down, a negative pitching moment characteristic. A nosedown pitch is a stabilizing moment, and a noseup pitch is a destabilizing moment. This is obvious when flying standard category FAA certificated aircraft, for if the power is reduced the nose of the aircraft will lower. This is a negative pitching moment and also a stabilizing moment. The aircraft merely rotates nosedown around the lateral axis to return to the speed for which it was initially trimmed.

2.4. Wing Pitching Moments

The aerodynamic pitching moment of the wing itself can be a major factor in longitudinal stability. If the airfoil sections of the wing possess negative pitching moment coefficients it can be noted from the moment equation that the pitchdown tendency increases as the square of the velocity. In other words, the faster the aircraft moves through the air, the greater the download requirement on the horizontal tail to control this moment. This is no basic problem in a stable aircraft, since as the airspeed increases there is an increased download on the horizontal tail surface, and it has to be trimmed out. The investigator should, however, have a clear understanding of these loads and moments, for the inflight loss of the horizontal tail surface could result in characteristic wing and/or fuselage separation or break. Also a number of modern general aviation aircraft have been designed with symmetrical airfoil sections and thus the stable nosedown pitch quality is zero as far as the wing is concerned.

2.4.1. Fuselage Pitching Moments

The fuselage and the nacelle configuration of aircraft also possess aerodynamic characteristics. These characteristics normally are determined from wind tunnel tests, and the shape may be modified somewhat to obtain a desired fuselage pitching characteristic.

2.4.2. Powerplant(s) Pitching Moments

Powerplant output and position of installation affects the longitudinal stability of an aircraft. Generally speaking, if the power is increased, a destabilizing influence is exerted on the aircraft, which means the engine or powerplant has the ability to establish a greater pitchup influence upon the longitudinal stability characteristics of the aircraft, as well as rolling and yawing. For this reason, it will be noted in numerous reciprocator engine installations that the thrust line is pointed to some small angle downward to five or six degrees. As power is applied with the thrust line pointing down slightly, a stabilizing influence results. All rotation takes place around the center of gravity, and if one refers to some of the current light twin jet business aircraft in which the engine is mounted in the aft fuselage area, it will be noted that these engines are also mounted at some specific angle, usually around three degrees to the longitudinal axis of the aircraft. Engines are mounted so that the forward end of the engine is higher than the aft end. Application of thrust has a nosedown or stabilizing influence on the aircraft.

The location of the engine also is a major factor in longitudinal stability. As an example, in the large four-engine jet transport in which the engines are mounted below the wing, and below the center of gravity, the low position of the engines causes a pitchup tendency during initial phase of takeoff. On the other hand, the light four-place amphibian aircraft, which has the engine mounted above the wing, has the problem of excessive nosedown pitch at full power because of the distance of the thrust line above the center of gravity.

2.5. Directional Stability and Control

As in longitudinal stability and control, an aircraft must possess desirable directional stability and directional controllability. Directional stability means that if the aircraft is subjected to a disturbing force or moment so that it is rotated around the vertical axis in one direction or another, the aircraft will display a heavily damped return to a straight flight condition. In the area of controllability the aircraft must be directionally controllable from minimum velocity to maximum velocity. In the slow speed regime, the aircraft must possess adequate directional control in the area of V_{mc}, and a full displacement of the rudder on multiengine aircraft is a factor in determining V_{mc}. In the other regime of maximum dive speed, the rudder vertical stabilizer combination must be capable of controlling the aircraft in a directional sense without excessive force or without oversensitivity.

2.5.1. Wing Yawing Moments

Since the wing is the all-important surface on the aircraft because it lifts the aircraft, lowers the aircraft, and turns the aircraft, the performance of these varied functions also produces some undesirable stability qualities in the aircraft. One of these is adverse yawing, that is, rotation of the aircraft around the vertical axis. Adverse yaw is the result of asymmetrical lift. In order to roll the aircraft to execute a turn, it is necessary to have dissymmetry of lift between the left wing and the right wing. If the lift on one wing is reduced, the drag is consequently reduced.

Considering the opposite wing, increasing the lift to produce this roll, drag is consequently increased. If a turn to the right is desired, the aircraft will yaw to the left, the opposite direction of the desired turn. This problem manifests itself with conventional ailerons or a spoiler system. In executing a turn to the right we spoil the lift on the right wing, utilizing the spoiler. This may have a tendency to reduce the drag and produce the same undesirable directional characteristics. The opposite may be true on certain aircraft. Crosswind landing might present a problem in the use of spoilers. The investigator must become familiar with the particular aircraft involved.

2.5.2. The Yaw and the Sideslip

The terms *yaw* and *sideslip* enter into the discussion of directional stability and control. The difference between a yaw and sideslip must be thoroughly understood.

The *yaw* is rotation of the aircraft around the vertical axis. In conventional signs for stability and control, when the aircraft rotates to the right around the vertical axis a right or positive yaw results.

The term *sideslip* means that the aircraft is slipping sideways.

To understand the difference between yaw and sideslip, visualize an aircraft in which full right rudder is applied; the aircraft yaws to the right, but it sideslips to the left.

As an interesting sidelight, a number of years ago the Air Force renamed the turn and bank instrument the **turn and slip indicator.** Since the wing turns the aircraft, the needle of this instrument is the turn indicator, and indirectly the bank indicator. The displacement of the needle in either direction indicates that the aircraft should be banked in that direction, therefore the needle indicates turn and bank. The ball, on the other hand, actually indicates whether or not there is a slip condition, that is, if the ball is in the center the aircraft is in equilibrium, it is not slipping. If the ball is displaced in one direction or another, it indicates that the airplane is slipping to the inside or the outside of the turn. This instrument records a yaw as well as a coordinated turn. As an example, in a yaw to the right using right rudder only, the needle displaces to the right; the ball is displaced to the left. It is this instrument's very sensitive response that poses the question: Is it possible to fly the modern jet transport or light executive jet aircraft at high altitudes under turbulent conditions when other attitude reference instruments have been lost?

Since this is the instrument which indicates the yaw and the sideslip, the investigator must have a clear understanding of the value of this instrument. As an example, if an aircraft executes an aileron roll on a point, what does the needle indicate? Another question, assuming that the aircraft is in a coordinated turn to the right: Is the aircraft yawing around a vertical axis? Is the aircraft sideslipping? In this case, the needle represents the fact that a turn is being executed, and if the ball is in the center, there is no yawing, no slipping. If a section of a wing has separated in flight, it is important to study the evidence of yaw and roll on the remaining structure. The aircraft is out of equilibrium at the time of separation, and will soon seek a state of equilibrium.

An inflight wing separation will cause a severe yawing of the aircraft, and it may cause a structural breakup of the vertical stabilizer. If it is a positive wing separation, there is a good chance that the wing may strike the tail assembly. This can easily be determined. If there is a negative wing separation, the tail may or may not separate. If it does, adverse yaw is probably the reason. A study of the breakup will show the sequence of separation.

2.5.3. Engine Torque

The subject of engine torque has been a catchall for all the undesirable torques that occur within an aircraft as a result of power application. In actuality, the torque taking place in a single engine aircraft is comprised of four individual twisting moments, each of which will be discussed in detail. Engine torque demonstrates the law of motion that for every action there is an equal and opposite reaction. The American-designed aircraft engine rotates the propeller to the right, which in turn causes the engine to rotate to the left, transmitting a counterclockwise torque to the aircraft around the longitudinal axis of the aircraft. This direction of engine rotation conforms to the automotive engine field. If the engine applies a left or negative torque to the aircraft structure, the aircraft will roll to the left, since it will be out of equilibrium. This roll, in turn, must be equalized by asymmetrical wing lift because the left wing will have to carry a slightly larger lift value than the right wing. The left wing will have to carry the slight increase in lift over the right wing. Consequently, this left wing will develop a slight increase in drag. This places the aircraft out of equilibrium around the vertical axis and therefore must be compensated for by a vertical stabilizer adjustment, or a rudder trim tab adjustment.

2.5.4. Asymmetrical Loading of the Propeller Disc Area

There is a second area of torsional moments around the vertical axis of the aircraft, caused by the propeller under certain low airspeed regimes and aircraft attitudes. If an aircraft is operated in a slow flight regime, in a high pitch attitude, and under power, the descending blade of the propeller will operate at a higher angle of attack than the ascending blade. This, in turn, will cause the right half of the propeller disc area to develop more thrust than the left half of the propeller disc area. This asymmetrical thrust will cause the nose of the aircraft to be yawed to the left. The magnitude of this yawing action is a function of engine output, aircraft attitude, and aircraft velocity. This is one of the primary factors which makes the number one engine critical on multiengine aircraft of American manufacture.

In an engine(s)-out regime, the engine(s) operating on the right side of the aircraft, under a low-airspeed condition, have the center of thrust moved outboard or farther away from the vertical axis of rotation, which passes through the center of gravity.

Conversely, with the right engine(s) shut down at the lower-speed flight regime, the centerline of thrust of the left engines will also move to the right, which in turn places the applied thrust closer to the center of rotation (vertical axis).

2.5.5. Slip Stream Rotation

A propeller creates thrust by accelerating a mass of air in one direction, in this case aft, in the aircraft. As this propeller accelerates the air, it also imparts a rotational motion to the air in the same direction as the rotation of the propeller. This should be visualized as a spring or spiral around the fuselage in a single engine aircraft.

If an aircraft is flying under cruise condition, this spiral or spring rotation around the fuselage should be visualized as a long, stretched-out spring in which the pitch is quite large. This spiral action of the air around the fuselage will strike the left side of the vertical tail surfaces. For this reason, the fin or vertical stabilizer is offset to counteract the angular relative wind at this point. The fin is offset under a cruise flight condition so that the rudder may fair in behind the vertical stabilizer to produce a minimum of drag. The flight regimes higher than cruise airspeed, with and without power, are normal points of confusion to many pilots, qualified or not.

Considering the flight regime above cruise speed, if a power-off dive is executed, there is a pronounced yawing or turning of the aircraft to the right, due to the rigging of the vertical stabilizers to the left. This requires a left rudder for coordination to put the aircraft back into equilibrium. If high power is used during this dive condition, the situation will be relieved somewhat, due to the slipstream rotation.

In the flight regimes below cruise flight, if the power-off condition is studied first, we see that in the absence of slipstream rotation the vertical stabilizer rigging will cause the aircraft again to turn to the right or yaw to the right. Therefore, considerably more-than-normal left rudder is required for rolling into turns to the left, and rolling out of turns from the right.

The areas just discussed are conditions existing in an aircraft and they are not directly associated with accident causal factors.

The third area is that of low airspeed and high power. To visualize this flight regime, the slipstream rotation might be pictured as a closely wound spring because of the low velocity of the aircraft. Then one can visualize a high dynamic pressure contacting the left surface of the vertical stabilizer system. This, in turn, will set up large negative moments around the vertical axis of the aircraft. It follows that the aircraft must be placed in equilibrium by large positive moments which are created by right rudder pressure application. This aerodynamic characteristic of aircraft results in many stall-and-spin-to-the-left type accidents.

2.5.6. Gyroscopic Action

The fourth area of torque around the vertical axis of the aircraft which involves directional control and stability is that of propeller gyroscopic action. In the average light single engine aircraft during a stall recovery, as the nose of the craft is lowered, additional right rudder is required to maintain a directional control. This is caused by the gyroscopic action of the propeller-engine combination. It is based on the principle of precession that a force applied to the rim of a gyroscope will be effective 90 degrees later from the point of application.

If the nose of the aircraft is lowered, this in effect causes a force application forward at the top of the propeller disc area and a force application aft at the bottom of the propeller disc area. In returning to the top of the propeller, if the force is applied forward at this point it will be felt gyroscopically 90 degrees later, or at the right side of the propeller disc area.

If the lower area of the propeller is examined, it moves from right to left as the force is applied aft. This force in turn will have a resultant maximum value when it is in the left side of the propeller disc. Therefore, the combination of these two forces will yaw the aircraft to the left.

An investigator must realize the difficulty an inexperienced or untrained pilot may have under such a flight regime. As an example, assume that a pilot executes a go-around from the final approach from a relatively low airspeed flight condition. Power is applied, and engine torque is directly applied to the aircraft structure to cause the airplane to rotate to the left or turn to the left. The slipstream rotation under a slow airspeed condition will reach a maximum, and therefore has the same maximum tendency to yaw or turn the aircraft to the left. If the nose of the aircraft is allowed to assume relatively high pitch attitude, asymmetrical thrust enters the picture, which will further turn the aircraft to the left, and if the pilot should inadvertently lower the nose rapidly, the aircraft will yaw or turn to the left.

All of these torques must be counteracted by adequate right rudder pressure to maintain a status quo in directional control. Improper control techniques such as the example above have resulted in aircraft rolling completely inverted and striking the ground in an unusually steep pitch attitude.

2.5.7. Asymmetrical Power

The problem of asymmetrical power has been a direct causal factor in numerous multiengine aircraft accidents. These accidents range from the very light twin aircraft to the large transport of either reciprocator or jet powerplant. Asymmetrical power connotes more than one engine with the thrust line located at some other point than through the vertical axis or the center of gravity of the aircraft. There is a type-certificated aircraft in production which is not in agreement with this statement. This is the Cessna Model 337, a twin-engine, push-pull type aircraft, in which the thrust line of both engines passes through the vertical axis of the aircraft. Consequently, loss of one engine does not result in a disturbing directional moment.

The loss of one or more engines on one side of the aircraft merely establishes a disturbing directional moment in a direction towards the loss of power. Under this condition, the aircraft will yaw, that is, rotation will be established around the vertical axis.

Referring to the slip indicator, the ball will be displaced from the center. This problem is counteracted by a moment in the opposite direction established by the application of rudder pressure. It is obvious that the application of rudder pressure develops a force, and this force works on a moment arm to the center of gravity of the aircraft. When this force or this moment arm generates a moment equal to the disturbing moment of asymmetrical thrust, the aircraft is again in equilibrium from a directional control standpoint. The ball in the turn and slip indicator will be centered if the wings are held in a level attitude.

It is immediately obvious that an aft c.g. condition is more critical than a forward c.g. position because of the length of the moment arm.

Asymmetrical power causes a problem primarily in the slow flight regime. This is the velocity referred to as minimum control velocity, or *V sub m sub c* (V_{mc}). In the type certification of a multiengine aircraft, this value must be established. The FAA in their V_{mc} requirements must know the minimum velocity for control, with the maximum power on the good engine(s) when the other engine(s) is in a simulated feather condition, with the center of gravity of the aircraft in the aft position. The aircraft will also be under a design gross weight condition.

Analyzing the overall situation for this test configuration, the aircraft is at a maximum design gross weight condition requiring a maximum lift coefficient or the minimum velocity at which the aircraft can sustain level flight condition. This requires that the aircraft be in a nose-high attitude, which will result in a maximum asymmetrical thrust of the propeller disc area. This is critical, with the right engine at maximum power. The thrust, therefore, is at its maximum distance from the center of gravity or the axis of rotation of the aircraft. The rudders, which must generate a force to result in a moment to counteract the disturbing moment, are at a minimum of aerodynamic efficiency because of the low velocity. The center of gravity must be in an aft condition to place the moment arm from the center of gravity to the center of pressure of the rudder at its minimum distance.

In essence, the aircraft is placed in a flight status where there is a maximum generation of disturbing moment under a condition of minimum correcting moment. The aircraft therefore is operating under a flight regime in which any decrease in airspeed will result in a loss of directional control.

Numerous accidents occur throughout the world annually as a result of attempts to fly below V_{mc}. The aircraft will normally perform two types of maneuvers under this flight regime. The first is a roll into the dead engine, a maneuver which usually results when the pilot merely holds the ailerons and rudders against the turn with no power change.

The second maneuver is that in which the airplane rolls violently or snaps away from the dead engine. This type maneuver is normally accomplished when the pilot abruptly closes the throttle on the good engine.

In analysis, the aircraft is set up in an ideal configuration for a spin into the good engine as soon as the throttle is closed on the good engine. Aileron and rudder are already in a position towards the good engine. It is not unusual for the investigator to find an aircraft which has spun in to the right with the left engine out, or vice versa.

2.5.8. Yaw Dampers and Rudder Boost

Although some straight wing aircraft have a characteristic of yawing left or right in normal flight, the advent of the sweptwing jet transport has resulted in a considerable number of problems of yaw due to the sweep back of the wing. This is caused by the location of the airfoil sections within the sweptwing, and any yaw left or right causes an aerodynamic deficiency in one wing, and an increase in aerodynamic efficiency in the other wing, which in turn aggravates the yaw and roll situation. For this reason, yaw dampers were developed to set up an opposite motion and/or damping effect to this undesirable yaw characteristic.

The yaw dampers are gyro-referenced devices and are usually part of the autopilot system or the rudder boost control system. Rudder boost means that the rudder pressures are so high, due to the size of the control surface, assistance is required to actuate the rudder. In an accident involving a directional control problem, the system checkout of the yaw dampers, rudder boost, and associated systems, is mandatory.

2.5.9. Ventral and Dorsal Fins

It is the consensus of airplane designers and aerodynamicists that the size of the vertical tail surface indicates the number of design errors within the aircraft. If one studies a number of designs in which the vertical tail area has been changed, he will find that the area, after a given design has been approved, will have been increased rather than decreased. One method of increasing the area is by the use of the dorsal fin which is located on the topside ahead of the vertical stabilizer. A lower fin has been installed on a number of aircraft, the ventral fin.

In discussing vertical tail surfaces, ventral and dorsal fins, the whole problem is that of vertical sideplate area of the fuselage. If one locates the mean aerodynamic chord, and in turn the center of gravity of the aircraft, which is approximately the quarter chord point of the mean aerodynamic chord, it will be found that there must be more vertical fuselage area aft of this point than in front. Otherwise, the aircraft will have a tendency to reverse in direction. It is for this reason that when numerous land planes are converted to seaplane operation the vertical sideplate area of the floats is such that it increases the fuselage area ahead of the center of gravity, and as a result, dorsal or ventral fins must be added to the aft fuselage area.

2.6. Lateral Stability

Lateral stability deals with stable characteristics or roll around the longitundinal axis of the aircraft. During any phase of time in which the rate of roll is increasing or decreasing around the longitudinal axis of the aircraft it is out of equilibrium in lateral stability. To illustrate: Visualize an aircraft flying straight and level entering a turn to the right; a 30-degree bank is maintained during this coordinated turn. As aileron pressure is applied to roll the aircraft to the right, there is a differential of lift between the left and right wings, and the aircraft rolls to the right. When a desired angle of bank is obtained, it is necessary to put the aircraft into conditions of equilibrium relative to wing lift. This means that the lift has to be the same on the left and right wings. This fact is easily demonstrated in flight, for in holding a constant bank attitude in a coordinated level turn, it will be necessary to hold slight opposite aileron. This is in effect equalizing the wing lift, since the outside wing travels at a slightly higher velocity than the inside wing.

2.6.1. Dihedral

The oldest and the most common method of obtaining some degree of lateral stability is accomplished by utilizing a dihedral angle.

This is the angle as measured from the horizontal lateral axis of the aircraft to the spanwise reference axis of the wing. When the reference line or plane of the wing is above the horizontal lateral axis, it is dihedral or positive dihedral. If these wing surfaces are below the horizontal reference axis, they are referred to as negative dihedral, cathedral, or antihedral. Although most aircraft possess dihedral, a few designs, particularly in the military, incorporate antihedral or cathedral, a design in which the tips are mounted lower than the root sections for a specific purpose, which will be covered later.

2.6.2. Types of Lateral Control Devices

Numerous control devices have been designed in the lateral control area, the most common being the aileron. Even today in the modern jet transport, varied devices are in use for the purpose of lateral control. In the high-speed sweptwing transport, the problem of twisting the wing from outboard aileron control cannot be tolerated at the normal high cruise speeds.

The three basic manufacturers of American sweptwing transports, the Convair, Boeing, and Douglas companies, use three different devices.

The Boeing Company on the 707 series, as an example, utilizes outboard aileron controls in a similar manner to our subsonic aircraft, provided the flaps have been extended a specific degree. Once the flaps are fully retracted, the outboard ailerons are locked in the faired position, and cannot be used. The aircraft then depends upon an inboard aileron for lateral control in conjunction with spoilers.

The Convair 880 and 990 series, on the other hand, are designed without an outboard aileron, and lateral control is obtained by the use of inboard ailerons.

The Douglas DC 8 utilizes an outboard aileron system in conjunction with spoilers, however, the outboard aileron is so torsionally loaded that at high speeds it is not possible to deflect the ailerons to produce undesirable torsional loading to the wing structure.

2.6.3. Roll Rate

Roll rate is the rate at which an aircraft is rolling around the longitudinal axis. It is usually expressed in technical reports in degrees per second or radians per second. This term is also used in the witness investigation for such problems as inflight wing failures, V_{mc}, etc., in which the witness, with the aid of a model can illustrate what he thinks the rate of roll or yaw of the aircraft was during the period of observation.

2.6.4. Spiral Divergence

Spiral divergence is a term used to explain a divergence of an aircraft around any one or combination of the three axes of rotation. It is a relatively simple maneuver but potentially lethal. In layman's language, a divergent spiral is commonly called the "graveyard spiral." The aircraft accidents associated with the graveyard spiral normally possess the ingredients of weather and unqualified flight personnel. Although an aircraft displays normal positive dynamic stability around the lateral axis, which is known as longitudinal stability, it is found that the aircraft is basically unstable around the longitudinal axis in lateral stability.

If an aircraft is not monitored properly, a wing may drop for one reason or another. If this takes place, the aircraft by normal aerodynamic process of slipping tends to follow the flight path direction. Once the wing has lowered below normal, a turn is initiated because of the geometry of flight. The right wing increases in velocity slightly due to the larger radius of turn. If this is not corrected, more lift on the right wing than on the left wing results, which steepens the angle of bank. The nose of the aircraft lowers, and the speed of the aircraft increases.

The next aerodynamic phenomenon which occurs is that as the airspeed increases, the download on the horizontal tail surfaces increases, since this surface is attempting to bring the aircraft back to a trim condition. As the turn is tightened by natural longitudinal stability characteristics, the bank will tend to steepen, and the spiral as a consequence will tighten. The aircraft therefore will spiral out at extremely high airspeed. The problem reaches a critical stage when the aircraft descends from an overcast, and excessive aerodynamic loads are placed upon the structure during recovery, with a subsequent failure.

2.6.5. Artificial Devices

Artificial devices are placed in the control system to give an artificial feel to the flight controls. These are associated normally with a boosted control system in an aircraft that possesses extremely high speed or size so that the aircraft cannot be operated by normal push-pull tube or cable-control system. These devices normally sense the dynamic pressure in certain areas on the aircraft, and this parameter is utilized to establish a feel load on the controls.

Part C — AIRCRAFT AIRWORTHINESS INVESTIGATION

Chapter III

STRUCTURES

1. Structures — General

Fortunately for the pilot and the flying public, structural failures cause only a relatively small number of aircraft accidents. In the air transport field relatively few accidents involving structural failure of a major component have occurred. In the light plane field a greater number of major inflight structural failures have occurred, most attributable to excessive loads imposed when the aircraft's limitations were exceeded by the pilot. Both types of aircraft have experienced landing accidents because of structural failures of landing gears. This phase of the problem continues to be important.

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

These problems challenge the accident investigator to ferret out answers against almost insurmountable odds. Whether he can or not depends on his training, experience, and perseverance.

Two basic questions face the investigator following an accident involving suspected 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 how to recognize the various fractures he will encounter. The first question requires a knowledge of investigatory procedures and techniques, and the latter that he have a knowledge of fracture analysis.

The "why-did-it-fail?" question is generally more difficult to answer, and an understanding of aircraft loadings is a prerequisite.

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

- Procedures and Techniques
- Fracture Analysis
- Aircraft Loadings

Only those procedures and techniques peculiar to structural failure investigations will be considered. It is assumed 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 fatigue failures and static failures. Recognition of different types of fractures is emphasized, since this point has been too often neglected. This phase is of chief interest to the investigator whose main task it is to ferret out the cause of failures from a pile of wreckage.

In another section, the overall problem of aircraft loadings is presented in considerable detail in the belief that the average investigator's knowledge of this subject is somewhat limited, and that a more thorough understanding of the aircraft loading problem will produce better investigations.

In arranging the material, it seemed logical and desirable to reverse the order of presentation, to begin with the subject of aircraft loadings, following with fatigue, static failures, and procedures, in that order. This was indicated since the first three topics are basic information and must be thoroughly understood before the investigator initiates his examination of the wreckage at the accident scene. The section on aircraft loads can be considered necessary background material; the sections on fatigue and static failures as basic tools for detecting failures; and general procedures of investigations as techniques employed to use the background information and the basic tools effectively.

1.1. Aircraft Loadings — Variations in Loading Application

The mode of load application has an extremely important bearing on the way a part fails in service. Any breakdown or typing of variations in loading applications is arbitrary, since in general the difference between types is one only of degree. Thus one mode of load application blends into another as rate of loading is decreased or increased. Or changes in the frequency of loading will result in a change in the mode. No hard or fast rule can be stated, however, for purposes of investigation it is sometimes convenient to look upon a particular loading as one type or another. For this reason, in the following discussion the various modes are arbitrarily divided into three types: static, repeated, and dynamic.

a. *Static Loading* — Static loading can further be divided into short-time static loading, and longtime static loading:

(1) *Short-time* — In short-time static loading, the load is applied so gradually that all parts are at any instant essentially in equilibrium, i.e., the simple, conventional stress formulas can be used directly. In testing, the load is increased progressively until failure results, and the total time required to produce failure is not more than a few minutes. In service, the load is increased progressively up to its maximum value, is maintained at that maximum value for a limited time, and is 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 design criteria. Loads imposed upon the aircraft by various maneuvers or by isolated peak gusts are generally considered as static loads.

(2) *Longtime* — In longtime static loading, the maximum load is applied gradually as before, but the load is maintained. In testing, it is maintained for time sufficient 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 longtime tests at the temperature prevailing under service conditions. This type of loading application is generally only important 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 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.

b. ***Repeated Loading*** — In repeated loading, the load or stress is applied, and wholly or partially removed or increased many times in rapid succession. This is the type of loading application which is associated with fatigue. Generally speaking, repeated loading implies a large number of load application. However, under certain conditions, repeated loading of only a relatively few cycles can produce an effect similar to a large number of cycles. This point will be explored further in the discussion on fatigue. The important point to remember is 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 psi is required to break a round bar of 2014-T6 aluminum alloy under static tension loading conditions. Yet this same bar would fail after about 100,000 cycles of a reversed bending load that produces a maximum repeated stress of only 20,000 psi. Cycles of this order of magnitude can be and often are encountered within the lifetime of an aircraft. In the aircraft field, atmospheric gusts and vibration produce a repeated type of loading. For some aircraft, maneuver loads are significant.

c. ***Dynamic Loading*** — In the two types of loading discussed, a state of equilibrium existed, i.e., the external loads were in balance with the internal loads. In dynamic loading, the loaded member is in a state of vibration and static equilibrium does not exist for a time. Loosely speaking, there are two classes of dynamic loading — sudden loading and impact loading.

(1) *Sudden Loading* — Sudden loading occurs when a weight or 'dead load," not in motion, is suddenly placed upon a member or structure. A beam would be thus loaded if a weight were suspended by a cord which allowed the weight just to tough the beam, and the cord was then cut. The stress and deflection so producted would be approximately twice as great as if the weight were eased 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 particular type of force or load being considered and upon the stiffness of the system. In the aircraft field, gust loads are forms of sudden loading, although as will be seen later, they are handled as static loads.

(2) *Impact Loading* — Impact is generally associated with motion, as when one body strikes another. Unusually high forces can be developed under impact loading. This type of loading has no direct place in aircraft design (a possible exception would, perhaps, be in design for crash survival) but it is important in aircraft accident investigation. Materials which ordinarily fail in a ductile manner under static loading can be made to fail in a brittle manner if the rate of loading is high enough. In this connection, the rate of loading has to be appreciably greater than 50 feet per second for this type of loading to be significant.

It should be remembered that even when an aircraft strikes the ground at high speed, because of elasticity in the structure, many of the parts are loaded at considerably lower rates than the impact velocity of the aircraft would indicate. Impact loadings occur in the air during structural breakup following, say, unusual maneuvering. The point to remember here is that very high forces can be developed by light objects or parts traveling at high speeds hitting other parts of the aircraft. The procedures used to distinguish impact damage will be covered later.

2. Materials

The complexity and variety of systems within a modern aircraft are matched by the complexity and variety of materials that comprise the makeup of the aircraft.

The Air Safety Investigator should familiarize himself 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 therefore recognize when technical assistance is required.

As an example, welding, brazing, and soldering are similar only in that they are methods of attaching metal to metal. As there are various welding methods, there are various brazing and soldering methods.

Aircraft structural materials encompass the fields of metals, plastics, fabrics, and woods. The Air Safety Investigator during his career will certainly become involved in one or more of these areas.

2.1. Materials Used in Aircraft Structures

The basic structural materials used in aircraft are aluminum alloys and steel. Wood has been used to a considerable extent in the past and some wooden components remain in service. Plastics and other nonmetallic materials are used in some applications and may become increasingly more important as their strength, aging characteristics, and dependability are improved and new ways of using them in combination with metals are developed. Composite materials are used in some structural applications and their use probably will increase in the future. Properties favorable for certain applications can be developed better with combinations of materials than with any single material. Examples of such combinations are (1) sandwich structural panels made with exterior surfaces of fiberglas or thin aluminum sheet and a core of aluminum honeycomb or plastic foam, (2) glass, graphite or metal fibers embedded in thin sheets of plastic which may be bonded together to form tubular or solid components, (3) solid bars of a metal, such as aluminum, reinforced with graphite or boron fibers.

There are numerous sources of additional information on these and other materials. One of the most comprehensive sources for metals is the Metals Handbook published by the American Society for Metals. Four volumes currently available are, Vol. 1, *Properties and Selection of Metals;* Vol. 2, *Heat Treating, Cleaning and Finishing;* Vol. 3, *Machining;* Vol. 4, *Forming.*

A few of the most common structural materials will be discussed in more detail in the following paragraphs.

2.1.1. Steel

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 maximum 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 contents, as follows:

Material	Carbon Content
Wrought iron	Trace to 0.08%
Low carbon steel	0.10% 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 and the Society of Automotive Engineers. 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 52100 steel has 1.00% carbon):

10XX	Plain carbon steel
11XX	Carbon steel with additional sulfur for easy machining
13XX	Carbon steel with about 1.75% manganese
23XX	3% nickel steels
25XX	5% nickel steels
31XX	1% nickel with some chromium
33XX	3% nickel with some chromium
40XX	0.25% molybdenum
41XX	Molybdenum steel with 1% chromium
43XX	Molybdenum steel with nickel and chromium
46XX	Molybdenum steel with 1.7% nickel
48XX	Molybdenum steel with 3.5% nickel
50XX or 50XXX	1.30% chromium steels
51XX or 51XXX	1% chromium steels
52XX or 52XXX	1.5% chromium steels
61XX	1% chromium steel with 1.15% vanadium
86XX	1.5% chromium, 0.5% nickel, 0.2% molybdenum
87XX	0.5% chromium, 0.5% nickel, 0.25% molybdenum
92XX	2% silicon steels
93XX	3.25% nickel, 1.20% chromium, 0.12% molybdenum
98XX	1% nickel, 0.8% chromium, 0.25% molybdenum

There are, of course, 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.

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

Alloy Designation	Carbon	Chromium	Nickel	Other Elements	General Class of Steel**
302	0.15 max.	18	8	--	Austenitic
310	0.25 max.	25	20	--	Austenitic
316	0.08 max.	17	12	2.5 molybdenum	Austenitic
347	0.08 max.	18	12	0.8 columbium	Austenitic
410	0.15 max.	12.5	--	--	Martensitic
440A	0.65	17	--	--	Martensitic
430	0.12 max.	16	--	--	Ferritic
446	0.20 max.	25	--	--	Ferritic
PH15-7 Mo	0.07	15	7	1.15 aluminum, 2.25 molybdenum	Precipitation Hardening
17-4 PH	0.04	16.5	4.25	0.25 columbium, 3.6 copper	Precipitation Hardening
AM350	0.10	16.5	4.25	2.75 molybdenum	Precipitation Hardening

* Remainder is iron and normal impurities.

** The austenitic steels are either nonmagnetic or weakly magnetic.

The strength and ductility, or toughness, of steels is controlled by cold working or heat treatment. In general, any process that increases the strength of a material will also decrease its ductility. 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 stainless steels, that cannot be hardened by heat treatment. Examples of this process are the cold rolling of sheet or strip material and the cold drawing of wire. Surface rolling and shot peening are also cold working processes.

Heat treatments are 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. 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 steels.

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

Quenching. 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 steel. Quenching is the hardening treatment used for quenched and tempered steels. It leaves most of these steels very hard and brittle, with undesirable internal stresses, so that they must be tempered before they can be used. In precipitation hardening steels quenching is used in softening or annealing treatments. Rapid cooling in such steels does not increase their hardness but instead prevents the precipitation hardening that would occur if they were cooled slowly.

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 it to room temperature. Tempering reduces the hardness, increases the ductility, relieves internal stresses, and, thus, increases the toughness of the quenched steel.

Stress Relieving. (sometimes called *Process Annealing*). A process similar to tempering but primarily to reduce internal stresses.

Solution Treatment. 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 dissolve precipitated particles and then cooled fast enough to hold the constituents of these particles in solid solution.

Precipitation Hardening (sometimes called *Age Hardening*). Hardening of ferrous and nonferrous alloys by precipitation of finely dispersed particles from a solid solution. This can occur at room temperature in some alloys, but usually the material must be heated. The temperatures used are considerably lower than those employed for solution treatments.

Case Hardening. 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.

2.1.2. Aluminum Alloys

Aluminum alloys are used in aircraft in both cast and wrought forms. Castings are generally more brittle than the wrought materials and they are used mainly for light load applications where ductility is not an important consideration. Wrought aluminum alloy products, such as extrusions, forgings, and rolled sheet, plate and bar stock, make up the great bulk of structural materials used in aircraft.

An alloy and temper designation is used for the identification of wrought aluminum alloys. A four digit numerical designation identifies the chemical composition and a letter followed by one or more digits identifies the temper of the alloy.

In alloy (chemical composition) designations the first digit identifies the alloy group, the second digit indicates a modification of a basic type of alloy, and the last two digits identify a specific alloy composition (except in unalloyed aluminum where the last two digits indicate the purity of the metal). The following table gives the designations for the alloy groups.

		Alloy Number
Commercially pure and high purity Aluminum (Aluminum content 99% and higher)		1XXX
	(Copper	2XXX
Alloys	(Manganese	3XXX
grouped by	(Silicon	4XXX
	(Magnesium	5XXX
major	(Magnesium and silicon	6XXX
alloying	(Zinc	7XXX
element	(Other element	8XXX

Most of these alloys contain other elements that significantly affect their properties. The nominal compositions of a few of the most widely used alloys are given below as illustrations of the number and amounts of alloying elements involved.

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

*Remainder is aluminum and normal impurities.

In aluminum alloys, as with steels, the properties of any specific alloy can be altered by work hardening (often called *strain hardening*) or 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 can be strengthened by heat treatment are precipitation hardened after a preliminary solution treatment.

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.

There are numerous subdivisions of the H and T tempers. The basic subdivisions of the T temper are given below as this is the most common designation encountered in aircraft materials.

- 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.
- T3 Solution heat treated and then cold worked.
- T4 Solution heat treated and naturally aged to a substantially stable condition.
- T5 Cooled from an elevated temperature shaping process and then artifically aged.
- T6 Solution treated and artificially aged.
- T7 Solution treated and then stabilized. This stabilization treatment provides additional control of some special characteristic of the material.

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

T9 Solution heat treated, artifically aged, and then cold worked.

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

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

2.1.3. 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 nonmagnetic (as are aluminum alloy and stainless steel), and thus precludes the use of a magnet for material separation and identification.

Titanium alloying elements are chromium, iron, manganese, vanadium, molybdenum, and tungsten, as an example, Ti-8AL-1MO-IV, Ti-6AL-4V, and Ti-6AL-6V-2Sn are three alloys which may be used on a supersonic transport. It is noted that these titanium alloys consist of aluminum, molybdenum, vanadium, and tin in varying percentages.

Additional information on titanium follows:

1. Titanium has been used as a structural metal only since 1952.
2. Unalloyed titanium is sometimes used for firewalls, bulkheads, compressor cases, shrouds and tailpipes.
3. 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.
4. Additional important alloys are Ti-5Al-2.5Sn, Ti-8Mn, Ti-7AL-4Mo, Ti-7Al-4Mn.
5. Use of titanium alloys is expected to increase rapidly 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. Boeing expects their use of titanium alloy parts to increase from 25,000 parts per month early in 1969 to 2,500,000 parts per month when the SST reaches the production line.

2.1.4. Copper

Copper is used in its basic form primarily as an electrical conductor and for oil and fuel line tubing. In this area of fuel lines, copper has been almost 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 under a condition of vibration, and for this reason has been eliminated from the fuel system in the carburetor to firewall area. Where copper and aluminum lines are attached to the powerplant, "s" curved lines and 360° spirals are incorporated to relieve the stresses from vibration.

Copper in annealed by heating in an air furnace to 1100°-1200°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, nullifying such information. 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. Either tin, aluminum or silicon are used as alloying elements in the various types of bronze.

2.1.5. 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, possess resistance to wear and excellent oxidation-corrosion resistance under extreme conditions of elevated temperatures and environments of corrosive conditions.

The development of the turbosupercharger at the beginning of WW II was dependent on the discovery of metals and alloys which would maintain adequate strength at temperatures of 1300°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 WW II, and coupled with improved design have lead to long times between overhauls (TBO's).

Another series of alloys are those which 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 possesses from 4.5%-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, possesses good strength properties, and most important, possesses excellent physical properties at elevated temperatures.

There are also thousands of noncommercial alloys. *Noncommercial* is a term generally applied to experimental alloys which are not produced in quantity for commercial use. These alloys are developed in both private and government sponsored research programs for many different purposes.

2.2. Crystallization

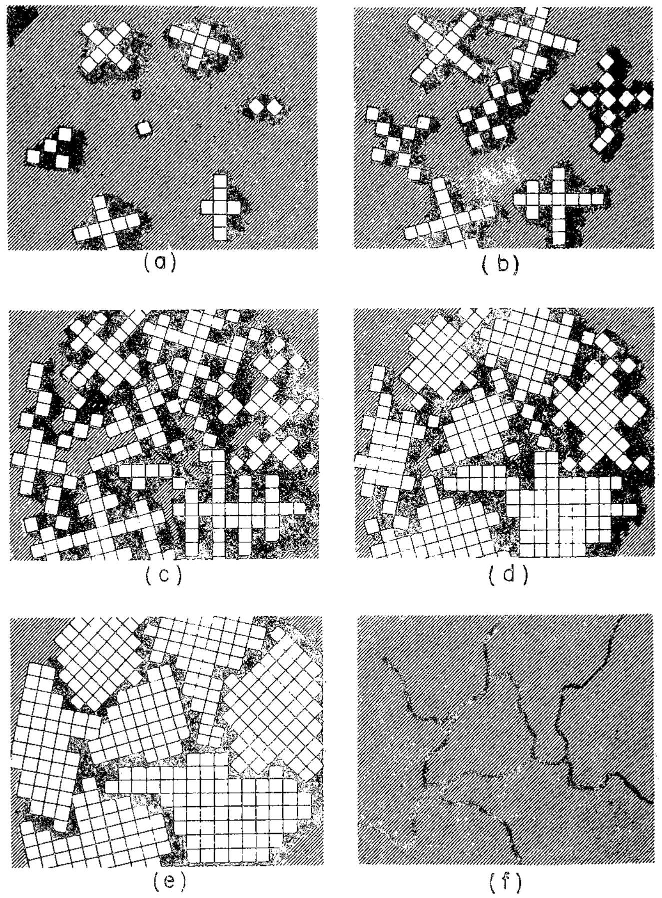
All metals are crystalline in their solid state; that is, they are made up of crystals or grains in which the atoms are arranged in a repetitive three dimensional pattern. Crystallization is the process by which metals solidify from a liquid state. It takes place by a mechanism of nucleation and growth. Reference Fig. C III-1 by Rosenhain. 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 crystallites then grow larger and larger, eventually growing together as the solidification process is completed.

The terms crystallization and recrystallization are two greatly mistreated words by the layman, the mechanic, and the untrained investigator. These words are used when a fatigue-type break is examined: "The metal crystallized and broke," or, "it was a case of recrystallization." The first expression, "It crystallized and broke," is technically correct, for at some time the metal was cooled from a liquid state to a solid – so it crystallized, and at some later time, it broke. These terms should be left to the metallurgist and his report.

2.3. The 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, therefore, is the result of a specific test, or one might say, the plot of a specific test, to provide 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



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). 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 crystal proper

Crystallographic Grain Growth of Metal

Figure C III-1.

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 pounds per square inch (psi).

The formula for stress is $S = P/A$, where S is the stress, P, the load in pounds, and A is the area in square inches. In the metric system, P could be in grams or kilograms, and A could be in square centimeters.

In the above case, where the rod carries a tensile load of 50,000 lbs., the rod has stretched in length to a certain degree. All materials will deform under a load condition, and since different materials deform differently, this is the purpose of plotting the stress-strain curve. When a material is deformed, it is under stress (S) and a strain (ϵ epsilon); you can't have one without the other. 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 of an inch, it is under a strain of 0.001 in. per in. ϵ (epsilon) = 0.001 in./in.

Referring to **Fig. C III-2**, note that stress is plotted on the vertical or Y axis (ordinate), and strain is plotted along the horizontal or X axis (abscissa). This is not a curve of a specific material, but is representive of a rather ductile material such as low carbon steel or 2024 aluminum alloy.

As a load or stress is applied, the material begins to deform or stretch if it is under tension. 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.

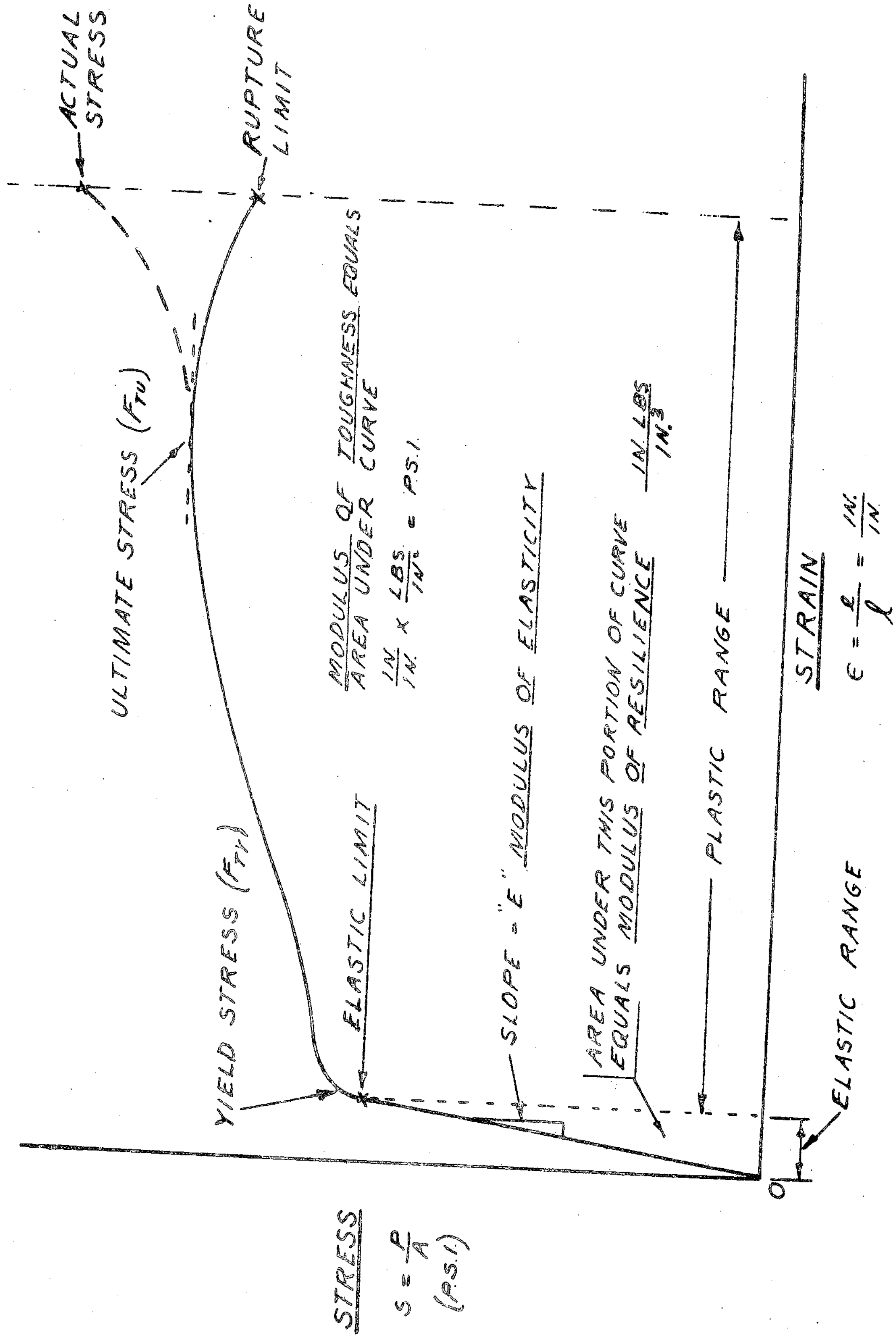
In studying the elastic range and the elastic limit, it is obvious that an aircraft structure should operate at some point on the straight line portion of this plot, not exceeding the elastic limit. The investigator should associate the design term ***limit load factor*** with the straight line portion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

If an individual desired to build an aerobatic type aircraft with a *limit load factor* of six, 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 degree 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 **Fig. C III-2**, note that the material will continue to sustain higher loads without separating or breaking.

3. Design

Present day aircraft are designed essentially for static loads. 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 of external and internal loads results. These loads fall into the short-time static loading category. At the present time precise methods for determining the life of parts or components under repeated loading are not available to the designer. Considerable research is underway to devise suitable methods to handle repeated loading. In the past, the designer has intuitively proportioned the parts so that the effects of repeated loading are reduced in importance. 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 in the static design criteria, or they have been small enough to neglect.



Stress-Strain Curve
Figure C III-2.

a. ***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 primary structure due to repeated loading or fatigue is of constant concern to the designer. Dynamic effects are being given closer attention today. Lower design load factors, higher speeds, 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 carefully examine all structural failures to determine if some other cause aside from exceeding the static strength is responsible for the failure.

3.1. External Aircraft Loadings

The actual loads imposed upon the aircraft structure during inflight and ground operation are extremely complicated in nature, and a rigorous analysis of the structure for the almost infinite number of loading combinations would be an impossible task. For practical reasons, therefore, the aircraft structure is designed for a relatively small number of simplified conditions, so selected to bracket all the critical conditions likely to be encountered by the particular aircraft in its useful lifetime. These loading conditions or requirements are set forth in the Federal Aviation Regulations for commercial aircraft. Transport aircraft are designed according to FAR Part 25 requirements. Light planes and larger aircraft in nonscheduled passenger and cargo use are designed 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 equilibrium.

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

3.1.1. Airloads

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

3.1.2. Maneuvering Loads

When an aircraft is flying in straight and level flight the lift forces on the wing and tail are in balance with the overall weight of the aircraft, and the aircraft is said to be operating at a one "g" level. If the aircraft is maneuvered by deflecting the control surfaces in one direction or another and until the aircraft reaches an equilibrium condition in its new altitude, the forces acting on the aircraft are either raised or lowered from their initial value 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 of control surface displacement and the speed entry into the maneuver. The higher the rate of change or the higher the speed, the higher will be the resultant load factor developed. If at any instance in a maneuver the acceleration forces are balanced by equal and opposite inertia forces, the airplane can be considered in a state of equilibrium and a static type of strength analysis may be made.

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 for the specified values. For transport type aircraft, where maneuverability is secondary, strength is provided for a limit positive load factor of 2.5. The load factor used in light plane design depends on the use to which the airplane is to be put.

Acrobatic type of aircraft, for which all maneuvers are permitted, are designed for a limit positive maneuvering load factor of 6.0.

Aircraft designed to the utility category requirements of FAR Part 23 are provided with a strength of 4.4 load factors. 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 only 3.8. Some of our modern light planes are designed to meet both normal and utility categories. 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 for the airplane. Any of these values can be exceeded by the pilot if he fails to observe the limitation of the aircraft. Very often people will ask, "Why can't you design the airplane so that the pilot cannot fail the structure by any flight maneuver?" The answer, of course, is that the aircraft could be designed in this manner, but that the cost in weight would be prohibitive. With some of our modern light planes, 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 failure within the aircraft maximum speed limitations.

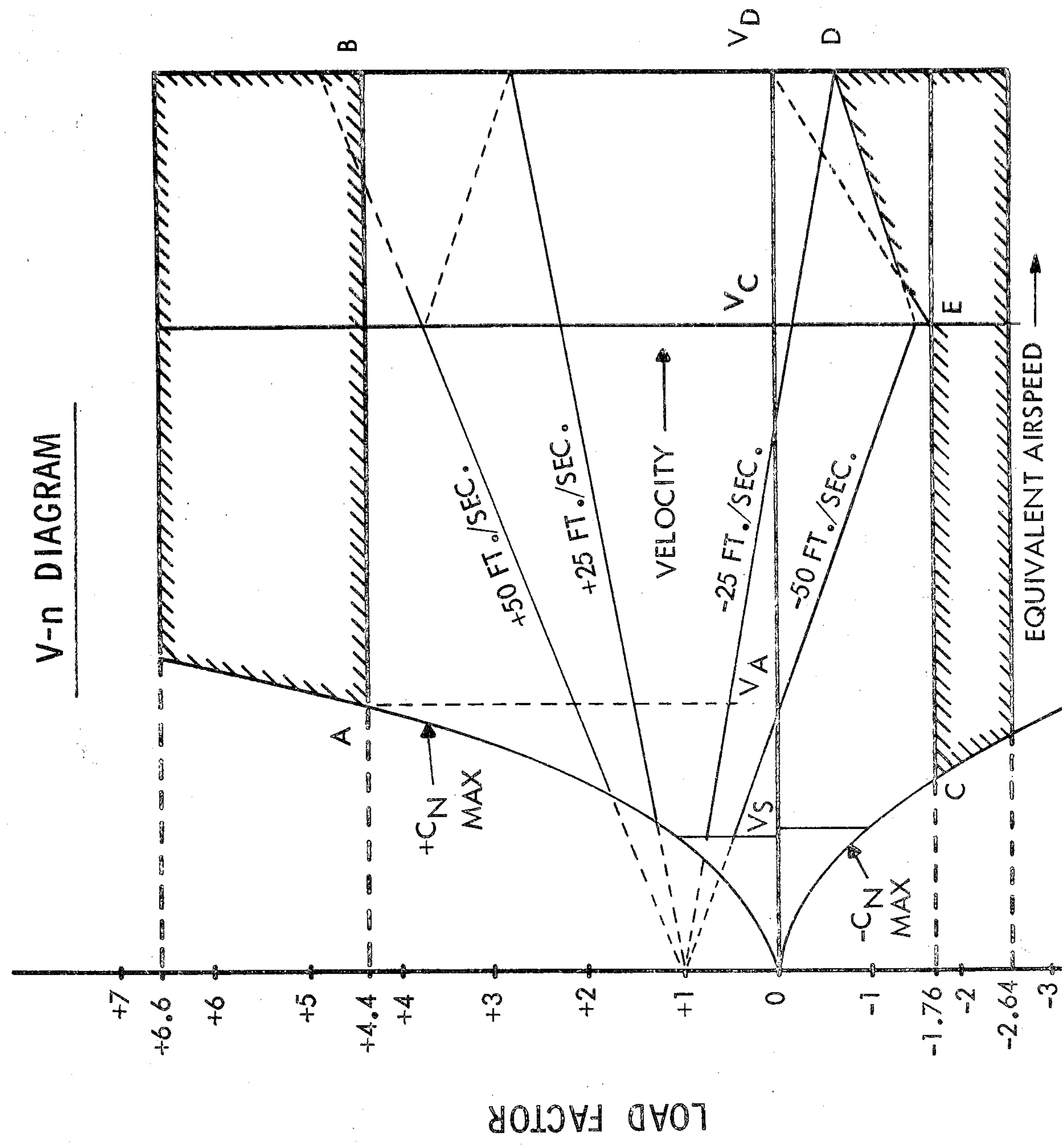
a. *V-n Envelope*

The basic maneuvering flight conditions 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. Such a diagram is shown in **Fig. C III-3.** This diagram shows all of the design conditions and is also useful in illustrating the operating limitations. The lines AB and CD represent the restricted positive and negative maneuver load factor which are limited to speeds within the line BD, the design dive speeds. 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 maneuver acceleration, 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 CD can be developed and the airplane will stall. Generally speaking, if the airplane is designed for the airloads resulting from the conditions A, B, D, and C, the corners of the diagram, it will be safe from a structural strength standpoint if it is flown within the specified limits 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. The various speeds used in design are also shown on the V-n diagram.

With the conditions specified on the V-n diagram as a basis, the designer balances the airplane for the various forces and computes the loads on individual components, such as the wing, tail, etc. These are then used to design the various members and components. From an investigation viewpoint, a thorough understanding of the significance of the V-n diagram is often useful in evaluating inflight structural failures. In this regard, it should be remembered that 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.

b. ***Tail, Flap, Aileron Loadings***

The tail loads determined by the procedure described in the previous subsection are referrred to as balancing tail loads. In addition to these balancing tail loads, maneuvering tail loads are prescribed in the pertinent regulations. The



V-n Diagram
Figure C III-3

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. The flap is designed for the loads imposed upon it when extended to any angle up to the maximum used. The ailerons are designed for the balancing loads while in the neutral position, and, also, the maximum displacement at speed, V_a.

In general, the horizontal tail, the vertical tail, and the ailerons are designed for full displacement at the maneuvering speed, V_a. This means, then, that these surfaces can be moved as abruptly as the pilot desires at speeds below V_a without danger of 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.

3.1.3. Gust Loads — Gust Envelope

All commercial aircraft are designed for the loads imposed by a ±50 ft./sec. gust at the cruising speed of V_c from sea level to 20,000 ft. and then reduced linearly from ±50 ft./sec. to ±25 ft./sec. at 50,000 ft.

At V_d speed, the gust factor is ±25 ft./sec. from sea level to 20,000 ft. and then reduced linearly from ±25 ft./sec. at 20,000 ft. to ±12.5 ft./sec. at 50,000 ft.

In addition, transport aircraft are designed for a ±66 ft./sec. gust at V_b from sea level to 20,000 ft. then reduced linearly from ±66 ft./sec. at 20,000 ft. to ±38 ft./sec. at 50,000 ft. (V_c is design cruise speed and V_b is design speed for maximum gust intensity). V_b is often referred to as the "rough air speed." (Reference is made to FAR Part 23 and Part 25.) The specified gust velocities are vertical velocities, and their effect on the aircraft is to produce a sudden change of angle of attack which increases or decreases the load on the wings (depending upon whether the gust is positive or negative). This produces incremental accelerations. These incremental load factors are added or subtracted, as the case may be, to the one "g" load factor and the aircraft is designed for the resulting loads. The diagonal lines on Fig. C III-3 are a graphical representation of the "gust lines" and show the load factor developed when a gust of a certain intensity is encountered at various speeds. In the illustration shown, the gust lines fall outside the maneuvering envelope at some points, and the gust loads are the controlling design loading factors. For some designs, these gust lines fall within the maneuvering diagram and are not critical. Whether gusts are critical or not depends upon the wing loading, W/S, and certain aerodynamic characteristics. In general, it can be said that the higher the design maneuvering load factor, the less likely is it that gusts will be critical. For this reason, then, gusts are more significant in transport aircraft design (where the maneuvering load factor is 2.5) than in acrobatic aircraft (where the maneuvering load factor is 6.0). In any event, the designer must check for gust loads as well as maneuvering loads, and select the critical loads for the particular design.

3.1.4. Distribution of Loads

To intelligently understand and evaluate inflight structural failures, the investigator should have some knowledge of the way in which the airloads are distributed on the various components. A few brief notes on this general subject are included in this subsection.

The lift on an airfoil shape is developed by a differential pressure between the upper and lower surfaces. For positive angles of attack, the pressure on the upper surface is less than ambient, which, in effect, means that there is a suction force acting away from the airfoil. For this same condition, the pressure on the lower surface is positive, or, in other words, the force is acting toward the airfoil. The center of pressure at high angles of attack is generally forward on the airfoil and moves aft as the angle of attack decreases. For this reason, wing failures resulting from violent maneuvers generally start at or near the leading edge or front spar, and are in an upward direction whereas failures while in a dive generally start farther aft on the airfoil, nearer the center or rear spars. Downward leading edge failures are also indicative of low angle of attack or dive conditions. These points are sometimes helpful in establishing the maneuver which produced the structural failure.

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 is 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. The distribution of loading on flaps, ailerons, and tabs is approximately triangular with the maximum unit loading occurring at the hinge line and dropping off to zero at the trailing edge.

3.1.5. Ground Loads

Ground loads are those loads which are imposed on the structure during the actual landing and during operation on the ground such as taxiing, braking, turning, etc. These loads are an important part of the overall design requirements. Whether a particular component or member will be designed by ground loads or airloads cannot generally be predicted until both are computed and compared with one another. In general, however, ground loading conditions are often critical for the wing structure inboard of the gear and for the fuselage structure. In all cases, of course, the landing gear attachment points are designed for the pertinent ground loads.

a. *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 in operation. For a tail wheel type of aircraft two basic conditions are used, the level landing and the taildown 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. 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.

(1) *Descent Velocity Considerations*

For transport type of 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 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 gear. A landing gear designed for a 10 ft./sec. descent velocity can absorb the shock of an unflared landing at 600 ft./minute. This is a hard landing. When landing gear failures occur, then they are usually associated either with fatigue failures, running over obstructions, or with free drops of the airplanes from considerable heights when the ship is "dropped" in from a stall.

(2) *Spin-Up and Spring-Back Loads*

When an aircraft initially touches down, the wheels are not rotating, and a considerable drag force is required to bring the wheel assembly up to the speed of the aircraft. This drag load is referred to as the wheel spinup load, and formulae are available for calculating its magnitude. The wheel spinup load is a very important design condition. In fact, considerable research has been conducted to determine methods for reducing this load. Oleo drag struts as used on some particular aircraft are examples of one method used successfully to reduce the effect of wheel spinup loads.

The wheel spinup 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 spinup load drops off to zero, and the gear springs forward so that at the instant of reaching the maximum forward deformation a dynamic springback load may be considered to consist of the inertia of the effective mass at the axle acting forward normal to the oleo. The springback load is an important design consideration.

b. *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 tie in the instant of 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.

c. *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 must be considered. In either case, however, the wing, fuselage, and other components are subjected to inertia forces, usually in a downward direction. The designer balances the airplane for all of the forces involved and checks out the structure strengthwise. In many cases, the landing conditions are more critical than the flight conditions for certain components.

3.1.6. Vibratory Loads

Vibratory loads are the most troublesome type encountered in aircraft design and, 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 sources of trouble and 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 mounts, cowlings, isolated skin panels, ribs, etc., are all 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 slipstream 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 frequencies and are more or less of constant amplitude. Overall loads, on the other hand (aerodynamic maneuver and gust loads, landing loads), are generally of lower frequency and of varying amplitude.

3.1.7. 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 into internal loads will be discussed.

a. *Ways in Which External Loads Are Supported*

Basically, the external loads are resisted internally by tension, compression, shear, torsion, and bending forces. In many applications, bending moments 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.

All of the 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 at the root. The designer's job then, is to provide sufficient structural material 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.

The torsion loads are resisted by the skin covering and by the spar webs. The torsion loads are distributed to the various 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, etc., special methods are used, but the basic concepts are essentially the same. The skin covering and spar webs are almost always designed as tension - field beams. This means that the sheet is permitted to buckle, but it should be remembered that even in this buckled state the sheet is still carrying 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 come out when the load is relieved. This is often a useful indication of excessive loads, and the direction of the buckles will tell 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 noseup 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 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 spar attachment joints with an overhang forward and an overhang rearward. The section of the fuselage aft of the rear spar attachment 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.

In examining the structure after an accident, it is desirable to be able to distinguish between inflight 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 primarily for tension and compression forces. If a piece of spar cap at the accident scene is found twisted, then it may be possible to say that that particular failure was secondary and occurred after the integrity of the wing was destroyed by other features. Similarly, if a member is known to carry only tension loads in its normal life and the part is found failed by bending or compression forces, this failure, likewise, can be said to be secondary. In this manner, it is possible to eliminate quickly a great many of the failed pieces. If it is difficult to decide whether a particular piece failed from impact or inflight loads, adjacent pieces are pieced together and their failure pattern studied. The basic tool in all of this work, however, is a thorough knowledge of how the structure carries the loads.

4. Failure Analysis — Fatigue

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.

4.1. Importance of Fatigue in Aircraft Field

Why is fatigue so important in aircraft design and operation? There are a number of reasons why fatigue is assuming more and more importance. In the early days of aircraft design, maneuver loads were the only ones used to design the airframe. Gust loads were completely disregarded in design until the 1930's. The strength that an airplane required in order to withstand the intentional maneuvers, such as sharp turns and pullups from a descending flight path, were adequate to cover gust loads. As transports grew larger, they were being maneuvered more cautiously, and the design maneuver loads were correspondingly reduced. It then became necessary to include in the design an investigation of the strength under the load imposed by single but severe vertical gusts. The gust design requirement is the one previously explained.

Unsatisfactory service experience with one aircraft has brought the fatigue problem to high priority, especially that aspect of the problem dealing with the cumulative effect of repeated gust loads. The emergence of fatigue as a design problem in airframes stems from the general trend of development in aviation.

Transports are becoming ever larger and more expensive; the operator is therefore forced to fly them longer until he can afford to retire them in favor of new models. Flight speeds are ever increasing. Longer life and higher speeds mean more miles covered; consequently, more gusts encountered. Higher airplane speeds mean higher loads for a given gust velocity. More refined methods of design for static strength bring about a reduction of the so-called hidden factors of safety. Finally, development in materials has brought about improvements in the static strengths of the materials, but unfortunately without proportionate improvements in the fatigue properties.

Although it may be dangerous to generalize, it is possible to note certain differences between transport aircraft and lightplane aircraft with regard to the incidence of fatigue. Because of the lower design load factor, higher speeds, operational procedures, etc., fatigue of major components such as the wing is an important consideration in transport design. In light aircraft, on the other hand, these same factors are such that the importance of fatigue is somewhat lessened. Fatigue failures of such items as lift strut fittings resulting in wing separation in flight indicate, however, that fatigue of major components in light aircraft cannot be ignored. Fatigue failures of landing gear components and their supporting structures occur in both types of aircraft with about the same incidence.

Some figures here will give you an idea of the importance of fatigue in aircraft. One authority has estimated that three-fourths of all failures in aircraft parts are caused by fatigue. Our experience in aircraft accident investigation work bears out this figure.

4.2. Basic Theory

Although fatigue failures have been occurring for a hundred years or more, the actual mechanism of fatigue is not clearly understood. Various theories have been proposed to explain the phenomenon, but as new experimental results become available most theories are being discarded. 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 very large number of crystals whose axes have a random orientation and whose elastic moduli vary depending upon the particular crystal axes being considered. When a small tensile stress is applied to a piece of the metal, the actual internal stress is not uniform because of the haphazard arrangement of the crystals and the different elastic moduli. It can be seen, then, that the shear 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, the localized stress reaches a value at some point in excess of the ultimate strength, and a crack starts. This will generally occur in a zone where much slipping has taken place. 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 slip bands, until the stress in the remaining section is sufficient to cause failure by tension.

The above can be summarized in the following manner:

a. No damage is observed microscopically.
b. Slip line formation takes place with gradual broadening of these lines.
c. The crack sets in most frequently at site of marked slip.
d. The crack continues to grow larger until the remaining uncracked cross sectional area is too small to withstand the applied load.
e. Complete failure occurs.
f. The final overload part of the fracture can be in bending or torsion, as well as tension, depending on the kind of load applied.

It cannot be emphasized too strongly that the above concept of the slip mechanism of fatigue has been quite firmly established.

4.3. Loadings

It has already been noted that fatigue is caused by repeated loading. The form of the repeated loading can be alternating bending loads, alternating axial loads, alternating torsion loads, or alternating shear loads. All of these forms of repeated loading are encountered in the aircraft field.

For the overall airframe, the repeated loading is produced in flight by alternating gust and maneuver loads, and on the ground by alternating landing loads. When an aircraft is flying in level flight, the wings are supporting a load equal to the weight of the aircraft, or putting it another way, the wings are stressed to 1 g. When the aircraft encounters a gust, the load is increased or decreased from the 1 g value, depending upon whether the gust is positive or negative. Surveys have shown that the frequency of negative gusts is about the same as positive gusts. It can be seen, then, that the load in, say, the lower spar chord varies about a mean stress of 1 g. The magnitude of the deviation from 1 g varies, since the airplane will encounter gusts of varying intensities.

4.4. Factors Affecting Fatigue Strength

The term *fatigue strength* is best explained by the fact that it depends on the material and the degree of stress concentration. *Fatigue life,* on the other hand, depends on the magnitude, frequency, and manner of loading. In this section some of the more important factors affecting the fatigue strength as it is related to material resistance are discussed.

While the designer does not exercise much control over the magnitude and frequency of loading which produce fatigue, he does control the design of the part. Judicious selection of material for a particular application, and careful attention to detail design offer the greatest promise today for the lessening or elimination of serious failures in service.

4.4.1. Factors Which Impair

In discussing factors which impair the fatigue life of a part, terms such as *smooth specimens, notched specimens,* and *stress concentration* are met. Since an understanding of what follows depends to some extent on a knowledge of these terms, a few words will be said about them first.

If a smooth, sound, polished specimen is put into a fatigue testing machine and tested at a particular stress, it will fail after a certain number of cycles. If a similar specimen, but with a notch machined into it, is tested at the same stress, the specimen will fail at a lower number of cycles than before. The ratio of these two figures is a measure of the effect of the notch and is called a stress concentration factor. The more severe the notch is, the higher is the stress concentration factor. Other factors are similarly compared with the smooth specimen values to arrive at a quantitative estimate of their effect on the fatigue strength of the material.

4.4.1.1. Stress Raisers

a. *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, scratches, threads, etc.

Almost all of the structural materials are sensitive to notches, and the fatigue strength of a part with a notch is less than for one without a notch. Within particular types of materials, there is evidence to indicate that the harder or higher tensile strength alloy is more notch sensitive than the softer alloys. This high notch sensitivity of high strength steels makes it imperative to use care in the design and maintenance of parts made from these steels in an effort to preclude failures.

Since notches as such cannot be completely eliminated from any design, we are forced to leave them and to 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 failures 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 these 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.

b. ***Decarburization***

Decarburization is the loss of carbon from the surface of a ferrous alloy as a result of heating in a medium that reacts with the carbon. The end result is a soft skin or "back" 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 siliconmanganese spring steels are especially susceptible to this effect. Decarburization does occur in other steels, however, and fatigue failures from the source 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. In service, decarburization frequently occurs in a notch, and the overall effect is to make the notch more severe.

c. ***Corrosion***

When a corroded part is subjected to repeated loading, the uncorroded fatigue life value for the metal is appreciably reduced. This follows since 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.

There is a corrosion condition in which splitting along the grain lines takes place. This is quite commonly observed in extruded members in which the material splits or flakes off. This is called exfoliation. This type of corrosive action can cause fatigue in certain installations in which the part is cycled with a variable tensile stress or an alternating tensile and compressive stress. The corrosion products work into a corrosion separation area and create a lever-fulcrum or wedging action. This results in a stress concentration at the tip or sharp edge of the wedge. This action could subsequently result in a separation of the part. This type of corrosion fatigue failure is not the same as a corrosion pitting type of fatigue failure. The latter is similar to machine-cut threads, stone nicks on propellers, or scratch marks.

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 tests figures 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.

Fretting corrosion can be considered as a special type of corrosion-fatigue. Fretting corrosion occurs when two parts are clamped, press-fitted or shrunk together and subjected to vibratory loads. In steel a red oxide powder is visible on the surface thus affected, while in aluminum or magnesium the powder is black. Fretting corrosion roughens the surface, inducing local stress concentration, and early fatigue failure results.

d. *Inclusions*

Many people believe that inclusions in a material are responsible for the majority of service fatigue failures. Actually only a minute fraction of service failures result from this cause. All metals have some 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 from his experiments 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 per cent 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 neglectful design. From the viewpoint of the accident investigator, the presence of inclusions is the last cause to be suspected.

4.4.1.2. **Internal Stress**

Internal or residual stress can combine with the external applied stress to increase or to decrease the actual maximum stress. The beneficial effect will be discussed later in the following section and only the injurious effect will be mentioned here. Certain heat treatment or fabrication methods can develop tensile stresses on the surface of a part. When the part is then subjected to repeated 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 results. In examination of a fracture, however, there may be little evidence pointing to the existence of a residual stress. It is then necessary to study the history of the part from the ingot to the finished piece. Heat treatment (including quenching practice), 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 excessive residual stresses.

4.4.1.3. **Clamping and Press-Fit**

Although the harmful effect of a sharp inside corner is rather universally 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. It has been shown experimentally that the endurance limit of a smooth shaft dropped from 88,000 p.s.i. to 45,000 p.s.i. when a collar was clamped to it. There has been a number of propeller blade failures due to clamping stresses. Failure in engine parts is quite often attributed to clamping stresses. Sometimes control system parts and helicopter shafting are similarly affected. 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.

4.4.2. Factors Which Improve

It almost goes without saying that by avoiding the pitfalls outlined in the previous section, the fatigue strength of a particular part will be improved. Good detail design and thoughtful manufacturing procedures will eliminate a great number of the difficulties which turn up in service. Aside from those general precepts, several things can be done to improve the fatigue strength of a part. Several of these are discussed below.

4.4.2.1. Surface Layers and Coatings

Under the section on decarburization it was explained how the soft carbon-deficient skin reduces the fatigue strength of a part. Under the section on residual stresses, it was explained how surface tensile stresses will also reduce the fatigue strength. The opposite of these two effects can produce a beneficial result. For 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. Aluminum Alclad sheet, on the other hand, has a lower fatigue strength than the unclad sheet, since the cladding of pure aluminum is much weaker than the core material. When a part has a hard case, the failure will in general occur in the core if the part is unnotched. If, however, the case is scratched, gouged, or otherwise notched, the failure may start either in the case or core depending upon the severity of the notch.

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. It has been found that the softer electroplating processes such as zinc, lead, and copper have no detrimental effect on the fatigue strength of steel when tested in air and actually increase the fatigue strength under corrosion conditions over the bare steel. Hard chrome or nickel plating, on the other hand, decreases the fatigue strength.

4.4.2.2. Cold Working

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 of the order of .025 in diameter.

Insofar as fatigue is concerned, cold working produces two beneficial results. First, residual compression stresses are induced. Second, the cold working aids the endurance limit. Extreme cold working, however, can have an injurious effect on a part. If the amount of shot-peening is not carefully controlled, the surface will be roughened and stress raisers induced. In addition, the notch sensitivity of the material may be increased sufficiently to counteract the beneficial effects.

Shot-peening is often used in aircraft design to lessen the effects of stress concentration due to changes in section. Rolled threads are a good example of cold working. In many applications, keyways, shafts, oil holes, bolt and rivet holes are cold worked to improve their fatigue strength.

4.4.2.3. Understressing and Overstressing

If a specimen is fatigue tested at a stress level just below the endurance limit for a large number of cycles, it will then be found that the endurance limit has been raised over the corresponding value for the virgin material. This procedure is called understressing.

If a specimen is fatigue tested at a stress slightly above the endurance limit for a relatively small number of cycles, an improvement similar to that produced by understressing may sometimes be noted. This procedure is called overstressing. If a still higher stress is used, or if a much larger number of cycles is used, then the fatigue strength of the piece will be damaged.

Both of these procedures have been used only in the laboratory so far. These effects are, however, some measure of moral comfort to the designer, since in operation, an aircraft encounters large numbers of small stresses and small numbers of large stresses. Also, the recuperative effect of understressing will offset, to some extent, the injurious effect of infrequent high overloads.

4.5. Recognition of Fatigue Fractures

As explained earlier, the term fatigue failure is normally applied to those fractures caused by repeated loading at a calculated stress considerably lower than would be required to cause failure under a single load application. The complete story of a fatigue failure is in almost all cases set forth on the face of the fracture. In other words, much valuable information relative to the magnitude and direction of loading and to the presence or absence of stress concentrations can be developed through a careful study of the fractured surfaces. Interpretation of the fracture, however, may not always be influenced by many variables. Some of the variables have been briefly touched upon in previous sections. Some contributing factors, like decarburization, can only be verified by laboratory examination. In many cases, on the other hand, the cause can be pinpointed in the field by careful study of the fracture alone.

Fatigue failures occur without perceptible ductility, as contrasted against static failures where considerable ductility or "necking down" generally takes place. This distinction is often helpful in isolating a part which has failed from fatigue. All brittle failures, however, are not necessarily fatigue failures and this feature must be used with other features to be described before a final determination is made. In addition, most fatigue failures (some torsion fatigue failures excepted) occur on planes which are at right angles or nearly at right angles to the loading. On a large number of parts the fatigue plane will be perpendicular to the axis of the load, and in the fatigue area the fracture will generally be in one plane. Irregular fractures, therefore, when the fracture slips from one plane to another and when these planes are very much different from a plane perpendicular to the loading or to the axis of the part, are very probably not fatigue fractures although close examination is often required to see if some small area on the fracture does not conform to the basic requisites. The two features of a fatigue fracture referred to in this paragraph are extremely useful in ferreting out a fatigue failure from a large number of failures. In fact, in those cases when the fractured surfaces are mutilated from subsequent damage, these features may be the only ones available to distinguish between fatigue and static failures. Having both halves of the fracture available so that the sections can be carefully fitted together and studied is almost a necessity in making determinations of this type.

As indicated previously, the most valuable information is contained on the fracture surface itself. The actual fatigue fracture surface is composed of two distinct regions: one smooth and velvety – the *fatigue zone;* and the other, coarse and crystalline – the *instantaneous zone.* The smooth velvety appearance of the fatigue zone is caused by rubbing of the mating surfaces as the crack opens and closes under repeated loadings. The coarse appearance of the instantaneous zone has given use to the erroneous "crystallization theory." For many years, in examining a fatigue fracture or in discussing them, people were accustomed to saying that the part had "crystallized." We know now that this erroneous belief sprang from the coarse appearance of the instantaneous zone. Metallurgically, it is untrue that the part or the metal in the part does crystallize under fatigue loading.

The first task, then, in searching out a fatigue failure is to look for the two distinct types of zones on the fracture – the fatigue zone and the instantaneous zone. In many fractures, more than one fatigue zone will often be found, indicating that several fatigue cracks had developed and were progressing at the time of the final failure. In each fatigue zone, the origin of the fatigue crack can be found by locating the center of radiation of the crack progression marks. These crack progression marks are variously known as *clamshells, oyster shells,* or *stop marks,* and are found in almost every service fatigue failure. It should be noted here that in some instances, the fatigue progresses without leaving distinctive wave markings, although in these cases the fatigue area can be identified by its smooth, rubbed, velvety appearance. In some cases, secondary techniques such as examining for absence of ductility and single failure planes approximately perpendicular to the loading or for the presence of stress concentration must be used to isolate a fatigue failure. Any suspicious or dubious fractures should be referred to a specialist for confirmation.

The many wave lines or bonds in a typical service fracture are caused by various degrees of rubbing as the crack either stops for certain periods or as it progresses at a varying rate under different stress levels. For this reason, the term "stop marks" as it is applied to the crack progression marks is perhaps more pictorial than the other two commonly used expressions since it indicates a hesitancy in the crack progression. Laboratory fatigue specimen failures very seldom show stop marks because the loading is almost always at a constant level.

In the following sections, the appearance of the fatigue fracture under various types of loading are illustrated, and the information that can be learned from an analysis of the markings is briefly discussed. It should be reiterated that fracture analysis as such is a complex problem and that this presentation cannot hope to cover all of the countless variations. However, knowledge of the material in these following sections should enable the investigator to recognize and diagnose the majority of service fatigue failures that he is likely to encounter.

4.5.1. Bending Fatigue Failures

Bending fatigue failures can be divided into three general classifications according to the type of bending load imposed, one-way bending, two-way bending, and rotary bending. Most severe bending fatigue will fall into one of these categories.

One-way bending results when a steady or a fluctuating bending stress is imposed on a part. Under this type of loading, the stress at one point on the outer edge of the piece, generally, is a maximum, and a fatigue crack will start here if the stress is above the endurance limit and if it is repeated long enough. Under two-way bending loading, the stress on both sides of the neutral axis is the same and, when the stress level and number of loadings are of the right order as before, cracks will start on either side of the part and progress toward the center. Rotary bending occurs when a part is rotated while under a bending loading. A typical example of rotary bending would be an engine crankshaft or a railroad axle under service loading.

When the stress level is low, the fatigue zone is large, and vice versa. Stress concentration affects the general curvature of the fatigue waves or stop marks. 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 stop marks increases markedly, and at the very high stress concentration the curvature becomes convex instead of concave. The displacement of the stop 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.

These general features, then, can be used to determine the type of bending loading applied, and, qualitatively, the stress level and presence or absence of stress concentrations. If the cross section under consideration differs widely from a symmetrical section, the actual significance of the markings as related to stress level and stress concentrations may be somewhat altered, but, in general, the same reasoning still applies.

4.5.2. Tension Fatigue Failures

Because of initial eccentricities in a part or because of eccentric loading, pure tension loading as such rarely occurs in service. Usually some amount of bending accompanies tension on axial loading. However, enough fatigue failures under predominantly axial loading do occur in service to warrant learning how to distinguish these failures from bending and torsional failures. Tension fatigue failures can generally be recognized by the manner in which the crack has progressed into the part. Parallel or constant curvature stop markings are characteristic of fatigue failures resulting from straight tension 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.

4.5.3. Torsion Fatigue Failures

Fatigue cracks usually progress in a helical direction from a rounded source of stress concentration like a corrosion pit or a hole, will tend to follow a line source (such as a tool mark or a longitudinal flaw in the material) in either the transverse or longitudinal direction. The direction of grain flow may also influence the direction of the initial crack in some materials. Fatigue stop 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 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 complete separation of the part. This characteristic can be used to isolate this type. In many service torsional fatigue failures, 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 torsion loading is present in the service application. In this regard, torsion fatigue should be suspected when examining failures of crankshafts, flap drive torque tubes, coil springs, splined shaft members, etc.

4.5.4. Static Failures

A static failure is defined as a failure resulting from one or a small number of load applications. The failure is characterized by permanent distortion or rupture of the member as a result of stresses in excess of the yield point of the material. This type of failure can be recognized by yielding over a considerable portion of the member in the region of the failure. The phenomenon is commonly referred to as "necking" in the failure of a conventional tensile test specimen. Impact loading may be considered as a special case of static loading where the speed of load application affects the magnitude of load.

4.5.5. When This Type Failure Occurs in Aircraft

Static failure will occur when loads in excess of the design loads are imposed on the aircraft or some component of the aircraft. In flight, this can happen when the aircraft is maneuvered too severely or at too high a speed. In landing or on the ground, this can occur when the aircraft is landed too hard or when the aircraft is taxied over an obstruction. The damage that results when an aircraft strikes the ground is of the static type, with impact loading being an important consideration.

4.5.6. Recognition of Common Fractures in Metal

The amount of distortion, yielding, necking, and size of the shear lips in a tensile separation depends on the ductility of the material. Very little, if any, of this 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 mate with one another or can be recognized as a pair.

4.5.6.1. Tensile

In a tensile failure, the fractured surface is ually made up of a series of planes inclined approximately 45 degrees to the direction of loading. In a thin part, such as sheet metal, there may be only one such inclined plane. Considerable local deformation or "necking" with a reduction of cross-sectional area is also generally evident. If the fracture is the result of pure tension alone, the two halves of the fracture will part cleanly and there will be no evidence of rubbing.

4.5.6.2. 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. 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 failures, the piece separates on oblique planes as in tension, except that there is rubbing of the two halves of the fracture during separation. In addition, in some materials there is a local increase in cross-sectional area where the material has yielded.

4.5.6.3. Bending

Bending is resisted by tensile forces on one side of the member and by compression on the opposite side. The apperance of the fracture in the respective areas is as outlined under tension and compresssion 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.

4.5.6.4. 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.

Shear buckling generally occurs in thin sheet metal such as wing skin or spar webs. The sheet will buckle in a diagonal fashion and the direction of force application can be told from the appearance of the buckle.

Failures of rivets, screws, or bolts in shear, are usually accompanied by elongation of the hole and there will appear behind the rivet a crescent shape open space. This result can be used to determine the direction of the shearing force.

4.5.6.5. 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 tubing members or a large open section, like the wing, torsion failures often occur as instability failures in a buckling manner. Again the direction of twist can be determined by close examination of the buckle.

4.5.6.6. Tearing Failure

Tearing failures in sheet metal, or heavier sections for that matter, generally occur in two distinct 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.

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. Examination of the fracture will disclose "herringbone" marks with the head of the herringbone pointing back to the origin of the tear.

4.6. Recognition of Common Fractures in Wood

Wood, like metal, fails in a distinctive manner under different loadings and 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, then, depends not only on the type of loading, but the direction of grain in the wood.

4.6.1. 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.

4.6.2. 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.

4.6.3. 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 has a flat carrot-like appearance.

4.6.4. Bending

Bending failures are essentially a combination of tensile and compression failures. In addition, there generally is considerable deformation of the fiber at the fracture in the direction of bending.

4.6.5. 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.

4.6.6. Torsion

Under torsion loads, wood will fail along the grain as in tensile failure and the apperance 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.

4.6.7. 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, therefore, the failing force on each ply must be compared with the others and the type and direction of loading arrived at. 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.

4.6.8. Glue Joint

Sometimes it is important to be able to determine if the glue joint or bonding two members has 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.

4.7. Recognition of Common Fractures in 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, then the outstanding thread ends which have a brushlike 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.

4.7.1. 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.

4.7.2. Teasing

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

4.7.3. Recognition of Common Fractures in Plastics

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 procedures 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". Rib marks are similar to the familiar fatigue clamshell or beach marks 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 other flow, the hackle marks will very often spread out in ray-like fashion from the flow.

4.7.4. 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, highly 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 discernible.

4.7.5. 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.

4.7.6. 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 effects 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 beach marks found in metal fatigue failures and are generally referred to in plastic fractures as "rib" markings.

4.8. Analysis — General Approach

The task confronting an investigator when he first arrives at the scene of an accident often seems bewildering. Yet 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 shortcut 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 cause is uncovered.

Every investigation requires the development of all related facts, and it is the investigator's task to decide which facts are pertinent and to develop these to the fullest extent possible. Thoroughness is a must in this type of work, and all evidence must be carefully sifted and each clue must be fully traced in order that all relevant points are brought to light. Throughout the investigation, the investigator should bear in mind that his report will be later evaluated by those responsible for initiating corrective action, and that toward this end his report should contain sufficient information to permit a complete evaluation to be made. The prevention of similar accidents depends for the most part on the investigator's success in determining the cause of the accident and this constructive goal should be a constant inspiration to the investigator in his work.

Only those procedures and techniques that are particularly useful in structural failure accidents will be presented in these sections. The standard procedures used in other types of accidents will be covered by other lectures.

4.8.1. Elimination Technique

When an aircraft crashes, there may be any one of a thousand and one reasons why it did so. The overall task, then 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, say, those possibilities associated with takeoff configurations or circumstances can obviously be eliminated. Or 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 and others are eliminated soon after it is established just how the aircraft contacted the ground. The general approach, then, 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.

4.8.2. Types of Structural Failure

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

4.8.2.1. Major Component Failure

As the title indicates, this category is associated with inflight 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 either (1) inadequate design strength, or (2) excessive loads imposed upon the components, or (3) deterioration of static strength through fatigue.

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 design 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 fairly closely supervised by FAA manufacturing inspectors, major manufacturing "botches" 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 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 an overcast in a typical weather accident. More often, however, 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.

Fatigue failures continue to be one of the major causes of structural failures of aircraft parts and components. This basic cause should always be strongly 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 either (1) inadequate design, (2) poor maintenance, (3) defective manufacturing, or (4) alternating loadings not anticipated by the designer. The majority of fatigue failures result from imperfect detail design and from improper installation or handling of the part. 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.

In addition to the three basic causes for inflight structural failure cited above, there is a special type of failure called flutter. This is an instability type of phenomenon involving a self-excited oscillatory system and its occurrence is dependent upon the interrelationship of the aerodynamics forces, inertia forces and elastic forces of the system. When 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 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 FAA have been demonstrated to be free from flutter when operated within their design limitations. However, flutter can occur in service, if the original configuration 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. In general, though, flutter is a remote possibility.

In the preceding paragraphs of this section, the basic causes and contributing factors which are associated with inflight structural failures of major components have been briefly noted. These points should be of assistance to the accident investigator in his 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 approximately in the 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, and some investigators have had considerable success in evaluating the significance of wreckage trails in accidents of this type.

In most accidents of this general category, more than one component fails and separates, and the investigator is faced with the difficult task of deciding which component failed first. The study of the wreckage distribution as described in the previous paragraph is a useful tool for this purpose. Other techniques are available for determinations of this type and these will be covered in subsequent sections.

4.8.2.2. Partial Failure or Malfunctioning

Accidents in this general category are by far the more difficult to investigate, since no obvious evidence, such as a wing being found two miles from the main wreckage scene, is usually available on which to make a rapid determination. 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 not rigged properly, 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 landing approach, and the aircraft dove into the ground in a near vertical attitude.

The general procedures used for accidents in this category are to follow routine investigatory practices, systematically checking out 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. Then, by careful study of the physical evidence, the true cause can be found. The reconstruction technique, explained in a later section, is most helpful at this stage of the investigation.

4.9. 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 detail wreckage examination, the investigator should have completed a general survey of the surrounding area, noting particularly 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 around the wreckage site. The general techniques used during this phase of the investigation are explained in other portions of this manual. In the following discussion, it is assumed that this phase has been fully explored.

4.9.1. General Examination

When the investigator arrives at the scene of a suspected structural-failure accident, one of his first moves 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 observing for 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.

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 making up the wreckage distribution chart. When this work has been completed, the detail examination of individual pieces of wreckage can be started. At this stage, particular attention should be directed toward significant smears, scores, indentations, toward the extent and type of damage, and toward the surrounding ground and the position of the piece relative to the ground. If, during this preliminary examination, the investigator observes any unusual smears, he should consider their significance and whether or not a laboratory examination is indicated. 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 since ash is easily disturbed. 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 as outlined above and additional examination of the wreckage may not be necessary. In some investigations, however, 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. Some of these techniques and procedures are explained in the following sections.

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, however, the suggested plan is a practical one and its use should insure the development of all significant facts within a reasonable length of time. Experience indicates that apparent shortcuts often lead to additional work. In this connection, it has been observed that investigators, who immediately after arriving on the accident scene start 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. Deviations from orderly procedures should only be tolerated as a last resort when unusual circumstances dictate such a course.

4.9.2. Examination for Smears, Scoring, etc.

In his preliminary examination at the accident scene, the investigator's immediate concern is to determine if a structural failure had occurred before impact. Toward this end, his chief interest initially is in separating ground impact damage from inflight 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. In the following paragraphs of this section some of the points that can be learned from a study of smears and scoring marks are presented in some detail.

A smear can be defined as a deposit of paint, primer, 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 inflight 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 pileup of paint will be found on the side of the protuberance away from the direction of the 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. If the investigator cannot make a preliminary determination and if he believes that the smears may contain valuable clues, he can resort to laboratory examinations. This type of examination 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, and this finding had an important bearing on the final flight path determination.

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. This type of determination is often helpful in many investigations. Score marks can often be used to establish that the damage occurred prior to impact and not afterwards. 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 inflight breakup.

Many other distinctive markings are often 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, the length and width of the cut, all provide indications of the propeller torque 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. In this case, 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. Further, it is sometimes possible to be misled by cutting marks produced by an axe or hacksaw used in the salvage operation, and the investigator should learn to be familiar with this type of marking and to distinguish this type from the others described.

4.9.3. 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. Thus, when a wing panel fails and detaches itself from the aircraft, very often the severed panel will strike and detach portions of the fuselage or tail section. The separation of the wing panel failure is generally referred to as the *initial* failure, whereas the fuselage or tail failures are referred to as *subsequent* failures. In addition, when the aircraft or its separated components strike the ground, substantial impact damage usually results. The investigator's task, then, is first to separate the inflight damage from the ground impact damage. Next, he must search out among the inflight failures the initial failure. Finally, he must isolate the exact cause for this initial failure.

In the preceding sections, background material has been presented for the investigator's guidance in developing all pertinent facts relating to structural failure accidents. As the various points are developed, the investigator should constantly integrate the new evidence. If the investigation has been proceeding systematically and if the detail 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.

4.9.3.1. 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. Thus, 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 differ 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. In general, primary-type failures are usually associated with the initial and subsequent inflight failures, while the secondary-type failures are more frequently associated with ground impact failures or damage.

4.9.3.2. Relating to Aircraft Attitude Just Before Accident

In the previous information on failures some of the more important procedures and methods for isolating the various types of failures have been presented. Thus, if the investigator had followed the procedures outlined he would be able, for example, to determine that the left wing panel had failed in flight. However, it still remains for him to determine 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. If the accident has been observed by ground or air eye witnesses, no great amount of work may be necessary to reconcile the structural damage to the flight attitude. When witnesses are not available, 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.

4.10. Laboratory Examination of Failed Parts

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 through the National Transportation Safety Board. At the present time, three government agencies are available to perform test work on failed aircraft parts for the Board. Wood parts are tested by the Forest Products Laboratory. Metallic parts are tested by the National Bureau of Standards. Most of the chemical analyses are performed by the Federal Bureau of Investigation Laboratory, although the N.B.S. also does some of this type of testing. On certain occasions, tests are conducted at the manufacturer's plant under NTSB 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 by the NTSB, and, 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 Safety Board's Technical Division staff in Washington coordinates all test work and evaluates the results as they are related to the particular accident. With regard to test work to be performed, it should be noted that only the Washington office of the Technical Division determines the agency to be utilized.

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: (1) tests on metallic parts for evidence of fatigue cracking, poor welding, substandard material properties, poor heat treatment, stress corrosion cracking, inadequate dimensional properties, etc.; (2) tests on wooden parts for evidence of inadequate glue bond, substandard material properties, moisture content, improper grain slope in splice connectors, etc.; (3) tests on smears, scores, cuts, etc., to determine the nature of the substance and direction of applied forces, etc.; and (4) tests on fuel and oil to detect presence of foreign substance or nonconformity with standard specifications.

4.10.1. 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, however, 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. In this regard, the investigator should include instructions relating to exactly what he suspects and what he hopes to establish 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 (1) when the part was installed in the aircraft; (2) total number of hours on the part; (3) time since overhaul or inspection; (4) whether any previous difficulty had been reported; and (5) other pertinent data which might throw a light on *how* the part failed and *why* the part failed. This type of information is extremely important to the technician who must arrange for and evaluate the results of the test program. 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 out the cause of a particular failure, it is almost 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 to 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. Wreckage Distribution

The wreckage distribution chart is one of the most useful tools the investigator can use, and often takes the place of or supplements a detailed writeup 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 placed at the accident scene. The significance of later findings often depends upon reference to the original wreckage distribution chart; and if one had not been 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, however, all major components, parts, and accessories should be listed, and suitable identifiable symbols or titles for each noted. The initial ground contact markings and other ground markings (made by propellers, fuselage, nose, wingtips, 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, too, should be noted on the distribution chart. Pertinent dimensions, descriptive notes, and locations from which the photographs were taken are additional items which add 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, control surface or system failures are some of the details which often can 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 closeup 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.

6. Wreckage Mock-Ups

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 (as it will be if the contact speed was high), the reconstruction job may be extremely difficult. The most positive means of identification is through part numbers which are stamped on most aircraft parts. Parts 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 normally flat pieces are often found curved, and normally curved pieces 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.

As indicated previously, 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 inflight 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. Inflight 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 closeups of significant details, should be made.

6.1. 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 is begun, a specific procedure should be followed; i.e., overall photographs made, wreckage distribution chart completed, a walk-around inspection made, and adequate notes made on the manner in which the various pieces were first found. Parts from the suspected area are collected, identified, and arranged on the ground in their relative positions. Major components such as the wing, tail, and fuselage are generally laid out separate from one another for ease of later examation. If the suspected area is at the junction of the major components, these areas are sometimes reconstructed separately. Individual cable runs with their associated bell cranks, idlers, and quadrants are usually 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.

6.2. 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 or the investigator's task may be considerably magnified. As in other stages of the investigation, shortcuts should be suspected until they are established as really being shortcuts.

When the reconstruction is done 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 mock-ups 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 mock-ups, 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.

7. Inflight Breakup

The aircraft accident investigator will become involved many times during his career in the study of inflight structural separations. These separations are normally the result of metal fatigue, improper design, or aerodynamic loading.

Metal fatigue in a separated part is quite easy to recognize at the accident scene, however, the investigator must ask himself the question, "Was it primary or secondary?" Generally speaking, an inflight breakup can be classified as to a primary break, and all other breaks then become secondary breaks or separations.

In determining the sequence of events of the breakup, the breaks or separations must be specified as to which is primary, secondary, tertiary, etc. For example, consider a single rotor helicopter in which a pitch link broke on one blade — the primary separation. The errant rotor blade then cut the tail boom off — a secondary separation. The tail rotor then fell to the ground, and a blade broke in an area where a fatigued surface was present — the tertiary break in the sequence. Although the cause of the fatigue zone must be thoroughly investigated, it must be placed in its proper perspective relative to the entire breakup pattern.

In the area of improper design, it has been found that the amateur-built aircraft, to the light general aviation aircraft, through the air carrier transport, redesign and modification have often been necessary as the result of findings in aircraft accident investigation. The investigator must not hesitate to question design deficiency when the facts lead to this area.

Although an aircraft certificated under the standard airworthiness requirements undergoes rather stringent structural testing before certification, the true test of its structural integrity comes as a result of use (and often misuse) during the routine day-to-day operation of its life.

In the area of aerodynamic loading, an aircraft will be subjected to two basic types of loads, those resulting from gusts, and those which are imposed by maneuvering. It is obvious that these two types of loading can occur at the same time, that is, high pilot or autopilot input loading at a time when the aircraft is being subjected to severe gust loading.

Structural separation or damage can occur from a flutter condition. This is an aeroelastic situation where the elastic properties of the structure react to aerodynamic loading so that a vibration takes place. Such items as repainted balanced controls installed without rebalancing; water trapped in the area of the trailing edge; airfoil change due to oil-canning of the metal; and fabric and tape separations, are examples of a C.G. change or aerodynamic change, respectively, which could cause control surface flutter.

It is not the problem of the investigator to determine the specific cause of the flutter, but to recognize the evidence of flutter and those things which could have caused it. Most aircraft manufacturers have a flutter specialist on their engineering staff, and if a flutter problem exists, experts in this field should be utilized.

An aircraft inflight breakup is an eventual incompatability between the applied aerodynamic load and the load sustaining capabilities of the structure. To analyze properly inflight structural separations, it is apparent that a working knowledge of aerodynamics and structural engineering is a must for the aircraft accident investigator.

In the subject of aerodynamics there should be familiarity with the gust and maneuvering load equations, stability and control, airfoil pitching moments, and center of pressure versus angle of attack. There must be a knowledge of flight control design, autopilot inputs, control surface balance, boundary fences, vortex generators, leading and trailing edge slots and flaps, and servo and antiservo tabs.

In the subject of structures there should be an intimate familiarity with construction techniques as well as an understanding from a practical aspect the meaning of moment of inertia, section modulus, and elastic axis. There should be a familiarity with the equations of stress dealing with tension, compression, bending, shear, and torsion.

To be able to analyze an inflight breakup, it first must be clearly understood how an aircraft is aerodynamically loaded in its normal configuration. Reference is made to **Fig. C III-29.** It must be remembered that an aircraft rotates around the center of gravity, and the center of gravity is located in close proximity to the quarter chord point of the mean aerodynamic chord. The center of gravity in this illustration may be assumed to be in the center of the aircraft and in the area of the spar outline. The arrow under the engine represents the weight of all items ahead of the C.G., and the arrow under the aft fuselage represents the weight of all items aft of the C.G. The sum of the moments of these two forces around the C.G., in addition to the wing pitching moment, if it is a cambered airfoil, will result in a nosedown pitch. This resultant nosedown pitch is prevented by a download on the tail. This download then brings the pitching moment to zero, and may be considered as that balancing force which places the aircraft in a state of equilibrium as far as pitching moments are concerned. The summation of vertical forces, as well as horizontal forces, must equal zero if a state of equilibrium is to exist. In considering the vertical loads, the two large arrows under the aircraft represent the total weight of the aircraft multiplied by the existing load factor. These downward forces, plus the download on the horizontal tail, represent the total *down* force, and this in turn must be brought to zero by an equal *up* force, which is the lift of the wing.

It must be remembered at this time that if the total weight exceeds the lift, or the total lift exceeds the weight, there will be an acceleration in the direction of the greater force. This is an important factor when considering the summation of horizontal forces, particularly thrust and drag, in this illustration. These would be the fore and aft forces. The lateral or left and right forces can be of major importance in inflight breakup if a yaw condition exists. Severe yaw is present after the loss of a wing, due to the differential in wing drag.

Assuming that there is no yaw in this case, the vertical forces, lift and weight, and the horizontal forces, thrust and drag, must be analyzed.

If a pilot loses control of an aircraft in weather, and the aircraft enters a spiral, the thrust will exceed the drag. This will be true whether the powerplant is at full throttle or at idle, for the weight of the aircraft, depending upon the nosedown attitude, can far outweigh the thrust of the powerplant. This will be true until the aircraft attains a velocity such that the drag equals the thrust. This will be terminal velocity for that particular nosedown pitch attitude. The specific term, *terminal velocity of an aircraft,* means the maximum velocity attained in a nosedown attitude such that the flight path is toward the center of the earth.

The modern civil aircraft is so clean aerodynamically that to test it for a terminal velocity would result in destructive airspeeds. This velocity has no practical value and is not a design requirement. Unfortunately, however, the air safety investigator will become involved in aircraft accidents where the aircraft has been subjected to airspeeds far in excess of those for which it was designed. This high speed flight regime is encountered normally in the loss of control under instrument flight conditions.

In studying Fig. C III-29, under a high velocity flight regime, if back elevator pressure is applied, the "g" forces increase as well as the horizontal tail download.

If the aircraft has a cambered airfoil (not symmetrical), the nosedown pitching moment increases as the square of the velocity. This fact also requires a larger download on the tail. This download capability of the tail is easily acquired, since it also reacts to the square of the velocity. It is designed this way to cause the nose to start to pitch up for stability reason. If the aircraft is in a spiral, this pitchup ability of the horizontal tail then merely tends to tighten the spiral. This maneuver, called the graveyard spiral is common in general aviation weather accidents.

The stresses must now be studied. The top skin of the horizontal tail (stabilizer or stabilator) is under a tensile stress with the corresponding compressive stress on the bottom. The wing in supporting the high "g" loads as well as the large horizontal tail download is deflected upward, and there is tensile stress on the bottom skin surfaces and compressive stresses on the top skin surfaces.

Since the fuselage is supported in the area of the C.G., there is a maximum bending moment at the C.G. This places the top of the fuselage (above the neutral axis) in a tensile stress, and therefore the bottom of the fuselage is under a compressive stress.

The question now is, "What can break?" The fuselage could break anywhere along its length, for it must be visualized as a bending beam supported only at the wing. Fuselage breaking is rather uncommon in the medium-to-small size general aviation aircraft. However, it has occurred, and can again occur, in large transport and bomber type aircraft.

There are two other areas where the structure will normally separate or break. These are the wing and the horizontal tail.

The first consideration is that of the wing separating first, as shown in Fig. C III-30. As the wing breaks, the aircraft is instantaneously changed from state of equilibrium to one of inequilibrium, particularly as far as moments of roll are concerned. The right wing with its high lift values will violently roll the aircraft to the left, while the left wing, which also possessed this large positive lift, will roll to the right over the fuselage.

In studying Fig. C III-31, it is quite common to find at the accident site that the separated wing has damaged or completely severed the tail of the aircraft.

A study of the wing primary structural breaks, and an elementary mockup at the site, will confirm this type of breakup. Bear in mind that the remainder of the aircraft may be demolished or burned beyond usefulness for investigative purposes.

There are two important facts that the investigator must realize for this type of breakup. First, the horizontal tail must have been on the aircraft in order to generate the high "g" loads necessary to cause a positive wing separation, and second, high speed flight was involved. (Some airspeed well over V_a, or maneuvering speed)

In the next common type of inflight separation, reference is made to Fig. C III-29 and 32. The aircraft is loaded normally as in Fig. C III-29, and either the wing will break up or the tail will break, normally down. The manner in which the horizontal tail separates is a matter of pilot technique or control input. There have been positive stabilizer separations, but these are rather rare. These are of the stabilizer-elevator combination horizontal tail in which the pilot applies such a violent download to the elevator hinge line (excessive control back pressure) that it in turn twists the leading edge of the stabilizer up around the elastic axis of the stabilizer. The torsional deflection is maximum at the stabilizer tips, and zero where the stabilizer attaches to the fuselage.

Regardless of how the horizontal tail is separated, the aircraft will respond as shown in Fig. C III-32. It will pitch violently nose down as shown by the curved arrows, the wing will encounter a high negative angle of attack as shown by the loading arrows under the wing, and the inertia of the aircraft will be in the direction as shown by the largest arrow.

The resultant separation is shown in Fig. C III-33, in which the wing incurs a negative separation. The investigator need not be surprised to find permanent set in the wings from positive loading, or contrary to the direction in which the wing separated. It must be remembered that before the horizontal tail separated it was subjecting the wing to high positive "g's" in an effort to break the wing upward. When the tail separated, the wing was instantly loaded in the reverse direction, and this loading does not remove all of the positive loading permanent set.

The investigator must visualize the wing, tail and fuselage combination as a wound clock spring. The faster the aircraft is flown, the tighter the spring is wound, and when some structural member of this combination breaks it is equivalent to suddenly releasing the spring. It is also analogous to a bomb exploding, for the structure possesses a high degree of potential energy, and when it separates there is a great deal of noise generated as the result of energy release. This is the reason a witness to an inflight separation will report that the aircraft exploded. He hears the noise as an explosion, and his imagination will then add fire and smoke, although the wreckage is free from soot and burning evidence at the accident site.

If there is evidence of inflight fire at the crash site, as well as of an inflight breakup, the scatter pattern must be carefully documented.

It must be borne in mind that an inflight separation of the primary structure is always a short-time-interval event, a matter of seconds, or fractions of seconds.

There must be no guessing or assuming in the study of an inflight breakup. Keep in mind that the aircraft obeys the laws of physics just as precisely during breakup as it does during normal flight. There should be no mystery about it, yet on the other hand, determination of the breakup sequence is not accomplished in two minutes at the accident site. It may require hours and hours of cogitation. Certain general aviation weather-type accidents are so common that a few questions by the investiga-

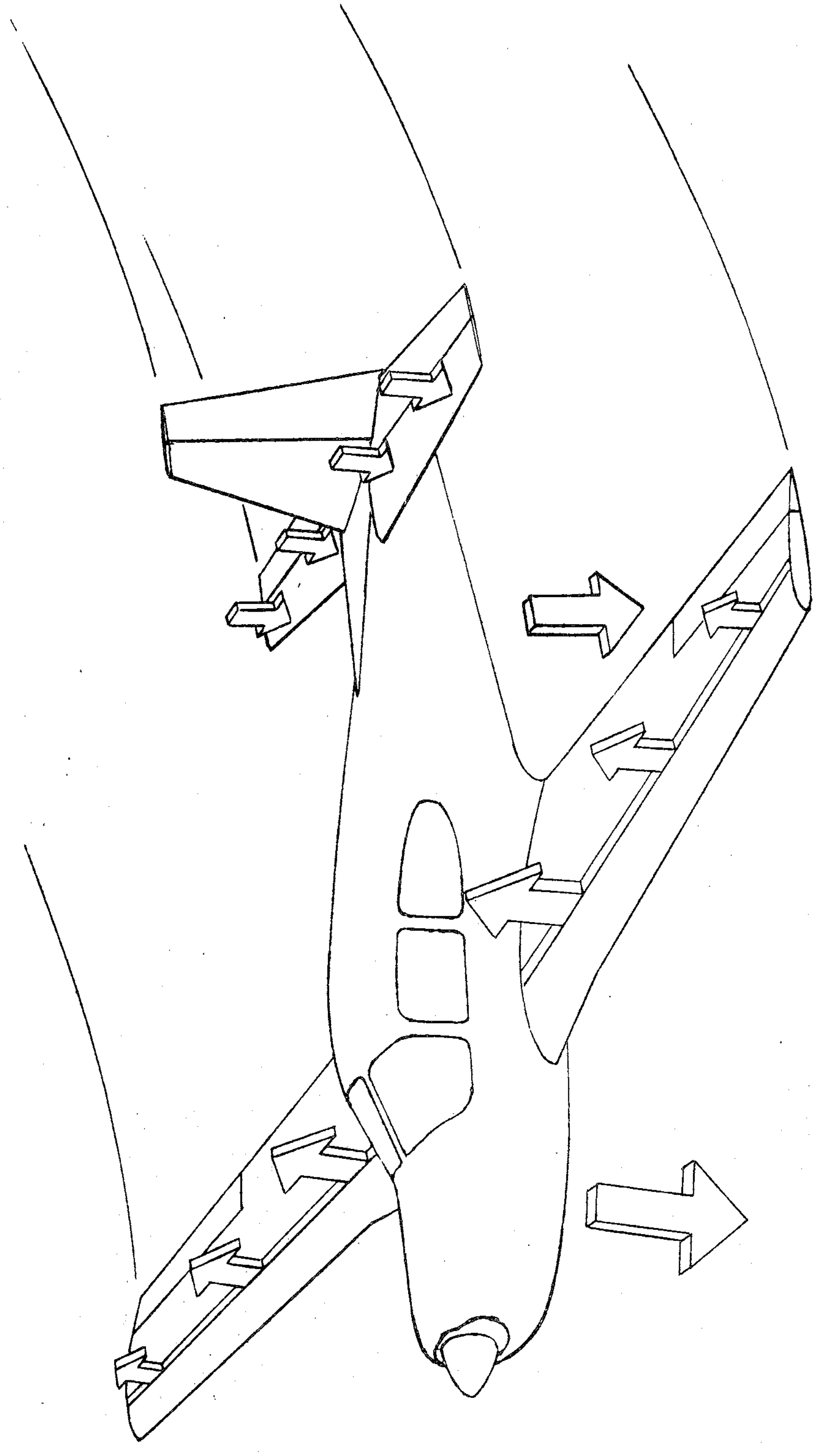


Illustration to show the loading applied to the wings, horizontal tail, and fuselage under a positive "g" flight regime, regardless of the aircraft attitude.

Figure C III-29

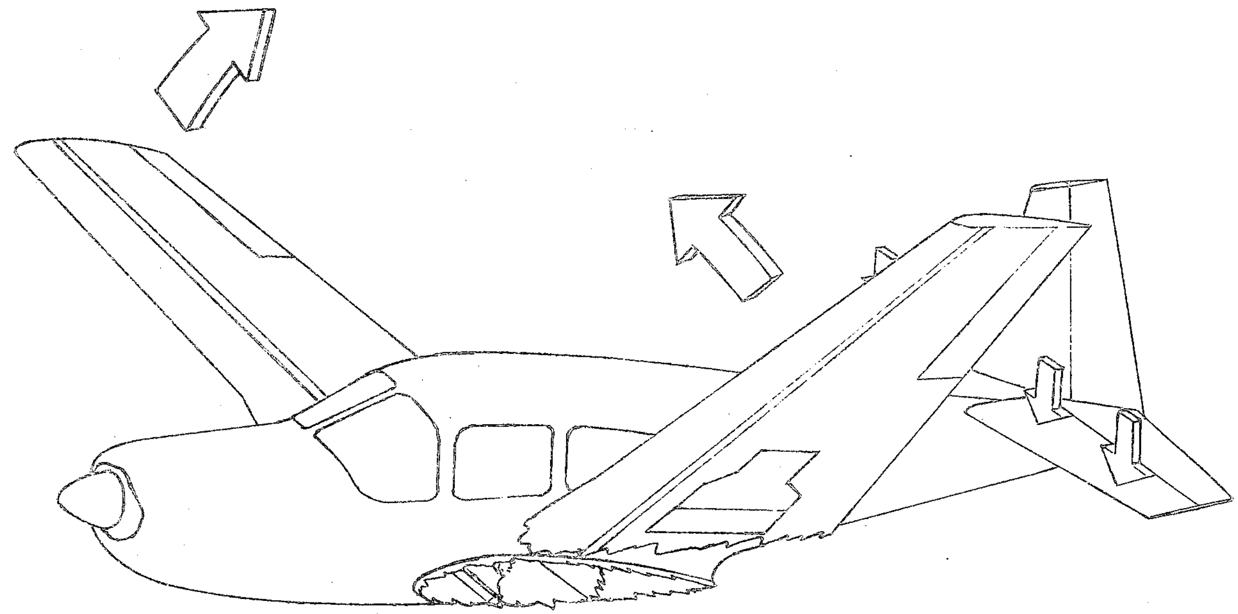


Illustration to show the relative aircraft motion subsequent to a wing separation.

Figure C III-30

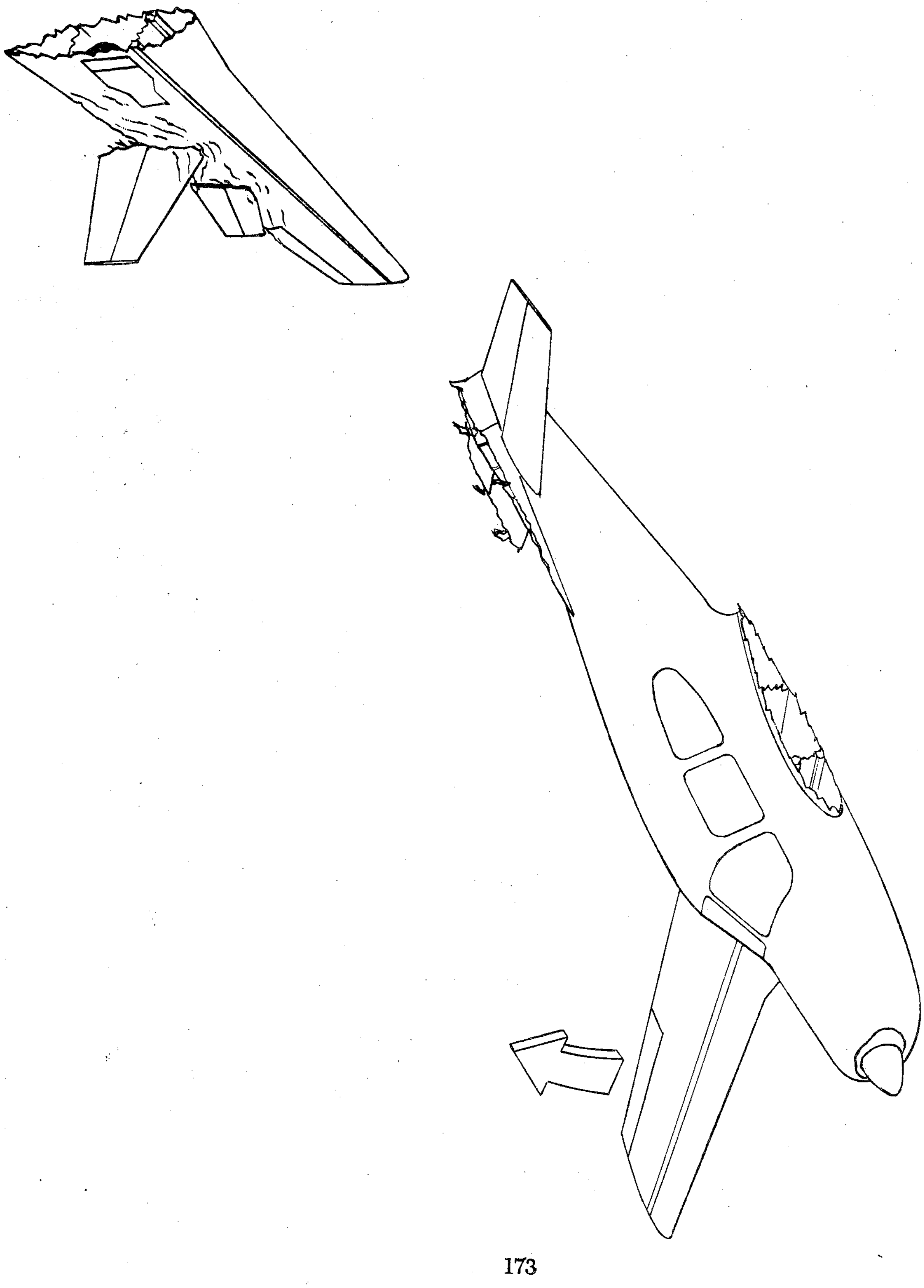


Illustration to show a possible tail damage as the result of a positive wing separation.
Figure C III-31

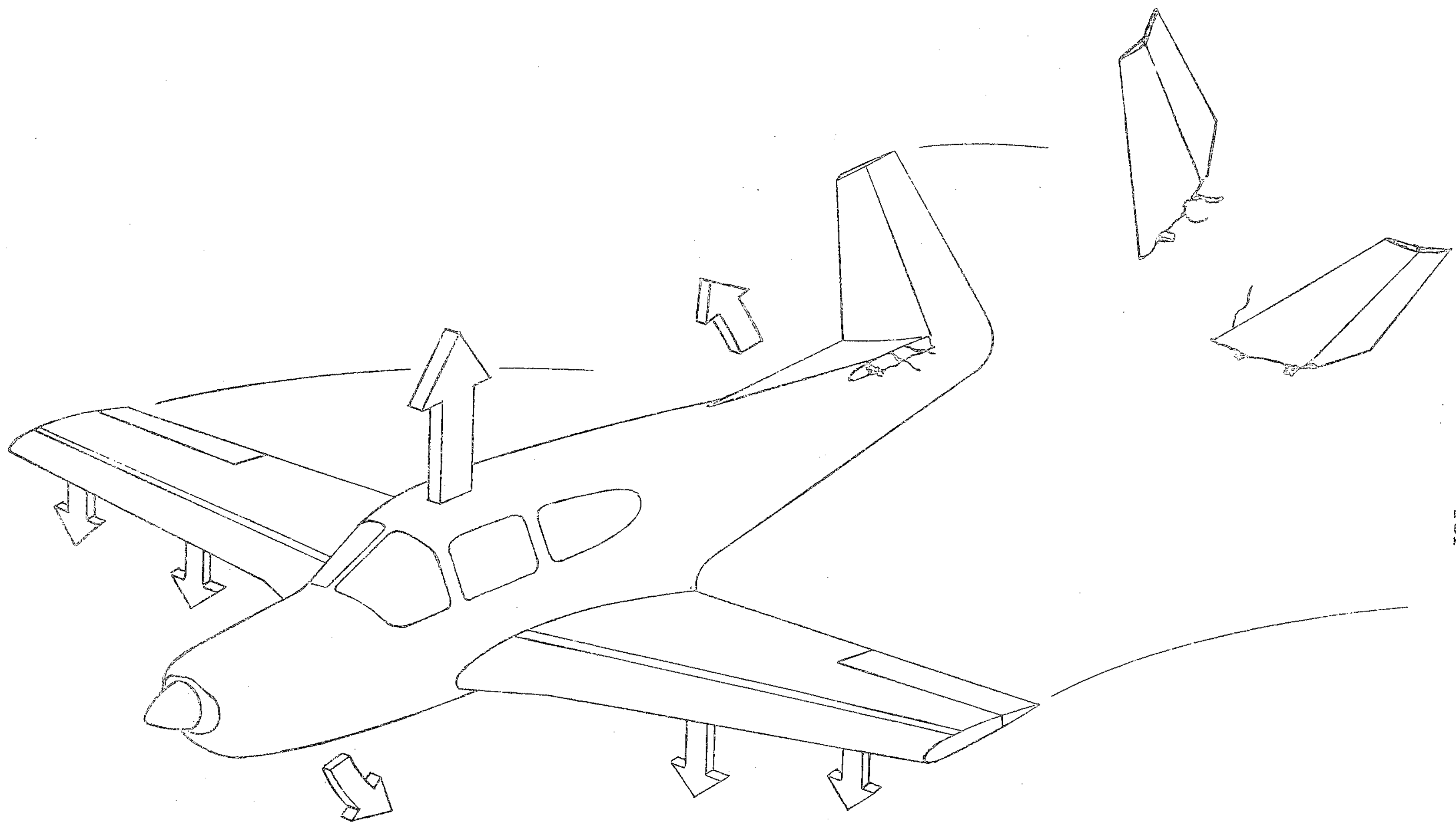
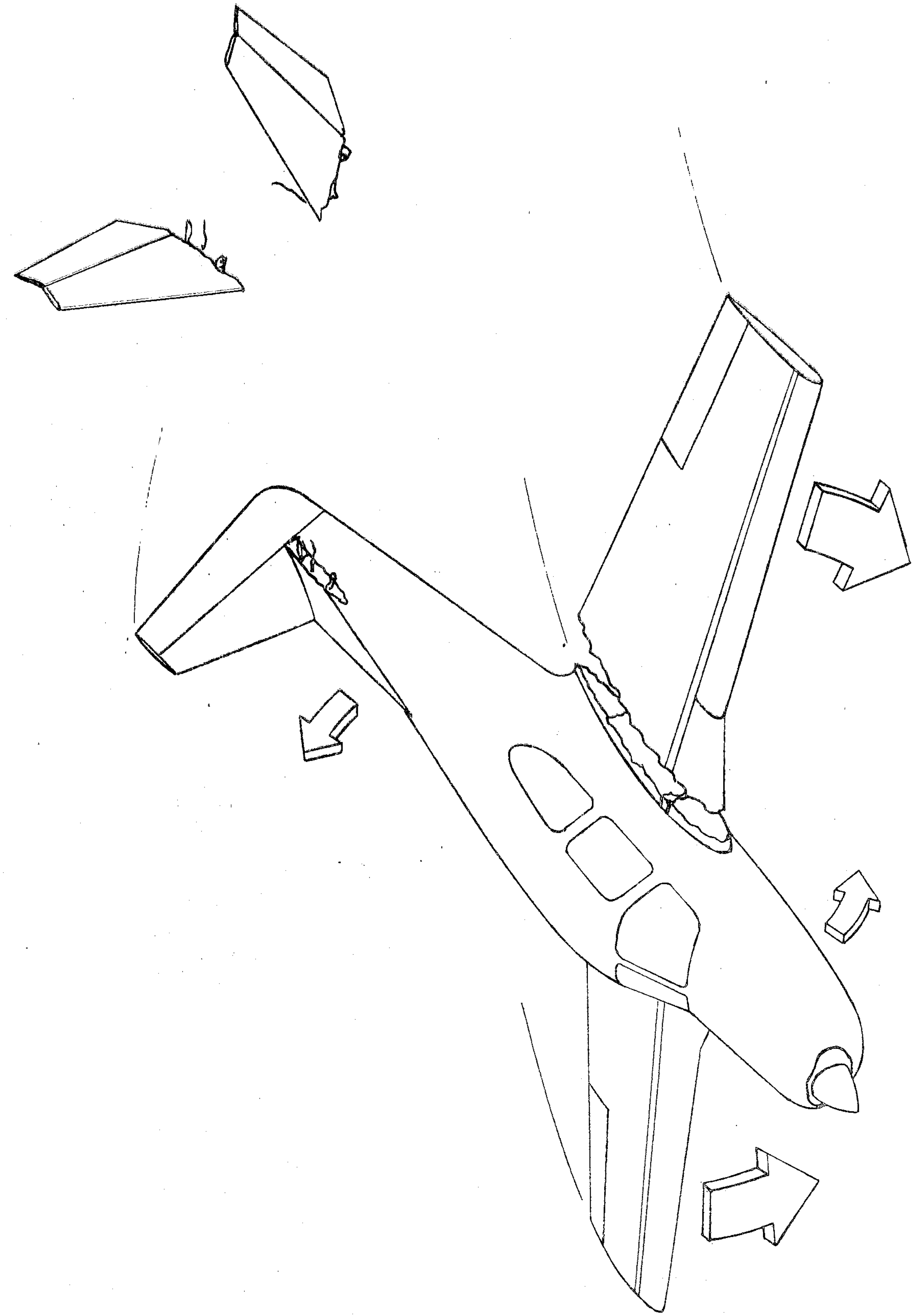


Illustration to show the immediate aircraft reaction as the result of the loss of the horizontal tail.

Figure C III-32



The loss of the horizontal tail results in a high negative wing loading as the result of a severe nosedown pitch.

Figure C III-33

tor by telephone can reveal how the aircraft broke up. This also makes for complacency, and complacency has no place in the investigator's line of skills. All of the evidence and factual information is at the crash site, therefore, premature disturbance or removal of the wreckage to another area can destroy valuable evidence.

8. Midair Collision Analysis

When two aircraft collide, significant marks are usually left on each aircraft. An examination of these marks will reveal many facts to the investigator, and an interpretation of these facts will enable a more comprehensive analysis.

Certain facts are normally learned by an investigator about the actions of at least one, and quite often both, aircraft prior to collision. Such information includes positions, times over known fixes, flight plan, true airspeeds, and ground speeds, witness observations, etc. It is important to obtain these facts since their use in evaluating the scratch marks assures more accuracy; accuracy in the evaluation is a function of the number and accuracy of the known facts.

In many collision accidents, certain facts may be available on one aircraft but not on the other. The evaluation of the scratch marks is the only way to determine these unknown facts. For example, if the speed of one aircraft is known, the speed of the other can be determined. Examination of the direction of the scratch marks aids in determining whether the aircraft were level, climbing, descending, head-on, etc.

The following pages treat one facet of the midair collision problem mathematically and graphically in order to provide the investigator with yet another tool in the never-ending search for the oftentimes elusive probable cause, thereby further promoting aviation safety through accident prevention.

8.1. Possible Paths of Colliding Aircraft

a. Both aircraft in level flight.

b. Both aircraft climbing.

c. Both aircraft letting down.

d. One aircraft level, one aircraft climbing.

e. One aircraft level, one aircraft letting down.

f. One aircraft climbing, one aircraft letting down.

g. In all of the foregoing cases, the aircraft may or may not be at the same speed.

h. In all of the above cases, the flight paths of the aircraft may be directly head-on, at an obtuse angle, a right angle, an acute angle, or directly overtaking. (Note: In a directly overtaking angle, obviously the aircraft could not be at the same speed if both are level, neither could a slower aircraft overtake a faster one.)

It is readily apparent that the possible paths of collision are limitless. In fact, in all probability, the aircraft will approach each other from different directions and different altitudes simultaneously. Significant scratch marks will in many cases be found on both the horizontal and vertical surfaces of each aircraft. Since direction and speed are involved, proper treatment of the scratch marks will enable the investigator to resolve his findings into force vectors and to solve the problems trigonometrically.

8.2. Rules of Thumb

a. Although the rules which apply to scratch marks are the same whether they are found on the horizontal or vertical surfaces, or a combination thereof, they have been placed in a one-plane surface for clarity.

Rule 1. There are only three possible planes in which two or more aircraft may operate. (1) The same plane as the horizontal surfaces wherein no relative horizontal movement takes place. (2) The same plane as the vertical surfaces wherein no relative horizontal movement takes place. (3) A third resultant plane wherein both relative horizontal and relative vertical movement take place simultaneously.

Rule 2. If the scratch marks on each aircraft slope in opposite directions with respect to their longitudinal axis, then the smaller angle between the longitudinal axis and the scratch mark is the one to be measured on each aircraft (Fig. C III-4).

b. Note that rules 2 through 4 are based on the fact that when two lines intersect, two angles are formed. In these bases, the two lines involved are the scratch mark and the longitudinal axis. Rules 2 through 4 will show which angles to measure and how to recognize various types of collisions.

Rule 3. If the scratch marks on each aircraft slope in opposite directions, as in Fig. C III-4, then each scratch mark was made in a direction proceeding from front to rear.

Rule 4. If the scratch marks are sloped in the same direction, then one aircraft overtook the other, and the larger angle between the longitudinal axis and the scratch mark is measured on the slower aircraft. The smaller angle is measured on the faster aircraft. (See Fig. C III-5.)

If the scratch marks are sloped in the same direction, one of the scratch marks had to be made in a direction proceeding from rear to front. The aircraft on which this mark appears is the slower aircraft. (See Fig. C III-5.)

Rule 5. If the sum of the scratch angles is less than 90°, the collision angle is obtuse (greater than 90°). (Fig. C III-6)

Rule 6. If the sum of the scratch angles is equal to 90°, then the collision angle is 90°. (Fig. C III-7)

Rule 7. If the sum of the scratch angles is greater than 90°, the collision angle is acute (less than 90°). (Fig. C III-8)

Rule 8. If the scratch angle on one aircraft is the same as the scratch angle on the other, then the speeds of the two aircraft are the same. (Fig. C III 9)

Rule 9. The larger scratch angle will always appear on the slower aircraft.

c. The following rules deal with scratch marks found on the vertical surfaces of each aircraft:

Rule 10. If the scratch marks on each aircraft slope in opposite directions with respect to their longitudinal axis, then the smaller angle between the longitudinal axis and the scratch marks is the one measured on each aircraft. (See Figs. C III-10 and 11.)

Rule 11. The scratch marks in rule 10 will always proceed in a front-to-rear direction on each aircraft.

Rule 12. If the scratch marks in rule 10 also proceed in a generally bottom-to-top direction, the aircraft collided in a relatively noseup attitude with respect to each other. (See Fig. C III-10). Conversely, if the scratch marks in rule 10 also proceed in a generally top-to-bottom direction, the aircraft collided in a relatively nosedown attitude with respect to each other. (See Fig. C III-11.)

Rule 13. If the scratch marks slope in the same direction, then one aircraft overtook the other, and the larger angle between the longitudinal axis and the scratch mark is measured on the slower aircraft. The smaller angle is measured on the faster aircraft. (See Figs. C III-12 and 13.)

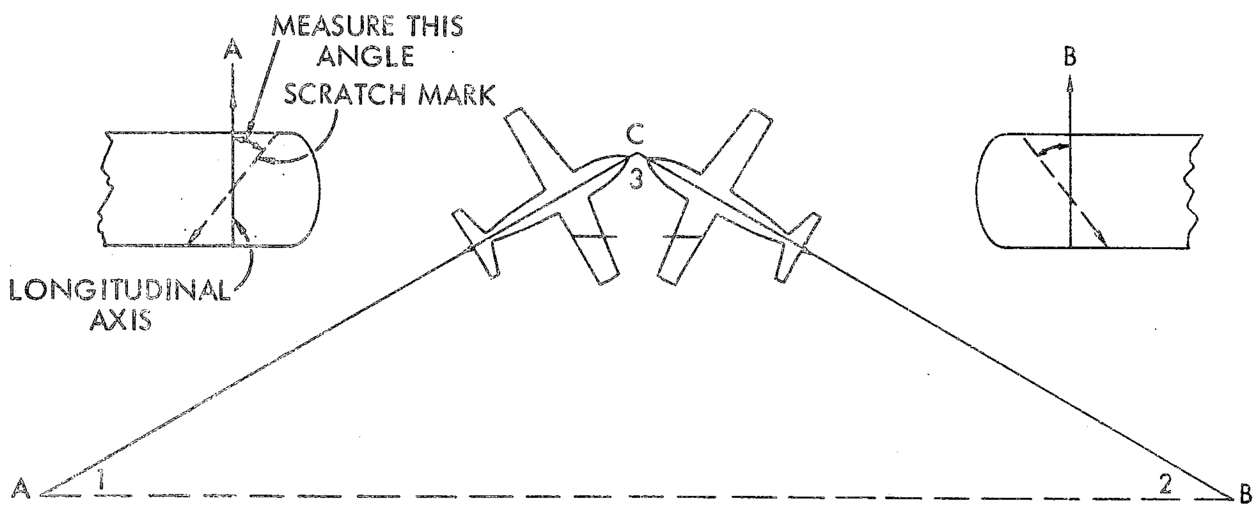
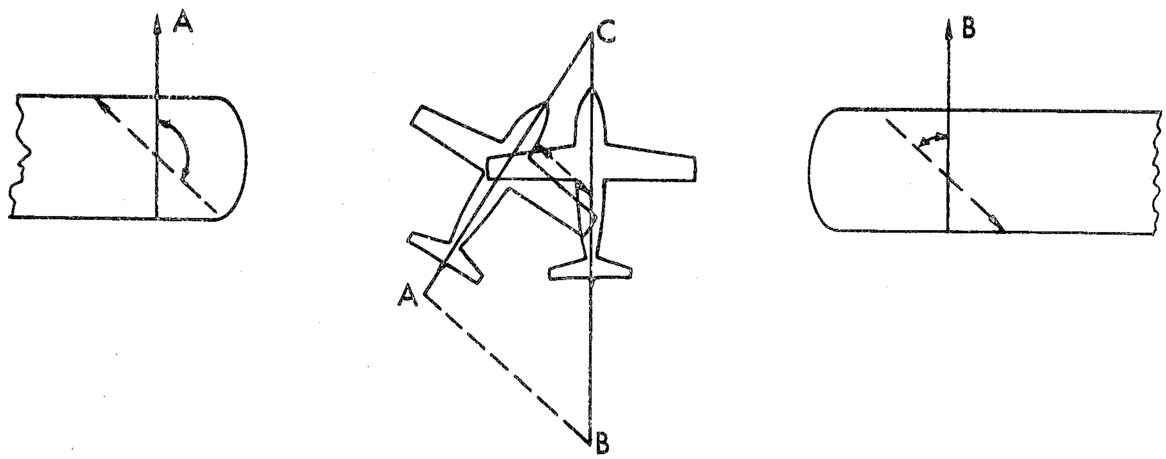
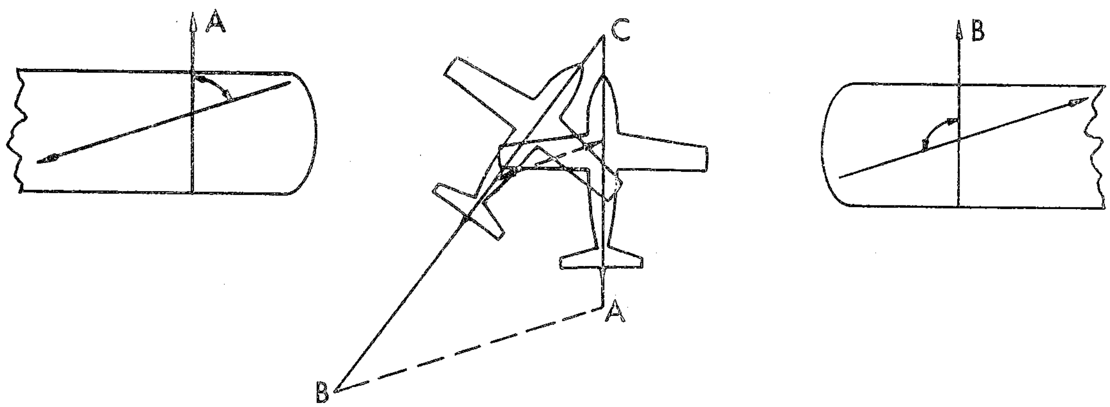


Figure C III-4.



FASTER AIRCRAFT APPROACHING FROM RIGHT



FASTER AIRCRAFT APPROACHING FROM LEFT

Figure C III-5.

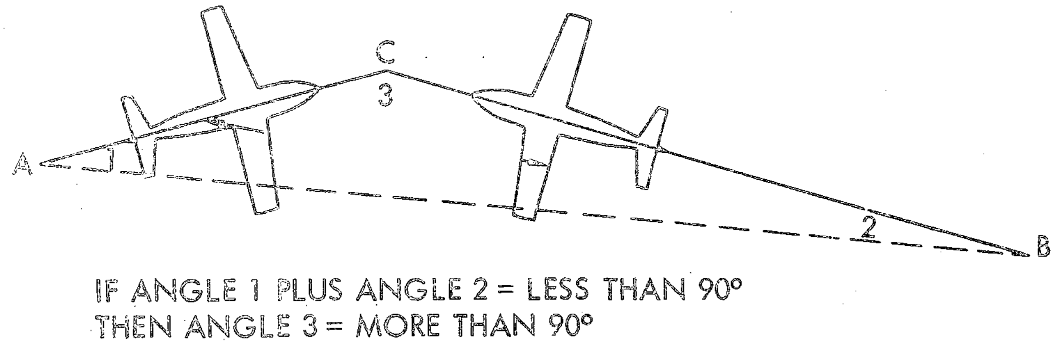


Figure C III-6.

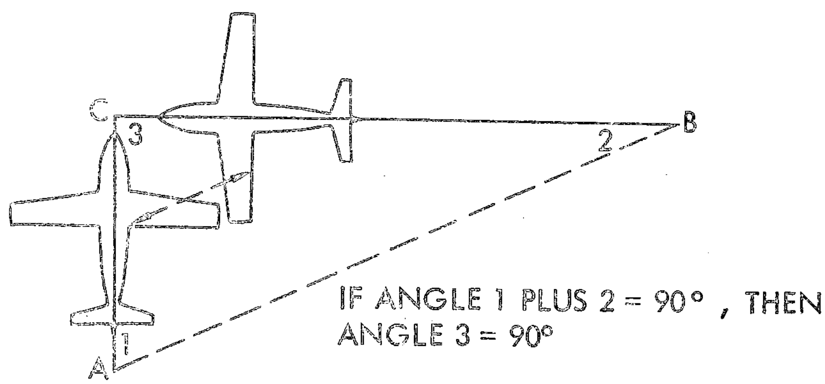


Figure C III-7.

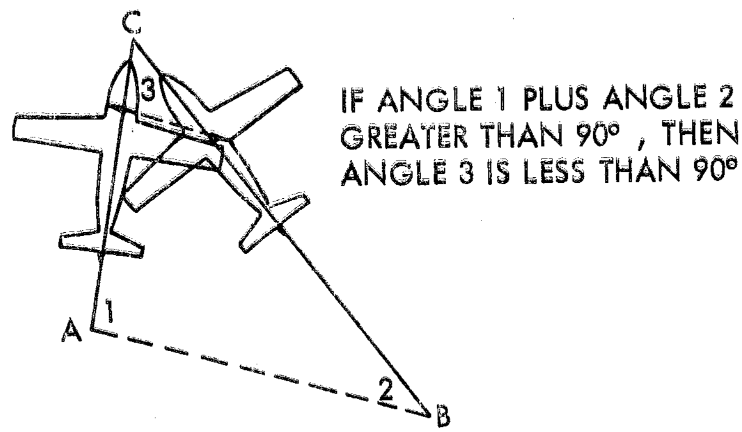


Figure C III-8.

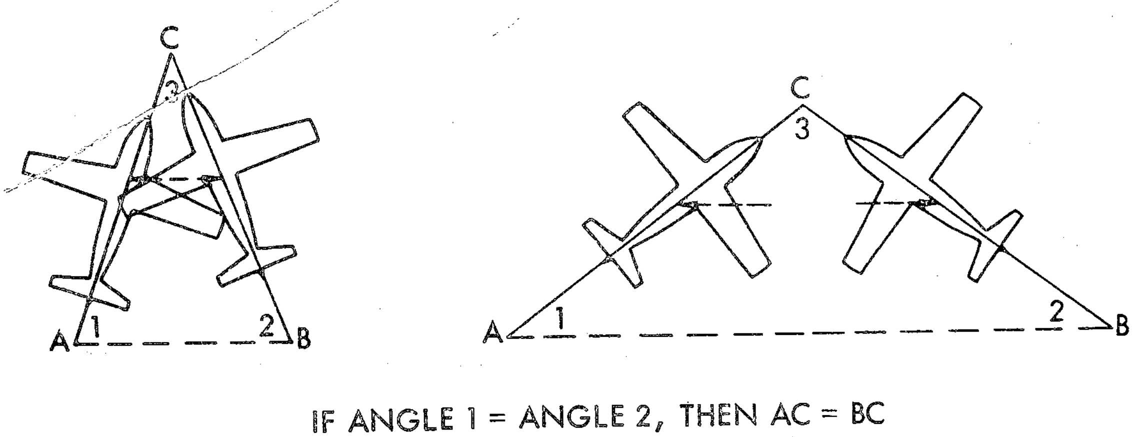


Figure C III-9.

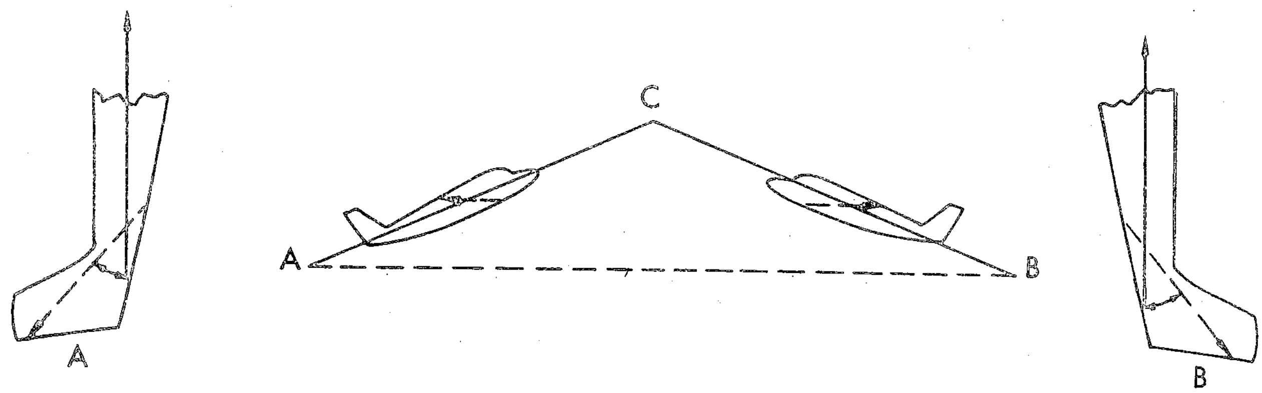


Figure C III-10.

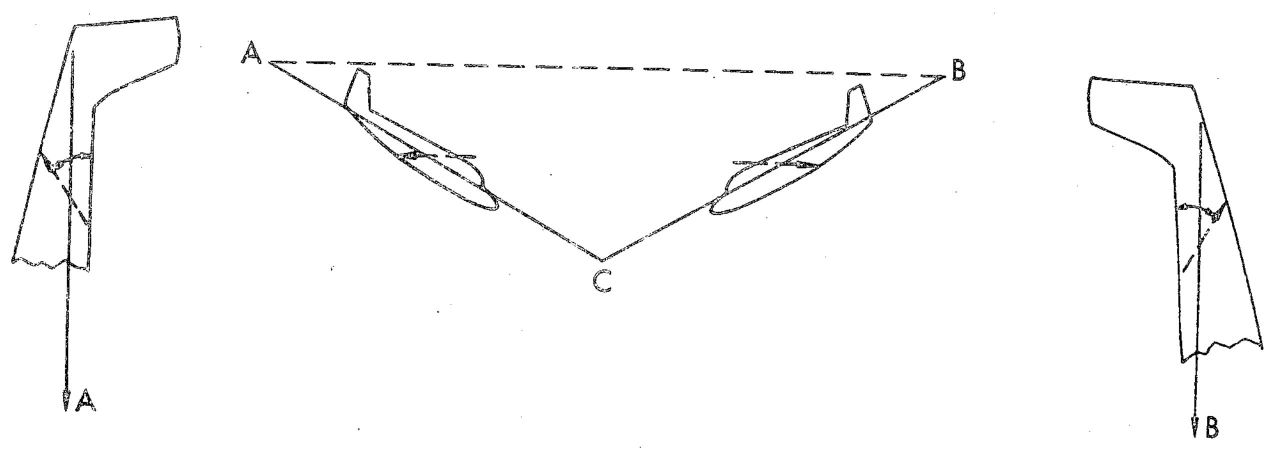


Figure C III-11.

Rule 14. In all cases under Rule 13, the **slower aircraft is the one having the scratch mark which was made in a rear-to-front direction. Conversely, the faster aircraft will have the scratch mark made in a front-to-rear direction.** (See **Figs. C III-12 and 13.**)

Rule 15. If the scratch marks on the slower aircraft also proceed from bottom to top, then that aircraft was above the other. Conversely, if the scratch marks on the slower aircraft proceed from top to bottom, then that aircraft was beneath the other. (See **Figs. C III-12 and 13.**)

Rule 16. If the sum of the scratch angles is less than 90°, the collision angle is obtuse. (See **Fig. C III-14.**)

Rule 17. If the sum of the scratch angles is equal to 90°, then the collision angle is 90°. (See **Fig. C III-15.**)

Rule 18. If the sum of the scratch angles is greater than 90°, the collision angle is acute. (See **Fig. C III-16.**)

Rule 19. If the scatch angle on one aircraft is the same as the scratch angle on the other, then the speeds of the two aircraft are the same. (See **Fig. C III-17.**)

8.3. Triangle Relationships

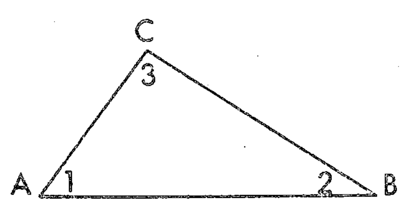


Figure C III-18.

(1) In any given triangle such as ABC above, certain formulas have been taken from trigonometry in order to solve the mathematical portions of scratch mark investigations. The first of these formulas is the law of sines:

$$\frac{AB}{\text{Sin of angle 3}} = \frac{AC}{\text{Sin of angle 2}} = \frac{BC}{\text{Sin of angle 1}}$$

(2) The second formula is the law of cosines:

(a)

$$(AB)^2 = (BC)^2 + (AC)^2 - 2(BC)(AC) \cos \text{angle } 3$$

(b)

$$(BC)^2 = (AB)^2 + (AC)^2 - (AB)(AC) \cos \text{angle } 1$$

(c)

$$(AC)^2 = (AB)^2 + (BC)^2 - 2(AB)(BC) \cos \text{angle } 2$$

(3) Another rule from trigonometry is used in the problems involving scratch marks in the situation where an overtaking aircraft approaches at an angle from the rear. This causes a scratch mark on the slower aircraft in such a direction that the obtuse angle must be measured. (See rule 4.) Since trigonometric tables do not show a sin for angle greater than 90°, a relationship must be shown in order to obtain the sin. The rule is as follows:

$$\text{Sin } X = \text{Sin}(180° - X)$$

In other words, the sin of 125° is the same as the sin of 55° (180−125).

8.4. Sample Problems

(1) Consider a situation wherein the scratch marks on each aircraft are parallel to their respective longitudinal axis. This is the simplest to compute. If nothing is known about the flight plan or witness statements as to whether the aircraft were approaching headon or overtaking, it is necessary to carefully examine the scratch marks to see if the direction in which the scratch marks were made can be determined from the wreckage. Quite often, pieces from one will be found in the other, which will tell the direction. Also, the direction in which the metal is torn is indicative. If the aircraft collided headon, the closure speed is simply the sum of their airspeeds. If one aircraft overtook the other, the closure speed is the difference between their airspeeds.

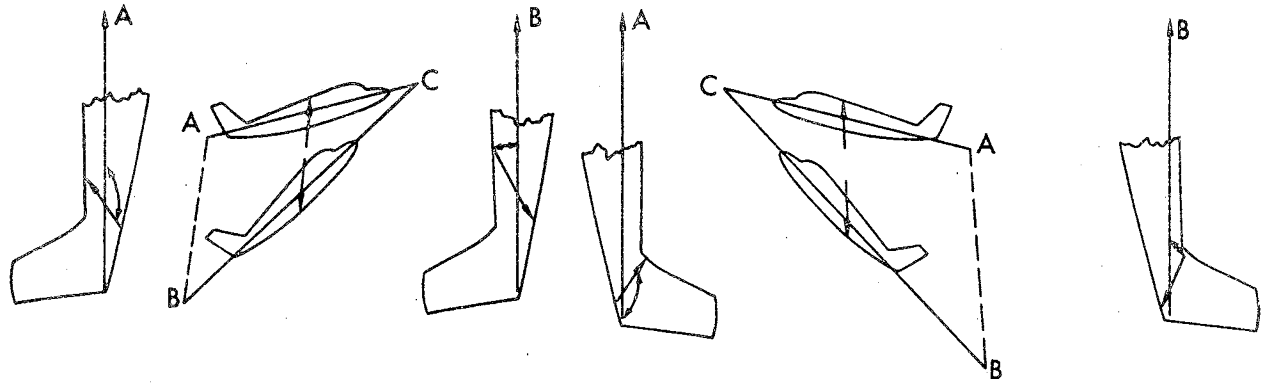


Figure C III-12.

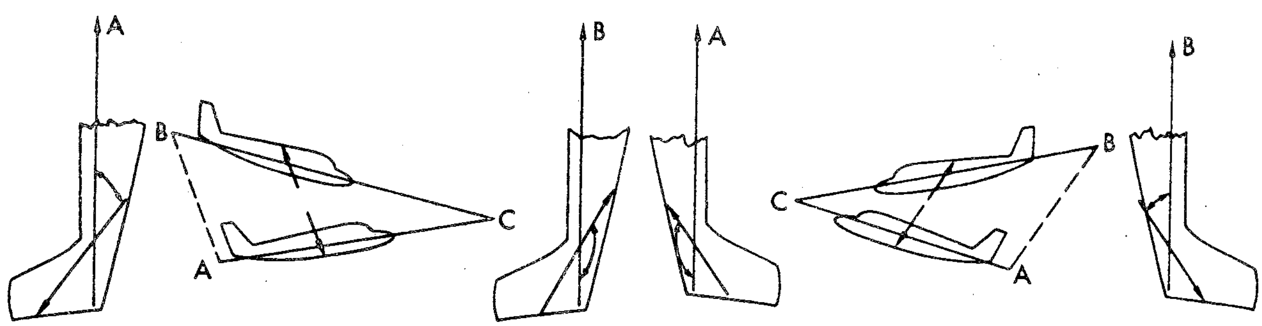
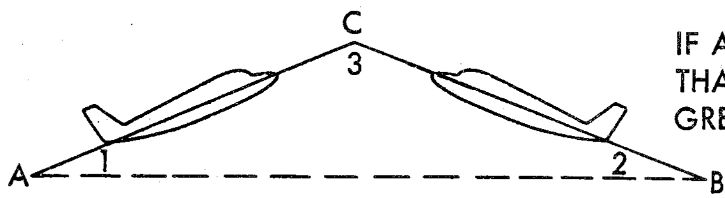
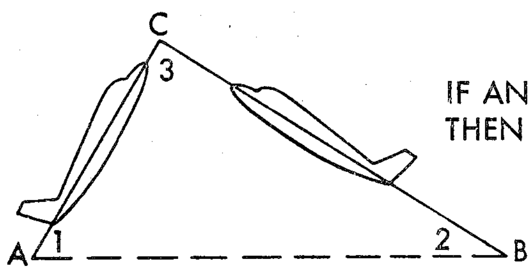


Figure C III-13.



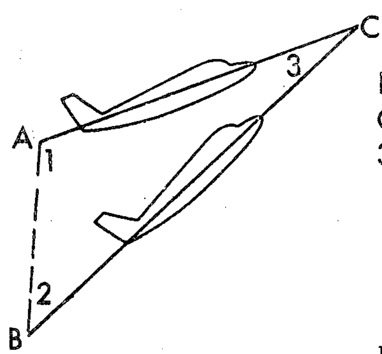
IF ANGLE 1 PLUS ANGLE 2 IS LESS THAN 90° , THEN ANGLE 3 IS GREATER THAN 90°

Figure C III-14.



IF ANGLE 1 PLUS ANGLE 2 = 90° , THEN ANGLE 3 = 90°

Figure C III-15.



IF ANGLE 1 PLUS ANGLE 2 is IS GREATER THAN 90° , THEN ANGLE 3 IS LESS THAN 90°

Figure C III-16.

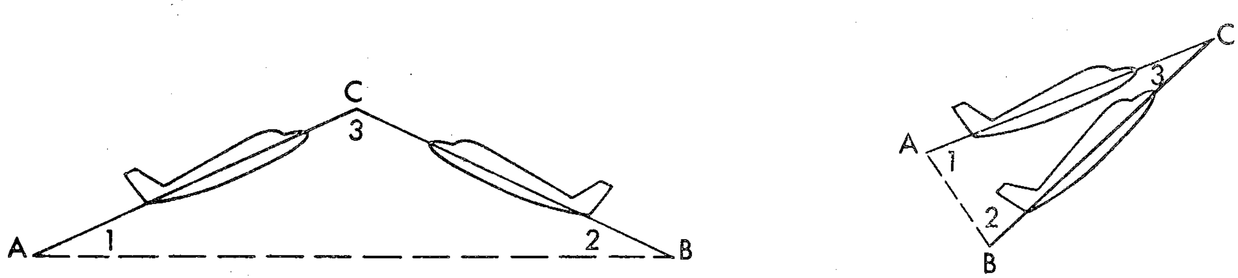


Figure C III-17.

(2) Consider the following situation:

(a) We know the following:

Aircraft A was traveling at 100 knots.

The scratch mark on A was in a front-to-rear direction and measured 45° from the longitudinal axis.

The scratch mark on aircraft B was in a front-to-rear direction and measured 32°.

The direction of the scratch marks tells us that the smaller angles are those measured (rule 2).

Both aircraft were at the same altitude.

(b) By examination we can deduce the following:

Since the sum of the scratch marks add up to less than 90°, the aircraft collided at an obtuse angle (rule 5).

Since aircraft A had the larger scratch angle of the two aircraft, it was slower (rule 9).

(c) Solution to find:

(1) The airspeed of aircraft B.

(2) The collision angle 3.

(3) The closure speed of the two aircraft.

Step 1. Using the law of sines, find the speed of aircraft B:

$$\frac{AC}{\text{Sin Angle } 2} = \frac{BC}{\text{Sin angle } 1}$$

$$\frac{AC}{\text{Sin } 32°} = \frac{BC}{\text{Sin } 45°}$$

Since the speed of aircraft A is 100 knots, then:

$$\frac{100}{.530} = \frac{BC}{.707}$$

$$BC = \frac{100\,(.707)}{.530}$$

BC − 133 knots (Airspeed of aircraft B)

Step 2. Angle 1 plus angle 2 plus angle 3 equals 180°.

Therefore:

Angle 3 = 180 − (angle 1 + angle 2)

Angle 3 = 180 − (45 + 32)

Angle 3 = 103° (Collision angle)

Step 3. Using the law of sines, find closure speed:

$$\frac{AC}{\text{Sin angle } 2} = \frac{AB}{\text{Sin angle } 3}$$

Since from step 2, angle 3 = 103° then:

$$\frac{AC}{\text{Sin } 32°} = \frac{AB}{\text{Sin } 103°}$$

Since the trigonometric tables do not go higher than 90°, we cannot use the figure 103°. However, we know that sin = sin (180−X), therefore:

$$\frac{AC}{\text{Sin } 32°} = \frac{AB}{\text{Sin } 77°}$$

$$\frac{AB}{.974} = \frac{100}{.530}$$

$$AB = \frac{100\,(.974)}{.530}$$

AB = 183.77 knots (closure speed)

(3) Consider a situation such as that in rule 4.

(a) The following is known:

The scratch mark on aircraft A was made in a forward direction and proceeding towards the left side of aircraft A.

The speed of aircraft A was 100 knots.

The speed of aircraft B was 150 knots.

The smaller scratch angle on aircraft A was 30°.

The larger scratch angle on aircraft A was 150°.

Because aircraft B was entirely consumed by fire, no scratch marks could be found on it.

Both aircraft were at the same altitude.

(b) By examination, the following can be deduced:

Since the scratch marks proceed forward and to the left side of aircraft A, then aircraft B approached from the rear and the right side. (WHY? We know that aircraft B was faster. Since the marks on A were made in a forward direction, then B approached from the rear. Since the marks on A also proceeded toward the left side of A, then B approached from the right side of A.)

The larger of the two angles between the scratch line and longitudinal axis on aircraft A is measured since it is the slower aircraft and was being overtaken; therefore, the angle used is 150°. In addition, since the sum of angles 1 and 2 are obviously greater than 90°, then the collision angle must be acute (rule 7).

(c) Solution to find:

What the scratch angles probably were on aircraft B.
The collision angle.
The closure rate.

Step 1. Using the law of sines we solve for the scratch angle 2:

$$\frac{AC}{\text{Sin angle 2}} = \frac{BC}{\text{Sin angle 1}}$$

$$\frac{100}{\text{Sin angle 2}} = \frac{150}{\text{Sin } 105°}$$

Since Sin 150° = Sin (180−150) then:

$$\frac{100}{\text{Sin angle 2}} = \frac{150}{\text{Sin } 30°}$$

$$\text{Sin angle 2} = \frac{100(\text{Sin } 30°)}{150}$$

$$\text{Sin angle 2} = \frac{100(.5)}{150}$$

$$\text{Sin angle 2} = .3333$$

Angle 2 = 19° 28′ (Probable scratch angle on aircraft B)

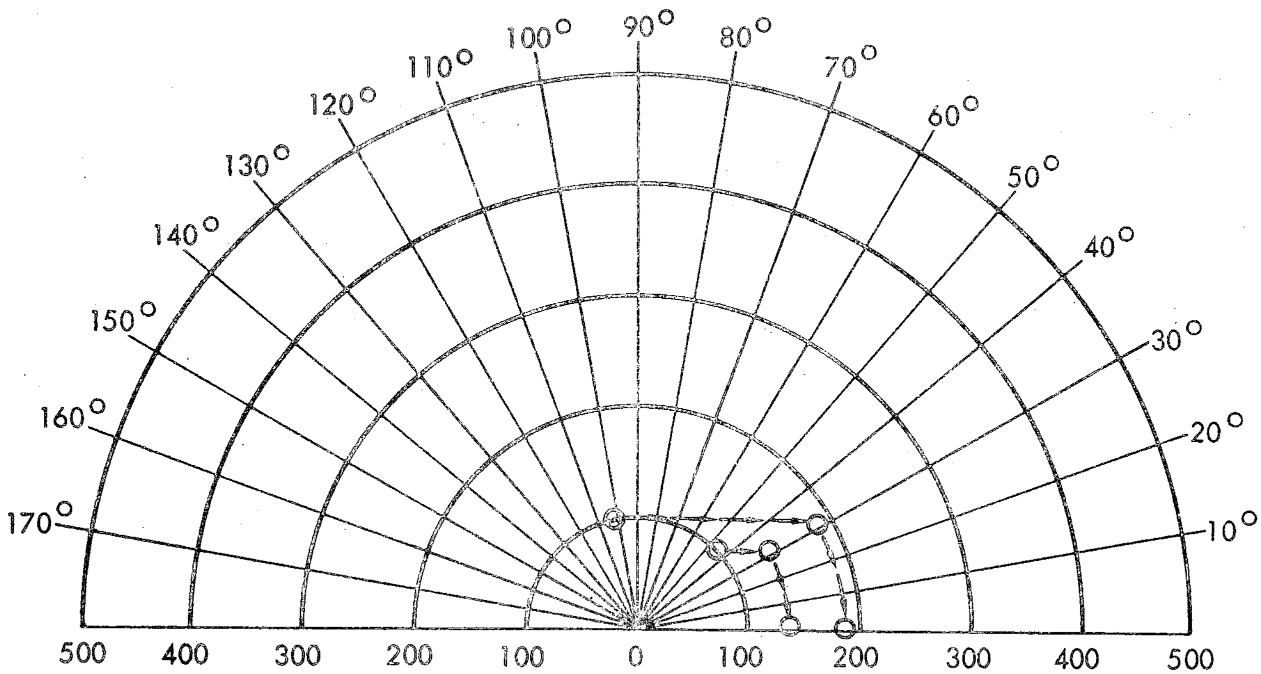


Figure C III-19.

SAMPLE PROBLEM

Given: Scratch angle on aircraft A equals 45°.
Scratch angle on aircraft B equals 32°.
Airspeed of aircraft A equals 100 kts.

Find:
1. Collision angle.
2. Speed of aircraft B.
3. Closure speed.

(1) Collision angle equals 180 − (45 + 32)
(2) Speed of aircraft B.
(Solid Line)

Enter the graph at 100 knots and proceed along the speed ring to 45°. Proceed horizontally to 32°, then along the speed ring to speed scale and read off 133 knots.

(3) Closure Speed (Dotted Lines)

Enter graph at 100 knots and proceed to 103° (Collision angle), then proceed horizontally to 32° (scratch angle on B), then follow the speed ring to the speed scale and read off 184 knots.

Step 2.

$$\text{Angle } 3 = 180 - (\text{Angle } 1 + \text{angle } 2)$$

$$\text{Angle } 3 = 180° - (150° + 19° 28')$$

$$\text{Angle } 3 = 10° 28' \text{ (Collision angle)}$$

Step 3. Using law of sines to find closure speed:

$$\frac{BC}{\text{Sin } 150°} = \frac{AB}{\text{Sin } 19° 28'}$$

$$\frac{150}{15} = \frac{AB}{.333}$$

$$AB = \frac{150(.333)}{.5}$$

$$AB = 99.9 \text{ knots (closure speed)}$$

(4) It can be seen that if the two aircraft were in the same vertical plane but in different horizontal planes (initially different altitudes, but converging), the problems may be solved in the same manner as illustrated.

The only difference would be that the scratch marks indicating the relative positions would be on vertical surfaces such as the sides of the fuselage, cowling, vertical fin, etc. Any scratches found on horizontal surfaces would probably be parallel to the longitudinal axis of each aircraft.

8.5. Graph

Working the problems out trigonometrically will obviously give more nearly accurate results than other methods. It is academic whether or not such extreme accuracy is needed in all situations. Therefore the following graph will provide a quicker method to arrive at the solution, in addition to eliminating the need for trigonometric tables (Fig. C III-19).

Using the same problem as the second problem under section 8.4 *Sample Problems,* we know that aircraft A was traveling 100 knots, the scratch mark on A was from front to rear and measured 45°, the scratch mark on B was front to rear and measured 32°.

8.6. Collision of Two Aircraft Operating in Different Horizontal and Different Vertical Planes

a. This condition, probably the most common, would at once appear to be a combination of the vertical and horizontal situations previously discussed. This is true with some modification, particularly in the speed category. If one or both aircraft are climbing or descending, then an adjustment must be made in order to arrive at a correct speed for the computation of the horizontal triangle. Once the horizontal triangle is solved, it is a simple matter to construct the vertical triangle and solve it. The effect of rate of climb or sink on horizontal speed is shown in the graph at the end of this section.

b. Consider that the horizontal aspects of the collision have been solved and the following triangle ABC has been constructed:

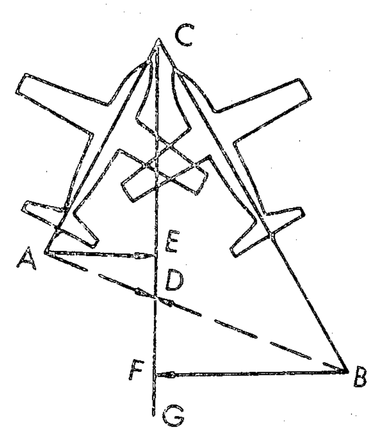
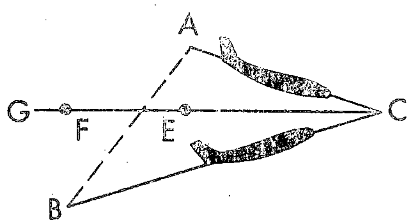


Figure C III-20.

The line CG is then constructed by bisecting the collision angle C. This divides the line of relative motion AB into two segments: AD and DB. From geometry it is known that these two segments are proportional to the sides of the triangle or AC/AD = BC/BD. In other words, as aircraft A progresses to the collision point C, it also moves towards aircraft B along the line AB to the point D. Likewise, aircraft B moves along AB to the point D.

The projection of the line CB (path of aircraft B) on line CG is represented by the line CF. The projection of the line AC (path of aircraft A) is represented by the line CE. The lengths of the lines CF and CE will be used in the following drawing to construct the vertical aspects of the collision. Two drawings are used, since it is possible that aircraft A was above B or vice versa.

The triangle which represents the vertical cross section of the flight is now solved in the same manner as the horizontal triangle.



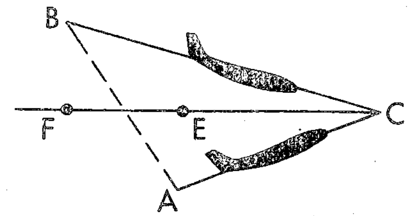


Figure C III-21.

c. We have now determined, through scratch mark analysis, both the horizontal and vertical aspects of the flight. The actual movement of the aircraft obviously took place in a plane which is the resultant of the horizontal and vertical planes. If the problem is constructed in three dimensions by placing a drawing of the horizontal movement perpendicular to a drawing of the vertical movement, the resultant plane is easily seen. The following drawing illustrates this:

The red triangle ABX is the resultant plane. The visibility that each aircraft would have of the other is now readily determinable.

d. The following graph, **Fig. C III-23** concerns the effect of vertical speed on horizontal speed. To use the graph, enter at the rate of climb (or sink), proceed horizontally to the curve of the airspeed of the aircraft (indicated or true), then proceed vertically, and read the horizontal airspeed component. In the sample illustrated, the aircraft is climbing at 4000 feet per minute, and the indicated airspeed is 150 knots. The horizontal component is found to be about 146 knots. It is obvious that if the horizontal speed is known and the indicated or true airspeed is known, then the rate of climb may be found. In other words, if any two of the three factors are known the third may be found by the use of the graph.

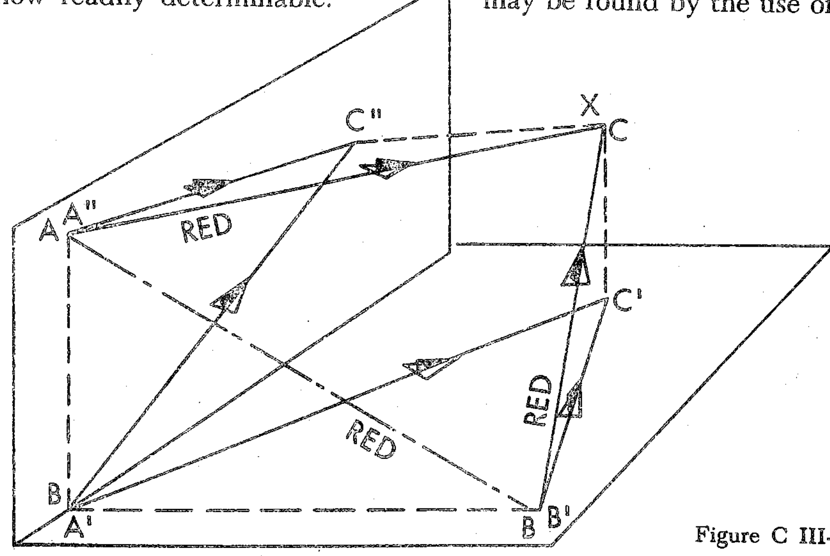


Figure C III-22.

EFFECT OF VERTICAL VELOCITY ON HORIZONTAL VELOCITY

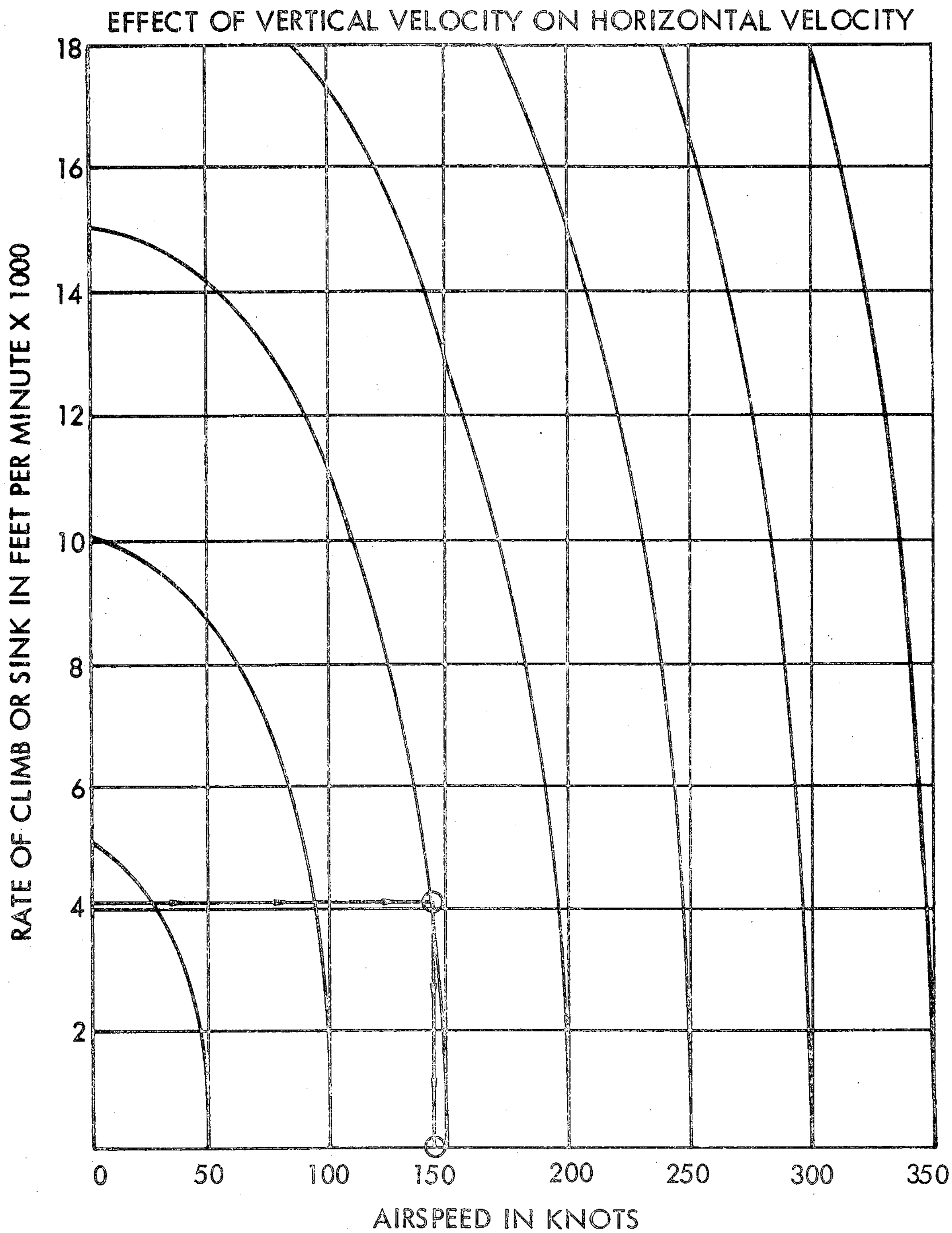


Figure C III-23.

8.7. Sample Problem Involving Simultaneous Horizontal and Vertical Movement

a. Known Facts about Aircraft A.

(1) From the ground speed computed from the times over two known radio fixes, the true airspeed along a horizontal path was found to be 100 knots.

(2) The reported altitude between the same two fixes established a rate of climb of 600 feet per minute.

(3) A scratch mark was found on the upper right wing of the aircraft which proceeded in a forward and inboard direction, making an angle of 60° with the longitudinal axis. Small pieces of metal were missing in a few places along the scratch mark, leaving holes in the skin.

(4) Scratch marks were found on the right side of the fuselage which proceeded in a forward-and-down direction, making an angle of 20° with the longitudinal axis.

(5) The aircraft was a Cessna 172.

b. Known Facts About Aircraft B.

(1) Scratch marks were found on the underside of the left wing and proceeded in an aft and inboard direction making an angle of 45° with the horizontal. Pieces of metal which matched the holes on the right wing of aircraft A were found still attached to the rivet heads along the scratch marks.

(2) Scratch marks were found on the left side of the fuselage and proceeded in an aft-and-upward direction making an angle of 9° with the longitudinal axis.

(3) The aircraft was a Beechcraft Bonanza.

c. Solution:

By using the knowledge from the facts and applying the principles of scratch mark interpretation, we hope to learn the manner in which the two aircraft came together. We will determine the unknown speed of aircraft B, and whether it was climbing, descending, or level. It will be possible to determine which pilot had the best opportunity to see the other aircraft.

(1) Let us first examine the scratch marks, and construct a drawing of their relationships.

(a) Horizontal scratch marks.

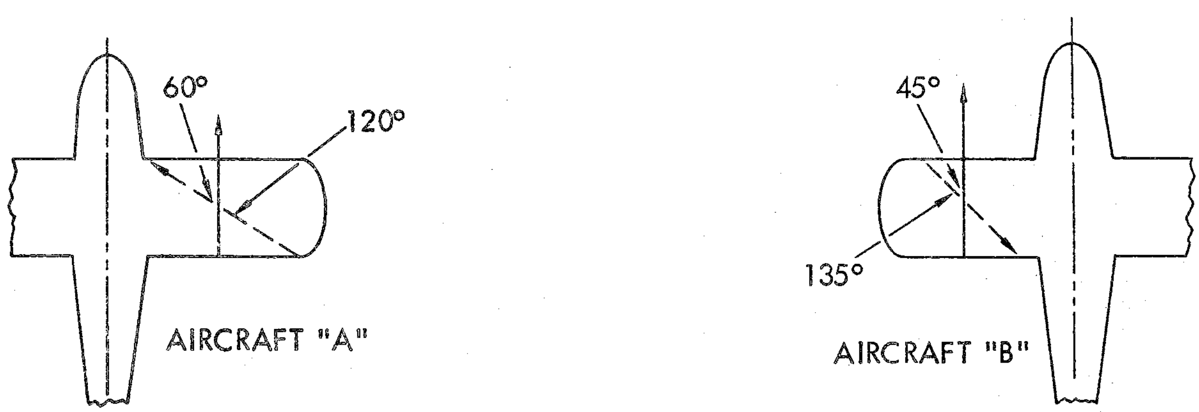


Figure C III-24.

From rule 4 it may be determined that aircraft A was the slower aircraft; it was overtaken by aircraft B, and aircraft B approached aircraft A from the right wing. The angle of 45° is the scratch angle to be used on aircraft B whereas the large angle of 120° is the scratch angle to be used on aircraft A.

(b) Vertical Scratch Marks

Figure C III-25.

From rules 13, 14, and 15 it may be determined that aircraft A was the slower aircraft, and that relatively, aircraft B was above aircraft A. The larger angle of 160° on aircraft A should be used and the small angle of 9° on aircraft B should be used. It is not yet known whether or not aircraft B was descending or level. We do know from the facts that aircraft A was climbing, and by consulting the graph we can determine that its true airspeed was 100.2 knots.

(2) Let us first construct a horizontal drawing of the probable collision path.

Label the triangle ABC with A being the slower aircraft, B the faster, and C the point of collision. Label the known facts such as the known speeds and scratch angles. We may now solve the triangle to determine the horizontal collision angle, the horizontal speed of aircraft B, and the closure speed of the two aircraft (line AB).

(a) Collision Angle

Angle C equals 180 − (120 + 45) which equals 15°.

(b) Speed of aircraft B.

Using the graph, **Fig. C III-19**, the speed of aircraft B is found to be 122 knots.

(c) Closure rate.

Using the same graph, the closure rate is found to be 36 knots.

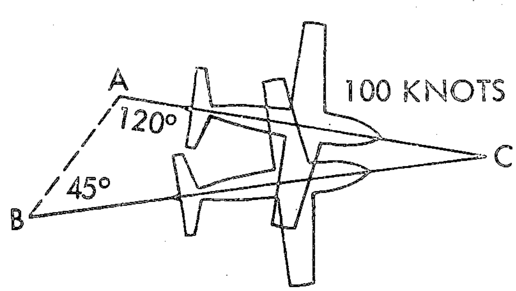


Figure C III-26.

(3) Using the same triangle, bisect angle C and construct as in **Fig. C III-20.** This is for the purpose of determining the vertical aspects. Label with the facts known and computed so far, i.e., Speed of A: 100 knots, Speed of B: 122 knots, Collision angle C: 15°, etc.

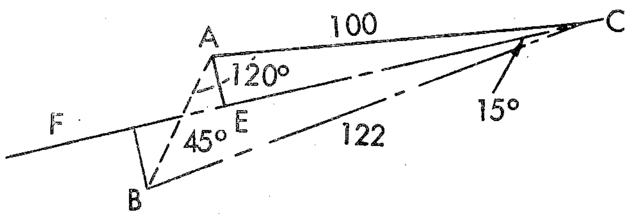


Figure C III-27.

If the drawing is made to scale, then the lengths of the lines CE and CF may be measured and converted to speeds. Otherwise they may be solved trigonometrically or by the use of the graph, **Fig. C III-23.**

(a) Line CE

Since Angle C is bisected, half the angle of 15° is 7½°.

$$\text{Cos } 7\tfrac{1}{2}^\circ = \frac{CE}{100}$$

$$.991 = \frac{CE}{100}$$

$$CE = 99.1 \text{ knots}$$

(b) Line CF

In a similar manner, Line CF may be found to be equal to 120.9 knots.

(4) Now the vertical triangle may be drawn and solved.

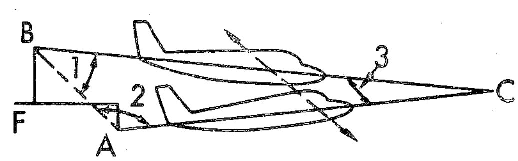


Figure C III-28.

About the above triangle we know the following:

(a) EC equals 99 knots.

(b) CF equals 121 knots.

(c) Angle 1 (scratch angle on B) equals 9°.

(d) Angle 2 (scratch angle on A) equals 160°.

(e) Rate of climb on aircraft A is 600 feet per minute. On the drawing this would be represented by the line AE. The rate of climb of aircraft B is unknown, but would be represented by line BF.

(f) Collision angle C equals 180 − (160 + 9) or 11:

(g) From the graph, **Fig. C III-23,** which deals with the effect of rate of climb on horizontal speed, the speed of line AC may be determined. This is found to be 99.5.

(h) From the scratch angle graph, **Fig. C III-19,** we can now determine the speed line BC. This was found to be 122 knots.

(i) The rate of closure (Line AB) is found in a similar manner to be 68 knots.

(*j) From the graph, **Fig. C III-23,** or mathematically, the rate of sink of aircraft B may now be determined since we know the horizontal speed CF and the descending speed BC. This was found to be 1800 feet per minute.

(k) From the same graph, the true airspeed of aircraft B may now be determined by comparing the rate of climb (1800 feet per minute) with 122 knots horizontal speed. The airspeed is found to be 123.3 knots.

*It is apparent from the calculations in this section that the difference between the indicated or true airspeed and the horizontal component is insignificant at the lower rates of climb or sink. In fact it is practically impossible to determine the differences on a graph unless the scale is so large as to be unwieldy. Therefore at the lower rates of climb, a higher degree of accuracy is possible by solving the triangles mathematically rather than by use of the graph.

(5) We can now summarize what we have learned:

	Aircraft A	Aircraft B
True airspeed	100.2 knots	123.3 knots
Rate of climb or sink	600 f.p.m.	1800 f.p.m. down

Position of aircraft B as viewed from aircraft A: 120° to the right and 160° above the nose. Above and to the right rear.

Position of aircraft A as viewed from aircraft B: 45° to the left and 9° below the nose.

Collision angles: 15° horizontally and 11° vertically.

Rates of closure: Horizontally: 36 knots, vertically: 68 knots.

Resultant closure rate: 43 knots (This is found by comparing the horizontal closure rate with the combined rates of climb.)

(6) Conclusions

Since aircraft A was a Cessna 172 which is a high wing monoplane type, the thought immediately occurs that the pilot or other occupants could not have seen the approaching aircraft B. An examination of the scale drawings of the Cessna 172 reveals that the pilot could not have seen aircraft B. However, the occupant of the right rear seat could have seen aircraft B if he had the occasion to look to the rear and upwards.

The pilot of the Bonanza, (aircraft B) by looking to the left and slightly down should have easily seen aircraft A, provided that weather conditions were such that visibility was good. Since the resultant closure rate was found to be 43 knots, the aircraft would have been .7 nautical miles apart, one minute before the collision or 1.4 nautical miles apart, two minutes prior to collision.

8.8. Summary

It is obvious that two aircraft will not necessarily follow unswerving straight lines for an indefinite period and ultimately collide. The scratch marks will reveal the relative positions at and just prior to impact. Therefore, the entire story can be determined only by considering the scratch marks with the other facts learned during the investigation.

The accuracy of the results is a function of the accuracy and quantity of factual information found. It is possible that scratch mark evidence will prove or disprove evasive action. Who was at fault may in most cases be definitely proved. It is incumbent on the investigator to accumulate all the facts possible.

In the treatment of the problems, certain other factors were omitted. For example, it is recognized that all aircraft do not climb nor descend along a path parallel to the longitudinal axis. This may be a function of flaps, climb speed, etc. Therefore, to be more accurate, the relationship of the longitudinal axis to the flight path must be considered, and a correction applied to the scratch angle. Similarly, yaw was not considered. This situation is quite possible with the V-tailed Beechcraft used in the sample problem or in the case of a multi-engine aircraft with one engine feathered.

In every case of midair collision, it is important that each step in the process of investigation be performed carefully. It is suggested that sketches be made to clarify each step of the solution as it is performed.

A suggested step-by-step procedure follows:

1. Determine whether or not one aircraft overtook the other or whether the collision was the head-on type.
   a. Either place a scratched piece from one aircraft beside the scratched piece from the other aircraft or simulate by a drawing.
   b. Place the pieces in such a position that the longitudinal axis of one aircraft is parallel to the longitudinal

axis of the other. Now view both pieces as though you were looking from the rear toward the nose of both aircraft.

c. If one scratch mark slants to the left and the other to the right, then the aircraft approached in a head-on manner and the collision angle was obtuse. Therefore, measure the smaller angle between the longitudinal axis and the scratch mark on each aircraft. (Rules 2 and 5)

d. If both scratch marks slant to the left, one aircraft overtook the other with the faster aircraft being to the right of the slower. (Rule 4, top illustration of Fig. C III-5)

e. If both scratch marks slant to the right, one aircraft overtook the other with the faster aircraft being to the left of the slower. (Rule 4, lower illustration of Fig. C III-5)

f. In both d and e, measure the larger angle between the longitudinal axis and the scratch mark on the slower aircraft, and the small angle on the faster aircraft. (Rule 4) The slower aircraft is the one on which the scratch mark was made in a forward direction. (Rule 4)

g. Follow the same procedure for the vertical scratch marks, if any.

2. Draw the triangles as described in the sample problems, preferably to scale.

3. Solve either mathematically or by the use of the graphs.

9. Fire Analysis — General

Fire is a universally recognized causal factor of aircraft accidents. Consequently, the investigation for fire becomes a logical part of every accident investigation. In general, a complete fire investigation consists of the following:

a. Determine that a fire did or did not exist.

b. Determine the crash/fire sequence.

c. Determine the inflight fire damage and its effect on equipment and flight.

d. Determine the source of combustibles and ignition.

e. Determine effectiveness of fire detection and firefighting equipment procedures.

f. Recommend preventive measures.

Fire can be the cause or it can be the result of aircraft accidents. Consequently, investigators of accidents involving fire are faced with the problem — which came first, the fire or the crash? This problem is compounded by the fact that inflight fires are usually followed by post-crash fires which can obscure or destroy the telltale evidence of the inflight fire. The crash/fire sequence problem can usually be solved through intelligent, persevering investigation. The techniques to be used are not unique, but require observation and common sense, combined with a knowledge of the characteristics of fire and of fire's effect on materials.

9.1. Approach —

From a theoretical standpoint, all of the physical evidence remaining is there in the wreckage, and an investigator should be able to extract the evidence without prior knowledge of the accident. Practically, the time required is not available. Also, man is not all-observing and some evidence is bound to be missed — possibly even destroyed — without recognition. Therefore, it is best if some specific possibilities, ideas, theories, items to check, etc., are outlined prior to examination of the wreckage.

Occasionally an accident occurs about which very little direct information is known. There may be no survivors, witnesses, or pilot statements. However, information may still be obtained from the airplane's maintenance history, pilot's record or habits, weather, or accident history of the aircraft type. The danger in these ideas or theories is that they may cause the investigator subconsciously to look only for evidence that will fit the theories and completely overlook evidence which he would otherwise recognize as significant. Human nature being what it is, some theories and ideas are bound to be formed. It is obviously better if these are based on all of the information available.

9.2. Preserving the Evidence

Eventually, all of the witness' statements, together with all possibilities and theories, must be proved or disproved to the extent possible, through examination of the wreckage. It is therefore of utmost importance that the evidence in the wreckage be preserved and that false evidence not be introduced. The type of evidence which is significant to the fire investigator may be extremely minute, unrecognized by untrained personnel, and easily destroyed. The location of a part by a foot or even inches, the attitude or exact position of parts, scratch marks through soot, etc., may be significant. Rescue of personnel involved is, of course, the first consideration. The degree to which further loss or damage occurs will depend largely upon the individual circumstances involved and the proper training and indoctrination of authorized personnel in attendance at the scene. A well-trained crash crew will be aware of the importance of minimizing aircraft damage, and will mentally log the damage which they must inflict to perform their duties. Preservation by photographic means should begin as soon as possible with immediate effort being concentrated on those objects or areas most likely to be affected by the fire or rescue operation. Subsequently, all physical evidence having a direct bearing on the accident should be photographed as found in the wreckage before being removed.

Restraint of curiosity seekers, scavengers, and others who might interfere with wreckage or the investigation procedure is a matter for the cognizance of those in authority at the scene.

Two problem areas in preserving the evidence are not always recognized. One is the cleanup of the firefighting chemicals or agents. It is quite apparent that the presence of these agents considerably hinders an investigation, and that after a preliminary examination it is desirable to remove them. It is fortunate that there is no rapid chemical action between the commonly used agents (CO_2, mechanical foam, dry powder) and the aircraft structure or fire residue. The removal problem is therefore a mechanical one of removing the agent without removing or displacing soot patterns, fire residue, fluid films, etc. No method of agent removal is completely satisfactory. Consequently, as much inspection as possible should be done before removal is attempted. In some cases, a light air blast will remove much of the dried materials with a minimum disturbance of the evidence. Light water sprays will also remove these agents, but some of the loose, sooty deposits will be washed away, and may be deposited on other parts and edges. High pressure solid water streams must be avoided, since they will remove more of the deposits and are capable of scattering the wreckage.

A second problem is the structural disassembly of the wreckage so that all of it can be inspected. Frequently the structure will be deformed and compressed so that it and entrapped equipment cannot be inspected. The complete engine may be so entrapped. This problem of structural disassembly is becoming increasingly more difficult as more and heavier gauge stainless steel is used. The parts are heavy and awkward to handle. During attempts to stretch them apart, they seem to have the characteristics of spring steel. Saws are extremely slow, and when the wreckage is compacted deeply, impossible to use. Usually oxyacetylene or electric arc cutting is resorted to. The result of the handling and cutting is the introduction of a great deal of false evidence, if not actual destruction of significant evidence. Oxyacetylene and electric arc cutting are burning processes which produce soot, metal discoloration, and material damage similar to an intense fire.

Such structural disassembly requires a high degree of patience and ingenuity, plus the use of special equipment to handle highly individualized projects involving various combinations of cutting, stretching, and nut and bolt disassembly. Because of the unavoidable damage, it is imperative that the disassembly be controlled by an investigator. Step-by-step photographs should be taken, and all areas and edges affected by cutting should be marked with a colored grease pencil so that there will be no doubt during reinspection of the wreckage.

9.3. Methods of Determining if the Fire Existed in Flight

a. Locate Parts Not Subjected to the Ground Fire

The solution of the crash/fire sequence problem is mainly one of observation and application of average common sense. The most logical place to start is to locate parts which were not subjected to the post-crash ground fire and to examine them for evidence of fire. Any evidence of fire would be positive proof of inflight fire. The evidence to look for is soot, heat discoloration, charred sealant, metal spray, etc. In this regard, it is imperative that the investigator have a knowledge of normal conditions of the airplane parts, for they may normally have the appearance of being subjected to fire.

Erroneous conclusions can also be reached from observing normal heat discolorations. The heat discoloration of materials is a function of time at temperature. The same discoloration can be achieved by exposure to a low temperature for a long period of time as will result from a high temperature exposure for a short period of time. The discoloration of titanium exposed to 600°F for 260 hours will be the same as that resulting from exposure to 1000°F for 15 minutes. The increased use of stainless steel and titanium results in higher normal operating temperatures. At these higher operating temperatures, the steel and titanium structures gradually acquire a blue discoloration.

One of the primary methods of determining whether a part has been subjected to ground fire is to note the location of the part in relation to the apparent ground fire area. Parts or molten droplets may be shed in flight and found along the flight path. Other parts may be thrown completely clear of the fire area by the force of the impact. Even parts found within the ground fire area may be free of ground fire damage. Frequently parts are buried under a protective covering of dirt, both at the initial point of impact and at the point of rest. Occasionally, the crash scene will be just a hole in the ground and the wreckage must be dug out from the sides of the initial explosion, with the parts protected from fire by the dirt covering. If the crash site is swampy or in water, the water may shroud the parts. In cases of prompt ground-fire fighting, the fire-fighting chemicals may shroud the parts, or parts may be below the liquid level of unburned fuel. In some cases, parts may be trapped or enclosed in other parts which shroud them from the ground fire.

The location of a part may not be completely decisive in determining whether or not it was subjected to ground fire, or what the crash/fire sequence was. Close scrutiny of the part may provide additional information. The evidence to look for is the relation of the effects of fire to the results of the mechanical disintegration. The existence of bright scratch marks, scuffs, and smears in the soot and discoloration would indicate that the disintegration took place after the soot and discoloration had formed from an inflight fire. Soot in torn edges indicates that the part was subjected to fire after mechanical disintegration. Usually, heat discoloration of torn edges and scratches also indicates that the part was subjected to fire after mechanical disintegration. This is not always true, for the residual heat remaining in the part after being thrown clear of any ground fire may be sufficient to discolor the exposed surfaces. This is more apt to occur with parts of large mass.

b. Soot, Heat, and Fire Patterns

Inflight fires other than electrical are usually the result of some failure or condition which releases a combustible fluid or vapors. The combustible may drift or flow a considerable distance and be widespread before reaching an iginition point, but once ignited, it will flash back to the source of the combustible and burn from there, producing a reasonably concentrated fire similar to electrical fires. The spread of the flame, heat, soot, and consequently, the fire damage, from the source of the fire is greatly influenced by the airflow in the region. The usual influence of the airflow is to confine effects of fire to the shape of a cone with the apex of the cone at the combustible source and expansion in the direction of the airflow. Obviously, the presence of structure and equipment downstream will alter the shape of the cone. This confining of the effects of fire results in outlines or patterns of soot and heat from which a great deal can be learned, primarily because the patterns formed in flight will not be the same as those formed by a ground fire. As explained above, the direction of the soot and heat pattern is controlled by the direction of the airflow across the parts. In flight, this is usually from forward to rear, but once the aircraft has come to rest, the direction of airflow across the parts will be changed for two reasons. First, the smoke and flame will rise vertically or be blown in the direction of the ground winds. Second, the disintegration of the airplane will cause the parts to be in random orientation. Also, the impact or continued ground fire may open fuel cells or other combustible containers, thereby providing a much broader source of combustibles and a fire whose limits are beyond the surfaces of the airplane so that a pattern cannot be detected.

A heat pattern consists of the deterioration and discoloration of the objects in the area. In order to detect the pattern, the investigator must know the effects of heat on various materials. As previously explained, the degree of these effects is a function of time at temperature, and the time of exposure must be considered in cases of a sustained fire. Usually, just a knowledge of the effects will enable determination of the relative deterioration and from this, the heat pattern. The effects are many. A few which have been found useful are:

(1) Glass cloth fuses at 1200°F.

(2) Cadmium plating starts to discolor at 500°F.

(3) Titanium melts at about 3300°F. Most titanium alloys melt in the range of 2800 to 3200°F, but a few start to melt between 2250 and 2800°F.
Most stainless steels melt in the range of 2500 to 2650°F.
Copper melts at 2000°F.
Brass bearings melt at 1600°-2000°F.
Aluminum alloys melt between 900 and 1200°F. This same range (900-1200°F) applies to magnesium alloys.

(4) Neoprene rubber blisters at 500°F.

(5) Silicone rubber blisters at 700°F.

(6) Zinc chromate paint primers start to tan at 450°F; are brown at 500°F; are dark brown at 600°F; and are black at 700°F.

(7) Stainless steel discolors starting at 800°F to 900°F from tan to light blue, to bright blue, to black with increasing temperature.

(8) Titanium discolors from tan, to light blue, to dark blue, to gray with increasing temperature.

(9) Titanium metal has a high affinity for gases when heated, and a scale will begin to form at 100°F. This scale increases thickness with time at temperature.

A soot pattern is formed as a result of the soot drifting with the airstream until it strikes an object to which it can attach itself by means of the unburned oils it contains and by electrostatic attraction. A point to remember is that soot will not attach itself to surfaces which are over approximately 700°F. Consequently, areas which show the greatest intensity of fire may contain little or no soot.

It may be necessary to reconstruct the airplane from the remaining parts in order to detect a pattern. If, following reconstruction of the airplane, there is a detectable pattern in the direction of the inflight airflow, an inflight fire is indicated. Conversely, if there is not continuity of pattern across lines of failure, the patterns were formed after disintegration.

The shapes of the patterns will be affected by any object which tends to shroud or block off another part. The shrouded part will show the general outline of the object doing the shrouding. If a part is found with such an outline, but the part which did the shrouding is not there, the pattern must have occurred before disintegration. Conversely, if both the outline and the shrouding part are found in relation, but the shrouding part is not normally in this position in the airplane, the pattern was formed after disintegration. An example of the latter would be the finding of clean surfaces upon unfolding a sooted part.

c. Heat Intensity

Heat intensity is another possible means by which the crash/fire sequence can be determined. It is becoming more prevalent as more of the higher heat-resistant materials are used. The flame temperatures of post-crash fires in which combustibles like gasoline, JP-4, lubricating oil, and hydraulic fluids are being consumed in still air, is normally in the range of 1600° to 2000°F. The flame temperature of inflight fires may be in excess of 3000°F due to the forced draft of the slipstream compartment cooling air. The effect of the forced draft is to cause the fuel/air ratio to be more nearly stoichometric. Therefore, when any parts which have a melting point in excess of 2000°F, like stainless steel and titanium, are found showing evidence of melting, it is a strong indication, but not conclusive, that it occurred in flight. The indication is stronger if the part is found in an area in which it appeared that the ground fire was not intense. It is not conclusive, because it is possible for the ground fire to exceed 2000°F. Strong ground winds may provide a forced draft, or peculiar piling of the wreckage may cause a chimney effect whereby the fire causes its own draft. In addition, materials like magnesium, which burn with an intense flame, may be present. Usually, the areas in which a flame temperature hot enough to melt stainless steel or titanium exists are very small, and are the result of some localized jet effect similar to a welder's torch.

d. Determine that Fire Conducive Conditions Existed.

Frequently, a failure or condition which would logically produce fire is found before any evidence of inflight fire is found. This circumstantial evidence should be proved or disproved by locating inflight fire evidence in the wreckage. Fire will not always occur when logic says it should. This is not due to trickery of fire, but because man cannot always accurately predict or determine the conditions on which logic is based. The odds, however, are in favor of logic and the discovery of such evidence will indicate a specific, relatively small, area for intensive investigation.

Such circumstantial evidence is almost infinite in variety. It could be almost any failure. It may be a burn-through of the engine, disintegration of high-speed rotating equipment, electrical shorting, etc. Incidentally, electrical arcing damage can usually be differentiated from fire damage. The damage from electrical arcing is very localized as to both metal removal and heating. The damage will have an eroded appearance and there may be metal splatter similar to that produced in arc welding. The strands of the copper wiring will be fused together, and usually little beads are formed on the ends. Such fusing does not occur from fire. The difference is probably due to the heating rate and intensity. When heated externally, the heating rate is relatively slow. This permits a scale to form on the surfaces of the strands and the scale prevents fusing. In addition, the intensity of most fires, particularly those on the ground, is not sufficient to melt copper.

A word of caution in regard to evidence which indicates that an inflight failure or fire condition existed: a ground fire or the impact may produce such evidence. An example is "B" nut line connections. It is not uncommon to find numerous "B" nuts, both steel and aluminum, only finger tight in the wreckage, thus indicating that an inflight leak existed. Loose "B" nuts may be caused either by mechanical damage or by fire. Loosening by mechanical damage is usually evident by the mechanical condition of the connector and its attaching lines. The loosening by fire is probably due to annealing and relief of the stresses which constituted the torque. If a "B" nut is more than a quarter of a turn loose, it is not the result of fire.

9.4. Conclusion

Quite frequently the crash scene will initially appear as though the ground fire had consumed all of the evidence and that any investigation will be futile; but through patient, ingenious application of all the known techniques it would be rare indeed if some information could not be obtained. Granted, it may not be easy technically or comfortable physically. Consequently, every effort should be made to improve existing, and to develop new, accident investigative techniques.

Part C — AIRCRAFT AIRWORTHINESS INVESTIGATION

Chapter IV

POWERPLANTS

1. Reciprocating Powerplants

Powerplants have established a record for reliability over the years, but they are just mechanical devices and subject to mechanical failure.

The investigation should never be closed until the investigator has proven the extent to which the engine has, or has not, contributed to the accident, with his proof documented beyond doubt as to his findings. The actual extent of the investigation will vary with the circumstances surrounding each accident. The investigation in all cases should be to the extent necessary to establish the condition of the engine and its components, whether or not they contributed to the accident, or were a causal factor.

1.1. Investigation at the Scene of the Accident

The investigation at the accident site occasionally can constitute the entire powerplants investigation. Many times an investigator can obtain enough information from the physical evidence at the site to establish that the engine did not contribute to the accident.

This section will deal with the items that usually can be checked and documented at the crash site. However, if the engine is so badly broken up that sufficient information cannot be obtained for complete documentation of the engine and accessories, the investigator may have to move the wreckage to a location with adequate facilities for a complete teardown and inspection. The procedures for the latter will be covered in section 1.2.

1.1.1. Fuel System

The fuel system investigation at the scene of the accident should be a determination of whether or not the engine was getting sufficient fuel, free of contaminants, to develop power.

1.1.1.1. Methods of Obtaining Samples and Why They May Be Necessary

If there is any doubt about the quality of the fuel or the presence of contaminants, samples should be obtained as early in the investigation as possible. If a complete analysis is required it is recommended that as much fuel as possible be obtained, up to one gallon. The sample must be collected in a clean container and drawn from a fuel tank sump if possible. If this is not feasible it may be possible to obtain enough from the settler bowl or a fuel line to the engine.

Care must be exercised when obtaining samples to keep from losing valuable evidence in the form of water or other foreign material possibly contaminating the fuel.

1.1.1.2. Investigation of Tanks, Plumbing, and Valves

During the investigation of the fuel system the quantities of fuel in each tank should be carefully documented. The fuel lines should be checked to ascertain whether or not there were any breaks, leaks, or clogged lines prior to impact. All selector valves and switches should be documented as to position and as to which tanks were in use. If possible, determine whether valves were functioning properly prior to impact. Check the boost pumps for proper operation and whether or not they were capable of operation prior to impact.

Some investigations may require close coordination between the Powerplants Group and others, such as the Operations and Systems Groups, for the investigation of such things as cockpit settings, valve positions, and components of the fuel system.

1.1.2. Oil Systems

Oil systems can furnish the investigator with some very valuable information concerning the engine condition and operating capability.

1.1.2.1. Methods of Obtaining Samples and Why They May Be Necessary

As in the fuel system, the oil samples should be obtained in a clean container. Care should be exercised in obtaining the samples to assure the collection of the first oil to leave the engine or tank. This is important, because if the amount of contaminants is small it may be flushed out with the initial surge of oil as the plug is removed. Samples should be taken from the engine sump, the main screen, or the tank. In some cases it may be advisable to take samples from all the sources.

1.1.2.2. Types of Possible Contamination and Their Significance

Some of the contaminants that may be found in the oil system are metallic or carbon particles, foreign fluids, and sludge. The metallic particles may be either ferrous or non-ferrous. Ferrous particles indicate some failure of steel parts within the engine and the size and shape can give a clue to what failed. The most probable sources of steel shavings or particles are cylinder walls, piston rings, and gears.

Non-ferrous particles usually indicate failure of sleeve bearings, bushings, pistons, or some other aluminum, magnesium, or bronze part of the engine.

Excessive sludge or carbon may cause oil starvation and engine failure. Foreign fluids such as water or fuel are not too prevalent in the oil system, but their presence, if excessive, will change the lubricating qualities of the oil and cause trouble in the engine. Another possibility of engine failure is the wrong type of oil. This can cause serious overheating and engine malfunction to the point of complete stoppage.

1.1.2.3. Inspection of Oil System Components Such as Pumps, Tanks and Plumbing

Engines with wet sump oil systems have internal plumbing, and therefore, the sump and screens are the best sources for contaminants. Some engines have a magnetic plug to trap steel particles. This plug is of great assistance in determining the type of metal particles found in the system; without the plug, a very close inspection would be required to differentiate between some of the non-ferrous materials and steel particles.

Oil tanks, plumbing, and valve position and condition are checked on dry sump installations as well as the sumps and screens. The same type of inspection applies to the oil system as described for the fuel system. Figure C-IV-1 shows an oil screen removed from an engine that failed in flight.

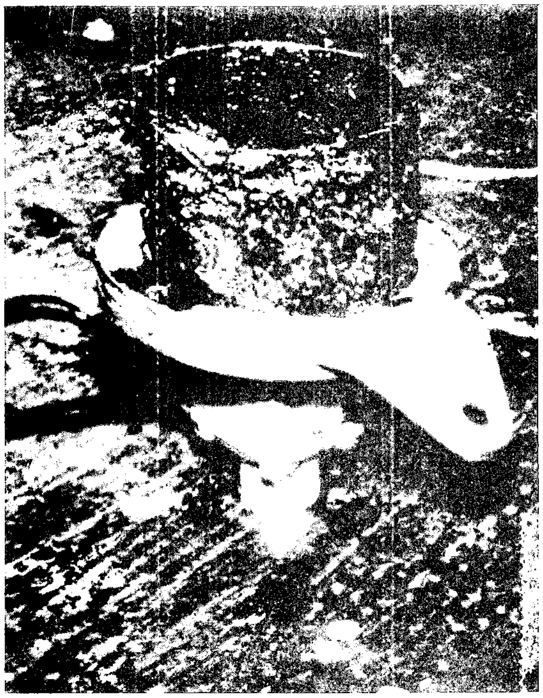
1.1.3. Ignition Systems

In many cases, the inspection at the accident site can eliminate the possibility of an ignition system failure. On the other hand, if there was a malfunction, it may be possible to determine the degree and the cause. This applies mainly to accidents in which the engine is not too seriously damaged and there are sufficient parts to make a determination.

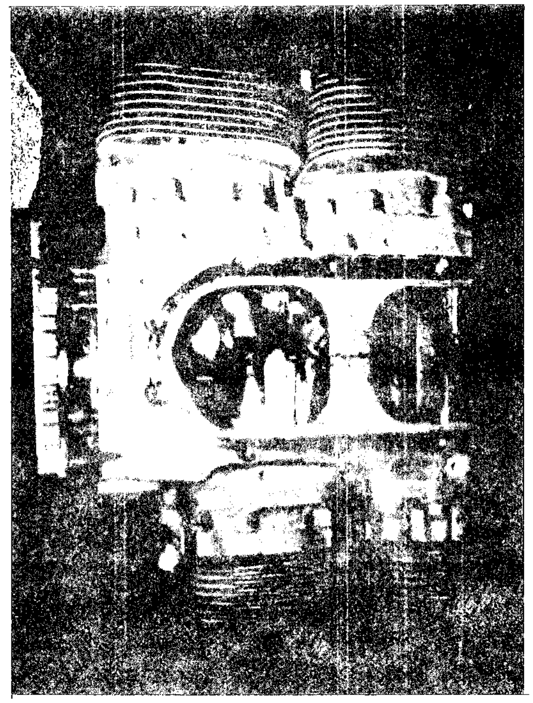
1.1.3.1. Magnetos and Ignition Timing — Methods of Checking Condition of Magnetos — Checking the Timing of Various Magnetos to the Engine

Magnetos and ignition timing can be checked by various methods, depending upon the engine and the engine installation. Most of the light engines have timing marks etched on the crankshaft flange. However, some engines have other means to check the timing to the engine. There are also patented timing devices. The investigator must determine the means of accomplishing this check for the particular engine under investigation.

There are several different types of magnetos, the most common types found on light aircraft are Eisemann and Scintilla. The method of checking the points and general



Screen from Franklin engine filled with helicopter clutch material.



Crankcase cover removed from Franklin engine at the scene.

FIGURE C IV-1

condition of the magneto varies, and this information is available in the manufacturers manuals. The best method for checking the opening position of the points is the use of a timing light. However, a 0.0015 feeler gauge or a piece of cellophane, the type used on cigarette packages, can also be used. The points should open when the number one piston is in the firing position.

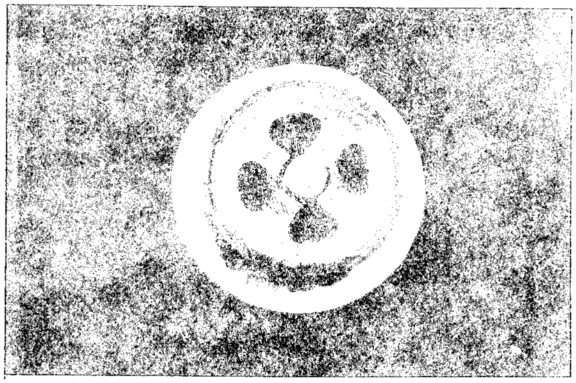
1.1.3.2. Checking the Harness — How to Use the Various Testers and What to Expect During the Test

The ignition harness should be visually checked for obvious signs of deterioration. If there is doubt, or circumstances dictate, a more thorough check can be made with a harness tester or a megohm tester. When using a tester on the harness, be sure to follow the operating instructions for the type of tester being used.

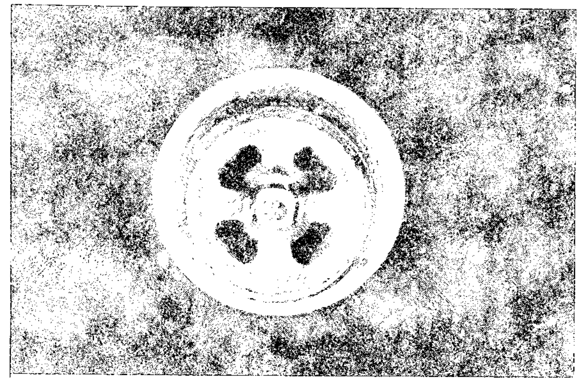
1.1.3.3. Spark Plugs — Methods of Removing Spark Plugs and Precautions to Observe — Checking Engine Compression and Significance — Spark Plug Indications and What They Have to Tell the Investigator

Spark plugs can sometimes tell an investigator a great deal about the engine and how it was operating prior to the accident. Therefore, when removing the plugs, take precautions to protect them from further damage. As the spark plugs are removed, check them for any indication of plug or engine trouble. Be sure to label them so that it can be determined later from which cylinder they were removed. If the engine can be rotated, it is best to remove one plug from each cylinder until a compression check is made.

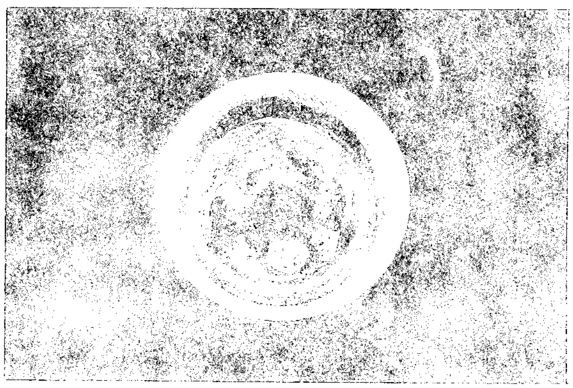
CAUTION: **Before rotating the engine, make sure ALL ignition leads are disconnected from the spark plugs.** It is emphasized that all

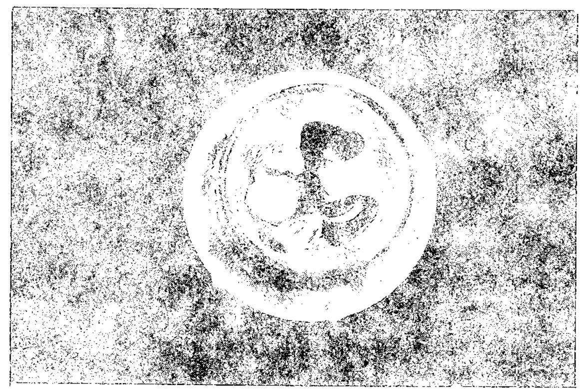


Ground electrode damaged by peening.

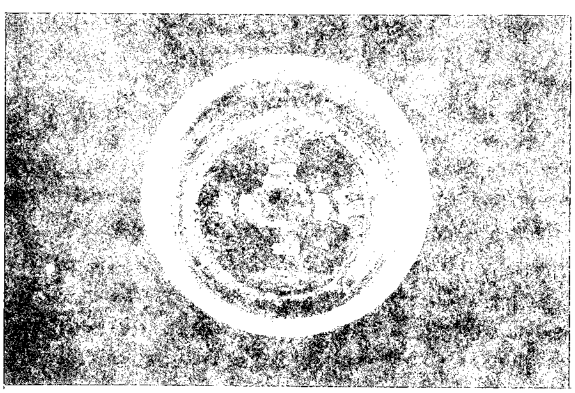


Grounded electrode. Caused by peening.

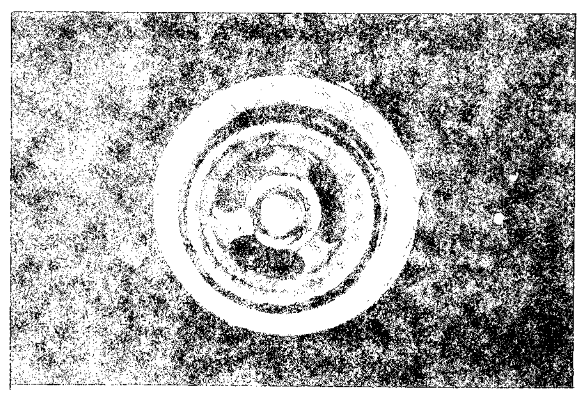




Badly damaged spark plugs. Damage is usually caused by a broken piston, valve, or piston ring.



Copper runout.



Badly eroded electrodes.

Figure C IV-2.

plugs should be inspected and labeled as they are removed. Some of the things an investigator should look for when removing plugs are: spark plug loose in the cylinder or overtorqued; deterioration of the lead; arcing between cable and contact spring; mechanically damaged electrodes; sprayed metal deposits on the nose of the spark plug; copper runout of the center electrode; badly eroded center electrodes; fuel fouling; lead fouling; oil fouling; and other-than-normal-colored deposits. Each of the above indications has a story to tell, and the more experienced the investigator, the better he will interpret the story spark plugs have to tell. **Figure C IV-2** shows some typical spark plug failures.

1.1.3.4. Compression Check

The compression check can be made by holding a thumb or finger on the spark plug hole while rotating the engine. This is not positive proof of the cylinder condition, but it can furnish the investigator with an idea of the condition of the cylinder.

1.1.4. Carburetion

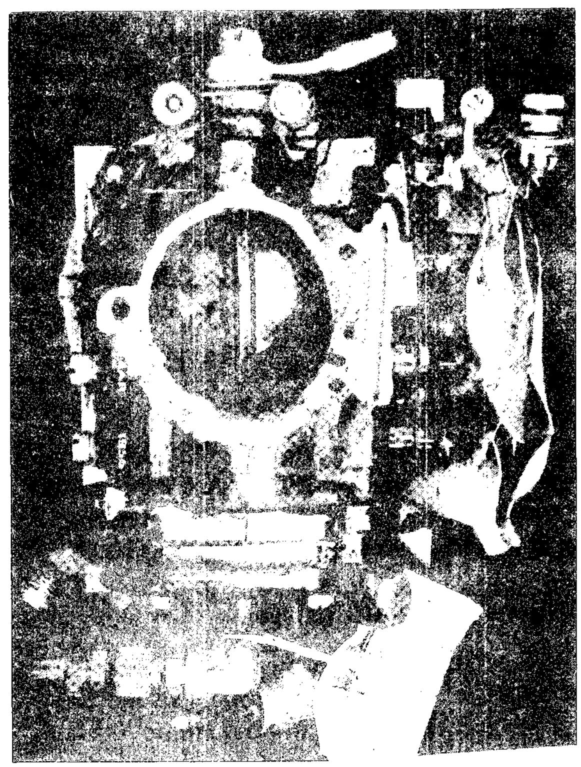
When engine stoppage may be a factor in the investigation, the carburetion system should be checked out as completely as possible. The principal factor here is whether or not the engine was getting the proper amount of uncontaminated fuel.

1.1.4.1. Float Type Carburetors

The float type carburetor can be checked quite easily if it is intact. All float type carburetors have an inlet screen that can be removed and checked for deposits of foreign material. The float chamber can be drained to check for water or other foreign material. Another check can be made simply by opening the carburetor to visually inspect the floats for leaking or sticking. The investigator should keep in mind that the floats may stick when the carburetor is assembled, but they may be perfectly free when disassembled.

1.1.4.2. Pressure Type Carburetors

Pressure type carburetors are more difficult to check at the scene than float type carburetors. The only checks that can be made conveniently are of the screen, availability of fuel to the carburetor, whether or not the poppet valve is free and clean (PS-5C), and for evidence of internal leaks. **Figure C IV-3** shows an unusual type failure in a pressure type carburetor. A cleaning rag was left in the air intake duct and during takeoff it blocked the carburetor inlet which led to a fatal accident.



Rag in PS-5C carburetor which caused a fatal accident.

Figure C IV-3.

1.1.4.3. Direct Injection

The unit used on the R-3350 engine is a master control that meters fuel to the injection pumps. These pumps furnish timed high pressure fuel to the injection nozzles. It is possible to check these units for proper operation at the scene.

1.1.4.4. Injection or Continuous Flow Types

Another type of carburetion used primarily on light horizontal opposed engines without internal superchargers is used by the Continental fuel injection system, the Simmonds continuous flow system, and others using the same basic principle. These can be checked for fuel at the fuel control and inlet screens; for leaks in the plumbing; integrity of the diaphragm in the fuel manifold; and nozzle condition.

1.1.4.5. Carburetor Ice Probability

Carburetor ice is always more or less of a problem for the investigator. The accompanying "Icing Probability Curves" (Fig. C IV-4 and 5) prepared by the NTSB, Bureau of Aviation Safety, shows the parameters to consider in determining the probability of carburetor ice. These curves should be helpful to the investigator if icing is suspected.

1.1.4.6. Induction Systems

The induction systems should be inspected for breaks, loose packing, or any leaks that would affect the engine operation.

1.1.5. Gear Train Continuity

Gear train continuity can be checked by turning the engine over, noting whether or not the accessory drives are rotating and whether there is a binding tendency in the engine.

1.1.6. Rocker Arms and Valves

The rocker arm and valve action can be checked quite easily if the engine can be rotated. The rocker box covers are removed to allow access to the rocker arms for observation of the action of the valve while the engine is rotated.

1.1.7. Cylinders and Pistons

If the compression check indicated malfunction in the cylinder, further checks should be made. A number of things could cause a loss of compression. A valve could be broken or stuck open, piston rings may be stuck or broken, the piston may be broken or burned through, the cylinder may be grooved, or a hole may be burned through the cylinder wall or the cylinder head. There are different ways to inspect the interior of the cylinder. Direct a light through one spark plug hole while looking through the other. If an inspection light is available, it can be inserted into the cylinder through the spark plug hole to give a great deal more light. This method should be used with care, however, because too much light causing a glare inside the cylinder may create difficulty during the inspection. A better method of inspecting the inside of the cylinder is the use of a borescope. Several models are on the market, some are better for particular types of inspection than others. It may be necessary to remove a suspected cylinder for a complete inspection. This can be done at the scene if necessary, if the engine is not going to a shop for a complete teardown and inspection.

1.1.8. Exhaust Systems

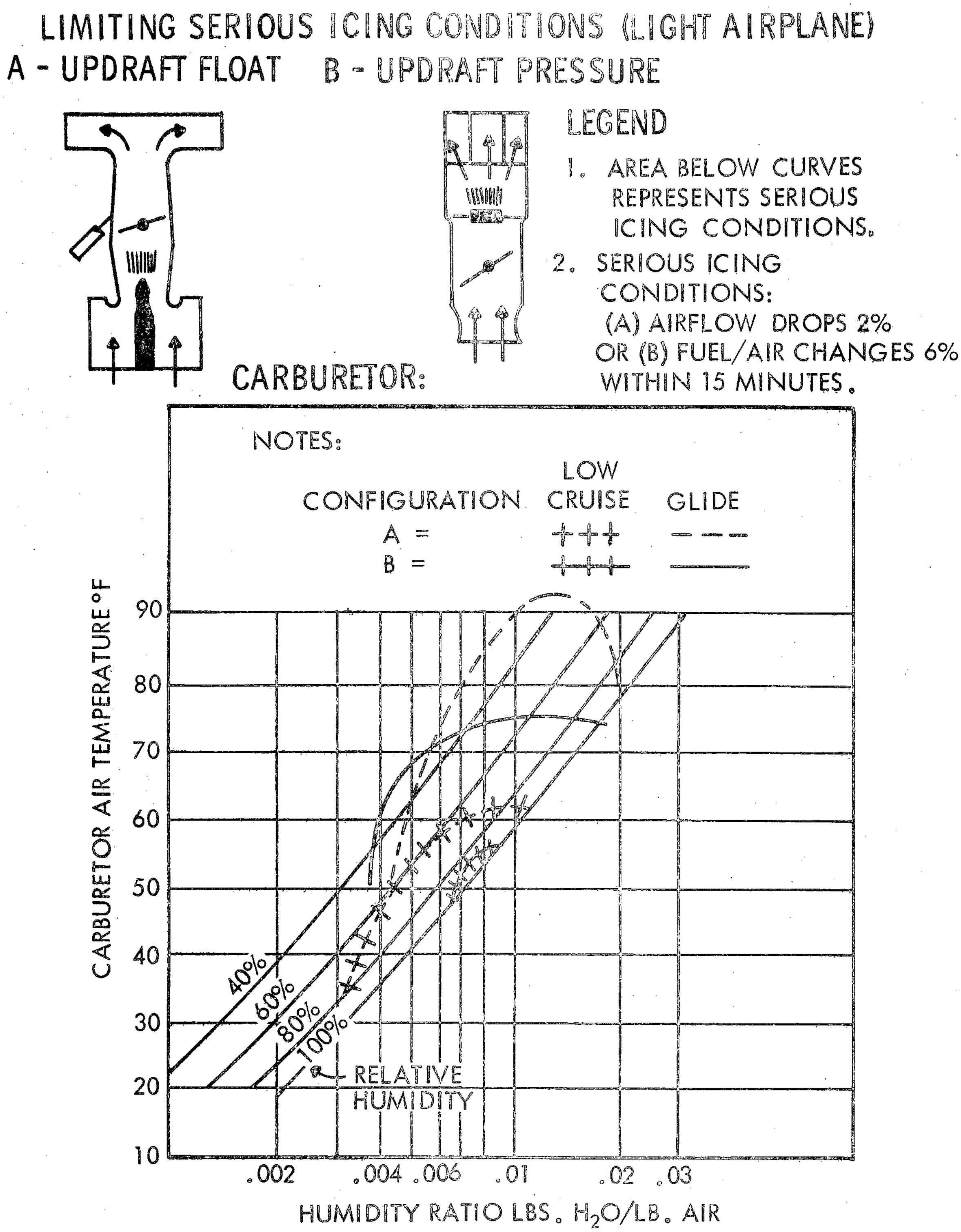
Exhaust systems should be checked for security, for any leaks that may have created a fire hazard, that may have introduced carbon monoxide into the cabin or exhaust gases into the induction system.

1.1.9. Possibility of Operating Engine

After the powerplant has been thoroughly checked and no indication of a malfunction has been found, it may be feasible and practical to run the engine. Before running the engine, insure that everything is in order, that nothing will be damaged, and that all precautions against fire are taken. All components of the engine should be as they were at the time of the accident. Cleaning or replacement of any component may invalidate the results of the runup.

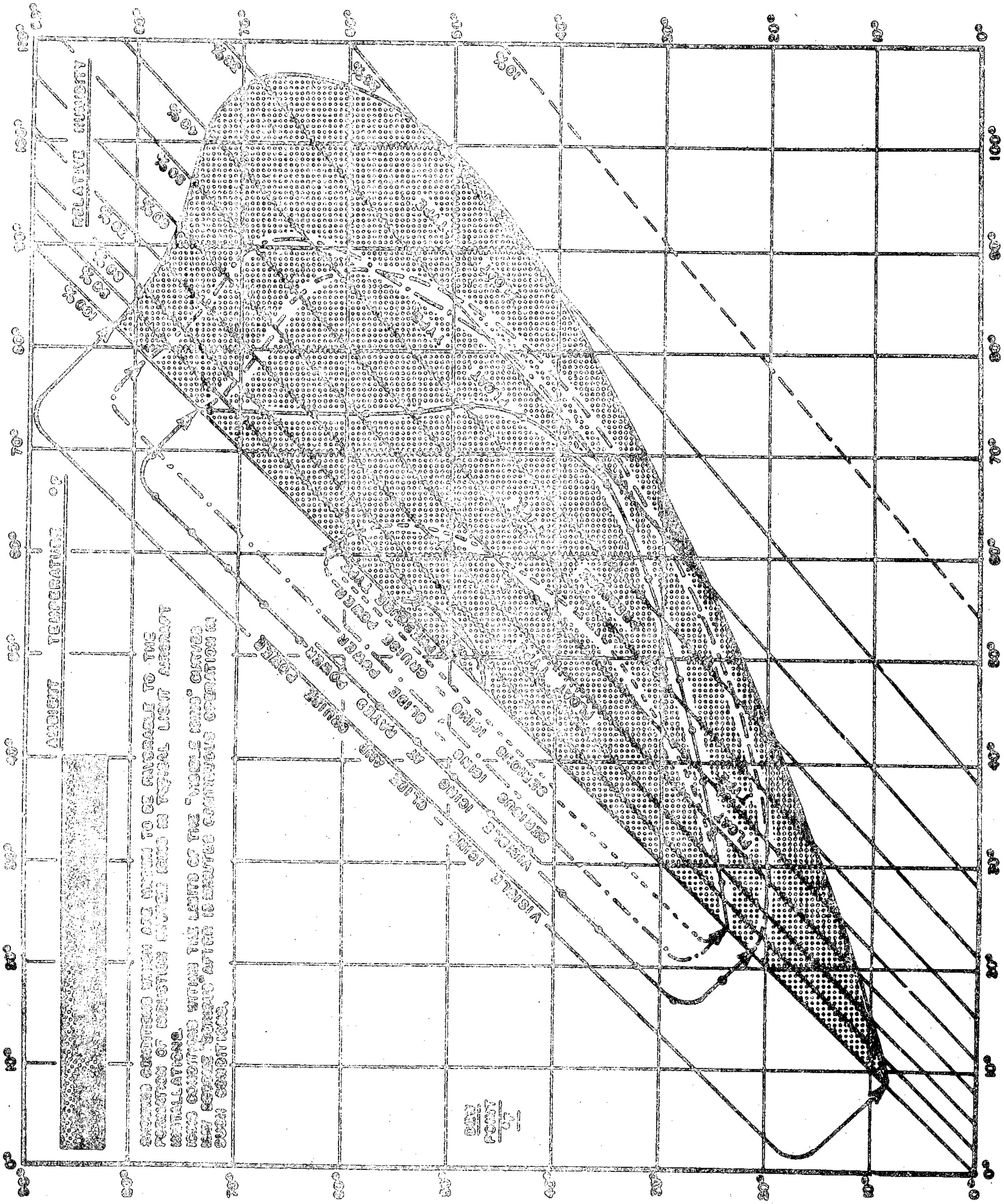
1.2. Complete Inspection of the Engine

Sometime during the course of the investigation the investigator must decide whether or not circumstances require a complete teardown inspection of the engine and/or component parts, accessories, etc.



Icing Probability Curves.

Figure C IV-4.

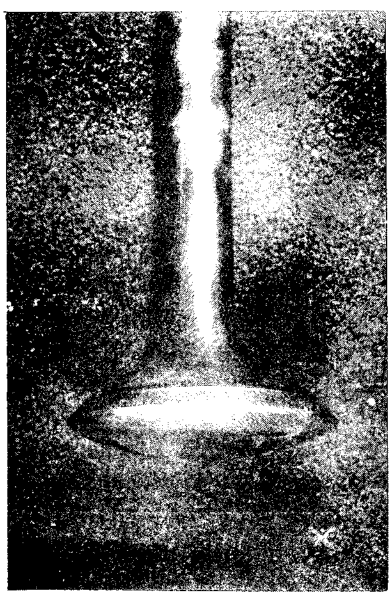


Icing Probability Curves.

FIGURE C IV-5.

Parts of a "swallowed" valve.

Failed valve.



Valve failure found during an investigation.

Magnification of the valve failure shown in figure to the left.

Figure C IV-6.

1.2.1. Facilities Available for the Inspection

If a complete teardown is necessary, the investigator will arrange for adequate facilities and the means to transport the powerplant to these facilities.

1.2.2. Requirement for Manuals and Other Technical Information

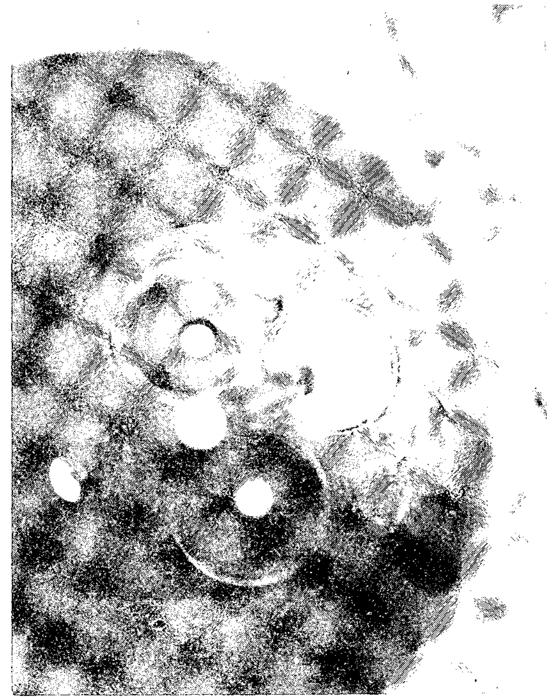
It is necessary for the investigator to procure the required manuals and other technical information and aids for the inspection. These vary with the circumstances and complexity of the investigation.

1.2.3. Inspection Procedures

Inspection procedures vary somewhat for different powerplants, therefore, the manufacturer's manuals should be used for the specific engine under investigation. The following are general procedures and items that will apply to most situations.

1.2.3.1. Cylinder Assemblies (Including Valves), Springs, Rocker Arms, Push Rods, Valve Lifters, Valve Seats, Cylinder Walls, Cooling Fins, etc.

Cylinder assemblies, including valves, valve seats, valve springs, rocker arms, push rods, valve lifters, cylinder walls, cooling fins, and flanges should be checked carefully if there is a suspected malfunction. The valves can be checked for carbon buildup on the stem that would cause sticking. Check valve faces for warping or burning, and the valve guides for excessive oversize or out-of-round. A groove pounded into the end of the guide by the valve spring retaining device is an indication of engine overspeed. If the valve has failed (swallowed valve), every effort should be made to determine the cause of the failure. See **Figure C-IV-6** for some types of valve failures. The valve seats should be checked for security, burning, and any evidence of a foreign object lodged between the valve face and seat. **Figure C-IV-7** shows the screw lodged between the exhaust valve and the seat, which resulted in a fatal accident.



Screw lodged between exhaust valve and seat.

Figure C IV-7.

The valve springs are in multiples of two or three, and if they are weak or one is broken, the resulting valve chatter could be a factor in an accident. Broken rocker arms will probably be found at the accident scene, but if the engine is moved intact to a teardown facility, determine the cause of any broken arms.

Push rods may cause a malfunction if bent or of improper length. One type engine has mixed-length push rods and upon occasion the wrong push rod has been installed.

Broken or "flat" valve lifters can cause an engine malfunction.

Check cylinder walls, heads, and cooling fins for signs of overheating; also check the walls for scoring, gouging, or out of limits. It may be necessary, because of a cylinder failure, to check the flange for warpage. In some cases, it may be advisable to investigate the history of the cylinder for previous abnormal operation possibly bearing on the failure.

1.2.3.2. Piston Head, Rings, Lands and Grooves, Pins, and Pin Boss

The piston may show signs of preignition and/or detonation. Burned pistons are indications of sustained high temperatures and overheating by preignition. The spark plugs from a cylinder which has experienced preignition may have copper runout from the center electrode. Detonation creates extremely high pressures and temperatures of short duration which show up as impact-type damage. The piston usually has a pockmarked appearance when the cylinder has been detonating. Prolonged detonation can destroy the engine.

The most common difficulty found in piston rings is sticking rings, where carbon building up between the ring and the groove prevents the ring from conforming to the contour of cylinder walls. This allows "blow-by" and excessive oil consumption. Another quite common occurrence is a broken ring (**Fig. C IV-8**), usually the top ring. If the engine continues to operate, the broken ring will break the top land, allowing the portion of ring and piston to damage the cylinder head, spark plug, and top of the piston. This is also one cause of scored cylinder walls.

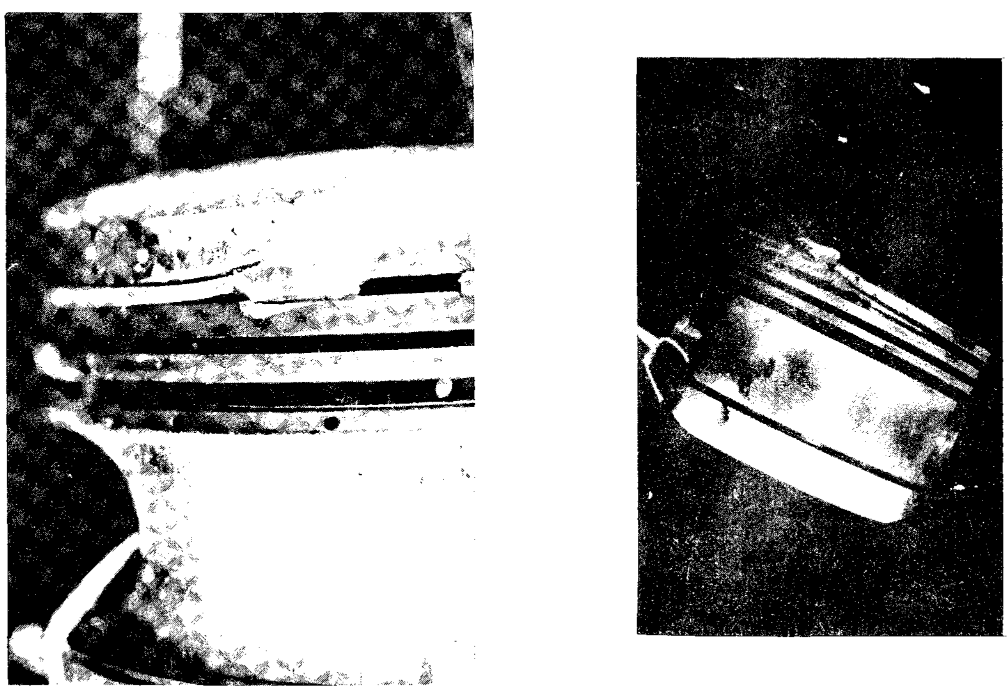
Occasionally, a piston pin will work loose and score the cylinder walls. This can happen when the aluminum plug wears out or a retainer clip works loose.

1.2.3.3. Connecting Rods and Bushings

The connecting rod and link rods are checked for damaged or excessively worn bushings and bearings. If a cylinder or piston failure occurs, the rod may be bent. Fatigue failures have occurred in rod caps and bolts. The investigator should ascertain whether the bolt broke or the cap and bolt were installed properly.

1.2.3.4. Crankshaft Alignment, Fillets, Tolerances, and Dampers

A bent crankshaft can cause engine failure. The investigator needs to determine whether



Pistons damaged by broken rings.

Figure C IV-8.

a bent crankshaft was the cause of the engine failure or was the result of the accident. Crankshaft journal and pin tolerances may furnish a clue to excessive wear in the bearings. Broken crankshafts usually result in extensive internal damage to the engine. Crankshaft fatigue failure usually starts in a fillet and may be due to a concentration of stresses caused by any number of reasons. Some failures have been attributed to improperly ground shafts or failure to nitride shafts requiring this process after grinding. The condition of the dynamic dampers is another clue to what was happening to the engine. The proper operation of the dampers is critical, and any change in their operation may set up critical torsional vibrations in the engine. Worn or damaged bushings or pins are the most common cause of this type of malfunction.

1.2.3.5. Camshaft or Camdrum Lobes and Cam Followers

A number of engine malfunctions have occurred because of cam lobe wear. In some instances, the engine was operated until the lobe was worn off completely. This situation caused the affected cylinder(s) to function erratically and eventually cease.

The cam followers sometimes wear severely, or at times become sluggish in their operation, and cause engine malfunction.

1.2.3.6. Accessory Drives and Bushings

Accessory drives have failed and caused engine stoppage. A number of different arrangements for accessory drives exist, from the simple direct drives in the small engines to the spring-loaded drives and clutches on large engines. The investigator is responsible for determining the type and arrangement of the accessory drives and bushings for the engine under investigation.

1.2.3.7. Crankcase General Condition, Bearings, and Bosses

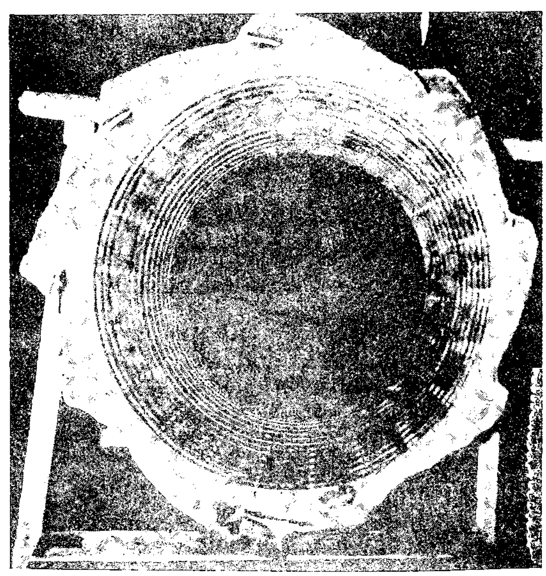
The crankcase very seldom is the cause, however, the condition of the case may provide a clue to the source of an engine failure. If conditions warrant, the bearing installation and oil passages should be checked.

1.2.3.8. Supercharger, Seals, Impact Marks, etc.

The supercharger section can reveal much information at times, i.e., during impact the impeller may be forced against the case, leaving marks that will aid in determining whether or not the impeller was rotating at the time of impact. An indication of no rotation is shown in **Fig. C IV-9.** Supercharger section seals have failed, allowing oil to enter the blower section and cause engine malfunction or failure.

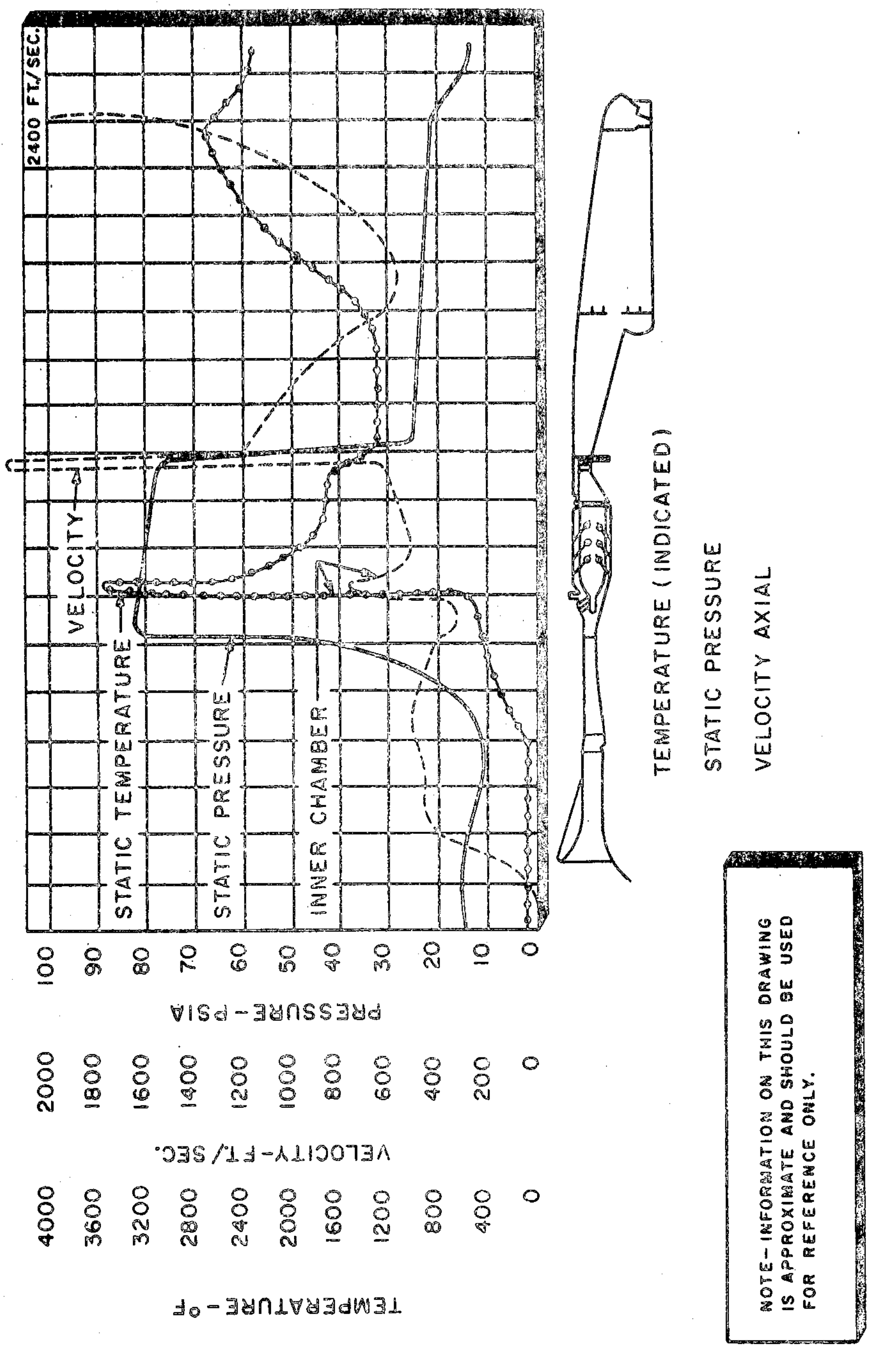
1.2.3.9. Turbosuperchargers

Cases of loss of power due to erosion of the backplates by exhaust gas have occurred.



Section of supercharger case showing impeller marks.

Figure C IV-9.



TEMPERATURE, PRESSURE, VELOCITY DIAGRAM MAXIMUM RATING

Figure C IV-10.

1.2.3.10. Engine Indications of Power at Impact

Circumstances surrounding the accident will sometimes dictate the extent of the investigation. The above points are general and not mandatory for every investigation. It is much better to be too thorough than to overlook an item that may be a causal factor.

The engine teardown many times can furnish evidence of power at the time of impact.

A number of things can furnish clues to the observant investigator. Among these are marks left by rotating parts.

The actual condition of the engine can also furnish clues, i.e., the engine seized due to lack of lubrication and/or overheating; broken crankshaft, connecting rod, piston or other components, accompanied by peening, hammering, or general mutilation of other parts of the engine. The investigator must determine whether engine damage was caused by impact forces or by engine failure in flight.

2. Turbine Engines

In order to be able to evaluate the gas *turbine* or *turbojet engine* in an accident the investigator must understand the theory of operation of the engine and its systems. He must know the operating procedures and limitations for the particular type of engine involved. This is an introduction to these areas and to investigative techniques which have been helpful in determining if an engine failure occurred. Examination of the engine and its systems may reveal not onl' the damage but the sequence of events, or more important, the *cause* of the damage.

2.1. Theory of Operation

The turbojet engine consists of four main sections: a compressor, a burner section, a turbine, and a jet nozzle. These four sections combine to accelerate the mass of air passing through the engine. This acceleration results in a propulsive force which obeys Newton's third law, which states that for every action there is an equal and opposite reaction.

Newton's second law states that a change in motion is proportional to the force applied. Expressed as an equation, force equals mass times acceleration ($F = Ma$). In terms of the turbojet engines, consider force as the net thrust and acceleration as the rate of change of velocity, then substitute velocity change in the equation in place of acceleration, net thrust is equal to the mass of the gases times its change of velocity through the engine.

The equivalent thrust horsepower of a jet engine can be approximated by determining the amount of work the engine is doing. Horsepower is defined as a force applied over a distance in a given time. Equivalent thrust horsepower depends on the thrust developed times the distance the engine moves in a g.. n time.

One horsepower is defined as 33,000 ft.lb./min. When ft./min. is converted to miles/hour, one horsepower becomes 375 mile lb./hour. From this is derived that equivalent thrust horsepower is equal to the net thrust times the aircraft velocity (in mph) divided by 375.

The change in velocity of the air passing through the turbojet engine is accomplished by taking in air, compressing and heating it, then exhausting it to the atmosphere through a turbine which drives the compressor. Compression ratios as high as possible are desirable as this will result in the highest velocity when the air is exhausted to the atmosphere. The efficiency of the compressor limits the pressure available. It is also desirable to heat the air as much as possible to allow for the maximum expansion and pressure rise which will be converted to velocity as the air passes out the exhaust duct and jet nozzle. Temperature, pressure, and velocity changes through a turbojet engine would look something like those shown in Fig. C IV-10.

The gas generator is the basic section of the gas turbine engine. This excludes the inlet duct and jet nozzle of the turbojet engine.

2.2. Compressors

The most common *compressors* used in the turbojet engine are the centrifugal, the axial flow, or a combination of these two. The axial flow compressor has the capability of deliver-ing a higher pressure than the centrifugal and is used in most large turbojet engines as well as in many of the smaller engines.

2.2.1. Centrifugal Compressor

The *centrifugal compressor* consists of three main parts, the impeller, the diffuser, and the compressor manifold. The compressor takes in air near the hub of the impeller which is rotating at high speed. The air is guided toward the outer edge of the impeller. The rotation of the impeller imparts a high velocity to the air which then flows through the diffuser which converts the high velocity kinetic energy into low velocity high pressure energy which is suitable for use in the combustion chambers. The design of the diffuser is such that it straightens the airflow and delivers it into the manifold with the least amount of energy loss. The manifold acts as the collector ring and delivers the air to the burners.

The compressor may be a single-face or single-entry or a double-face double-entry design. The double-face impeller can handle the same airflow with a smaller diameter than the single-face. The double-face impeller requires a plenum chamber where the air is collected and fed to the rear face of the impeller.

Multiple stage centrifugal compressors are sometimes used where the air from the first stage is fed to the subsequent stages to increase the final output pressure of the compressor.

The efficiency of the centrifugal compressor may run as high as 75% to 80% up to a compression ratio of 4 to 1. Above this ratio the efficiency drops off. For this reason the centrifugal compressor is limited in use to a compression ratio of 5 or 6 to 1.

2.2.2. Axial Compressors

The *axial flow compressor* is one in which the flow of air and compression takes place parallel to the rotational axis of the compressor. The axial flow compressor has capabilities of producing higher pressures than the centrifugal compressor and does not have the problem of changing airflow direction.

The axial flow compressor is made up of a series of rotating "rotor" blades and stationary "stator" vanes which are concentric with the axis of rotation. As the flow proceeds through the compressor, the cross section area for the mass decreases in proportion to the decreased volume as compression takes place from stage to stage. This may be accomplished by decreasing the cross section of the compressor and case or by maintaining the cross section of the compressor case and enlarging the compressor rotor.

After entering the air inlet duct the air passes through the inlet guide vanes and is directed onto the first stage of the compressor. The air is flowing in an axial direction as it enters the rotating compressor blades. It is deflected in the direction of rotation by these blades. When the air leaves the compressor blades it is slowed by the stator vanes and a pressure rise takes place. The succeeding stages of the compressor each result in an increase in the air pressure. The increases in air velocity through the compressor stages are virtually eliminated by passage through the stator stages. This results in the velocity of the air being approximately the same as it leaves the compressor as when it entered.

The axial flow compressor is capable of compression ratios of 16 to 1 or higher, therefore, it is used in most high performance jet engines.

Compressor blades are designed as airfoil sections and have a range of rpm where their efficiency is the greatest. Since jet engines are not operated at constant rpm settings, the shape of the airfoil is a compromise to allow operation under varying conditions. At some rpms the front stages of the compressor are capable of pumping more airflow than the rear stages are able to handle. There are two principal ways of taking care of these conditions. One is to have variable inlet guide vanes and stators in the first several stages of the compressor. This arrangement changes the angle at which the air is introduced to the front stages of the compressor and limits the mass of air which these stages can process.

Another method for handling the conditions which are encountered at part throttle operation and during starting is to split the compressor into two independent rotors. Each part of the compressor has its own turbine and operates at its own best speed. The front or low pressure compressor is normally not governed, but is allowed to operate at the speed necessary to furnish the required airflow to the rear or high pressure compressor. The size of the high compressor rotor and the increased air temperature of the mass it processes allow this compressor to operate at higher rpm without the problem of the blade tips reaching sonic speed.

Split compressors also have some advantages during starting. It is necessary to have a starter only large enough to turn the high pressure compressor and accessories.

2.2.3. Compressor Stalls

Compressor stalls are a characteristic of axial flow gas turbine compressors. Compressor stalls may result from compressor blade damage, however, it is more common for the stall to occur when the pressure ratio through the rear stages is insufficient and these stages become choked. As the airflow slows without a corresponding reduction in rotor rpm the blades stall. Compressor stalls are undesirable as they result in a loss of thrust.

In order to reduce the stall potential of the compressor, air is bled off the center of the compressor to unload it under those conditions where the likelihood of a stall is the greatest. Air bleed ports are located between the compressor sections of dual compressor engines. These bleeds are automatic, usually controlled by engine rpm to open at start and during low thrust operation. When the bleeds are open they allow for greater airflow through the front stages of the compressor and reduce the pressure on the rear stages.

2.3. Diffusers

After leaving the compressor, the air passes through a *diffuser* which reduces the velocity of the air in preparation for entry into the burner at a low velocity so that proper burning will take place without blowout of the flame. Diffusion takes place through the compressor exit guide vanes which are located aft of the rearmost compressor stage in the diffuser section of the engine where the air passages are increased to convert the kinetic energy into static energy.

2.4. Burner Section

The *burner section* provides for the addition of heat to accelerate the mass flow through the engine to provide the required thrust and the power to drive the turbine. The temperature of the gases leaving the burner section and entering the turbine must not exceed the allowable turbine inlet temperature. For this reason, the burners are designed to contain all of the burning within the burners. The combustion chamber which is contained in the burner section may consist of individual cans, annular or can annular chambers. No matter which design is used, only 25% to 30% of the total volume of air entering the chamber is used for the combustion process. The remainder of the air is used for cooling the burner surfaces and to mix with the burned gases to cool them before they enter the turbine section. Anything which causes a reduction in mass flow without a fuel reduction, i.e., a compressor stall, will result in an overtemperature condition as the cooling air is lost.

2.5. Turbine Section

The turbine extracts kinetic energy from the mass flow out of the burner section. This energy is converted into shaft horsepower to drive the engine compressor and accessories. Approximately 70% of the energy available is used to drive the compressor with the remaining available for thrust.

As the gases leave the burner section they pass through the turbine nozzle. This is a section of stationary vanes which discharge the flow of gases onto the turbine wheel at the proper angle and velocity. There is a turbine nozzle for each turbine wheel. The turbine nozzle area is a critical area of turbine design. An increase in area will cause the turbine to lose efficiency, while a decrease in area results in increased back pressure and higher temperatures from the burner section. There may be an exit nozzle aft of the rear turbine wheel to straighten the gas flow as it enters the exhaust duct and jet nozzle.

2.6. Exhaust Duct and Jet Nozzle

The *exhaust duct* connects to the engine aft of the turbine section, and is used to direct the gases to the jet nozzle. The duct also is designed to increase the velocity of the mass as it moves toward the exhaust nozzle. The rear opening of the exhaust duct is the exhaust nozzle. The size of the orifice determines the density and velocity of the gases as the exhaust leaves the engine. The size of the orifice is determined by engine design and is quite critical. Anything which causes the area to be altered will affect engine performance and exhaust gas temperature.

Various mechanical devices such as noise suppressors or thrust reversers may be attached to the exhaust nozzle as conditions require.

2.7. Post-Accident Engine Examination

The purpose of the *post-accident engine examination* is to determine if the engine was capable of producing thrust, and if so, if it was producing sufficient thrust to sustain flight at the time of the accident.

The estimation of thrust production by the jet engine is based upon the rotational speed of the engine at the time of the accident. The thrust of the typical jet engine does not go up directly with engine rpm, i.e., it is not uncommon for a jet engine to produce approximately 50% thrust at 80% to 85% rpm. For this reason, it will be necessary to differentiate between low, mid-range, and high rpm if an accurate assessment of thrust is to be made.

When examining the jet engine for evidence of rotational damage, it is necessary to view such damage with the extent of impact damage in mind. An engine which has no impact damage or internal failure will not exhibit rotational damage even though the engine may have been operating at full power at the time of the accident.

The amount of energy which is stored in the rotating parts of the engine is dependent upon the mass of the rotating parts, the force being exerted on the turbine, and the rotational speed of the engine. This energy will have to be absorbed to bring the rotating parts to a halt. The amount of energy absorbed will be evidenced by the rotational damage. It is well to remember that the energy necessary to stop the rotating parts increases as the square of the rotational velocity.

The components of the various engine systems may furnish evidence to support a thrust estimation made from the rotational damage. Some items to be checked are positions of the fuel control, fuel flow transmitter vane, throttle, fuel shutoff valves, variable inlet guide vanes or stators, and bleed valves. For engines equipped with afterburners the position of the afterburner fuel control and the variable exhaust nozzle should be checked. Knowledge of the particular engine systems is vital in establishing the power at impact. Current engine manuals are necessary for proper understanding and evaluation of the evidence found.

Some of the conditions which have been found to be of value in estimation of engine speed are:

1. For low speed/thrust conditions –
   a. Rotor, stator, and turbine blades bent in a random fashion. Blade corners relatively square and undamaged.
   b. Limited passage of debris through the compressor section.
   c. Localized and limited evidence of interference between the rotating and non-rotating parts. Where there has been interference, the marks will be rough with possible impressions of the blades on the case.
   d. Gouge marks will usually be restricted to small areas and may be quite deep.
   e. The overall damage indications in the engine will be non-uniform in nature.
2. For high speed/thrust conditions –
   a. The general damage to the rotating

parts will be more severe and uniform throughout the engine. There will be extensive evidence of contact between the rotating and non-rotating parts.

b. Blades and stators will exhibit physical damage with corners rounded and the blades fragmented if the engine case does not rupture. The blades will be bent opposite to the direction of rotation and may be broken off or torn out of the rotor. If the case fails, the blades and stators will not usually have as extensive physical damage.

c. Debris will be found throughout the engine. Metal from the compressor rotor or case may be plated onto the forward surfaces of the hot section parts of the engine.

d. Turbine buckets will be bent opposite to the direction of rotation. Evidence of rubbing on the turbine wheel, turbine nozzle, or exhaust cone will be highly polished with bluing from frictional heating.

e. The turbine shaft may be sheared. The engine manufacturer will be able to furnish the strength of the shaft to resist shearing. Generally, a sheared turbine shaft is an indication of rpm in excess of 75% when the impact angle is steep, 45° or greater.

The amount of engine breakup from impact is not a direct indication of rpm at the time. It should be remembered when the wreckage is spread over a long path that the engine speed may change between the time of initial impact and final impact. This will be especially true in the case of a mild first impact followed by a subsequent more violent impact. In such a case, signs of both high and low rpm are common. The entire engine should be integrated into the speed estimate without undue reliance on a single indication.

The turbojet can produce thrust only when there is burning of fuel in the burner section, therefore, the engine examination should include an assessment of the temperatures in the hot section of the engine. The hot parts of the engine cool rapidly when the engine is shut down. For this reason, the deposits found in the hot section can be a very good indication of operating conditions at the time they were deposited. A smooth, even coating of metals from the compressor section are indicative of engine operation at or above the idle range. Any metal striking the hot section 8 to 10 seconds after the fuel flow has stopped, or the engine flamed out, will be rough in apppearance and will not adhere tightly to the hot section components. Metal deposits located on the exhaust gas temperature thermocouples or in the exhaust duct are indicative that the engine was operating in a temperature range above the melting point of the material deposited. The pattern of deposits on a variable nozzle is useful in determining the nozzle position at the time of engine failure. Compressor stalls will usually result in a reduced amount of metal being fused to the hot section parts even though burner operation is continuing. This is due to the reduction in mass flow to transport the metallic debris.

Other materials such as grass, wood, or dirt may also give an indication of the temperature at the time they were deposited in the hot section.

2.7.1. Engine Failure Versus Impact-Caused Failure

Engine failure may be the cause of the accident as well as the result of the accident. It will be necessary to separate pre-impact failure or damage from impact damage during engine examination.

Some of the more common types of engine failure are compressor failure, compressor stall damage, turbine failure, turbine disc failure, and bearing failure.

Compressor failures may involve the compressor disc or the individual compressor blades. These failures may be said to result from foreign object damage as any part becomes a foreign object when it is displaced from its proper place in the engine. Foreign objects may be introduced from outside as well, coming in through the air inlet or even parts from the air inlet. It will be necessary to determine if the compressor failure is cause or effect. A bearing failure which allows the compressor rotor to shift will result in compressor failure. A compressor failure may start at any point and progress forward, aft, or both. For this reason, a thorough examination is necessary to determine the initial failure point. In flight, compressor failure will usually result in more damage to the compressor blades and stators than will result if the failure is preceded by ground impact.

Foreign objects entering the engine through the air inlet are frequently knocked or thrown forward as they encounter the first stage of the compressor. An examination of the rear of the inlet guide vanes may reveal an identifying mark left by the foreign object. It should be remembered that objects do not have to be large or hard to cause compressor damage. Engine failures have occurred when rags, paper or plastic sheeting have been ingested.

Compressor stalls do not often result in compressor damage. The damage will be found in the hot section of the engine if a stall has lasted for any length of time. The first air loss during a stall is of the cooling air while combustion air continues to be furnished. This results in rapidly rising temperature throughout the hot section. This type of overtemperature results in holes being burned in the exhaust nozzle guide vanes and burning of the tips of the turbine buckets. If the stall persists, burning of flame cans and exhaust gas temperature thermocouple loops will result.

Overtemperatures without compressor stall are to be considered also. One of the main inputs to the fuel control is rpm of the compressor rotor. Any condition which slows the rotor or reduces the mass flow through the engine without a throttle lever reduction will result in the fuel control adding more fuel in an attempt to regain the lost mass flow.

The most common turbine failures involve the turbine buckets. The buckets may fail from overtemperature, excessive rpm, or from failure of the serrations at the base of the bucket. The turbine bucket failure resulting from a non-stall overtemperature will occur at the midspan of the bucket after stretching or necking-down of the bucket has taken place. Complete and immediate engine failures do not usually occur as the result of turbine bucket failures. The loss of one or more turbine buckets will result in an increase in temperature as the fuel control adds more fuel to compensate for the loss of efficiency of the turbine.

Turbine disc failures result in a portion of the disc breaking out as the result of thermal shock or overtemperature of the turbine disc. A turbine disc failure usually results in bearing failure in a very short time, due to the massive unbalance which is created. Catastrophic engine failure results almost immediately. Many times failure of the engine mount will occur following a turbine disc failure.

Bearing failures which occur in operation are different in appearance from those which are caused by impact. Impact-damaged bearings will not show signs of overheating, and the damage will be localized. Those which fail in operation will be discolored from the heat; they will be distorted with galling and scraping of the balls or rollers. Most bearing failures are the result of loss of oil which results in severe overheating and ultimate failure. The thrust bearings (ball) are the most susceptible to failure as they are loaded fore and aft, as well as radially.

The question may arise: Which failed first, the bearing or the oil pump? This can usually be resolved by an examination of the lubrication system. If there is contamination throughout the system the oil was probably circulating, therefore, the pump was operating and the bearing failed first, causing the oil pump to fail from metallic contamination. If the system does not have extensive contamination, the pump or pump drive probably failed first.

Fuel and oil samples (as much as possible up to two gallons) should be collected for analysis to determine if the fuel and oil are of the proper grade. Contamination can also be detected if present. Spectrographic analysis of the oil can identify the type of metal contamination, which is an aid in identifying the source of the metal.

3. Propellers — Need of On-Scene Investigation and General Discussion of Various Props and Information That May Be Obtained

Propeller examination is one of the most important phases of the powerplant investigation. In many instances, careful examination of the propeller can furnish valuable clues to the condition of the engine at impact. The propellers should be examined and documented as soon as practicable because excess handling or moving of the propellers can cause movement of dome positions or additional damage that may destroy very important evidence.

There are a few general indications pertaining to the documentation of all propellers. The blade angle should be established if possible, the nature and location of breaks, bends, scratches, and other marks, and the general condition of the propeller should be documented.

3.1. Hamilton Standard Propellers

The Hamilton Standard propeller can furnish the investigator with much information in most cases. The rpm can usually be established on propellers having Woodward electric heads by measuring the distance from the parting surface of the head to the rack position. If the governor head is badly damaged, the rack position may be duplicated on another head from which measurement can be taken to establish the rpm setting. **Figures C IV-11 through 13** show the governor and **Fig. C IV-14** is a chart for determining rpm setting. Because of the gear train arrangement, movement of the rack within the head is very unlikely, even if the head is completely separated from the aircraft.

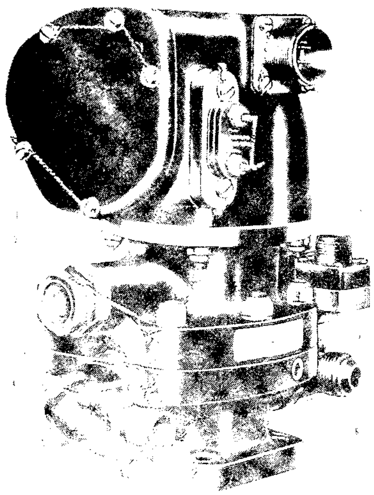
Cable controlled governor position is usually meaningless unless the system is intact and not moved by impact forces.

3.2. Dome Indications, Stop Rings, and Gear Teeth

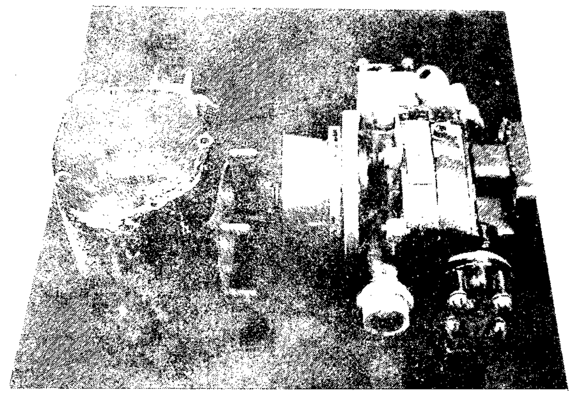
Domes should be removed at the scene of the accident to obtain all the information possible before the propeller is moved or the cam position changed. In non-reversing propellers, the upper stop ring is the high pitch stop and the lower is the low pitch stop. Arrows on the stop rings point to the high and low pitch settings respectively. If the movable stops are against either the high or low pitch stops, the propeller is either feathered or at full low pitch. If the movable stops are somewhere between the fixed stops, the angle can be determined by removing the high pitch stop and rotating it clockwise until it is against the rotating cam stop, replacing it on the serration, and reading the angle indicated by the arrow. The stop ring can be removed by prying gently around the ring with a screw driver. (**Fig. C IV-15.**)

On the reversing (without feather lock) propeller the reverse ring is the lower ring and the feather ring is the upper. On the propeller with feather lock, the feather ring is in lower and the reverse ring in upper position.

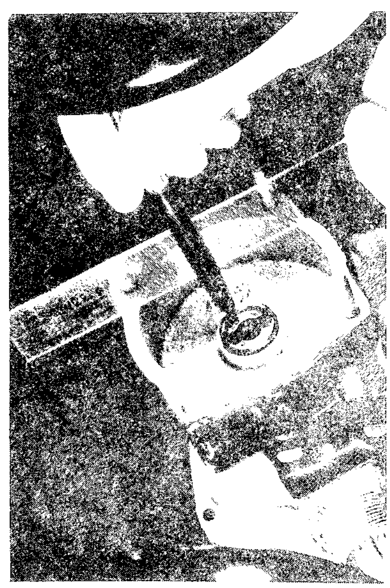
Regardless of which type dome is to be read, the procedure is basically the same; remove the upper ring and reposition it to its fixed stop. Count the number of ring teeth that the ring was moved. If the ring was moved to the feather stop, subtract the number of ring teeth from the feather angle; this will be the position of the dome in relation to blade angle in degrees. If the ring was positioned to the



Stepmotor Governor Assembly.
Figure C IV-11.

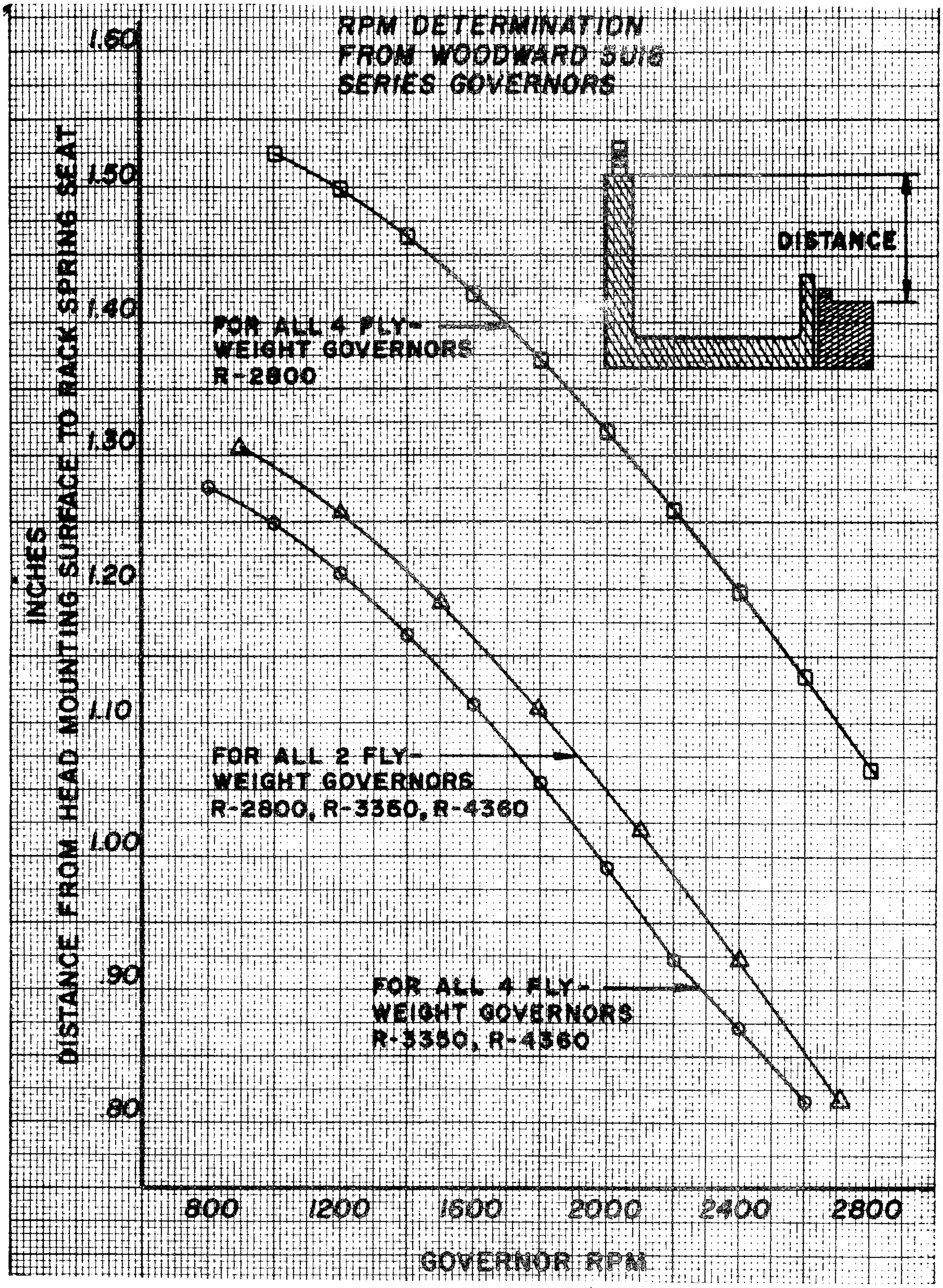


Stepmotor removed from governor.
Figure C IV-12.



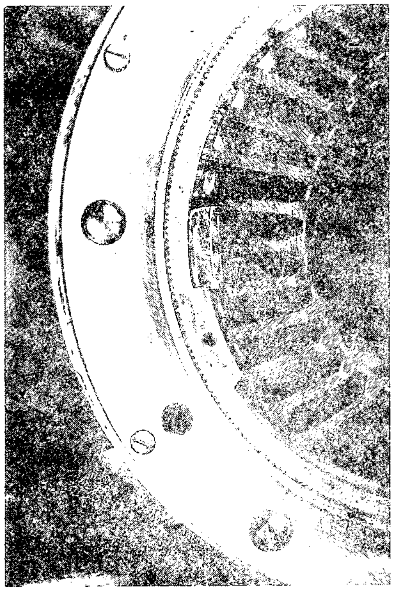
Location of measuring points. Parting surface to rack in stepmotor head.

Figure C IV-13.



RPM Determination from Woodward 5U18 Series Governors.

Figure C IV-14.



Non-reversing propeller stop ring set to determine blade angle. Arrow points to 50°

Figure C IV-15.

Reversing propeller stop ring set to determine blade angle. Index (25°) directly opposite tooth marked 35°

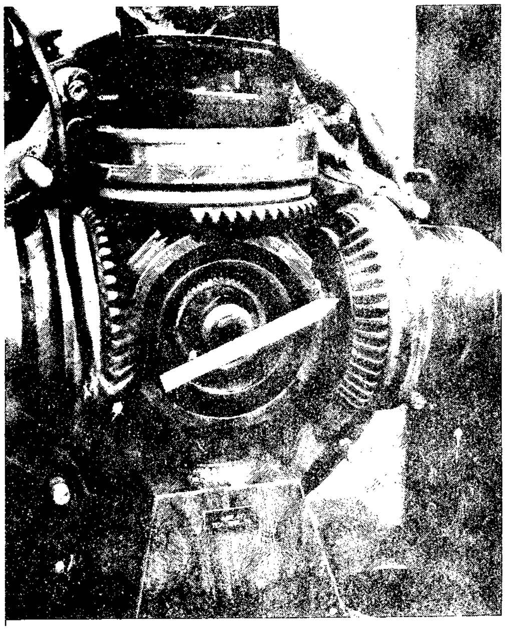
Figure C IV-16.

reverse stop, subtract the number of teeth from the reverse angle which will result in the blade angle in degrees. (**Fig. C IV-16.**)

3.2.1. Shim Plates — Methods of Obtaining Blade Angle — Significance of Information Obtained

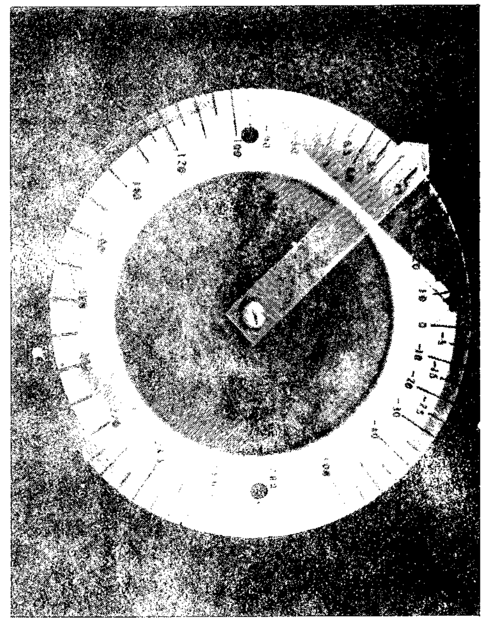
Under some conditions blade angle, determined by impact marks on the shim plates, is more significant than dome positions. This indication can be obtained only on Hamilton Standard propellers with aluminum blades. On these propellers a shim plate which turns with the blade when pitch changes occur is positioned between the blade butt face and spider shelf. The spider surface on which this shim plate bears is so shaped that a portion of the shim plate is not supported. When the blade is subjected to impact, straight line indentations and sometimes breaks occur on the shim plate along the straight line edge of the non-supported portion. By positioning the shim plate correctly on the blade butt face, then rotating the blade about its longitudinal axis until the straight line indentations or breaks are aligned with the straight edges of the spider shelf, the blade is in the same position as when the marks were made. Locate the blade angle markings which may be on the barrel lip or segmental gear, and at the index mark read the blade angle. If more than one indentation is found on the shim plate, determine the blade angle represented by each and note the relative depth of each. This method, if carefully executed, should be accurate within plus or minus two degrees. However, markings of all blades should be checked for angle and noted. Usually the most pronounced mark represents the initial mark.

When removing the blade from the spider, care should be exercised to recover the shim plate and maintain its relative position with the blade. The check must be made with the same dowels engaged in the shim plate holes and the same shim face against the blade that existed in service; otherwise, completely erroneous readings will result. The blade angle may also be obtained by the use of a blade angle protractor. (**Figs. C IV 17 through 20.**)



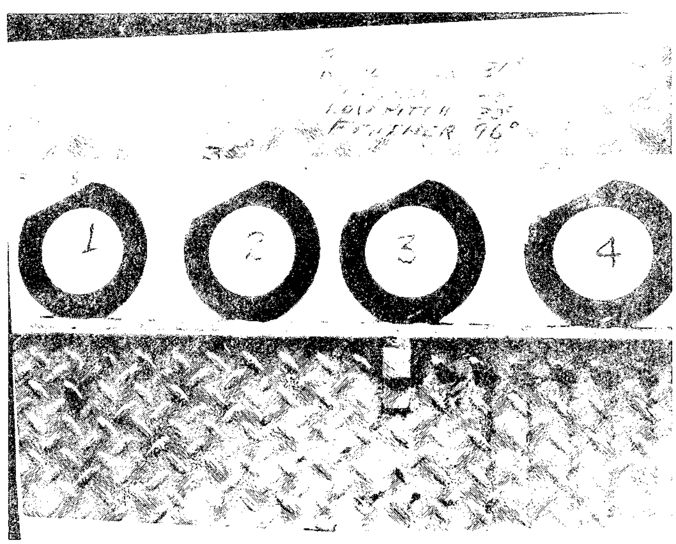
Shim plate location on Hamilton Standard propeller.

Figure C IV-17.



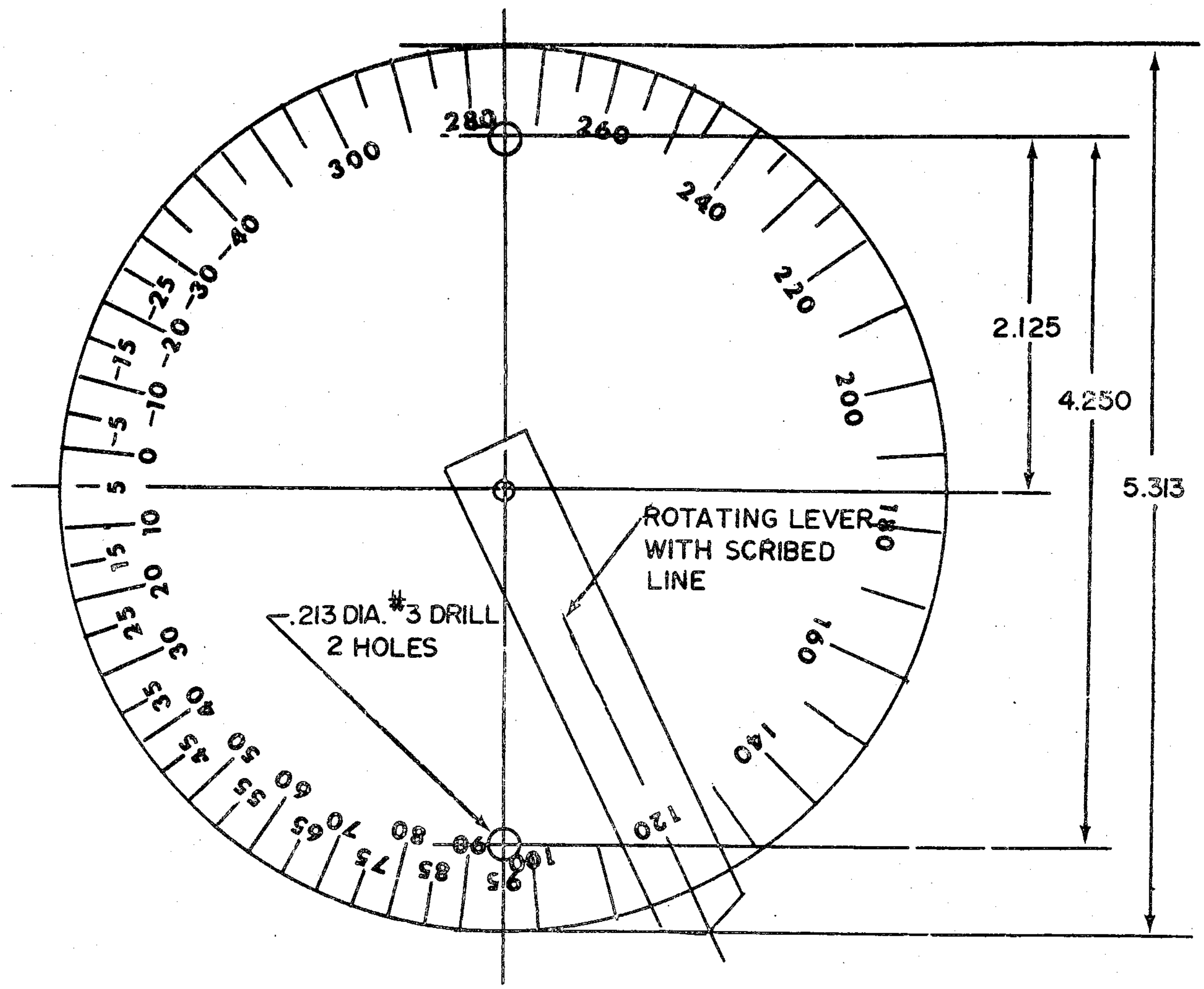
Protractor positioned on shim plate.

Figure C IV-19.



Shim plates from a damaged propeller.

Figure C IV-18.



The sketch shows a plexiglass protractor (3/32″ thick) drawn to "E" shank shim plate dimensions. It has proved extremely useful for interpreting shim plate impact markings with regard to blade angle. Protractor graduations have been scribed in relation to the centerline between dowel pin holes, taking advantage of the fact that the angle formed between this centerline and a line parallel to the chord of blade section at reference station is 5 degrees on all "E" shank blades.

The impact blade angle can be obtained by placing plexiglass protractor on the shim plate and aligning dowel pin holes. The scribed line on the lever arm is then held perpendicular to the shim plate line of impact. Blade angle is then read directly.

Protractor has been graduated a complete 360 degrees since it may also be used for interpolating number of degrees blade has rotated in bushing in the event of screw and dowel pin shearing.

Dietzen Model 1931AP, 6″ circular protractor or equivalent may be used.

Figure C IV-20.

Approximate blade angles at impact of Hamilton Standard steel blade propellers have been determined by matching brinnel marks caused by the blade retention balls on the barrel arm and blade butt races. Like indications can be determined in some instances by marks on the rotating cam and blade segmental gears. Translation of such markings into blade angle usually requires that a propeller be set up in a shop. The blades are rotated until the appropriate gear teeth are in mesh, and then the blade angle is measured.

The important thing is to obtain blade angles from as many indications as possible. Appreciable differences in indications can usually be resolved in the analysis of the accident.

3.3. Curtiss Electric Propellers

It is relatively easy to determine the blade angle on a Curtiss electric propeller, The methods for making the determination and the reliability of the findings are discussed in the following paragraphs.

3.3.1. Brake Mechanism

The brake mechanism, when properly adjusted, retains the power unit in a fixed position, and there is little possibility of its being rotated by impact forces. However, in analyzing electric propeller blade angles, it is important to remember that if current was applied to the power unit subsequent to impact, the blades would have been moved to a completely irrelevant position.

3.3.2. Limit Switches and Cams

Determination of blade angle at impact is relatively simple. The limit switches and actuating cams are located in the hub end of the speed reducer. It is difficult to remember the identity of the various limit switches, therefore, manual material should be referenced when establishing blade angle. The relative position of the cam lobes with respect to the limit switches establishes the blade angle. If the cam has opened the low pitch limit switch, the blade angle is the low pitch setting, and if the feather limit switch is open, the propeller is feathered. The intermediate angles are proportionate to the distance between the cams and limit switches. As in the case of the hydromatic propeller, blade and power gear damage, as well as barrel arm and blade markings, can assist in establishing blade angle at impact.

3.3.3. Curtiss Synchronizer

On installations using the Curtiss synchronizer, rpm can be determined from the master motor Teleflex control rack position if it has not been destroyed by impact. Master motor rpm is related directly to engine rpm.

3.4. Aeroproduct Propellers

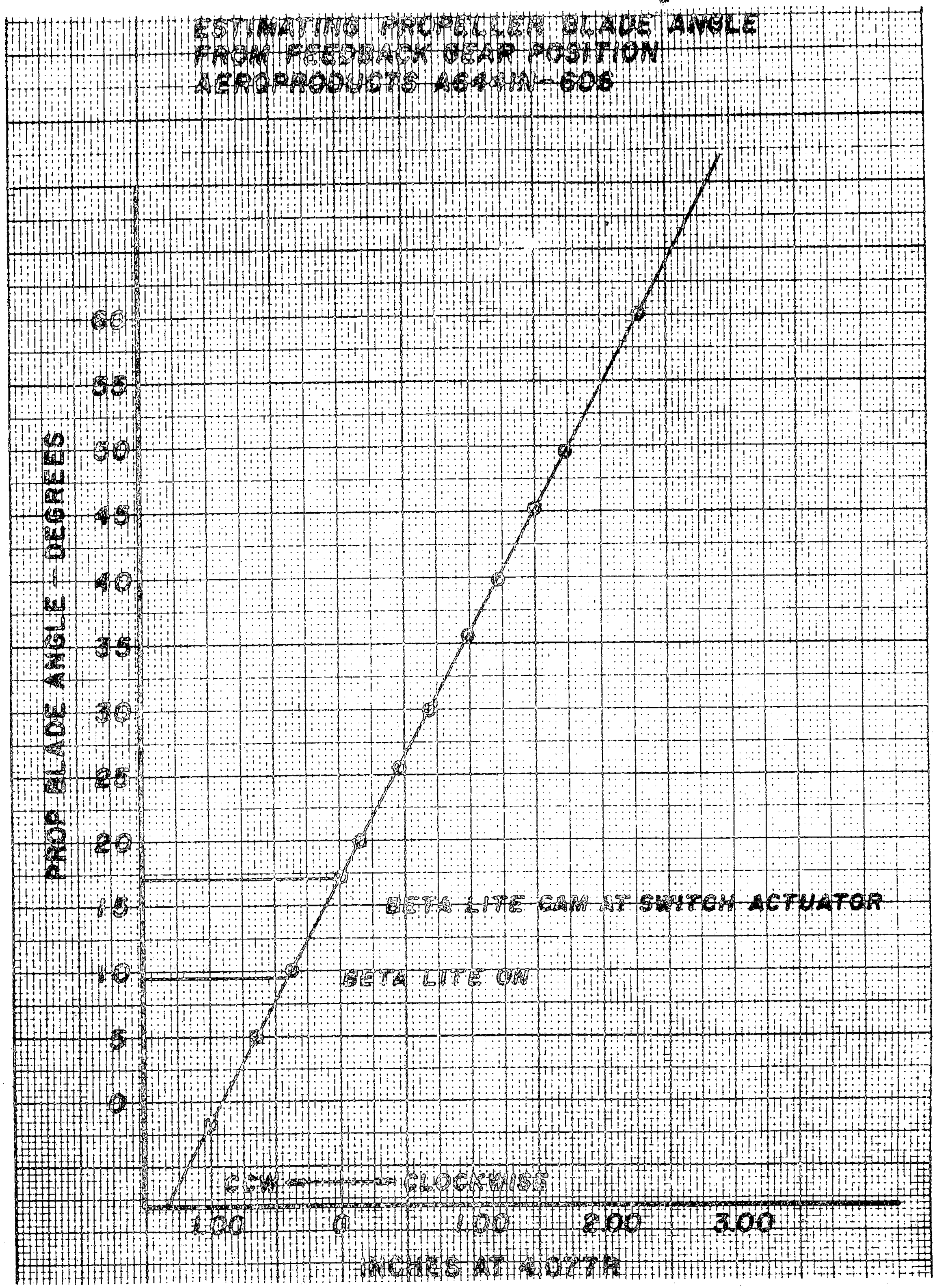
The Aeroproducts propeller presents somewhat more of a problem to the investigator than the Hamilton Standard. There is a need for manuals and a shop with the proper facilities to determine the blade angle. Every effort should be made at the scene to secure and document all the evidence available.

3.4.1. Beta Light Switch and Cam and Feedback Gear Position

One method of determining blade angle is from the feedback gear position and beta light cam and switch actuator. **Figure C IV-21** shows blade angles for positions up to 2¼″ (60°) at a radius of 4.077 inches for the A644IN-606 propeller.

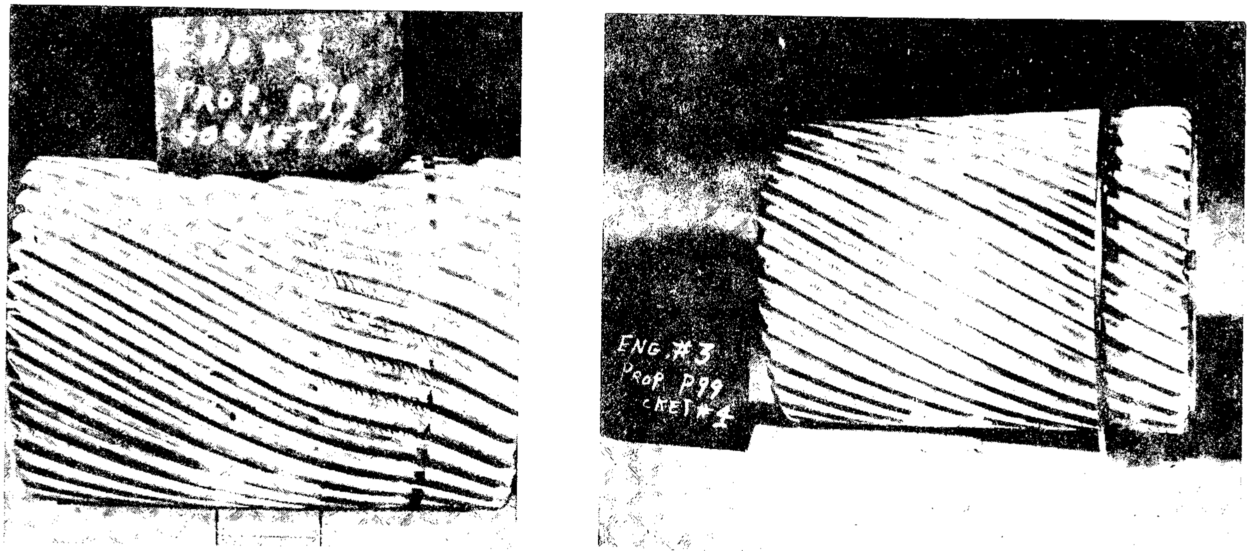
3.4.2. Spline Position at Impact

Many times the blade splines will be marked at impact, and by carefully measuring the position of these marks and duplicating this po-



Propeller Blade Angle from Feedback Gear Position.

Figure C IV-21.



Splines from an Aeroproducts propeller showing blade position at impact.

Figure C IV-22.

sition on another propeller, the blade angle can be determined within reasonable limits. (Fig. C IV-22.)

3.5. Rotol Propellers

A method of determining blade angle of a Rotol propeller in the field has been developed. With the use of a ¼-inch drill, chisel, and hammer, cut a hole, or window, lengthwise in the cylinder approximately ½ inch by 2 inches through which to view the operating piston and ground fine pitch stop. A measurement of the distance the piston is off the stop can be correlated to the blade angle (0.034 inches equals 1 degree).

3.6. Light Aircraft Propellers

Light aircraft propellers give the investigator very little information relative to the blade angle setting at the time of impact.

3.6.1. Beech Electric Propellers

The only light propeller that can furnish blade angle setting is the Beech electric propeller. The blade angle is changed by an electric motor through a gear train which allows little possibility of movement at the time of impact. If this mechanism is not destroyed, the position of the drive gear with respect to the high and low pitch stops can be duplicated on another propeller and the blade angle measured.

3.6.2. Other Light Propellers

Most of the light constant speed and feathering propellers are operated by governor oil pressure against a spring or compressed air. When oil pressure is lost at impact, the blades will be moved by the spring or air pressure, making any plade position unmeaningful. The investigator should be aware of the fact that if the propeller is in the feathered position it does not necessarily mean that the engine was shut down prior to impact. These propellers can drift into the feathered position after impact even though the engine was developing power at the time of impact. To make a determination in this case requires very careful inspection of the propeller and engine.

3.7. Significance of Blade Condition — Including Bends, Scratches, Breaks, etc.

The condition of the propeller should be carefully documented as to bends, marks, scratches, and breaks. No single item can furnish definite proof as to what was taking place at the time of impact, but many times all of the evidence available can fall into a pattern that will either indicate or eliminate a causal factor. All parts of the propeller, and particularly the blades, should be accounted for. A determination should be made as to whether or not a portion of the blade separated in flight.

3.8. Speed Determination from Slash Marks

In some cases the speed of the aircraft at impact can be determined by propeller slash marks. The distance between the first two marks should be used, because, as the propeller rotation slows down, the distance between subsequent marks will increase. The formula for determining speed is:

$$\text{MPH} = \frac{D \times A \times R \times N}{88}$$

$$\text{Knots} = \frac{D \times A \times R \times N}{101}$$

Where: D = Distance between first two slash marks

A = Number of blades

R = Propeller to engine ratio

N = RPM

Example: Distances between slash marks as measured at the accident site were 2.2′, 2.3′, 2.8′, and 3.4′. Engine rpm established to be 2600.

Propeller reduction ratio 0.50:1

Number of propeller blades 3

The formula would be:

$$\text{MPH} = \frac{2.2 \times 3 \times 0.50 \times 2600}{88} = 97.5 \text{ MPH}$$

4. Controls and Accessories

It is impossible to cover all the circumstances and conditions that may be encountered in this phase of the investigation. However, some common things can be accomplished. It is the investigator's responsibility to establish the facts pertaining to each accident. Discussed below are a few of the items important to the investigation of powerplant controls and accessories.

4.1. Cockpit Controls Settings and Their Significance

Engine cockpit controls positions should be carefully documented. On a large-scale investigation this may be done by a Group other than the Powerplants Group, therefore, close coordination must exist between the Group taking the cockpit readings (usually the Operations Group) and the powerplants investigator. Little credence should be given to the settings because they may have moved during the accident, however, they may help establish a pattern confirming other evidence found during the investigation.

4.2. Engine Controls Positions and Their Significance

The position of the controls on the engine should be documented. The same consideration should be given here as was noted for the controls in the cockpit. If control linkages have been broken, the types of breaks and condition of these linkages should be noted.

4.3. Fuel Pump

Fuel pumps should be checked for operation if there is any doubt about a malfunction. If the pump is to be run, it should not be cleaned or disassembled prior to running. After bench testing, disassembly may be in order if the pump failed to operate properly. **Figure C-IV-23** shows one type of failure. The clip which holds the sliding vane against the rotor failed, allowing the vane to move away from the rotor.

4.4. Vacuum Pump, Types and Problems

Vacuum pumps are of two types. The older, wet pump type, should be checked for proper lubrication and signs of overheating. Engine fire has been attributed to improper vacuum pump lubrication. Another type, the dry vacuum pump, will cause trouble if an oil seal is leaking, allowing oil to enter the pump. If loss of the vacuum system is suspected, the pumps and entire system, which includes the plumbing, regulators, check valves, fittings, and relief valves, should be thoroughly investigated.

4.5. Generator

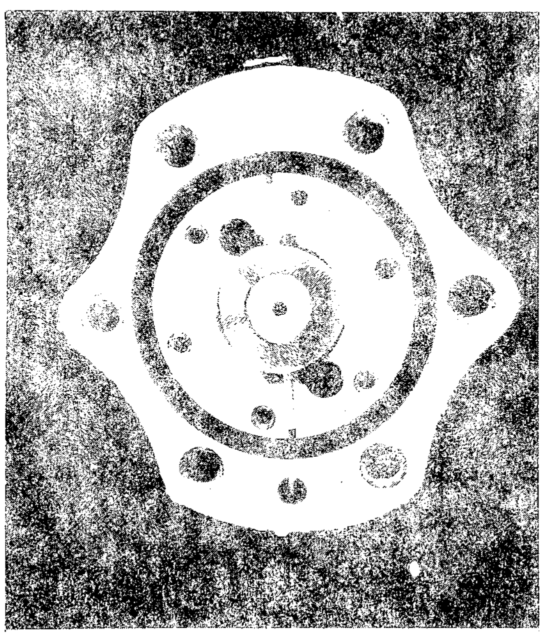
The generator is part of the electrical system, however, since it is installed on the engine the Powerplants Group may investigate its condition. Detailed investigation is discussed in chapter V, section 3.

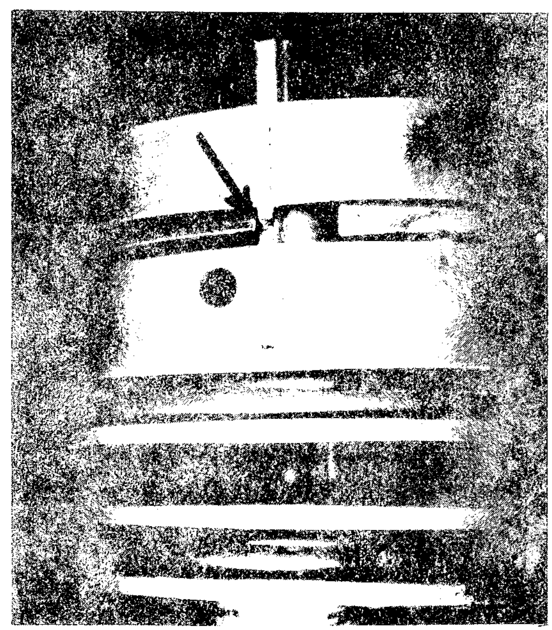
4.6. Magnetos

Magnetos and ignition systems are of two types, high tension and low tension. The main differences between the two systems are the location of the transformer coils and the length of high tension leads. Figures in **C IV-24** are schematics showing the basic differences in the systems.

4.7. High Tension

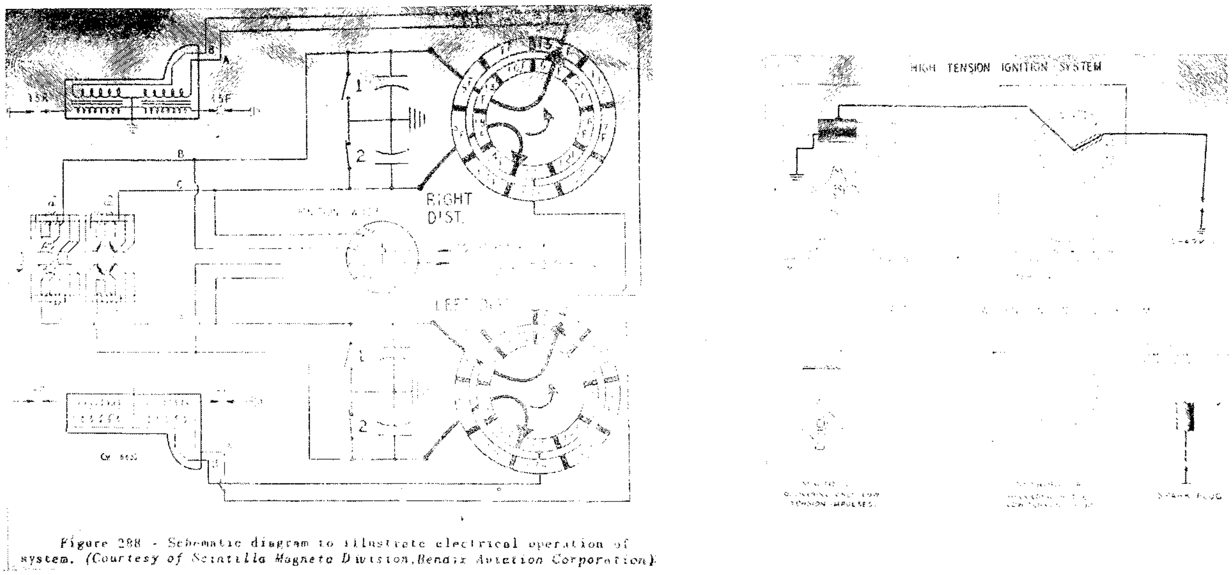
The high tension magneto generates the low voltage and transforms it into high voltage inside the magneto, requiring high tension leads from the magneto to the spark plug. Some installations have a double magneto, where a single rotating magnet generates a primary circuit in two separate coils to obtain dual ignition. This system utilizes separate distributors. The main disadvantage of the double magneto is that if the rotor or "mag" drive fails, the entire ignition system ceases to function.





Left photo shows vanes in position against the rotor. Right photo shows clip off the rotor (arrow) resulting in pump failure.

Figure C IV-23.



Schematics showing basic differences in the systems.

Figure C IV-24.

There are several different makes of magnetos, high tension leads, and connectors. The investigator needs to be aware of the possible failures and how to determine the condition of the various components.

4.7.1. Low Tension

The low tension magneto generates the primary current only. The primary current is carried by low tension leads to or near the cylinder heads where the secondary coil is located. (**Fig. C IV-25.**) This allows a very short high tension lead to the spark plugs, reducing the problems encountered with the high tension harness, particularly at high altitude.

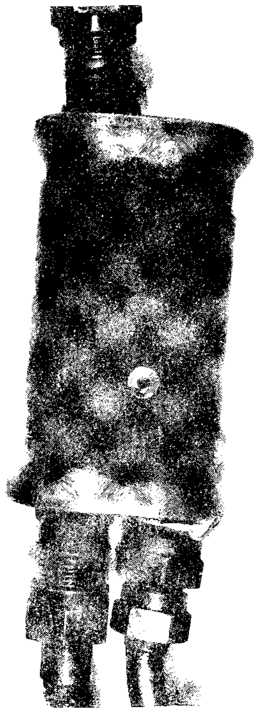
Inspection of the ignition system includes checking harness installations, coils, points, condensers, seals, and the general condition of all components. The entire system can be bench checked; however, the investigator should be aware that no matter how good the bench check seems to be, the actual service operating conditions cannot be duplicated.

4.8. Carburetion

There are several types of carburetion systems and each type has its peculiarities and area of malfunctions. This section cannot cover all possible problems that may be encountered, but will deal with those areas of the greatest probability of malfunction. One of the best checks that can be made on any carburetor is the flow test. Field investigation of carburetors, especially pressure or injection types, should normally be restricted to a visual inspection. The investigator in some instances may decide a teardown to be more advantageous than a flow check. In either case, the teardown or the flow check should be conducted at a properly equipped facility by highly qualified personnel, to preclude loss of existing evidence. The carburetor should be checked before it is cleaned or repaired, otherwise the test will be ambiguous. If it is necessary to make some minor repair, this should be taken into consideration when evaluating the findings.

4.8.1. Float Type Carburetors

The float type carburetor is rather simple to disassemble and inspect. Care should be exercised during the disassembly to protect any evidence that may be present. The float chamber should be checked for contaminants of any sort and for plugged jets. Metal floats may show evidence of fuel in the chamber. If they



High voltage for low tension **ignition** system.
Cutaway of high voltage coil.

Figure C IV-25

are "hydraulicked," or "caved in" (**Fig. C-IV-26**), it is pretty reliable evidence that the chamber contained fuel at the time of impact. Normally, except in the case of extreme heat, leaking floats will contain some fluid for a long period of time after the accident.

4.8.2. Pressure Type Carburetors

The investigator should have the proper facilities and background knowledge of the particular carburetor to make a determination of the findings, if, after flow test, it is necessary to disassemble pressure type carburetors. If the investigator is not qualified, he should have the assistance of a qualified technical assistant. In any event, the investigator should be aware of what should result from the tests and checks.

4.8.3. Continuous Flow — Including Bendix, Simmonds, and Continental

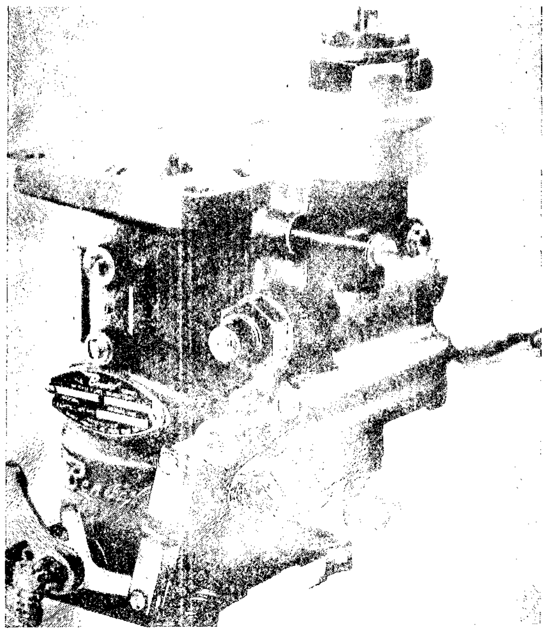
The Continental injection system and continuous flow systems have very few items to malfunction. If the pump is supplying the fuel

"Hydraulicked" metal floats.

Figure C IV-26.

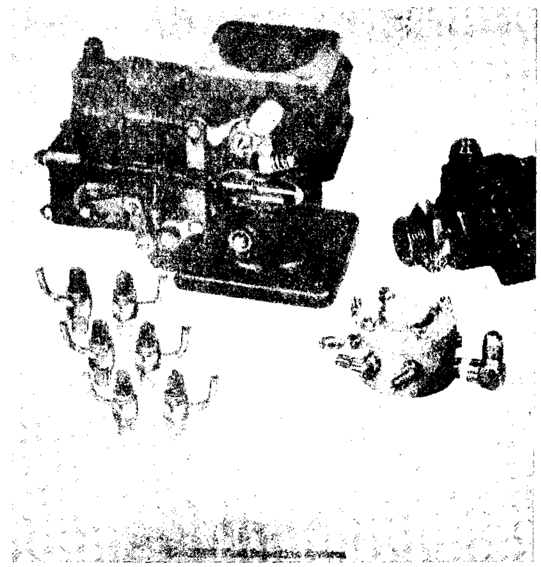
to the fuel control valve at the proper pressure, then the control valve operations should be checked. The fuel manifold diaphragm should be checked for leaks, the injection nozzles for stoppage, and all fuel lines for breaks or stoppage.

Figures C-IV-27 through 30 show examples of this type of carburetion.



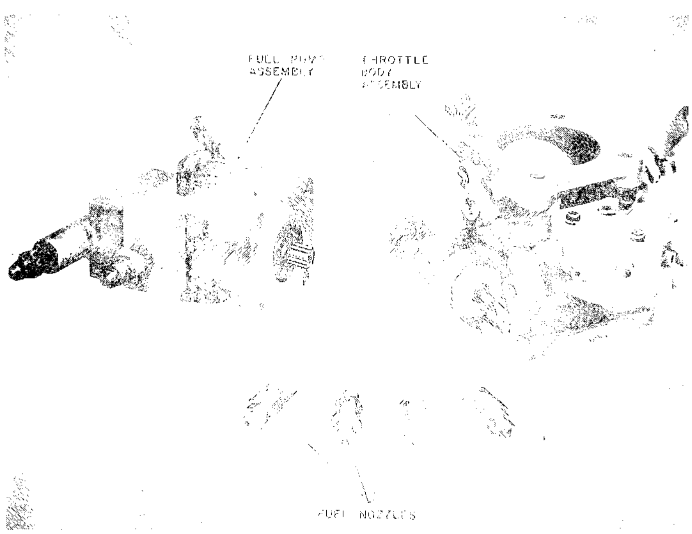
Bendix Servo Unit RS-10.

Figure C IV-27.



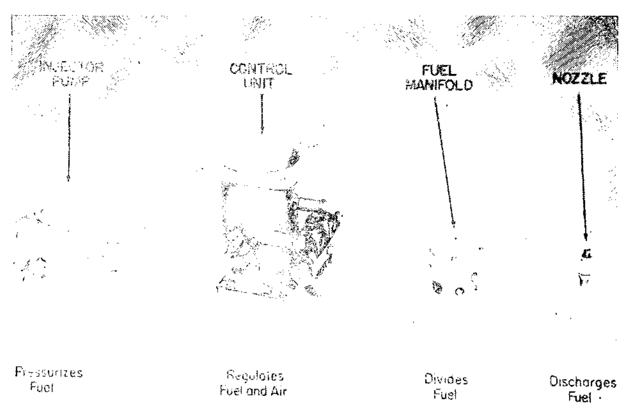
RS 5BD2 Fuel Injection System.

Figure C IV-28.



Simmonds Continuous Flow

Figure C IV-29.



Continental Fuel Injection Units.

Figure C IV-30.

PART C — AIRCRAFT AIRWORTHINESS INVESTIGATION

CHAPTER V

SYSTEMS

1. Systems — General

The aircraft systems investigator in any accident is faced with a difficult task in dealing with the physical evidence. His responsibility is to determine conditions and capabilities of each system prior to the accident. His task is further complicated because many of the systems components will be spread over the wreckage path, or will be partially or completely disrupted by fire. Difficulty arises from the diversity and complexity of systems, particularly those installed in large, turbine-powered aircraft, executive types, and some of the more advanced aircraft in the light twin field. Because all systems can be related to three basic areas, it is imperative that the investigator have a good working knowledge of hydraulics, pneumatics, electricity, and electronics to adequately investigate and analyze the available facts.

The investigation and analysis of the more complex systems, their capabilities prior to an accident, and their possible contribution to causal factors, must necessarily be definitive and exhaustive.

1.1. Approach

Because of the diversity in each system, depending on the particular aircraft involved, discussions will be general in nature and will cover the basic components in each system. It is very important to learn not only how to investigate each system but also to understand why it should be done in a certain way and what the end result should be.

The systems investigation will be presented in the light of the catastrophic or team-investigated accident. This type of accident usually involves large, transport-type aircraft. Although these aircraft contain complex systems, discussion will be general. The techniques of investigation and analysis as explained here will be equally applicable to small aircraft. The discussion of organization will be helpful to the investigator who handles a small accident by himself.

The systems to be covered are:

- Hydraulics
- Electrical/Electronics
- Instrumentation
- Radio Communications and Navigation
- Pneumatics
- Fire Detection and Protection
- Oxygen

1.2. The Importance of Attitude

The importance of attitude cannot be overemphasized. This one factor is the difference between a thorough investigation and one that is mediocre or incomplete. It is also reflected in the analysis of the facts developed.

Experience has shown that some investigators tend to predispose the areas of importance based on preliminary evidence developed during the first days of an investigation. This is a pitfall which must be recognized and avoided at all costs.

It does not suffice to catalog and document the wreckage in a purely mechanical routine. The investigator must think and plan, always keeping an analytical eye on the wreckage, otherwise some vital point may be missed which could provide valuable assistance in the analysis of causal factors.

Some investigators look at a system with the thought in mind, why bother with this? It couldn't have caused the accident. This is fuzzy thinking since there is no way of knowing whether or not it had a bearing on the accident until that system is investigated thoroughly. Only then can the extraneous be separated from the pertinent evidence in determination of bearing on the accident, and whether or not operation was normal prior to the accident. Even more important, the information developed from one system may be used to prove or disprove the integrity of another system or component.

Aside from the primary task of seeking causal factors in the accident at hand, the investigator has an equally important obligation to discern conditions and circumstances which point to an accident in the making. Many very important recommendations for corrective actions result because the investigator looks below the surface. Conditions are brought to light when the investigator makes a thorough examination of all areas, conditions which will be unnoticed if the investigator does not take a realistic approach to a systematic and thorough investigation.

1.3. On-the-Scene Investigation

The investigator must develop a firm plan of action for the investigation. While each accident has unique circumstances and the wreckage layouts differ, a good plan of action will be flexible to accommodate these variances.

Following the team organizational meeting, the Systems Group should assemble for introductions and a short briefing prior to departure for the scene. Suggested items to be covered at this briefing are:

- Introduce each Group member giving name, title, business address and phone number, and local contact.
- The basic ground rules for the investigation will have been laid by the IIC at the organizational meeting, however, it is well to emphasize the high points of security, teamwork, responsibilities of each person, and the plan of action. Above all, stress the importance of thoroughness and teamwork. Emphasize the need for staying together as a team and not diverting to activities of other groups. All pertinent developments will be covered in the progress meetings.
- Explain the method of documenting evidence and how notes are to be kept. Assign one of the members the task of keeping the official notes. The Group Chairman is responsible for the completeness and accuracy of the official notes which he controls at all times.

1.3.1. Conduct of the Investigation

The wreckage probably will be widespread and many systems components will be scattered, therefore, the Group should tour the accident site to determine the location of the systems components.

The Structures Group will plot a wreckage distribution chart. In the interest of time and to assist the Structures Group, the Systems Group Chairman should detail his members to identify and tag all loose systems components, beginning at the point of initial impact and working through to the end of the wreckage path.

Meanwhile, the chairmen of the Systems Group and the Operations Group, together with one other person knowledgeable on the particular aircraft, should begin cockpit documentation. No one is allowed in the cockpit area until this documentation is complete. The Systems Group Chairman is responsible for keeping the notes and they will be included in his factual report. These notes are to be duplicated as soon as possible for dissemination to the other Group chairmen, the technical coordinators, the IIC, and the interested party coordinators.

Following cockpit documentation, the Systems Group Chairman checks progress of locating and tagging systems components; he checks the areas covered to insure that nothing has been overlooked. He surveys the wreckage for any systems components in danger of being covered up or damaged further by activities of the Structures and Powerplants Groups, coordinating with the other Group chairmen to protect these components.

During the wreckage survey, the Group Chairman determines which systems or components should be documented first. This will be contingent upon the intentions of the other Groups to disturb or remove wreckage which may contain systems components. These items of mutual interest should be covered first to expedite the activities of the other Groups. It is advisable to take as many photographs as possible throughout the wreckage before it is disturbed. This applies particularly to hydraulic and electrical actuators, etc., to show positions in relation to other pieces of wreckage. Early photographs can be very important since many items will not be the same after removal from the wreckage.

1.3.2. Documentation of the Wreckage

When the location, identification, and tagging are complete, gather the Group members and brief them on the areas to be covered. Five or more men cannot conveniently work in the same area simultaneously, therefore, it is advisable to separate into groups of two and work more or less independently. Each group should be given a specific assignment. The Group Chairman then periodically checks the progress of these groups.

An explanation of documentation is necessary. The Group Chairman will have assigned to him at least one person who has never participated in an investigation. To most of these novices documentation means merely cataloging items, with a general description. These individuals should be instructed in the proper methods prior to beginning this phase.

Documentation is in the form of written notes of complete sentences. Short, cryptic notes are completely unacceptable. Some individuals look at a unit which has been exposed to fire and write ". . . destroyed by ground fire." This is absolutely meaningless! The exterior may be intact, the internal mechanism may be ruined by the heat, but the unit certainly is not destroyed. Another example of a meaningless statement is to refer to damage of a unit which has been crushed between heavy structures as "destroyed by impact."

If the investigator is to have complete documentation on which to base his factual report and analysis, he must describe the actual condition of a component or system. This means that dents, gouges, scuffing, scoring, smears, cracks, and failures must be written into the notes; otherwise, the investigator cannot properly describe these conditions in his report. One way to consider documentation is to write the notes so that the investigator could add only headings and titles and ask his secretary to write up his report. This may be extreme, but experience has shown that it is imperative.

Make certain that the groups working independently realize the importance of complete documentation. It will be necessary to check their progress periodically to insure accuracy and completeness. At the end of each day the Group Chairman collects all notes. In most cases, a duplicating machine will be available in the command post or conference room. Each day's notes should be duplicated as soon as possible, preferably each evening, for distribution to the Group members, the technical coordinator, and the IIC. Some time should be set aside daily or every other day, to discuss the accumulated notes and Group progress. Each member should be encouraged to make suggestions and each should be in agreement with the evidence and facts developed.

When the on-scene phase of the investigation is complete, the Group Chairman assembles his Group to consider the rough notes and to discuss the investigation. An agreement must be reached between all members that everything in the Systems area has been covered and properly documented. The technical coordinator is to be included in this meeting since he shares the Group Chairman's responsibility for a thorough and complete investigation. He should concur with the Group that everything has been covered.

When it is definite that the Systems Group has completed the on-scene phase, the rough notes are consolidated and grouped in the order of presentation in the Group Chairman's factual report. The notes should be completely rewritten in final form with additional details and comments deemed necessary. These will constitute the final official notes for the Group, and the factual report will be based on these notes. When the notes are completed they are to be duplicated and distributed to the Group members, the technical coordinator, and the IIC prior to disbanding the Systems Group. **THE NOTES MUST BE IN THE HANDS OF THE GROUP MEMBERS BEFORE DISBANDING.**

1.3.3. Coverage of Aircraft Systems

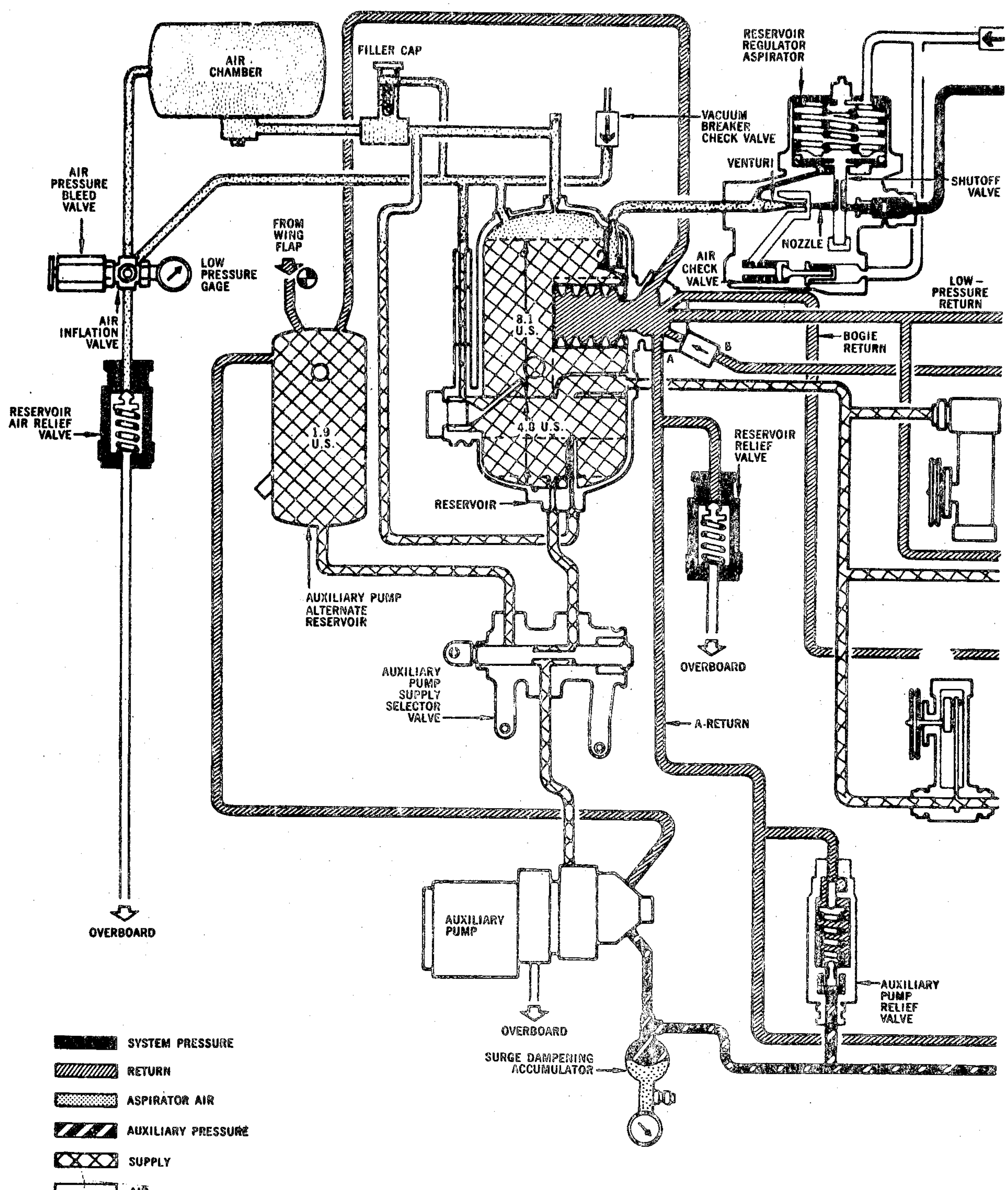
Determine the components included in each system by reference to the appropriate detailed schematic diagrams, and make every effort to account for all components. Note typical jet transport hydraulic power system schematic (Fig. C-V-1 and -2). Be sure to include basic information from component data plates for identification and location of components in the system such as:

- Nomenclature
- Manufacturer
- Specification Number
- Part Number
- Serial Number

Some components having the same part number are used in several places within some systems. This is true especially of some hydraulic pressure and control units and various valves in the air conditioning system. It will be necessary to obtain a current list of components from the air carrier to show which units were installed in the aircraft and where. If the aircraft is new, this information may be obtained from the airframe manufacturer. See example of hydraulic power system component location list (Fig. C-V-3).

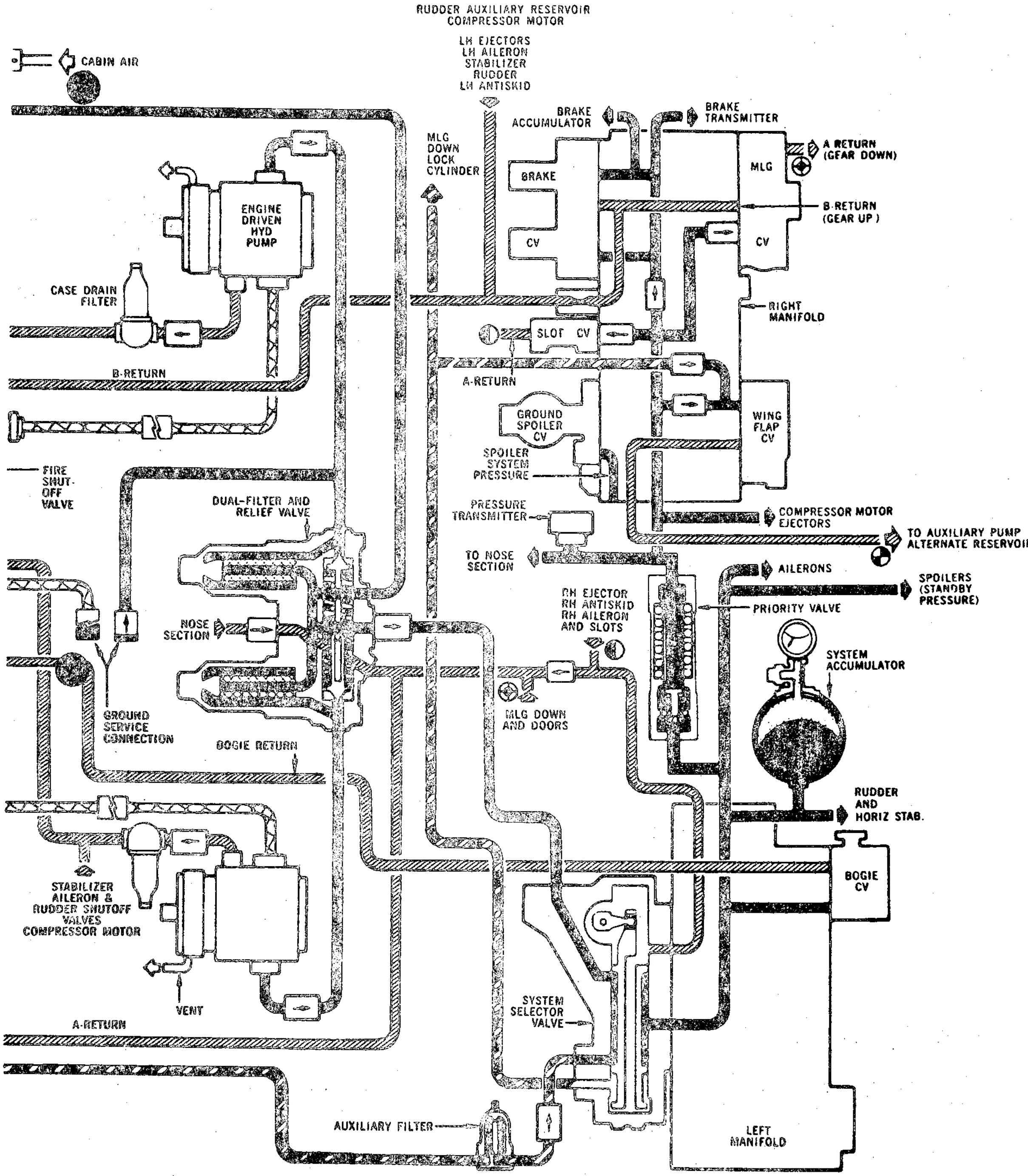
Many of these components are electric-motor operated; it is very important that the position of associated valves or actuators be documented. Once the component is identified by its location in the system, the position of movable parts can be related to a particular function of the system to enable the investigator to determine its prior operating condition. This will also be applicable to certain cockpit motor and gear-driven instrumentation which will hold position when electrical power is removed.

As an aid to covering all systems in a thorough manner a checklist has been prepared for the systems investigator. This list is only a guide to be used at the investigator's discretion, depending on the circumstances involved. Every effort should be made to cover each system in a systematic manner.



Schematic of Hydraulic Power System.

Figure C-V-1.



Schematic of Hydraulic Power System.

FIGURE C-V-2.

COMPONENT LOCATION LIST

HYDRAULIC POWER SYSTEM — 29

MAIN HYDRAULIC SYSTEM SELECTOR CONTROL LEVER	FLT ENGINEER CONTROL PEDESTAL
ENGINE DRIVEN PUMP SWITCHES LH & RH	FLT ENGINEER INSTR PANEL – UPPER AFT
C/B "ENG 2 & 3 HYD PUMP BYPASS"	DC BUS 1
AUX PUMP SWITCH & IND LIGHT – CAPTAIN'S	OVERHD SW PANEL, UPPER CNTR
AUX PUMP SWITCH – FLT ENGINEER'S	FLT ENGINEER'S INTR PANEL, UPPER AFT
C/B "AUXILIARY HYD PUMP CONTROL"	CABIN BUS 4
AUX HYD PUMP FUSES (3)	ELECT POWER CENTER, AFT END, IN GENERATOR CONTROL PANEL BAY
AUX HYD PUMP POWER RELAY	EPC AFT EQUIP PANEL
AUX HYD PUMP CONTROL RELAY	EPC AFT EQUIP PANEL
MAIN RESERVOIR HYD QTY INDICATOR	FLT ENGINEER INSTR PANEL, UPPER AFT
MAIN RESERVOIR HYD TEMP INDICATOR & WARN LITE	FLT ENGINEER INSTR PANEL, UPPER AFT
C/B "MAIN RESERVOIR HYD OIL TEMP & QNTY"	DC BUS 1
MAIN RESERVOIR LO-PRESSURE WARN LITE	FLT ENGINEER INSTR PANEL, UPPER AFT
MAIN SYSTEM HYD PRESSURE INDICATOR	FLT ENGINEER INSTR PANEL, UPPER AFT
C/B "MAIN & SPOILER HYD SYS PRESS IND"	INSTRUMENT SECONDARY
ALTERNATE (EMERG) HYD RESERVOIR LOW LEVEL WARNING LITE	FLT ENGINEER INSTR PANEL, CENTR FWD
C/B "ALTERNATE (EMERG) HYD RES LO LEV WARN LITE"	INSTR SECONDARY (C/B CALLED: "MAIN & SPLR HYD SYS PRESS IND")
MAIN HYD SYSTEM PRESSURE XMTR	NLG WELL, UPPER AFT RH END
LH HYDRAULIC MANIFOLD	LH MLG WELL, UPPER FWD INB'D
MAIN HYD SYSTEM SELECTOR VALVE	LH MLG WELL, ON TOP OF LH HYD MANIFOLD
MAIN HYD SYSTEM PRIORITY VALVE	LH MLG WELL, UNDER LH HYD MANIFOLD
EDP SYSTEM DUAL FILTER & RELIEF VALVE	LH MLG WELL, FWD LOWER RH
MAIN HYD SYSTEM (OR "FLIGHT CONTROL") ACCUMULAT.	LH MLG WELL, UPPER CNTR RH
AUX PUMP FILTER	LH MLG WELL, BELOW LH HYD MANIFOLD
MAIN RESERVOIR GRAVITY FILL PORT & AIR CHAMBER	LH MLG WELL, FWD END, OUTB'D
MAIN RESERVOIR RELIEF VALVE, PRESS GAGE, BLEED VALVE & AIR FILL VALVE	LH MLG WELL, FWD END, INB'D
RH HYDRAULIC MANIFOLD	RH MLG WELL,UPPER FWD INB'D
MAIN RESERVOIR: WITH XMTR FOR KTY, RESISTANCE BULB FOR TEMP, SWS FOR RES LO PRESS, OVERTMP	LH AFT WING ROOT, OUTB'D
MAIN RESERVOIR RETURN FILTER & ASPIRATOR RETURN FILTER	INSIDE OF RESERVOIR
MAIN RESERVOIR ASPIRATOR & NEGATIVE RELIEF VALVE	TOP OF RESERVOIR
AUXILIARY HYDRAULIC PUMP	LH AFT WING ROOT, INB'D
AUX HYD PUMP SUPPLY SELECTOR VALVE	LH AFT WING ROOT, FWD OF AUX PUMP
AUX HYD PUMP SURGE DAMPER ACCUMULATOR	LH AFT WING ROOT, AFT OF AUX PUMP
AUX HYD PUMP RELIEF VALVE	LH AFT WING ROOT, ADJ TO MAIN RES
ALTERNATE (EMERG) RESERVOIR, WITH LOW LEVEL SWITCH & SIGHT GLASS	LH AFT WING ROOT, INB'D
GROUND SERVICE CONNECTIONS	LH AFT WING ROOT, ADJ TO MAIN RES, BOT OF WING
EDP CASE DRAIN LINE FILTERS LH & RH	AFT FACE OF REAR SPAR, FWD OF INB'D FLAP SECTION. ACCESS BY LOWERING FLAPS
FIRE SHUTOFF VALVES LH & RH	WING LEADING EDGE AT#2 & #3 PYLONS – ACCESS THRU INSPECTION PLATES
ENGINE DRIVEN PUMPS, LH & RH	#2 & #3 ENGINES – MOUNTED ON AFT RH SIDE OF ACCESSORY DRIVE (BOT OF ENG)

Figure C-V-3.

CHECKLIST FOR SYSTEMS INVESTIGATORS

The following checklist is a guide to full coverage of all systems and will be applicable to all aircraft from light planes to jets. The list covers primary systems, secondary systems, and subsystems. The breakdown within each system covers the functions in the order of supply, pressure, and activity. This arrangement is simpler than that contained in the ATA Code for aircraft systems.

I. HYDRAULIC POWER SYSTEMS

A. *Reservoirs*

1. Check amount of fluid remaining and obtain samples for contamination count.
2. Check for proper fluid type by color. If questionable, obtain sample for analysis.
3. Check pump fluid suction shutoff valves for position.
4. Examine valve control (mechanical, electrical) for integrity.

B. *Hydraulic Pumps*

1. Engine-driven pumps – Check mounting, hoses, case drain filters, pump drive, evidence of fluid, pump condition; test if possible.
2. Electrically-driven pumps – Same as for engine-driven pumps.

C. *Pressure Manifolds or Modules*

1. Check condition, valve positions, plumbing attachments and electrical connections.

D. *Pressure Regulators*

1. Check condition and pressure setting if necessary.

E. *Accumulators*

1. Determine air preload, fittings, condition of pistons or diaphragms.

F. *Selector Valves and Control Valves*

1. Determine positions of landing gear, wing flap, leading edge device and spoiler selector valves. Examine valve control linkage for integrity. Check the same on control valves.

G. *Hydraulic Actuators*

1. Actuating cylinders – Check condition; measure piston rod extension.
2. Flight control units – (boost packages and power control units) – Examine for condition and attachment to structure; measure piston rod extensions; determine position of valves and external linkages, condition of plumbing attachments, and electrical connections.

H. *Filters*

1. Examine for evidence of entrapped contaminants such as metal, dirt, pieces of seals, etc.

II. ELECTRICAL POWER SYSTEMS

A. *Engine-Driven AC and DC Generators*

1. Test if possible.
2. Examine for evidence of internal arcing, overheating or burning
3. Examine for evidence of rotational scoring on armatures, commutators and pole pieces.
5. Examine brushes for abnormal wear or breakage.

B. *Constant Speed Drives*

1. Test with mated generator if possible.
2. Examine for condition and evidence of internal lubrication.
3. Examine for evidence of being engaged or disengaged.

C. *Inverters (Normal and Emergency)*

1. Test if possible.
2. Examine same as generators.
3. Examine control box components such as frequency and voltage regulators, capacitors, wiring, etc., for condition and security.

D. *Engine-Driven Alternators*

1. Test if possible.
2. Examine same as generators and inverters.

E. *DC Electrical Distribution Circuits*

1. Examine bus bars for loose or damaged studs and terminals. Look for evidence of arcing or burning.
2. Examine current limiters and fuses attached to bus bars for continuity or or non-continuity.
3. Check voltage regulators, reverse current relays and power contactors for condition, and evidence of electrical distress.
4. Examine relays for condition of arcing or welded contacts, etc.
5. Examine all electrical wiring for evidence of electrical overheat, burning or arcing. Examine loose circuit breakers by identifying prior electrical function and checking for condition.

F. *AC Electrical Distribution Circuits*

1. Examine and test voltage regulators, transformer-rectifiers, etc., if possible to determine their capabilities.
2. All other items same as in DC circuits.

G. *AC and DC Electric Motors*

1. Examine for evidence of electrical malfunction or operation at time of accident. Look for evidence of rotational scoring the same as in the generators and inverters.

III. ELECTRONIC EQUIPMENT

A. *VHF Communication Transmitters and Receivers*

1. Examine these units for frequencies selected prior to accident. Test if possible or necessary.

B. *VHF Navigation Receivers*

1. Examination same as for VHF Communications.

C. *Distance Measuring Equipment*

1. Recover the unit and determine the distance reading from examination of the distance mechanism module. Also determine the channel to which the unit is tuned. Determine the existence of a VORTAC station in the area of the accident to compare with the channel selected.

D. *LF Navigation Receivers*

1. These units can only be tested to determine the preselected frequencies since there are no indicators on the equipment.

E. *Automatic Pilot Components*

1. Attempt to determine whether or not the autopilot was in use at the time of the accident.
2. Test each component if possible or necessary to determine its condition.

IV. INSTRUMENTATION

A. *Flight Instruments*

1. Altimeters — Check the barometric scale settings to compare with the information last provided to the flight crew. Log the position of the drum-pointer combination or three needles on the older altimeters. Note the condition of the internal mechanisms.
2. Heading Indicating Instruments — Log the magnetic headings indicated.
3. Log the positions of needles and other indicators on all other flight instruments for further reference.

B. Engine Instruments

1. Identify previous locations and positions for loose engine instruments and log all indications.

C. Flight Engineer Instruments

1. Log all indications, settings, etc.

D. Vertical and Directional Gyros

1. Identify functions (No. 1, No. 2, Auxiliary, etc.) and describe condition. Remove gyro rotors and examine rotors and cases for evidence of high speed rotational scoring. Keep in mind that gyro rundown can be as little as six or seven minutes or as much as 13 or 14 minutes.

E. Compass Systems

1. Examine fluxgate transmitters if ininstalled. These units have gyros in them. Flux valves provide little or no information and are generally used in conjunction with directional gyros. Check gyro rotors and cases for rotational scoring.
2. If necessary and possible, test the compass amplifiers along with the other heading sources.

F. Pitot and Static Pressure Systems

1. Account for the pitot heads and static port panels. Examine and log their conditions. Examine plumbing for loose fittings and electrical connections and elements for pitot and static heaters.

G. Vacuum System

1. Test vacuum pump(s), filters, plumbing, and thermal relief seals if possible. Record conditions.

H. Flight Data Recorder and Voice Recorder

1. These items (if installed) should be recovered as soon as possible and shipped to the Engineering Division, NA-96, Bureau of Aviation Safety, National Transportation Safety Board, Washington, D.C., 20428.

V. PNEUMATIC SYSTEMS

A. Engine Bleed Air System

1. Bleed air shut-off valves – Check position (open or closed).
2. Flow modulating valves – Log position.

B. Air Conditioning Packs

1. Pack shutoff valves – Log positions.
2. Air cycle machines – Check for evidence of rotational damage.
3. Cabin air-mixing valves – Check for position.
4. Freon system – Check compressor and motor.
5. Pack ram air doors – Check position of doors and door actuator motors.
6. Check positions of all valves and condition of ducting. Look for evidence of exposure to fire, heat, or smoke.

C. Thermal Anti-Icing System

1. Check positions of wing isolation valves.
2. Examine all valves and ducting for evidence of exposure to fire, heat or smoke.

VI. WINDSHIELD ICE AND RAIN PROTECTION

A. Windshield Wipers

1. Note positions and condition of wiper arms and blades.
2. Check condition of wiper motors, resistors, drives, etc.

B. Rain Repellent System

1. Check repellent container for being discharged.

VII. FIRE DETECTION AND PROTECTION SYSTEMS

A. *Fire Detection System*

1. Determine the operational capabilities of the detection system if possible.
2. Check light bulbs in fire warning lights for burned out filaments at the time of the accident.

B. *Fire Protection System*

1. Check the fire extinguisher bottles for being charged or discharged.
2. Check the fire extinguisher selector valves for the engines to determine if they are open or closed.
3. Check plumbing for evidence of leakage or stoppage.
4. Account for all portable CO_2 and H_2O extinguishers and determine whether or not they have been discharged.
5. Check the condition of the thermal and system discharge discs for the main extinguisher bottles, and log the findings.

VIII. OXYGEN SYSTEMS

A. *Crew and Passengers Oxygen Systems*

1. Account for and examine the oxygen supply cylinders for the crew and passengers. Log whether or not they are empty.
2. Account for the portable or walk-around oxygen cylinders in the same manner.

IX. MISCELLANEOUS

A. *Light Bulbs*

1. Account for and examine as many of the exterior lights as possible to see what can be found regarding their use just prior to the accident. These lights can be very helpful. Interior light bulbs should also be examined.

1.4. Testing Systems Components

During the on-the-scene investigation the investigator will probably determine that certain items from the various systems should be given special attention, and he will arrange to conduct functional tests to determine the true condition and capability of these components.

1.4.1. Choosing the Facility

The investigator determines the need for testing to evaluate the true condition of the component. By consultation with representatives of the airframe manufacturer and the air carrier (at the scene), the Group members, and the technical coordinator, the investigator selects the facility which can best accomplish the testing. Facilities suggested in the order of desirability are:

a. The air carrier overhaul and maintenance base.
b. The airframe manufacturer.
c. Component manufacturer.
d. A nearby FAA certificated repair station.
e. Facilities on a nearby airport.

Circumstances dictate the choice of the facility which can handle all testing contingencies. Many manufacturers are willing to test their products for confirmation of quality.

1.4.2. Conducting the Tests

Upon selection of the facility, arrangements should be made for the management to schedule the desired activities. If more than one facility is involved in multiple tests, arrange an itinerary which provides for travel time between tests. The itinerary includes the name, address and telephone number of the facility, the person to be contacted, and the date and time of the tests.

It will not always be convenient for the regularly assigned Group members to witness all of the tests. Each Interested Party coordinator should be provided with an itinerary so that representatives can be selected to witness the tests if the regular Group member cannot attend. Secure a listing of these representatives as early as possible.

Assemble the witnessing group for briefing. Explain the type of testing planned and provide a copy of the test procedures for each member of the investigating group; these become a party of the official Group notes. See sample hydraulic gear pump test procedures (**Fig. C-V-4**). The official notes for the tests will be kept by the Group Chairman. Make a list of all members, their job titles, and the party represented. The test procedure should be reviewed to insure that it will cover the desired range of testing. Some items on standard test procedures will be unnecessary; the group should agree on items to be eliminated.

HYDRAULIC GEAR PUMP

Model No. 011227-100-01

TESTING

A. Preliminary Information.

Unless otherwise specified, the following conditions shall apply to all pump tests.

(1) Rotation – Counterclockwise viewing pump drive end.

(2) Inlet pressure – 5 to 7 PSIG.

(3) Inlet temperature – 26.67°C to 48.89°C (80°F to 120°F).

(4) Hydraulic fluid – MIL-5606.

(5) Pressure and flow measurement accuracy – ±2% at points designated.

(6) Horsepower required (max) – 25 HP

NOTE: Tests shall be conducted in the order given in the following paragraphs.

B. Break-In Run. Run-in the pump for a minimum of one hour as follows:

(1) With the pump running at 2280 RPM for ½-hour minimum, gradually increase pressure to 2000 PSI in increments of 500 PSI, allowing pump to run-in for several minutes at each pressure. Reduce pressure to 500 PSI for at least one minute to allow for internal cooling.

(2) Gradually increase pressure from 2000 PSI to 3000 PSI in increments of 200 PSI, allowing the pump to run-in at each pressure for several minutes. During this phase of run-in, reduce pressure to 500 PSI several times to prevent overheating of internal parts.

(3) Gradually increase pressure from 3000 PSI to 3750 PSI in increments of 200 PSI, allowing the pump to run-in at each pressure for several minutes. Reduce pressure to 500 PSI for cooling periods as required. Limit continuous running at 3750 PSI to 1 to 2 minutes maximum.

(4) Repeat steps (1), (2) and (3) at a speed of 3800 RPM. Include low pressure cooling periods as required.

(5) If the torque at 3000 PSI exceeds 26 ft-lbs, continue run-in until this torque is met.

C. Rated Speed Run. To conduct the rated speed run tests, operate the pump at 2000 ±90 PSI with the speed and inlet pressures as follows:

(1) 20 minutes at 3800 RPM with 10 PSIG inlet pressure.

(2) 20 minutes at 3800 RPM with 8-in. Hg absolute inlet pressure.

(3) 20 minutes at 4200 RPM with 5 to 7 PSIG inlet pressure.

There shall be no leakage of air into the pump through the drive shaft seal as determined by observing the condition of the pump delivery line fluid. Leakage past the drive shaft seal shall not exceed 3 cc/hr (cubic centimeters per hour).

Figure C-V-4.

HYDRAULIC GEAR PUMP

Model No. 011227-100-01

D. Proof-Pressure Run. Proof-pressure run the pump as follows:

(1) Operate the pump for 1 minute at 3800 RPM with 3750-PSI outlet pressure.

(2) After 1 minute, sharply increase pressure to 4000 PSI. Do not maintain 4000 PSI for more than ±2 seconds.

NOTE: Reduce pressure to 500 PSI and allow to run at least 3 minutes before performing the capacity and torque test.

E. Capacity and Torque Test. Perform the capacity and torque tests on the pump at the settings shown in Table 1.

Table 1

CAPACITY AND TORQUE TEST TABLE

RPM	DISCHARGE PRESSURE	INLET PRESSURE	MINIMUM CAPACITY (GPM)	MAXIMUM TORQUE
3800	400	24-30 in. Hg. Abs	8.7	5 ft-lbs.
2280	3000	24-30 in. Hg. Abs	5.1	26 ft-lbs.
2280	1000	24-30 in. Hg. Abs	5.4	10 ft-lbs.

Table 2

Trouble Shooting

Trouble	Probable Cause	Remedy
Air bubbles in pump delivery line.	Drive shaft seal worn or scored.	Repair seal or replace drive shaft.
Excessive pump leakage during proof pressure test.	Improperly installed or blown packings.	Replace packings.
	Low torque valve on nuts.	Torque nuts to values specified in Table 3.
High torque and pump too hot for normal run-in.	Run-in not completed or sufficient.	Complete run-in or run-in for a longer period of time.
	Clearances of internal parts too close.	Replace parts worn beyond the limits of Table 2.

Figure C-V-4a.

TEARDOWN — INSPECTION — HYDRAULIC PUMPS

Ser. No. .... Mfg. Ser. No. 220399 R&R Stock 25039

TSO 4252 Plane No. 7742 POSITION #3

Eng. No. .... Removal Date 11-6-67 Rec. Date ....

Removal Remarks ....

Test Before Overhaul ....

Test Remarks See other side.

2850 PSI/ CC 3050 PSI/ CC 2950 PSI/ CC

For Shop Use: Damaged, Failed or Worn Parts

.... Overheat	.... Inlet Flange	.... Valve Plate	.... Ft. Radial Brg.
.... Contaminated	.... Outlet Flange	.... Block	.... Thrust Brg.
.... Coupling Shaft	.... Rear Cover	.... Pistons or Rods	.... Tail Rad. Brg.
.... Housing	.... Control Piston	.... Drive Shaft	.... Pin Brg.
.... Mtg. Flange	.... Control Cyl.	.... Compensator Spool	.... Pintle Brg.
.... Shaft Seal	.... Linkage	.... Compensator Sleeve	.... Yoke
.... Edv. Valve Spool		Rotating Group	.... Replaced
.... Edv. Valve Sleeve		Rotating Group	.... Reconditioned
.... Edv. Valve Ball		Rotating Group	.... Same

Service Bulletin and/or E.M.O.

Overhaul Remarks and/or Abnormal Conditions This unit was overhauled on 3-1-66, and returned to 0 time.

Unit was returned to the shop on 11-22-66 to have the coupling shaft replaced. Shop checked and returned to service at 1817 TSO. Unit was returned to the shop on 9-14-67 for visual and shop check due to metal being found in the case drain filter. Subsequent checks were OK and unit was returned to service at 4119 TSO.

....................O.H. ....................S.C. Mechanic.................... Date....................

Figure C-V-5.

TEST RECORD Page 2

VARIABLE DELIVERY PUMPS

MFG. P/N AS 66651 L 8 Mfg. S/N 220399

R&R # 25039 O.H. or S.C. By HEM Date

Date Test By

E.D.V. & Compensator Leakage During Unit Test

Port "C" CC/Min. Port "C" CC/Min.

Port "B" & "C" CC/Min. EDV Energized

REGULATION INCREASING PRESS

OUTLET PSI	CASE PSI	GPM	RPM	BOSS T PRESS
500	45	23.	3600	
2850	45	22.	3600	
2900	45	22.	3600	
3020	45	0	3600	

LIMITS
R&R 25039
25090
@ 3600 RPM

Press 3000 - 3050 Press 2950 - 3000

Delivery at 3600 RPM and 2850 PSI 21.79 - 23.17 GPM 14.71 - 15.42 GPM

YOKE MOVES @ 2900 PSI

Yoke Centers 3020 PSI

Blocking Valve Leakage Five (5) cc/Min.

Shaft Seal Leakage one drop in 10 minutes

INTERNAL LEAKAGE

(a) 2850 PSI 1400 cc/min Solenoid De-energized OK

(b) 3020 PSI 1550 (2650) cc/min 18 & 27 V Test OK

(c) 2900 PSI 1975 cc/min

EDV ENERGIZED BLOCKING CLOSES 400 PSI Min Opens 1200 PSI MAX

Blocking Valve Leakage 0 (5) CC/Min.

Foot Valve Relieves 95 (90-100) PSI

Shaft Seal Leakage None DPM

Pump Rejected

Internal Leakage cc/Min. EDV Removed cc/Min.

Shaft Seal Leakage Blocking Valve Leakage

Foot Valve Setting

External Leakage Location

Unit Preserved with Skydrol 500 and Capped

Figure C-V-5a.

All test results should be recorded on standard forms used by the facility. See sample of one facility's test form (Fig. C-V-5). **The originals are to be retained by the Group Chairman.** Any discrepancies noted during the tests are to be written into the official notes with an explanation of their bearing on proper operation of the unit. If the nature of the discrepancy warrants, the unit should be disassembled following completion of the tests to ascertain the cause, and this information is included in the official notes. When the tests are completed, assemble the group to discuss the official notes and test results. When there is agreement that the notes and test results represent a true, factual picture of the unit condition, the Group Chairman requests that the facility duplicate the notes and test results so that each member of the investigating group and the facility will have a copy prior to disbanding. When tests are completed, the investigator will arrange to return the parts to the owner.

1.5. Writing the Factual Report

The factual report is extremely important because it is a direct indication of the thoroughness of the investigation. This document becomes a matter of public record; it must be completely factual. All facts must be described in a manner which will give the reader a clear picture of the physical condition of each system and component.

The report should be narrative in style, with similar items described in the same paragraph where possible. Where damage to like components differs, describe separately to provide a more readable report.

The order of reporting each system should follow the suggested checklist. For instance, all items pertaining to the hydraulic system should be grouped under the heading "Hydraulic Power Systems." This will preclude the reader's searching through the report for items relating to a particular system.

When describing tests performed on various systems and components, show the title and number of the test procedure used. The test procedure itself and the test result forms need not be made an exhibit unless necessary to show proof in a causal area. The results should be covered generally in the text of the report. Only substantial discrepancies need be covered in the report, with some form of factual explanation.

The report includes strictly factual information. Some facts concerning the physical evidence will be derived from "analysis" based on experience. These irrefutable facts may be included in the factual report but care must be exercised to insure that statements based on pure analysis or conjecture are reserved for the analysis report. The factual report must contain sufficient factual evidence to support the findings and conclusions set forth in the analysis. If the factual report is properly presented, the reader will have a clear picture and understanding of the condition of the systems and components, and an inkling of what the analysis will reveal.

Occasions will arise following completion of the on-the-scene and testing phases of the investigation when certain items will present a need for further exploration. When this occurs after the factual report has been completed and mailed to the Group members for comments, a supplemental factual report is necessary. Unless the additional exploration yields controversial evidence, this report need be sent to the Group members only for information.

The factual report is not a Group activity. This is the Systems Group Chairman's report of the investigation. The report is sent to Group members for comments on technical accuracy of the notes taken at the scene and during tests. These should be reviewed for validity. The Group Chairman decides whether the comments should be reflected in the report. Those that clarify the report may be included by means of an errata sheet.

1.6. Analysis and the Analysis Report

The analysis is the result of your investigative effort. If the investigation has been accomplished properly, the analysis will be definitive. If not, time is wasted and there is little concrete evidence on which to base the analysis.

Approach the investigation with the question: What can I get out of the physical evidence? The task is to prove or disprove the integrity of each system prior to the accident, to relate the facts to the accident cause. List all of the facts, conditions, and circumstances relating to the problem and eliminate them systematically. The sole remaining item is the answer.

If it can be proved by analysis of all the facts and conditions that a system was operating properly at the time of the accident, then one area of the problem is eliminated. This process must be carried through each system until integrity is definitely proved or disproved. Show proof with supporting facts developed by painstaking investigation.

1.6.1. The Analysis Report

The clear and convincing analysis report is the instrument which proves the depth of investigation. It is the most effective tool in the determination of the probable cause of an accident.

The order of presentation in the analysis follows the factual presentation system by system. An opinion or conclusion should be included as to the prior operational capabilities of each system. Certain factual information will necessarily be borrowed from the factual report, however, it is not necessary to repeat the factual report.

The method of presentation is in the order of a short factual discussion, followed by a truly analytical study resulting in a valid conclusion. The analytical process leading to an opinion or conclusion must be shown in the discussion. Each short factual discussion and the attending analysis should flow smoothly in logical sequence, in narrative form, leading the reader to a conviction that the only logical conclusion has been reached.

1.7. Summary of Factors in Investigative Planning

Consider the following items as the investigation proceeds. They will aid in planning a systematic investigation.

a. Develop a definite but flexible plan of action, with a checklist to insure that each system will be thoroughly covered.

b. Learn all the known facts about the accident without theorizing.

c. Do not presuppose the importance or unimportance of any system. Cover all of them.

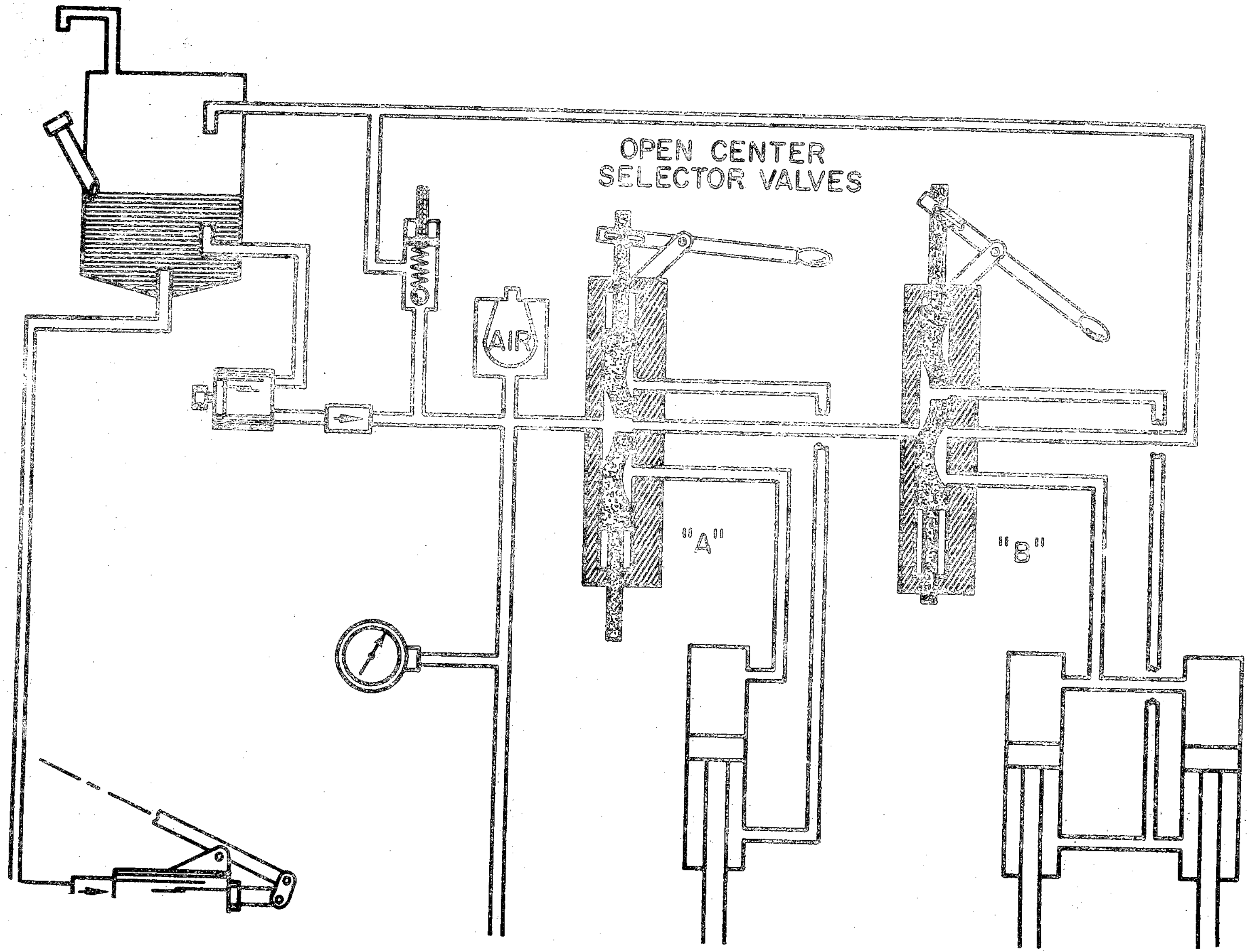
d. Coordinate Group members as a team to preclude independent investigating.

e. Insure complete and accurate documentation by notes and photographs.

f. Coordinate activities with the other Group chairmen.

g. Investigate the systems and components as they are, to get best results. Do not hurry to move wreckage.

h. Protect small parts such as instruments or other small components which may be needed for further examination. Plastic bags can be very useful for this purpose.



Open Center Hydraulic System.

Figure C-V-6.

A small card or tag bearing the identification and description of the item as found can be enclosed for future reference.

i. Fluid samples may be collected in clean glass jars.

j. Keep an open mind, be receptive to new ideas and different approaches to collecting evidence. Look for evidence of "an accident in the making" as well as for causal factors for the accident at hand.

k. Arrange for component testing as far in advance as possible to allow the testing facility to schedule personnel and equipment.

l. Above all, think and plan ahead. Develop an efficient plan of action.

2. Hydraulic Systems — General

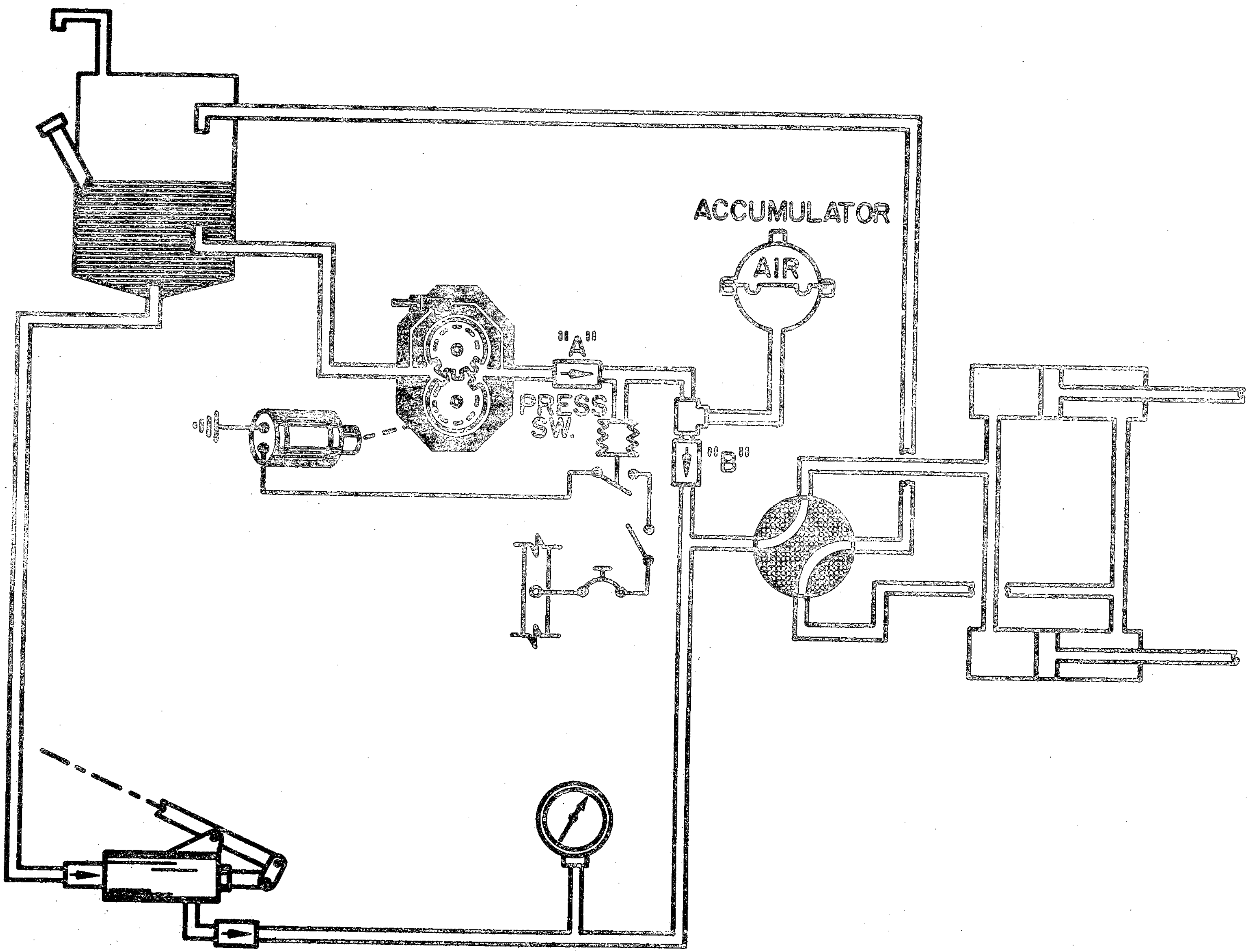
There are two basic hydraulic system designs, the open and closed center types. Some designs incorporate a combination of the two.

In the open center system the entire pump output is directed back to the reservoir when all selector valves are in the neutral position. In this way the pumps are not loaded and the entire system is more or less relaxed. This condition exists until some form of hydraulic service is required, such as operation of the landing gear or wing flaps. When some selector valve is activated, the return line is blocked and pressure begins to build up rapidly to the value for which the pressure regulator is set. A surge chamber is provided in this system to absorb rapid pressure buildup. This unit is similar to an accumulator but it does not store fluid under pressure. As soon as the required hydraulic service is satisfied and the selector valve returns to the neutral position, the return line opens and once again the total pump output is directed back to the reservoir, with a resultant drop in pressure. This system design allows the pumps to be unloaded at all times when no hydraulic service is required. For this reason no system bypass valve is needed in this system. See Fig. C-V-6 for a simplified schematic of an open center hydraulic system.

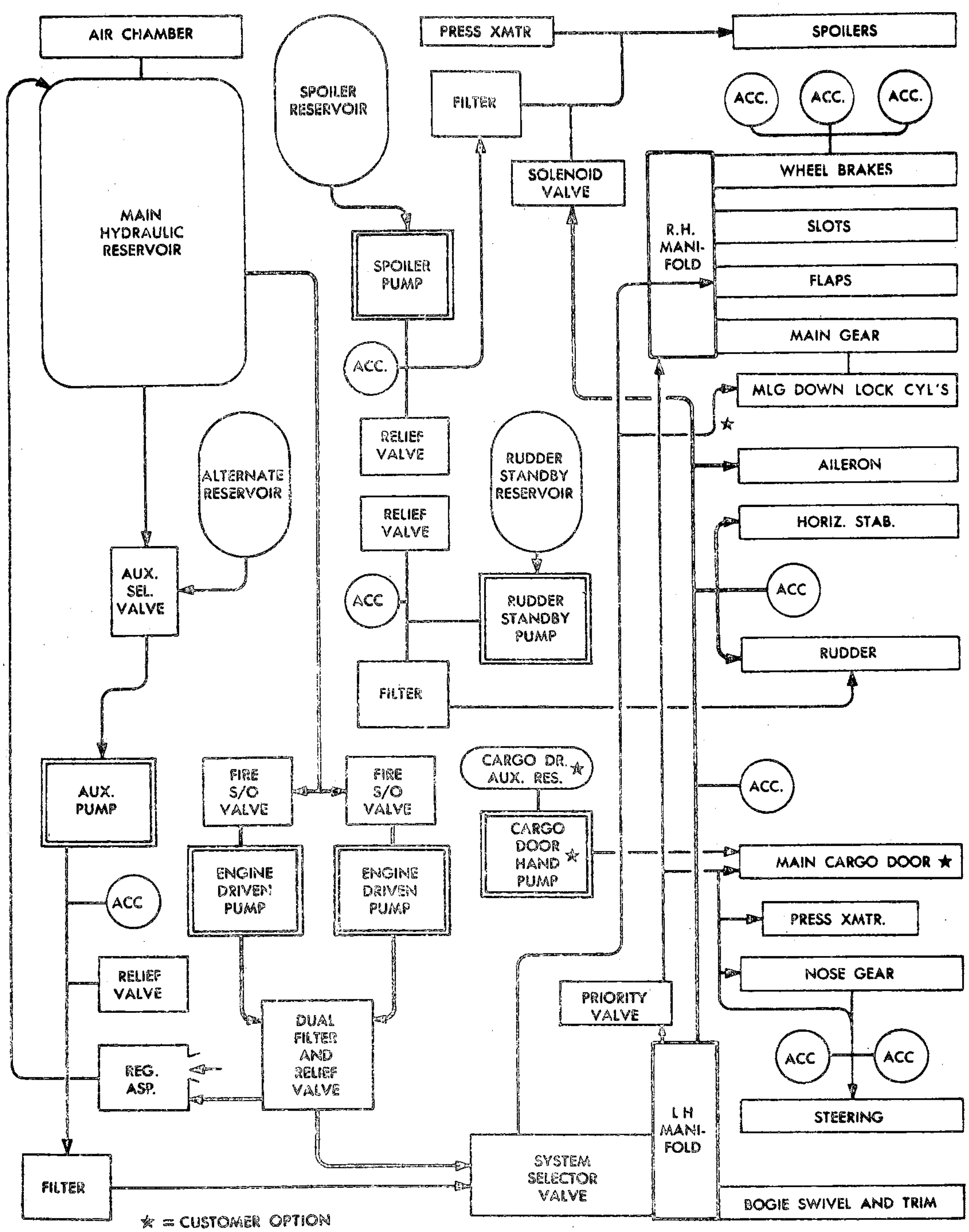
In general, the closed center system maintains constant system pressure. It incorporates an accumulator to store fluid under pressure and requires a system bypass valve to unload the pumps when no hydraulic service is required. This usually occurs in the cruise flight regime. All modern turbine-powered aircraft incorporate the closed center system. The turbine-powered aircraft do not have a system by-pass valve, therefore the pumps are always loaded. This is necessary because constant hydraulic service is required to operate the hydraulic flight control components. This is also found in piston-engined aircraft using hydraulically-boosted flight controls. See Fig. C-V-7, a simplified schematic of a closed center hydraulic system.

The hydraulic pressures involved range from 1000 psi on the small piston engine transports to 3000 psi for the large transport aircraft. The SST will utilize a 4000 psi hydraulic system. Plumbing consists of flexible hoses and aluminum tubing for the lower pressures, while the higher pressure systems utilize special high pressure hoses and stainless steel tubing for the pressure side, aluminum tubing for the return side. Stainless steel tubing with brazed fittings will be used throughout the SST hydraulic system.

All types of hydraulic system designs will be encountered. Before examination or testing of any hydraulic component it is important that descriptive information and diagrams of the particular system and its units be studied. A block diagram and partial system description



Closed Center Hydraulic System.
Figure C-V-7.



Hydraulic Power Supply – Block Diagram.

Figure C-V-8.

HYDRAULIC POWER

The hydraulic power system is a 3000 psi closed-center (always pressurized) system using Skydrol 500A. This fluid was selected because of its wide range of operating temperatures and its resistance to fire. The seals, packings and hoses are of Skydrol resistant material. The plumbing consists of stainless steel for all pressure lines, stainless steel for return lines ⅜ inch diameter and under, and aluminum alloy for remaining return, vent and suction lines. Exceptions occur in exposed areas where all lines are stainless steel. MS flareless fittings are used with both high and low-pressure lines having a diameter of ¾ inch or less. For all lines over ¾ inch in diameter, AN and Rubber-Tek fittings are used. In addition, AN flared fittings are used where components are difficult of access and frequently uncoupled at hoses.

Four hydraulic power systems are installed on the aircraft:

General (Normal) – two engine-driven pumps, mounted one each on the inboard engines, which supply power to the general systems.

Auxiliary – one electrically driven pump which supplies power to the general systems or, under alternate operation, to the flap system and main gear down lock cylinders only.

Spoilers – one electrically driven pump which supplies pressure to the ground and flight spoiler systems only.

Rudder Standby – one electrically driven pump which supplies pressure, on a standby basis, to the rudder only.

In addition to these power sources, the ground spoiler system can be operated with pressure available from the general or auxiliary systems. Also, the main cargo door system may be pressurized by a hand pump.

For the General, Auxiliary and Alternate systems, engine-driven and auxiliary pump pressure is routed from the pumps to a system selector valve. The system selector valve is controlled by the hydraulic system selector lever in the flight compartment. From the system selector valve, fluid is distributed to the using systems and power manifolds.

Partial System Description – Jet Transport

Figure C-V-9.

for a jet transport are shown in Figs. C-V-8 and **C-V-9.** It is not necessary to study the system's operation in great detail, but a general knowledge of its operation is essential for a thorough investigation.

2.1. Visual Inspection and Documentation

Examine the hydraulic system from five angles: supply, pressure, control, protection, and application.

2.1.1. Reservoir and Fluid

Fluids can be discussed at length, but the main concern is the type of fluid required in the aircraft involved. The investigator should be familiar with the favorite hydraulic fluids in use and recognize them by color. These are the "5606" (red mineral oil), and the newer synthetics, Skydrol 7000 (greenish or yellowish-green), and Skydrol 500A (purple). **Figure C-V-10** presents a partial list of hydraulic fluids and seals with some rule-of-thumb notes.

Fluid samples should be obtained for at least a visual examination. Be sure to use clean glass containers for samples which can be taken from various points in the system, primarily from the reservoir or filter bowls. These samples should not be taken from actuators or other components since draining fluid from these sources may introduce contamination or destroy some evidence if malfunction of the unit is suspected.

Clean the exterior of the point of sampling to preclude contamination. Visual examination to detect the presence of gross contaminants approximately 40 microns in size, pieces of seals, metal, etc., might indicate the need for a contamination count and fluid analysis. **Figure C-V-11** shows a typical contamination report for a sample of Skydrol 500 hydraulic fluid. Notes should include the color and appearance of the fluid and whether or not any gross contaminants are present.

The manufacturers of hydraulic components can perform a hydraulic fluid contamination count and basic fluid analysis, but if Skydrol hydraulic fluid is to be completely chemically analyzed, it is recommended that it be sent to the Monsanto Company.

A check of the hydraulic schematic will reveal the number of reservoirs in the aircraft; some have as many as three. Locate the reservoir, check for fluid remaining, and document the amount. Examine the drain valve for the proper safety. Document external/internal damage of the reservoir, external fittings and plumbing, and internal filters. **Fig. C-V-12** shows a typical jet transport hydraulic reservoir and reservoir filter.

2.1.2. Pump Suction or Shutoff Valves

These valves are normally located on the bottom of the reservoir or just below. There may be one, or two, depending on the number of pumps being served by the reservoir, and they may be electrically or manually operated. Document the physical condition and position of the valve, for if found in the closed position, the reason must be determined.

If an electrically-operated valve is found open, it may be concluded that no problem which would require it to be closed existed prior to the accident. If, on the other hand, the valve is found closed, it was activated prior to the loss of electrical power at some time before the accident. Such a finding indicates a problem with the hydraulic system and points to the need for more detailed investigation in this area.

A manually-operated valve presents a little more difficult proposition. This valve may or may not be safetied in the open position. It may be cable-operated or activated by mechanical linkage. If the valve is found open and safetied, then probably no problem existed.

TYPES OF HYDRAULIC FLUID AND SEALS IN GENERAL USE

NON-FIRE RESISTANT TYPES

FLUID	COLOR	BASE	TYPE OF SEAL USED WITH	MAX. TEMP.	REMARKS
A.C. 3586	BLUE	VEGETABLE	NATURAL RUBBER	275°F	OLDER AIRCRAFT
A.C. 3586D (D = VIS. CHG.)	BLUE	VEGETABLE	NATURAL RUBBER	275°F	OLDER AIRCRAFT (CORROSIVE)
A.C. 3580	RED	MINERAL (PETROLEUM)	NEOPRENE OR BUNA-N	275°F	FORMERLY MIL-H-3580 OR AN-O-366 (VIS. CHG.)
MIL. H 5606	RED	MINERAL	NEOPRENE OR BUNA-N	275°F	
MIL. H 5606B (TYPE III)	RED	MINERAL	NEOPRENE OR BUNA-N	275°F	SUPER CLEAN FLUID

FIRE-RESISTANT TYPES (WILL BURN WHEN DECOMPOSED)

FLUID	COLOR	BASE	TYPE OF SEAL USED WITH	MAX. TEMP.	REMARKS
*SKYDROL 7000	GREEN	ESTER	EPR (ETHYLENE PROPYLENE RUBBER) OR BUTYL, TEFLON, NYLON, MYLAR	225°F	PRIMARILY IN PISTON TYPE A/C
SKYDROL 500A	PURPLE	ESTER	EPR (ETHYLENE PROPYLENE RUBBER) OR BUTYL, TEFLON, NYLON, MYLAR	225°F	PRIMARILY IN JET A/C (Lower viscosity for high alt.)
MIL. F 7083 (HYDRO-LUB)	YELLOW	WATER AND ETHYLENE GLYCOL 60/40	NEOPRENE OR BUNA-N	275°F	U.S. NAVY FLUID (Almost discontinued) (P2V-ONLY)

SPECIAL FLUIDS

FLUID	COLOR	BASE	TYPE OF SEAL USED WITH	MAX. TEMP.	REMARKS
SHELL #1 A.C.	YELLOW		AIR CONDITIONING & PRESSURIZATION UNITS	275°F	MIL. H 5606 Sometimes substituted
UNIVS 54			AIR CONDITIONING & PRESSURIZATION UNITS	275°F	
SKYDROL 7000	GREEN		AIR CONDITIONING & PRESSURIZATION UNITS	225°F	
SPACE AGE FLUIDS – SST – COOLANOL 45 – ORONITE 8200			ELASTOMETRIC METAL SEALS	350°-400°F	SST PROGRAM

GENERAL RULE OF THUMB

VEGETABLE BASED OILS WITH NATURAL RUBBER SEALS

MINERAL BASED OILS WITH SYNTHETIC RUBBER SEALS (NEOPRENE OR BUNA-N OR AN.-NUMBERED SEALS). SKYDROL OR ESTER BASED FLUIDS USE SYNTHETIC SEALS, (ETHYLENE PROPYLENE OR BUTYL) THESE HAVE NO AN.-NUMBER: THEY DO HAVE AN. SIZE DESIGNATIONS.

SOURCE: MONSANTO * CURRENTLY IN PRODUCTION

Figure C-V-10.

CONTAMINATION CONTROL DEPARTMENT
LABORATORY REPORT

Sample No. 2 Date Rec'd 11-29-65 Date Tested 11-29-65 11-30-65

Sample of Skydrol 500 System Elevator Control Left side Stand No. Return port #1

Part No. 68000-5003 Serial No. 337

Sample taken by Leo J. Anderson

Particle count per 10 ml sample. Method Visible Light Microscopy 60X & 150X

............cu. ft.

Micron size range	Particle count	Max. allowable	Type
5-10 5-15	Appears to be graphitic and rubber Too numerous		Metallic, Non-Metallic, Magnetic
10-25 15-25	"		"
25-50	31,064		"
50-100	1,090		"
Over 100	260		"
Fibers	3 Metallic Fibers 17		"

Inspected by H.E.M.

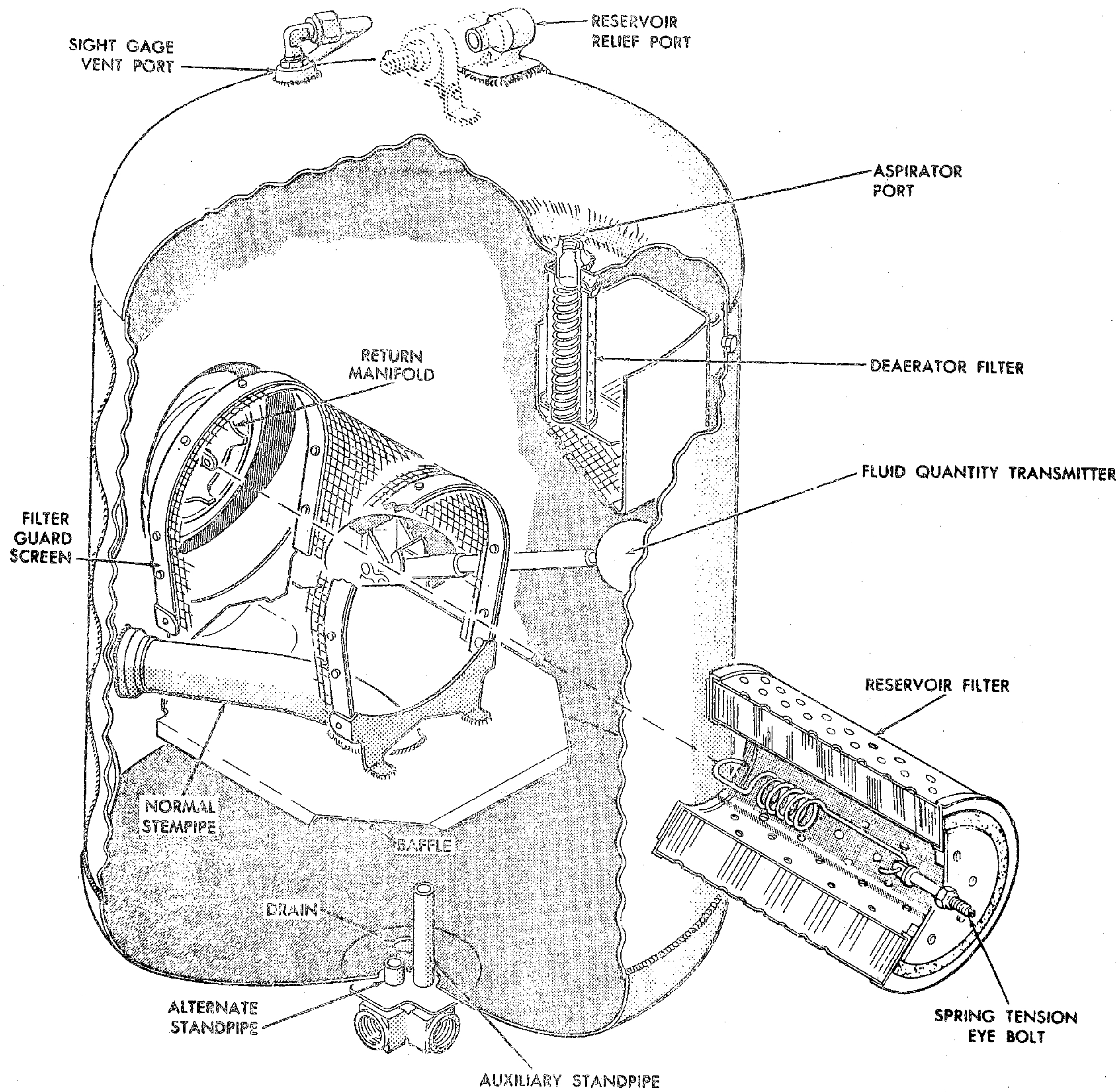
Remarks: Glass spheres 25μ to 85μ Dia
Three 100μ to 125μ (oblong)
525 counted. This count is not added to above counts

Largest metallic 550 Microns

Accepted by J. G. M.
Source Inspection

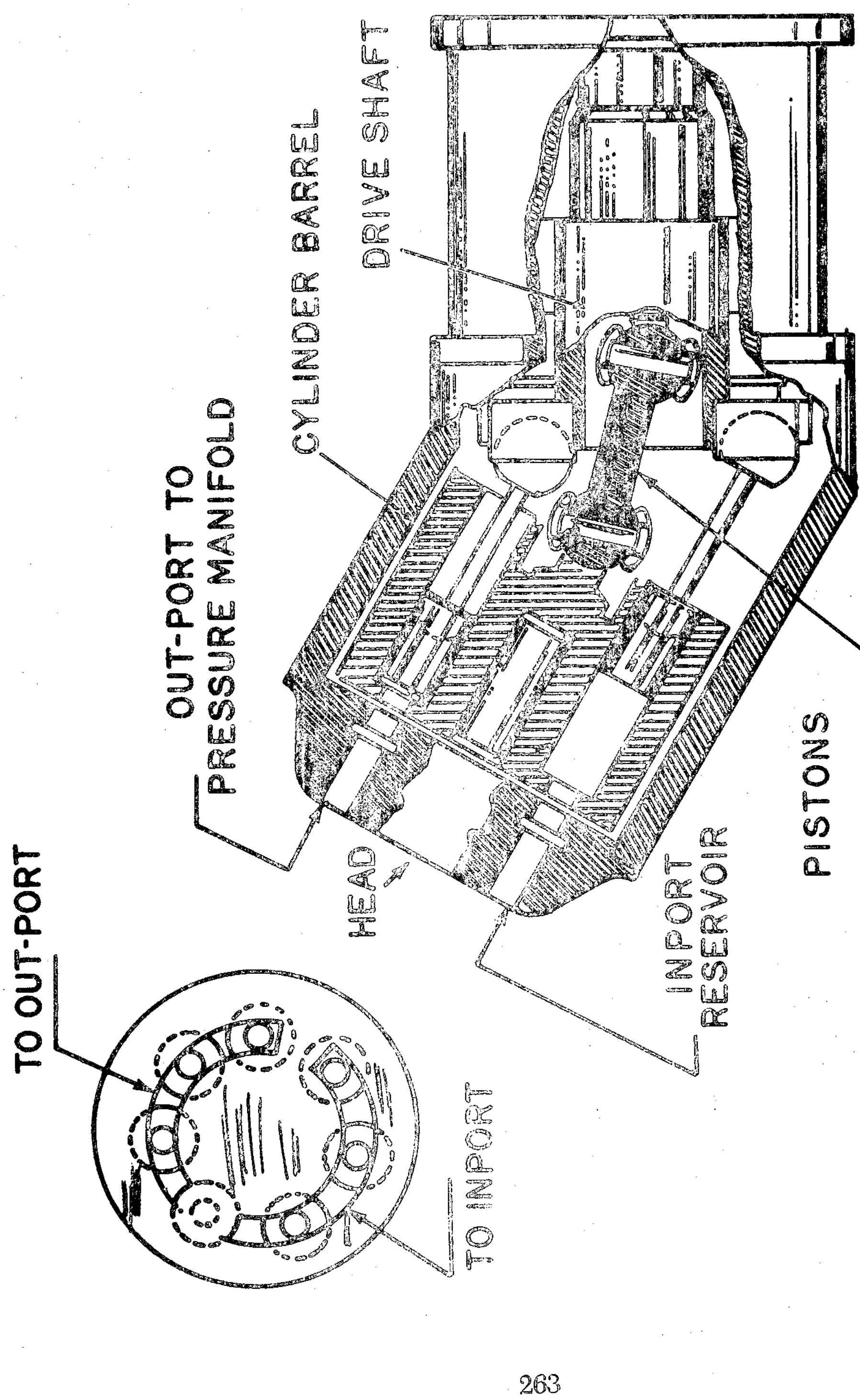
Submitted by H. E. M.
Contamination Control Dept.

Figure C-V-11.



Jet Transport Hydraulic System Reservoir and Filter.

Figure C-V-12.



Piston-type Pump Cross Section.

Figure C-V-13.

If the valve is closed and the safety is broken, investigate further. Examine the valve exterior for signs of foreign object damage affecting the safety and the closed position of the valve lever.

As in the case of all cable-operated or mechanical linkage-operated devices, it is possible that breakup of the aircraft could pull or jerk the cable or activate the linkages, thereby breaking the safety and driving the valve to the closed position. These areas must be thoroughly checked to determine whether or not the valve was intentionally closed prior to the accident. Document this further investigation thoroughly so that it can be readily proved in the factual or analysis reports.

2.1.3. Hydraulic Pumps

The hydraulic pump most common in aircraft today is the positive displacement piston-type. These pumps have a constant or variable output, depending on the system design requirements. See typical piston-type pump cross section (Fig. C-V-13).

The primary or utility system pumps are usually driven by the engines, but may be driven by electric motors as in the case of the L-188. The auxiliary system pumps are driven by electric motors. The hydraulic system schematic shows the number of pumps installed and the application of each.

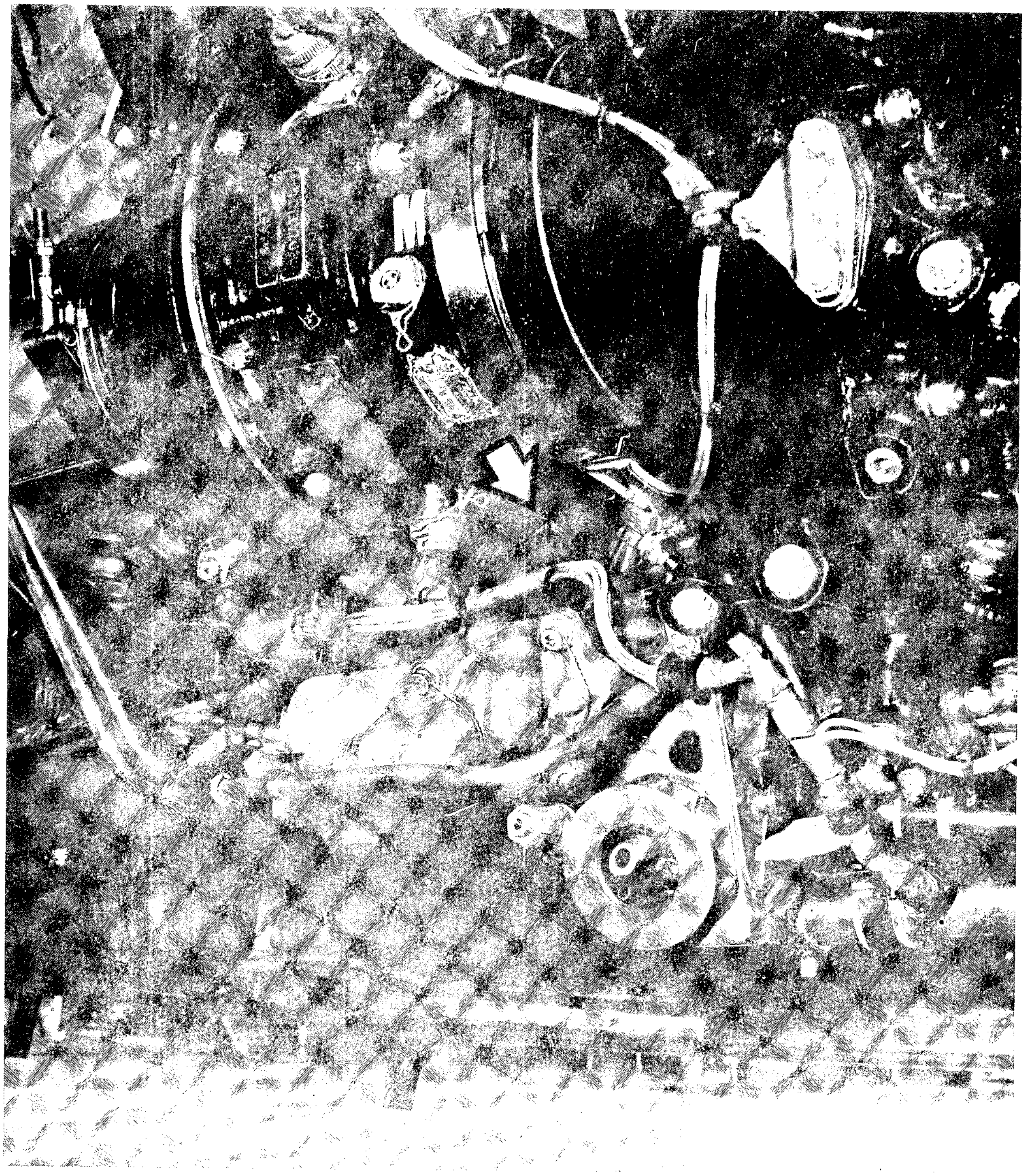
If the pumps are still mounted on the engine or gear box, begin documentation with complete identification. List the information on the data plates. Describe the external condition of the pumps, hoses, fittings, and the condition of the mounting. A suitably labeled photograph of the pump as found is recommended. See Fig. C-V-14 for example of a piston-type pump, from a jet transport, which failed in flight. Examine the pumps for evidence of overheat or exposure to externally applied heat, smoke, or fire. If such evidence is found, examine the area surrounding the pump mountings for similar evidence which will aid in determining the source of the condition.

After external examination and documentation the pumps should be removed for examination of the drive mechanism. The condition of the spline drive and the engine or gear box drive gear should be documented and photographed.

When documentation is complete and a functional test is anticipated, wrap and label the pumps for protective storage until the tests are arranged. If no tests are planned, remove the end plate from the pumps so that the pistons are visible. Rotate the spline drive coupling by hand in the normal direction of rotation and check the action of the pistons. If all rotate without evidence of internal distress, the pump is intact. Document the results, and note the condition of the pistons. If internal damage is evident or the pumps are binding, they should be completely disassembled. Note the presence of fluid, its color, appearance, and evidence of gross contaminants. Figure C-V-15 shows failed #2 engine-driven piston hydraulic pump partially disassembled, showing internal failure from four hours of operation without hydraulic fluid.

If the pumps are separated from their normal mounting, describe the degree of damage at the separation point. Compare the pump serial numbers with the maintenance records to identify the original position of the pumps, and document the conditions in the area of normal location. If the pump shows evidence of fire or heat, but there was no evidence of this condition in the normal location, the heat or fire occurred after the pump separated. Heat damage would be secondary to the crash, a fact which should be kept in mind when any loose systems components are examined.

Investigate the pumps operated by electric motors in the same manner as the engine-driven pumps. Fully document the condition of each unit. When feasible, these motor-pump assemblies should be tested to verify operation of the motors. If the motor and pump are separated, functional testing will be impossible because the pump will not be mountable in the test bench. However, it should be possible to test the motor. The pump should be partially disassembled to check operation of the piston while rotating the pump drive by hand.

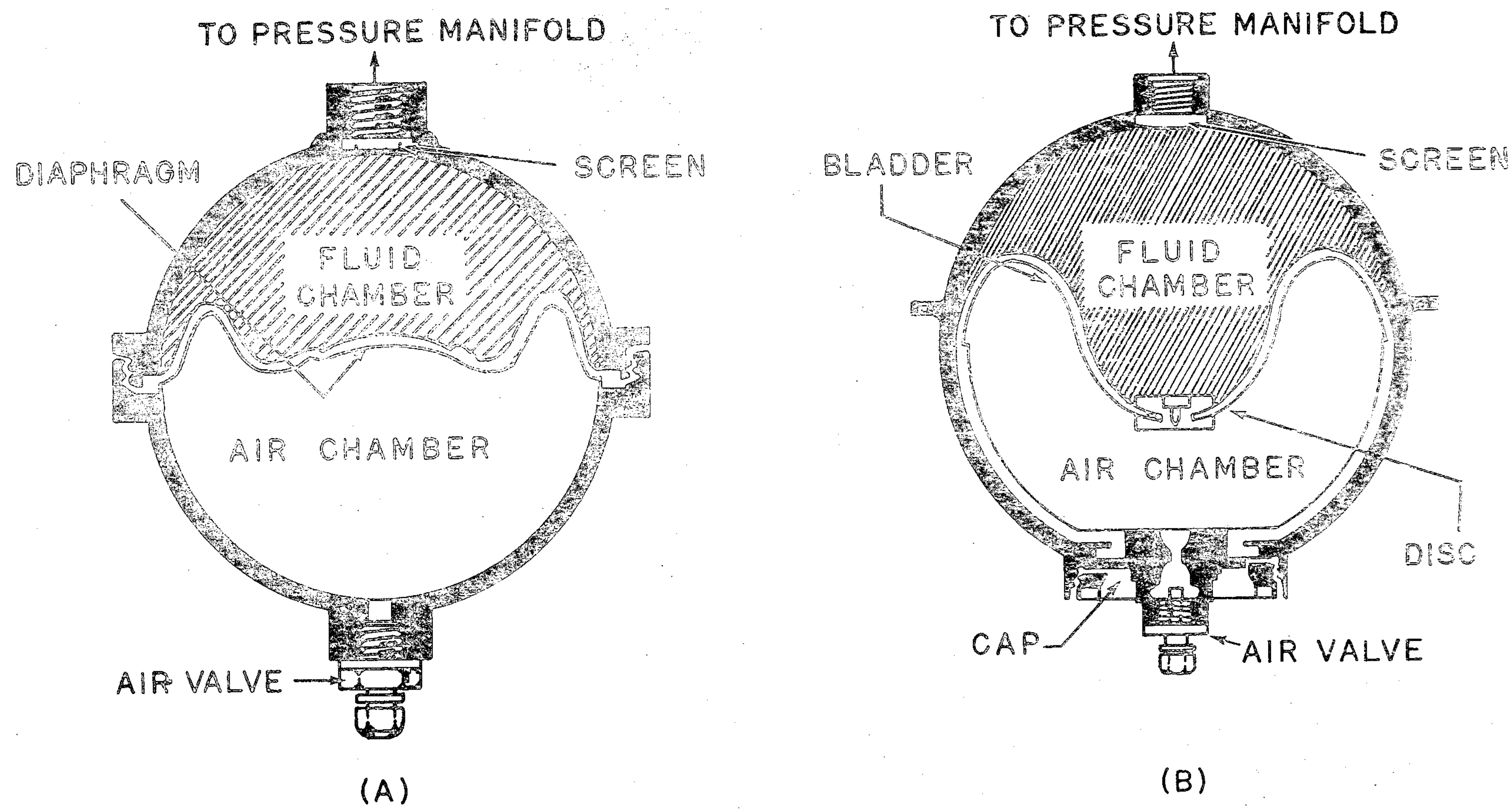


View of accessory section, lower inboard side #2 engine showing hole in #2 hydraulic pump.

Figure C-V-14.

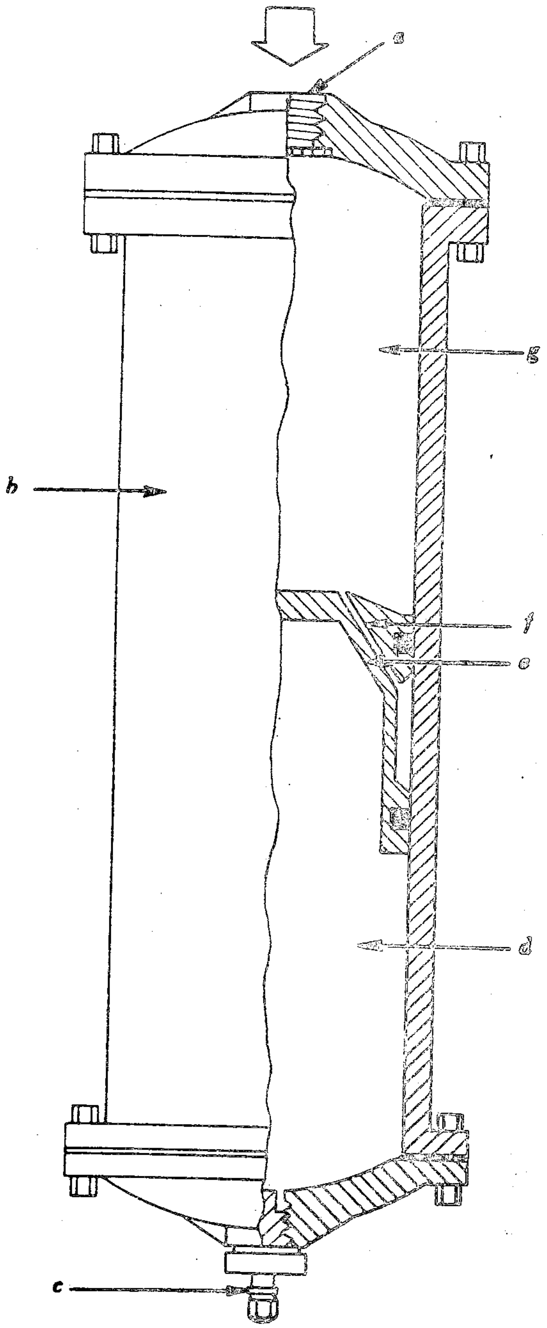
No. 2 engine-driven piston hydraulic pump (partially disassembled) showing internal failure. Nóte hole in pump case, and failed spline drive wired to top of pump.

Figure C-V-15.



Spherical Hydraulic Accumulators.

Figure C-V-16.



a. Fluid port
b. Cylinder
c. High-pressure air valve
d. Air chamber
e. Piston assembly
f. Drilled passage
g. Fluid chamber

Piston-type Accumulator.

Figure C-V-17.

2.1.4. System Control Components

These components are comprised of the pressure regulator, pressure relief valves, and the various check or non-return valves installed throughout the system. In general, these units yield no visual evidence of internal condition. However, they should be located, identified, and their external physical condition documented. These components can be tested if they are undamaged, but tests will probably not be conducted unless need is evidenced. These items should be kept in protective storage until testing is proved unnecessary.

2.1.5. Accumulators

The purpose of the hydraulic accumulator is to store fluid under pressure, in fact, the first output of the hydraulic pumps is used to charge these accumulators. The initial pressure for operating a selected hydraulic actuator is provided by the accumulator. The hydraulic schematic shows the number of accumulators in the system and specifies the air preload necessary for proper operation.

Accumulators are of two basic types, spherical and cylindrical. The spherical type is equipped with either a bladder or a diaphragm to separate the air preload from the fluid. See cross-sectional views of two spherical accumulator types, (Fig. C-V-16). The cylindrical type has a floating piston which separates the air and the fluid. See cross-sectional view of a cylindrical accumulator (Fig. C-V-17). In some installations an air-charging valve is mounted on one end of the accumulator. In others, the accumulator may not be readily accessible for charging so a line is connected to a remote air-charging valve. This installation usually has an air pressure gauge teed into the line.

All accumulators must be accounted for, identified as to their original location in the system, and examined. Document and photograph the external condition. If the accumulator is intact and the air-charging valve is present, handle the unit carefully; the air preload may still be present in amounts from 150 psi to 1000 psi. This unit should be removed from the wreckage for personal safety reasons and should be taken to a hydraulic shop where the proper tools and trained personnel are available.

If the unit is damaged, or if the air-charging valve or line is missing, and it is certain that the air preload is dissipated, disassemble the unit to examine the bladder, diaphragm, or piston. Check the spherical accumulator for evidence of internal leakage by looking for fluid on the air side. Some fluid may slop into the air side when opening the diaphragm type but if operating normally it should be dry. The bladder type should be completely dry inside. If there is a hole in either of these types, the air preload will be dissipated into the system, causing sluggish operation.

The cylindrical type accumulator may not be so easy to check for definitive evidence of leakage since the piston is free to move in either direction and will have some fluid on the walls of the cylinder as a matter of course. Documentation of this type accumulator should include the position of the piston at the time it was recovered, and whether or not it was free to move within the cylinder.

If the spherical accumulator does not yield evidence of internal leakage and there was an air charge present, conclude that it was capable of proper operation. If it is intact with no air charge present, then it can only be assumed (with lack of contrary evidence) that the air charge was present and the unit was working properly prior to the accident. However, it is well to hold this in abeyance until the remainder of the system has been examined. The only effect of a "flat" accumulator (no air preload) on the system is the slightly longer time required to operate a hydraulic component, but this should be kept in mind.

The cylindrical accumulator may be a little more helpful to the investigator. Because these units are generally located on or near heavy structure, it is possible that the cylinder may be crushed or otherwise deformed during aircraft breakup. This deformation traps the piston in its last position. If it is trapped at the fluid end, conclude that hydraulic pressure was lost prior to the deformation and the air charge pushed the piston to that end. It proves that an air charge was present prior to the accident. If the piston is trapped at the air end of the cylinder, conclude one of two things, either that the air preload was absent and the piston was being held in this position by hydraulic pressure normally, or that the air preload was lost due to aircraft breakup before hydraulic pressure was lost. Make a final determination when the rest of the system has been examined.

2.1.6. Hydraulic Manifolds and Pressure Modules

The hydraulic system manifold, such as used on many aircraft, contains such items as the pressure regulator, system pressure relief valve, system pressure bypass valve, and one or more hydraulic component selector valves. This unit is the distribution center for hydraulic service. It may also contain filters.

The hydraulic pressure modules serve somewhat the same purpose in the turbine-powered aircraft and contain some of the same components, with the exception of hydraulic selectors which are located separately but are supplied by the modules. Some of these modules have a manual shutoff valve while others will have an electric motor-operated shutoff valve.

Every effort should be made to recover and identify these components. Document and photograph their condition, noting especially the positions of any valves. The filters should be inspected visually for any contamination and this inspection should be documented.

2.1.7. Component Selector Valves

Account for all selector valves (landing gear, wing flaps, spoilers, etc.) and document their positions and condition. This can be related to the positions of their respective units later as the investigation progresses.

2.1.8. Actuators

The positions of various hydraulic actuators can be very important. The majority of actuators consist of simple cylinder and piston assemblies, others consist of hydraulic motor-operated jackscrews.

The cylindrical actuators should be measured for piston rod extension, and their overall condition documented. In many cases the piston rods are bent, and their surfaces exhibit some impression caused by forcible contact with the edge of the cylinder. These conditions are caused by impact forces acting against fluid trapped in the cylinders. Since the fluid will not compress, the rods bend. Measuring the extended length of the rods gives information on the position of the component operated by that actuator, when compared with a normal piston rod extension. See photograph (Fig. C-V-18) of bent hydraulic actuators from a jet transport involved in a catastrophic accident.

All actuators do not move in the same direction to extend or retract a component. Some pull the piston inward to extend, and push it outward to retract. Always be certain of the direction of movement before making a statement.

2.1.9. Filters and Contamination

There are many types of filters; they may be throwaway or reusable types. They also vary depending on the desires of the particular operator, but they all serve the same purpose to keep the hydraulic system free of harmful contaminants. Fig. C-V-19 shows a jet transport main hydraulic filter with broken clamp ring which caused hydraulic system failure.

If the hydraulic system is found intact, it is advisable to conduct an operational check using an auxiliary source of hydraulic pressure in lieu of the engine-driven pumps. This might occur in an incident or minor accident, in which case, wait until the check is completed before removing the filters for examination. The reason for this delay is obvious, for when a circuit is opened and hydraulic fluid is lost, air enters the system. The entrapped air may then alter the operating characteristics of the system. Pressure bleeding may result if the air is entrapped and unable to return to the reservoir.

When checking the system filters, foreign material such as bronze, steel, or synthetic rubber may be found, which indicates failure of system components. The degree of failure is proportional to the quantities of materials lodged in the filters. Steel particles are indicative of failure of bearings in the piston-type hydraulic pump and failure of gear teeth in the gear-type pump. Bronze particles indicate failure of the rotating cylinder block of the piston-type pump and in the case of the gear-type pump, the bronze is from the inner surfaces and bushings. Synthetic rubber particles indicate failure of packings, such as chevrons and o-rings in the actuating units, and may also indicate failure of the seals in the selector valves.

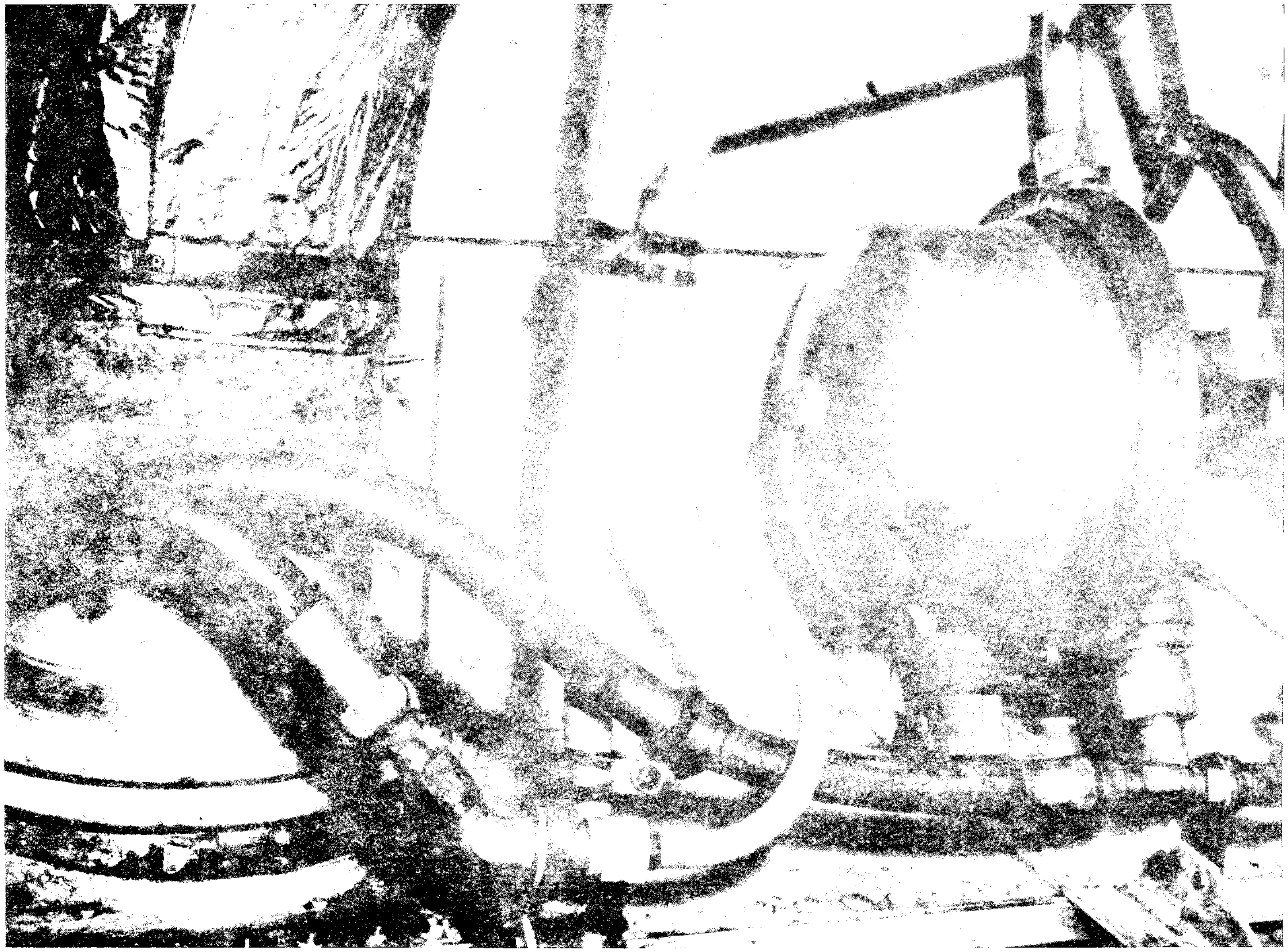
Addition of improper fluid to the hydraulic reservoir can result in the destruction of packings and seals and in the formation of gum deposits which lead to the plugging of small restrictions and orifices. Some of this gummy deposit may be found in the filter along with portions of seals and packings, which brings the condition of the fluid under suspicion. Filter units are prime locations for obtaining fluid samples for examination. Minute bronze particles in suspension in the fluid are not necessarily an indication of failure of a unit but generally result from the normal wear-in of the hydraulic pumps.

2.1.10. Plumbing

In most cases, the hydraulic system plumbing is severely disrupted. Stainless steel and aluminum are used in both tubing and fittings. As much of this plumbing as possible should be recovered and examined. Examine the tubing for evidence of chafing. If such evidence is found, determine prior location of the tubing to learn why chafing occurred. Such determi-

Figure C-V-18.

Measurement of hydraulic actuator rod extension length can help determine component position.



View of main hydraulic filter, left wheelwell, showing failed clamp ring (arrow).
Figure C-V-19.

nations can result in recommendations for corrective action, thereby alerting the FAA and the operators to a poor if not dangerous condition. Examination of the plumbing becomes even more important if it is known that the aircraft had previous hydraulic problems. Look for conditions of bursting tubing which would indicate high pressure failure. If such evidence is found, determine the location of the line in the aircraft and why it burst. See examples of tubing burst by high pressure and by high temperature (Fig. C-V-20, 20a, 20b).

Examination includes a security check for proper tightness of fittings while still connected to a component. Any damage or unusual condition of the fittings or tubing should be documented by photographs as well as by notes. When fittings have been exposed to heat or fire, they may be loose or only finger tight. This evidence should be documented with appropriate notes, but do not arbitrarily conclude that the fitting was improperly tightened. Heat will loosen fittings, therefore, reserve judgment until all of the facts concerning the system are accumulated.

During examination of fittings look for cracks or other overtorquing indications. With the fittings in use today most leakage is the result of improper installation, malformed or damaged seats or seals, etc.

2.2. Functional Check or Test of Component or System

Through consultation at the scene, decide definitely the need for testing of each component to determine its operational capabilities. A fullblown operational test of a unit may be indicated, or a plain functional test may suffice.

If the unit to be tested is still attached to the airframe, great care should be exercised during its removal because:

- Hydraulic fluid can be very toxic. Take every precaution to protect the skin, and the eyes in particular, because the fluid causes extreme irritation. This cannot be overemphasized.
- It is advisable to cap, plug, or otherwise seal off each port as soon as each fitting is disconnected, to avoid contamination of the unit.
- Care must be exercised to prevent further damage which might render the unit unfit for testing.

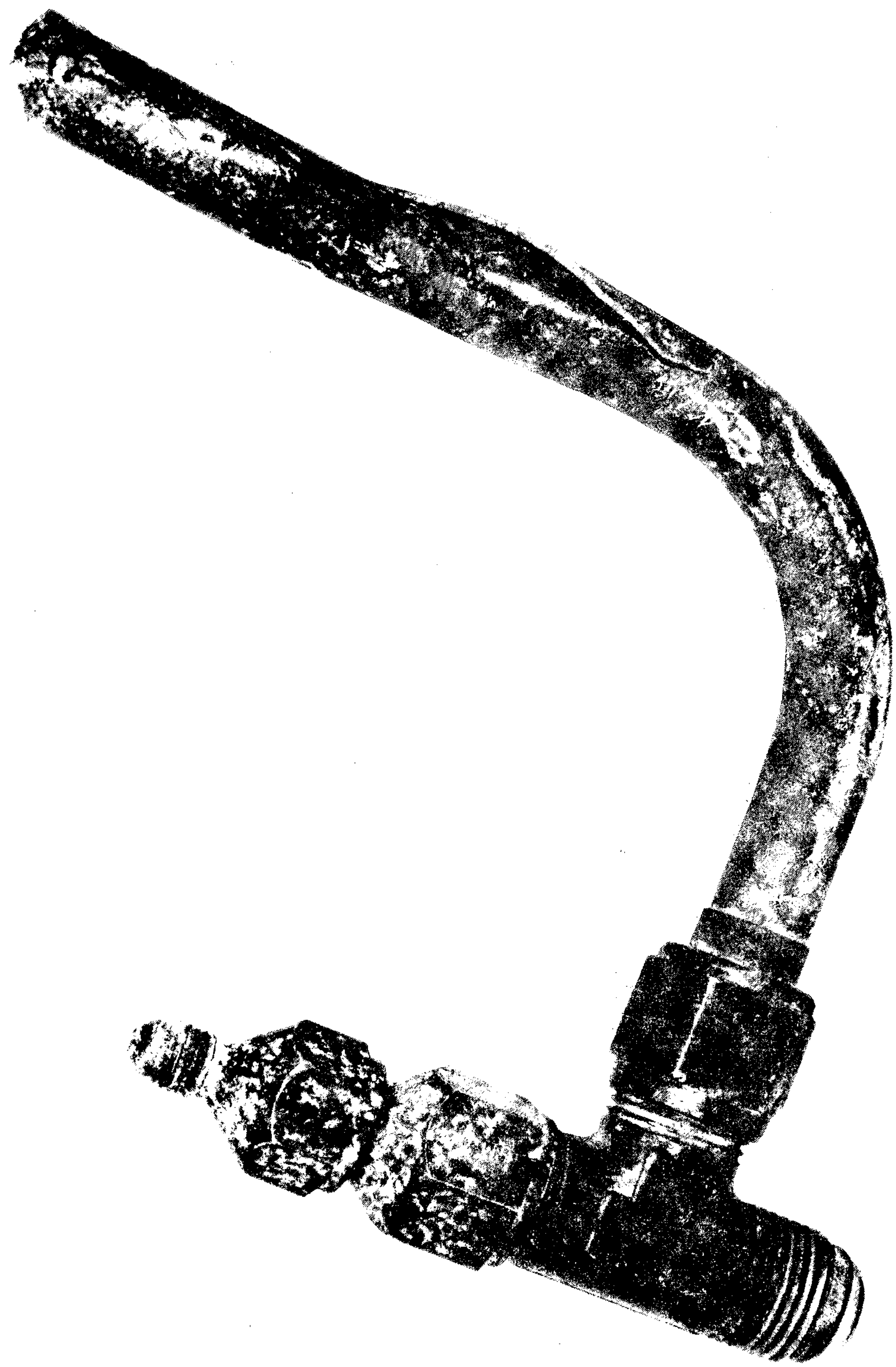
If the unit is to be shipped to the chosen facility or to a local NTSB or FAA office for holding, each large unit should be packaged separately and securely in a wooden box. Smaller items can be packaged together if adequately identified and properly separated. Each box should be given an identifying mark and listed for later reference and accountability.

It is important that the first fluid from the return port be caught because this is part of the operating fluid and it will contain any contaminants. If the unit was not operating properly prior to the accident because of foreign matter lodged in a valve or passage, this material may be flushed out by the new fluid and lost in the process. Be certain that each sample is properly identified by source. Notes should reflect the manner in which the sample is obtained and its appearance.

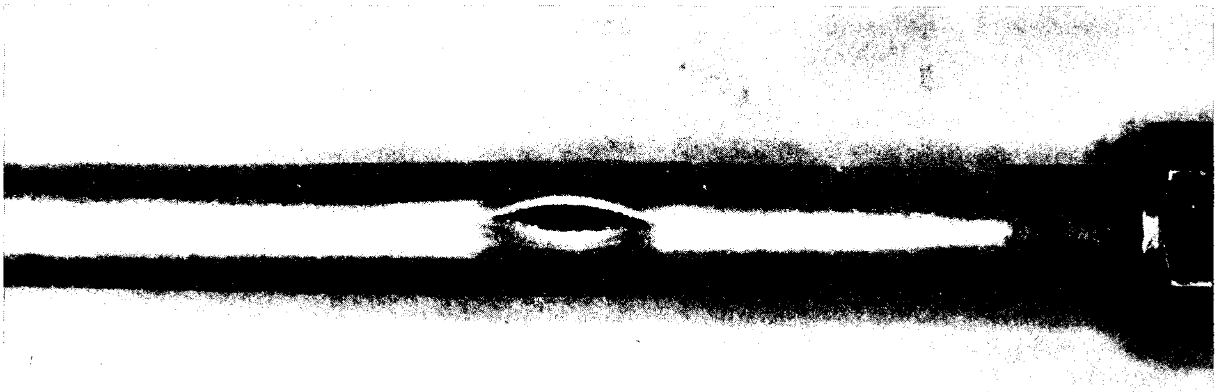
As the testing progresses, stop the proceedings at any time a question arises about the test methods or results. Be sure that everything about the test is understood and that the desired information is secured. One questionable action can render the test results invalid and necessitate another test.

Take time for a consultation when a discrepancy appears, to determine its effect on the unit operation and the operation of the aircraft. Cover the discrepancy in the notes with an explanation of its effect. Plan for disassembly of the unit for further examination following the tests if a significant discrepancy is found. The cause of such discrepancies must be determined, thoroughly documented, and photographed.

A nearby airport or Air Force Base hydraulic shop can accomplish tests of items which do not require a fullblown test, but on which a determination of operability is desired. Find one that uses compatible fluids and pressures, so that settings on pressure regulators and pres-

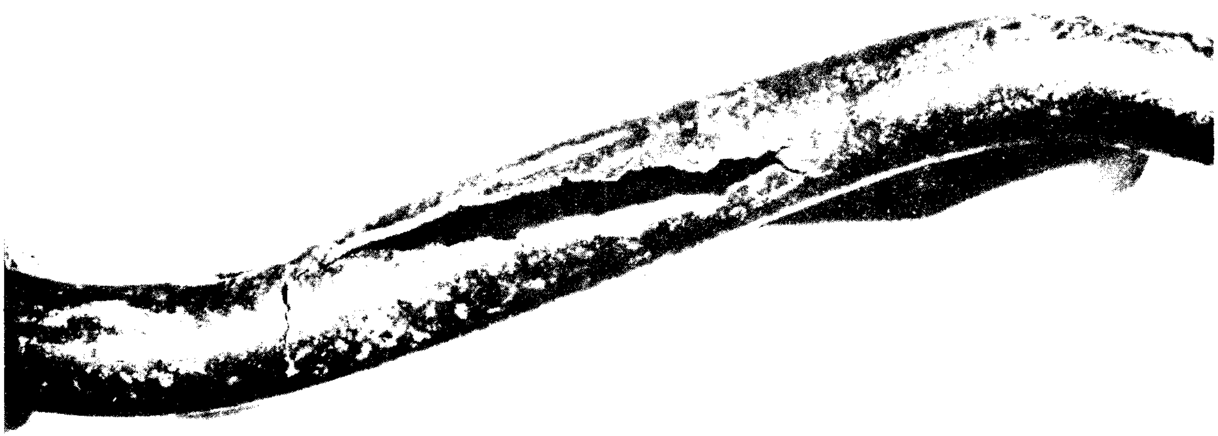


Failed hydraulic tubing X 3/4.
Figure C-V-20.



Unused tubing shown after subjection to hydraulic bursting test at room temperature X 1.

Figure C-V-20a.



View of longitudinal break in the tubing shown in Fig. C-V-20. X 1.

Figure C-V-20b.

sure relief valves can be determined. Simple check valves can be examined visually.

When tests are completed, assemble the Group for consultation. Discuss the notes and the general test results. Everyone should concur in the assessment of the actual condition and capabilities of the unit tested. Give everyone concerned, including the facility representatives, a copy of the notes and the test results, then arrange for disposition of the unit. It is advisable to arrange for disposition of the unit prior to departure from the accident scene. Learn where the unit is to be stored if it is separated from the other wreckage.

2.3. Disassembly and Minute Inspection of Components

Disassembly and minute inspection of components in the field will be considered, since disassembly at a testing or overhaul facility is performed by trained personnel using the proper tools. Booby traps which may be encountered will be discussed.

Exercise caution when actuating a loose hydraulic actuator or other movable unit because sufficient fluid may remain in the unit to spurt from an open port. This fluid is toxic and should be kept away from clothing, skin, and eyes.

Some hydraulic units contain powerful springs, particularly those incorporating pressure relief valves. If these units are opened for any reason, be sure to take restraining precautions to prevent injury to nearby personnel.

Items which require disassembly in the field are accumulators, selector valves, and other units which may be unfit for testing or on which testing may not be desired. Adequately explain the reason and the steps taken for disassembly when writing up the notes. Photographs are desirable to record the internal condition of the units and to supplement notes. Disassembly may be partial or complete, depending on information desired.

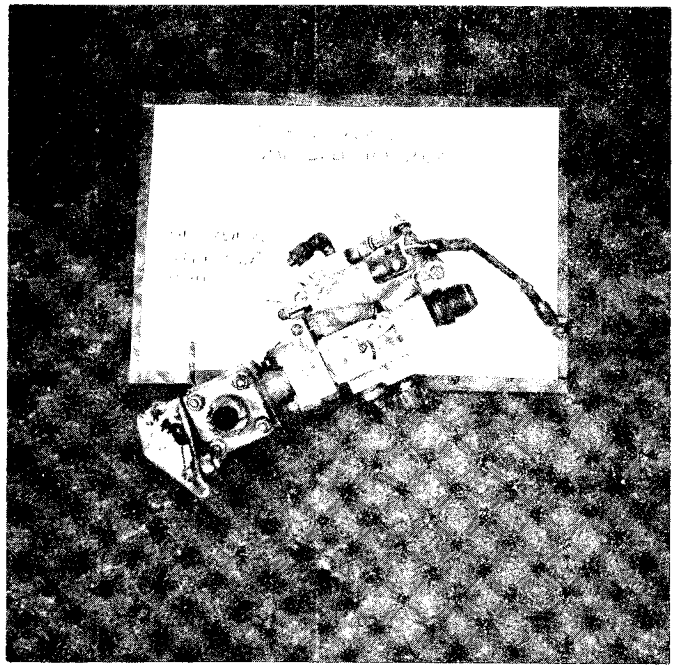
Spherical accumulators are simple to disassemble if they are the diaphragm type because the hemispheres are bolted together. Be sure that no air preload remains before you start. The bladder type will be difficult to handle because the bladder is secured by a large screw-type plug which requires the use of a special tool. This is also true of the cylindrical type which has large end caps screwed on.

If there is no other way to open one of these units, it may have to be cut into with a saw in order to view the interior. A visual examination will reveal any free fluid on the air side of the diaphragm or bladder, and will also show the position of the floating piston. No free fluid should appear on the air side of the piston.

Selector valve positions are not often reliable indications of the positions of their related components at the time of an accident, particularly when impact forces and disintegration of the wreckage are severe. Many of the selector valves and bypass valves are remotely controlled from the cockpit by cable and bellcrank systems or are electrically actuated. Settings of mechanically controlled valves are apt to be changed should the mechanical operating linkages be torn from the valves. Solenoid operated valves, if in the energized position at the time of impact, will be found in the spring-return position after impact forces have destroyed the electrical system unless physical damage prevents return action. Valves which are operated by electric motors will remain in their final settings when the electrical power is lost.

Some selector valves will have to be opened completely in order to determine the position of slide valves inside. On others, the slide may be viewed directly through open ports to see which one is closed off. Prior to disassembly, mark the positions of any movable linkage or slide valve extensions for reference after the valve is opened. These marks allow relation of the valve position to the pressure and return ports. Notes and photographs should cover the external condition of the valve before disassembly. **Figure C-V-21** shows the emergency hydraulic pump selector valve from an airline jet transport with the selector slide valve in the "normal" "brake-only" position, giving evidence of no hydraulic distress.

After the valve position is documented, use a magnifying glass (8-10x) to examine the edges the slide valve lands and the edges of the



Emergency hydraulic pump selector valve.

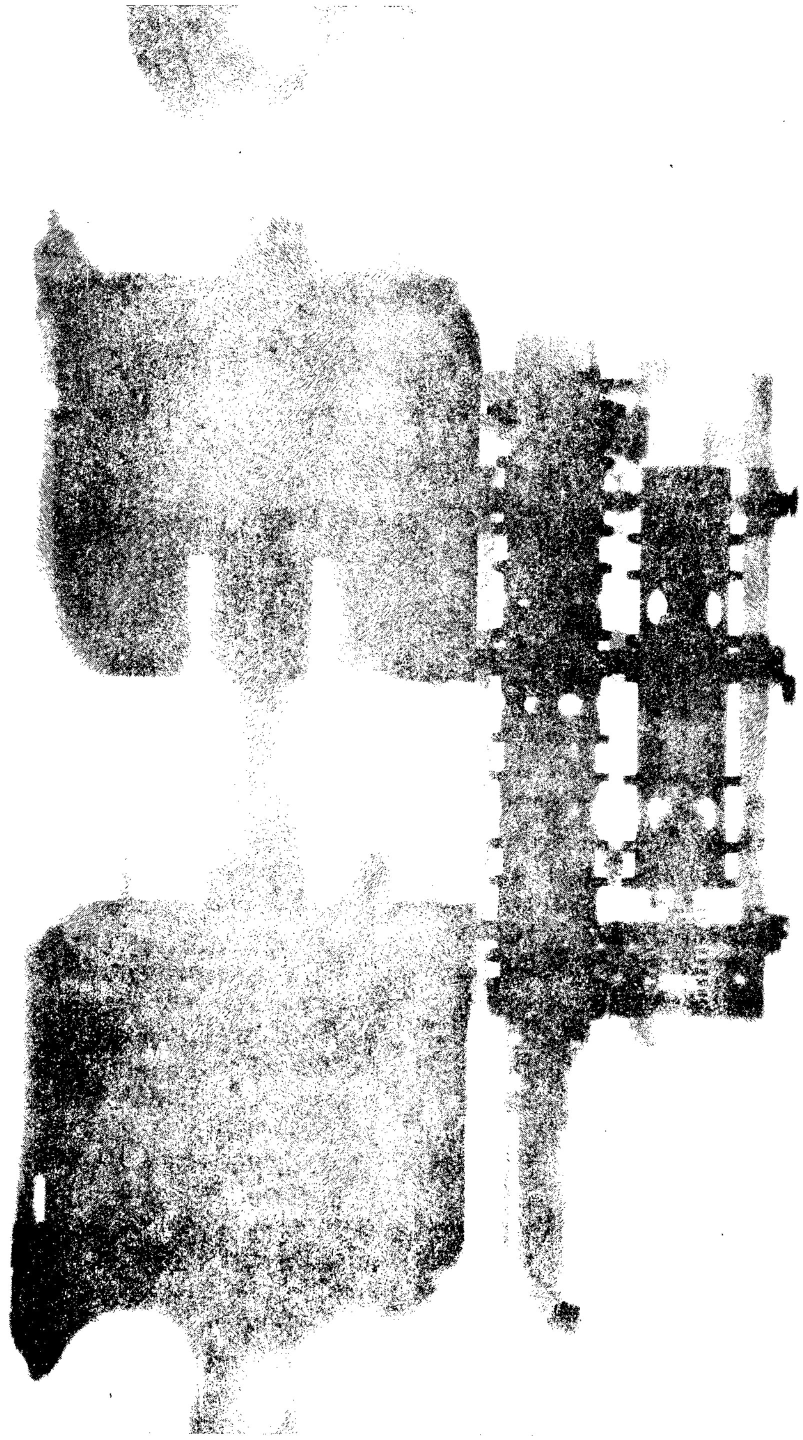
Figure C-V-21.

pressure ports for evidence of cavitation. Monsanto research has discovered that the damage in this area will appear as a localized area of erosion on the sharp edges of the slide valve lands and a corresponding area of the fluid ports, which leads to internal leakage and sluggish operation.

If selector or control valves are damaged to the extent that they cannot be disassembled, and determination of their internal condition and position is considered important to the investigation, they should be X-rayed. This should be done by a competent aviation or industrial X-ray technician. Many airframe and component manufacturers, airline shops, general aviation repair stations, and military facilities have the necessary equipment and personnel to accomplish this work. Interpretation of position of slide valve lands and ports will require use of technical data or the services of an expert on the particular system. **Figure C-V-22** is an X-ray view of the rudder power control slide valves and actuator from an airline jet transport aircraft.

Rubber or teflon seals should be examined for conditions of swelling, cutting or rolling, and for deterioration or breaking. The condition of the internal mechanism should be documented, and any unusual conditions photographed.

All of the above applies to internal examination of the hydraulic pumps, pressure modules, etc. Disassembly, partial or complete, may be necessary in order to determine positions and conditions. This can be done at times in the field using simple tools, or it may be necessary to use some nearby shop facilities. Any hydraulic pump or motor should also be checked for bearing wear, looseness or damage. Wherever, or however it is accomplished, remember to take every precaution to prevent injury to personnel. Document and photograph first, then open the unit and document again. If an opened unit yields evidence of importance, make sure that it is put into protective storage.



X-ray view – Rudder power control slide valves and actuator from jet transport.
Figure C-V-22.

3. AC and DC Electrical Systems — General

Until the advent of the turbine-powered aircraft the basic electrical power system was DC, with inverters to supply the necessary AC power for certain components. The majority of turbine-powered aircraft now incorporate engine-driven, constant-speed alternators referred to as AC generators; the necessary DC current is supplied through transformer rectifiers, an arrangement which provides systems designers with several advantages, to wit:

- AC can be increased or decreased very efficiently through transformers.
- AC systems use simpler units which are less prone to trouble than DC devices.
- AC components, including the miles of wire required for operation, are lighter.
- AC components operate more efficiently at high altitude.

For review, a few basic AC electrical principles of aircraft generating systems are outlined in Fig. C V-23, 23a.

Discussions will include supply and pressure provided by the generators, inverters and alternators; control, provided by transformers, rectifiers, and the regulators for voltage and frequency; protection, provided by fuses, circuit breakers, and other protective devices; and the electrically-operated components. Use of volt-ohm-milliammeters (VOM) at the scene will be briefly considered.

As in all systems investigation, request a detailed copy of the appropriate schematics, in this case, the schematics for AC and DC power distribution. This will provide the basic components of each system, and show how they are tied together. Also request from the owner/operator a wiring diagram manual for the aircraft which will indicate the wire numbers and gauges as well as electrical connector identifications. These will be necessary for proper circuit tracing. Figures C V-24, 24a show a simplified AC and DC electrical power system schematic for a typical jet transport, an explanation of abbreviations, and a list of basic components.

The term *electrical connectors* should be used instead of *Cannon-plug* because the latter is a trade name, and this type of connector is now produced by many manufacturers.

Generally, electrical distribution is centralized near the cockpit or beneath the floor, and except for engine-driven units, search for electrical system control components can be narrowed by digging into this part of the wreckage. Almost all the electronic components are located in racks in the cockpit or beneath the floor. Figure C V-25 diagrams the location of electrical and electronic equipment in a typical turboprop transport. Search for systems components along the wreckage path may yield a few of these, and many of the electrically-operated units such as fans, motors, etc. Examine them internally for evidence of electrical overheat or burning, and externally for other conditions such as exposure to heat or fire. Examine the area where these components were originally located to determine the extent of the damage.

Throughout this portion of the investigation, thoroughly document the internal and external condition of all the parts associated with the electrical and electronic systems. Make notes regarding the fitness of these units for later testing, and take usual precautions for protective storage.

3.1. Visual Inspection and Documentation — Generators, Inverters, Alternators

DC generators are driven directly by the accessory section of the engine or by an engine-driven gear box. If the relatively undamaged generators are still attached to the drive mechanism, plans should be made to test them for proper output. If nearby facilities are unavailable, and the Powerplants Group intends to take the engines to an overhaul facility for teardown, it may be expedient to coordinate with the Powerplants Group to have the generators tested, and provide the Systems Group with the notes on these tests.

Request from the owner/operator appropriate overhaul specifications to be used as guides for the tests if the testing is the investigator's

Alternating Current Frequency

In an a-c circuit, the voltage and current build up from zero to a maximum of one polarity, then decay to zero, build up to a maximum value of the opposite polarity, and again decay to zero. This sequence of build up and reversal is called a "cycle," and the number of times it occurs in one second is called the "frequency." It is illustrated by a typical sine wave *(see figure 24).*

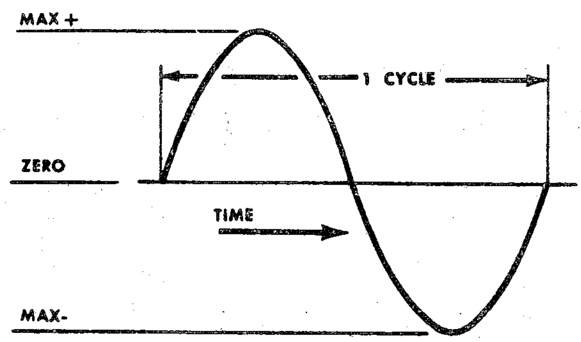


Figure 24 — Typical Sine Wave

In a conventional generator, the frequency is dependent upon the speed of rotation and the number of poles in the generator. Two poles must pass a given point on the stator every cycle. Consequently,

$$\text{Frequency (cps)} = \frac{\text{RPM x Pairs of Poles}}{60}$$

For example, with a 6-pole generator operating at 8000 rpm,

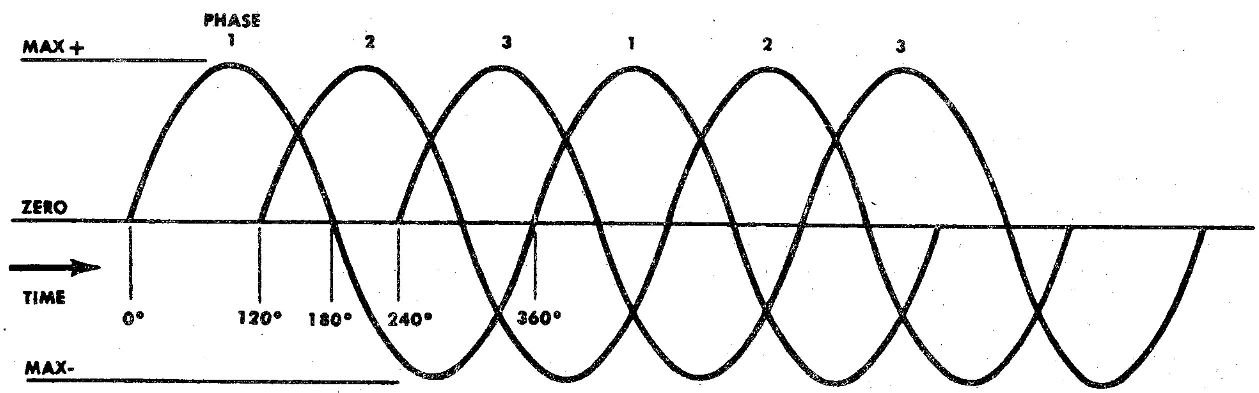
$$\text{Frequency} = \frac{8000 \text{ x } 3}{60} = 400 \text{ cps}$$

For aircraft constant frequency systems, 400 cps has been adopted as a standard.

Phase

The term "phase" indicates the number of alternating currents being carried simultaneously by the same circuit. "Degrees of phase" is used to designate the cyclic difference between the multi-currents in one circuit, as discussed below under "Phase Sequence."

A single alternating current is termed "single-phase" current, whereas several currents differing in phase are "poly-phase" currents. In a 3-phase system, the three currents differ in phase from each other by 120 electrical degrees, as shown in figure 25. A 3-phase generator may be considered as three single-phase generators combined in one machine.



Combined Sine Waves of a 3-Phase System

Figure C V-23

It takes a minimum of three wires to deliver 3-phase service, but the number of wires in a service should not be confused with the number of phases. For example, a home may have 3-wire service but in all probability it is single-phase, 115/230 volts.

Phase Sequence

In figure 25, phase 1 starts at zero and builds up from there. Phase 2 starts to build up 120 degrees later, followed by phase 3 which starts 240 degrees from time zero. Thus, the sequence is 1 - 2 - 3. At 360 degrees from time zero, phase 1 completes its cycle and the procedure is repeated.

The output terminals of generators are marked to show the phase sequence, and these terminals are connected to load busses which are marked accordingly. Usually they are labeled A - B - C to indicate a sequence of 1 - 2 - 3.

Volt-Amperes

In an a-c circuit, the product of voltage x amperage is volt-amperes (va), frequently referred to as "apparent power." In a d-c circuit, this product is "power" expressed in watts. Kilovolt-amperes (kva) is volt-amperes expressed in thousands, that is, 20,000 va = 20 kilovolt-amperes or 20 kva.

Winding Connections

The windings of 3-phase electrical apparatus such as generators, motors and transformers can be connected in various configurations, including the "wye" illustrated in figure 26. The phase-to-phase voltage of the wye-connected unit is the vector sum of the voltage of two windings, which is the square root of 3 multiplied by the phase-to-neutral voltage. For example, if the phase-to-neutral voltage is 115 volts, the phase-to-phase voltage equals $115 \times \sqrt{3}$, or 199+ volts. This is usually indicated as 115/200 volts.

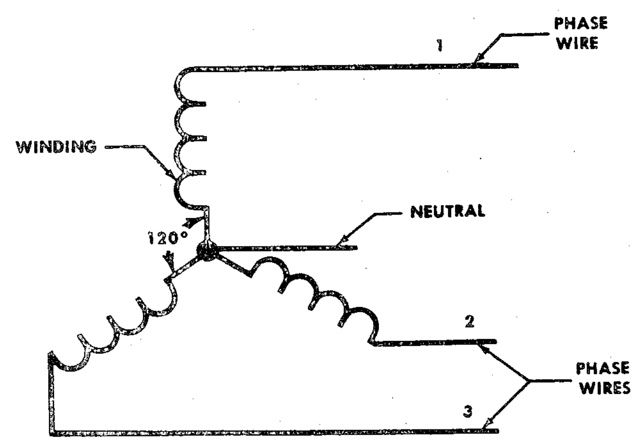
Most aircraft generators are wye-connected, with the neutral attached to structure. Then they are referred to as 3-phase, 4-wire generators, the aircraft structure serving as the fourth wire.

When making power feed connections to frequency-sensitive apparatus such as a 3-phase motor, the phase sequence must be a specific way in order to obtain the desired operation. That is, a 3-phase motor will rotate in a certain direction when properly connected, but will rotate in the opposite direction if any two of the three power feed wires are reversed, or if the phase sequence of the power source is reversed.

The neutral point of a wye-connected 3-phase motor may or may not be grounded. If one of the phase wires of an *ungrounded-neutral* motor is disconnected, with the two remaining wires connected, it operates as a single-phase motor. It will not start, but if already running it will continue to run provided the load is not too great. With any appreciable load, it will soon overheat, and obviously it will overheat if a single-phase start is attempted.

On the other hand, if one of the phase wires of a *grounded-neutral* motor is disconnected, leaving the two remaining phase wires and the ground wire connected, it will operate as a variation of a 2-phase motor. It will start, run and carry load, although its load-carrying capabilities will be somewhat reduced from that as a 3-phase motor. For this reason, certain of the important load motors, (such as fuel pumps) have a grounded neutral and are so designed as to perform reasonably well when operating 2-phase.

Single-phase service can be obtained from a wye-connected 3-phase system by connecting between two of the phase wires or connecting from one phase wire to neutral (or to ground), but 3-phase service cannot be obtained directly from a single-phase system.



Wye-Type Winding

Figure C V-23a.

Summary of System Components

For quick reference purposes, a summary of the electrical power system components is presented on pages 2 and 3. The operational simplicity of the system is apparent in the schematic *(see figure 1)*, and the table of terminology on page 3 indicates the origination and significance of abbreviations used in the schematic. By means of the numerical cross reference provided between this table and the basic component list on the same page, it is possible to tell at a glance the characteristics of any given component in the schematic.

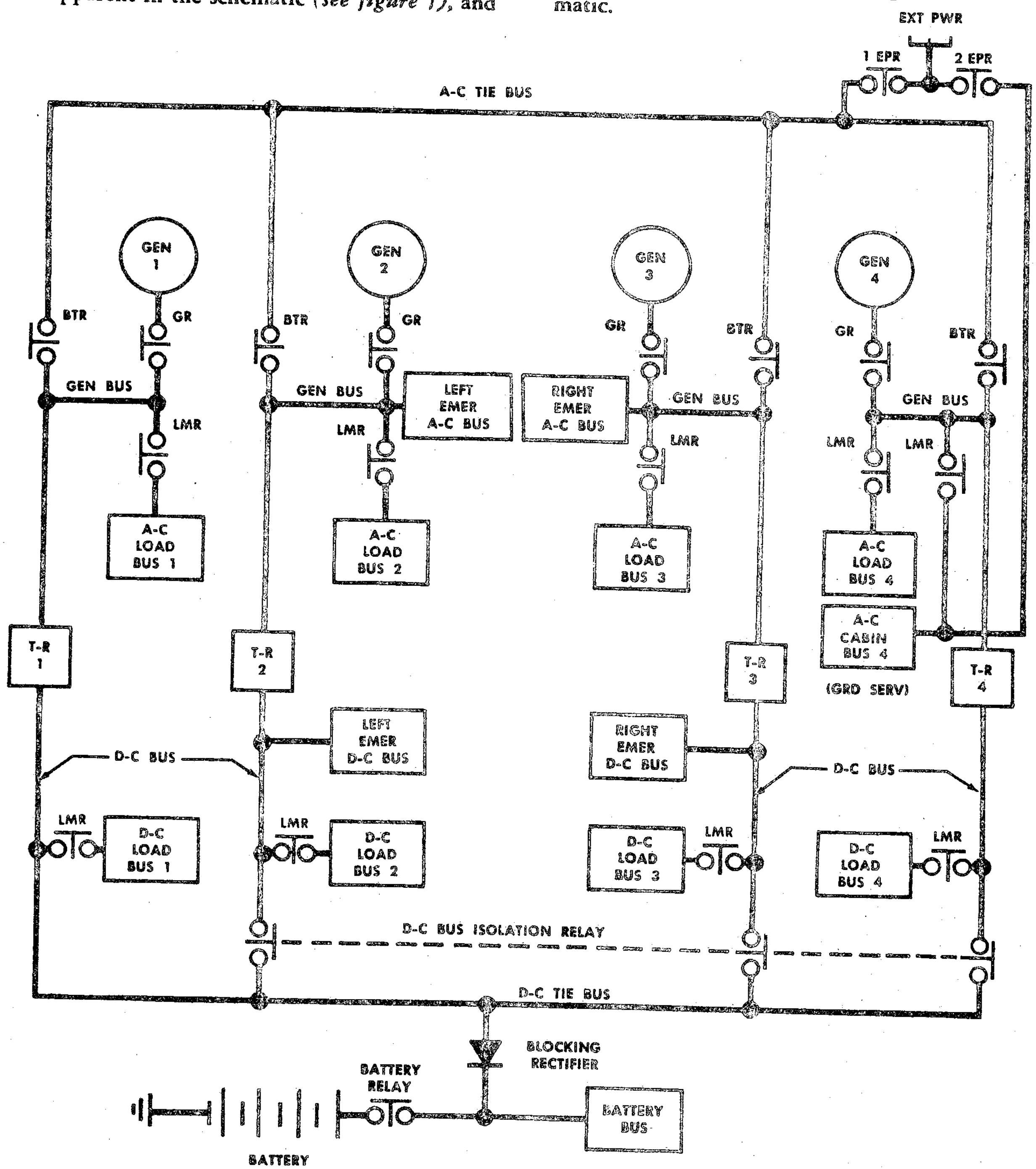


Figure CV-24. ***Schematic of the Electrical Power System***

TABLE OF TERMINOLOGY

Abbreviation	Item	Function of Item	Item Number in Basic Component List
B	Bus	A primary conductor to which other circuits are connected.	—
R	Relay	Controls (i.e., opens and closes) an electrical circuit.	—
GEN	Generator	Converts mechanical energy into electrical energy.	1
GR	Generator Relay	Connects the generator to its bus.	3
BTR	Bus Tie Relay*	Interconnects the generator busses, through the a-c tie bus.*	3
EPR	External Power Relay	Connects external power to the airplane system.	4, 5
LMR	Load Monitor (Shedding) Relay	Provides a means of disconnecting the load bus from the system.	6, 8
T-R	Transformer-Rectifier	A static device for converting alternating current to direct current and lowering the voltage.	15

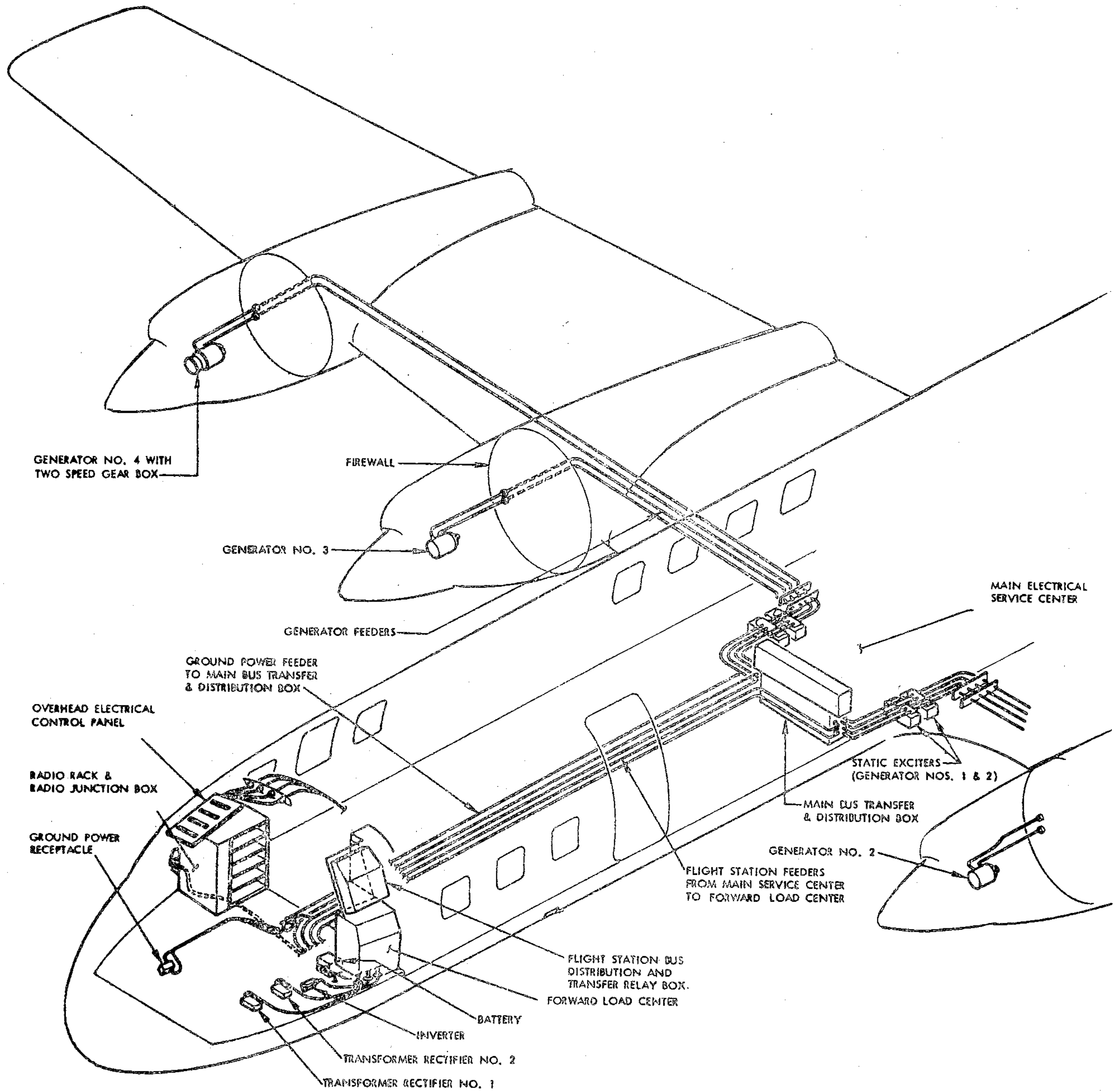
*Note:
Confusion from use of the words "tie bus" and "bus tie" can be avoided by considering the origin of terminology: The a-c tie bus is a bus that ties the four generating systems together; a bus tie relay is a relay that ties its (generator) bus to the a-c tie bus.

Basic Components:

1. Four 3 phase, 120/208 volt, 400 cps, wye-connected generators, 20 or 30 kva each, depending on customer requirements. (The meaning of a wye winding is explained on page 58.)
2. Four hydromechanical generator drive transmissions.
3. Eight 175 ampere, 200 volt, 3 pole latching relays (four generator relays and four bus tie relays).
4. One 175 ampere, 200 volt, 3 pole non-latching relay for external power connection to the a-c tie bus.
5. One 50 ampere, 200 volt, 4 pole non-latching relay for external power connection to cabin bus 4 ground servicing loads.
6. Four 50 ampere, 200 volt, 4 pole non-latching relays for a-c load shedding and for operation of the bus power failure lights.
7. One 50 ampere, 28 volt, 3 pole non-latching relay for d-c bus sectionalizing.
8. Four 50 ampere, 28 volt, 1 pole relays for d-c load shedding.
9. Four magnetic-amplifier voltage regulators for the generators.
10. Four generator control panels containing field relays, protective relays, automatic paralleling controls, etc.
11. Four frequency and load control panels for frequency control and real load division.
12. One tuning fork control panel for close frequency control.
13. One bus protection control panel for all components not typical for the four generators.
14. A total of 40 current transformers for metering, load division, differential protection of the generators and generator leads, etc.
15. Four 50 ampere, unregulated transformer-rectifier units; 200 volt, 3 phase a-c input, 28 volt d-c output.
16. One 24 volt nickel-cadmium battery (5 or 10 ampere-hours, depending on individual customer requirements).
17. Generating system control panel at the systems engineer's station.

Table of Terminology Basic Components

Figure C V-24a.



LOCATION OF ELECTRICAL EQUIPMENT

Figure C V-25.

responsibility. Insure that the equipment to be used is suitable for handling the particular generator involved.

Two types of inverters may be on the aircraft, with three-phase or single-phase outputs of 400 cycles and 115 volts. A single-phase output example is the NESA inverter which provides power for heating the windshields on certain aircraft. Inverters utilize DC power for excitation, therefore they can provide valuable evidence of the condition of the DC power system at the time of impact. **Figure C V-26** is a photograph of inverter armature with short gouge on commutator made by brush holder, indicating non-rotation, no DC power available. Some aircraft may use a static inverter, an electrical, non-rotating device which provides AC power.

For inverters capable of test, shop testing is required, and the air carrier should be able to accommodate. If not, the FAA Coordinator will supply the address of an FAA certificated accessory repair station.

The AC generators are operated directly from a constant-speed device mounted on an engine-driven gear box. If the double unit is intact, it should be taken to an appropriate testing facility. Again, it may be expedient for the Powerplants Group to cover these tests and furnish the results to the Systems Group. **Figures C V-27 and -28** present a simplified schematic of a constant speed drive system and controls, with a brief description of purpose and operation.

The aircraft may be equipped with emergency single-phase inverter or engine-driven alternator to supply power to certain flight instruments in case of failure in the normal AC power system. These should be tested to determine their capability for supplying this emergency power, and thoroughly documented as to external condition, security of attachment to the engine or gear box, and the condition of the attachment area if the unit was separated from its normal location. Relate the external damage to original environment to aid in the determination of fire in flight or after impact. This possibly may help to determine the point of electrical power loss.

Testing will provide sufficient data to analyze the capabilities of components prior to the accident. Should the units test satisfactorily, assume only that electrical power was available if the engines were running. It will be necessary to investigate fully the remainder of the system and related operating components before reaching a conclusion about the integrity of the total electrical power system prior to the accident. The units must be disassembled to determine the reason for discrepancies noted during the tests.

If units are unfit for testing (they may be broken apart with the armature or rotating field in one location and the remains of the case and its windings in another), match each rotating portion with its case to determine original position on the engines or gear boxes. The Maintenance Records Group can supply an identification list of accessories with serial numbers.

Note especially any signs of heavy rotational scoring or scuffing on the rotating portion of the units which might be the result of contact with field pole pieces or broken brush holders during the jarring impact. Check the color of the commutator; it should be a lustrous light-to-dark chocolate brown. Examine the commutator for signs of arcing, roughness, and copper bridging. Copper bridging causes shorting between segments and unwanted paralleling of some armature windings, resulting in lowered output which will cause the voltage regulator to call for a higher field current until the limits of its control function are reached. These conditions result in lowered efficiency of the generator and possible damage to the regulator or field windings.

Examine brushes for abnormal wear, breakage, loose leads; bearings for lubrication, roughness, looseness, or other evidence of wear. Bad bearings permit the armature to wobble and possibly contact pole pieces or field windings, causing other internal damage. Be alert for evidence of electrical fire or overheat in the internal wiring.

Heavy scoring or scuffing in a rotating device is generally sufficient to prove rotation at impact, but is not prima facie evidence of electrical output. All the facts must be analyzed

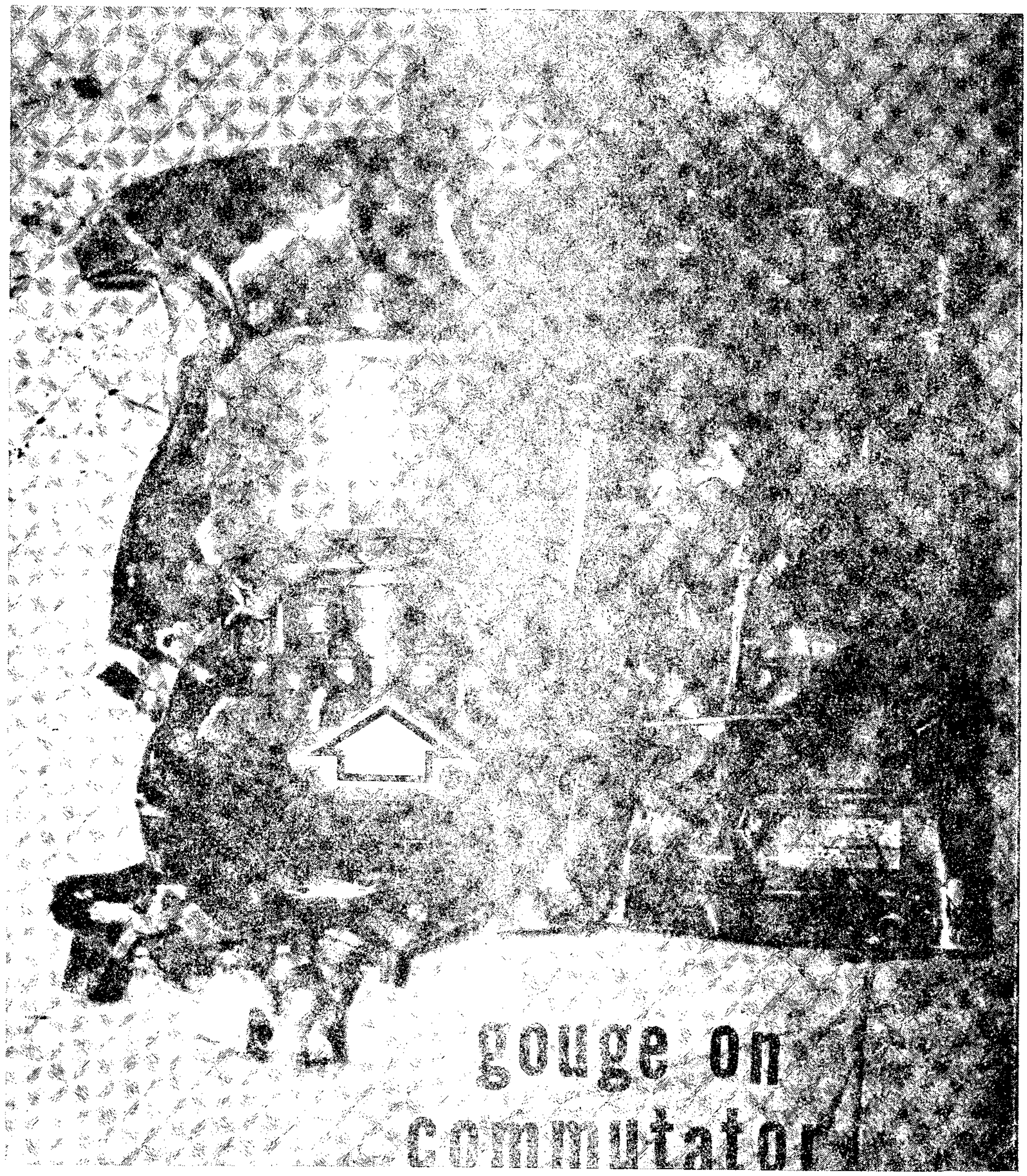


Figure C V-26.

AC POWER SYSTEM COMPONENT DESCRIPTION

Generator Drive System

The generator CSD is mounted between the engine and the generator. Its function is to receive the varying engine speeds – from idle to takeoff RPM – and convert them to one fixed output RPM so as to drive the generator at a constant speed. Maintaining a constant generator speed (frequency) is highly important because not all AC loads operate from a variable frequency. Also, if generator frequency varies, parallel operation is not possible.

The CSD contains a variable displacement hydraulic pump and a constant displacement hydraulic motor. The engine drives the pump, the pump drives the motor, and the motor drives the generator. The rate at which the pump supplies oil under pressure to the hydraulic motor determines the ratio at which speeds will be added to the input, and therefore the speed at which the generator will rotate. A speed governor maintains a constant output RPM (similar to the prop governor on DC-6/7 airplanes). A limit governor acts as a protective device against abnormal output speeds. When output RPM drops below a prescribed value the limit governor closes a speed switch which disconnects the affected generator from its load bus by opening the generator breaker. (This is what trips the generator breaker on every engine shutdown.) If output RPM exceeds a prescribed value the limit governor closes the same speed switch which trips the respective generator breaker. In this case the generator breaker cannot be reclosed until CSD rotation is stopped and the limit governor automatically resets itself.

The CSD has its own independent oil system. The oil serves as hydraulic fluid, lubricating oil, and coolant. The oil is the same as that used in the engine. The oil cooler is a "surface" type mounted in the fan air exhaust duct on the left side of the engine. The integral reservoir is pressurized to about 10 psi by 9th stage air from the respective engine. An oil level sight gage is on the forward end of the unit.

An electrically actuated disconnect is installed between the engine drive and the CSD. This is to facilitate disconnecting the CSD in the event of a CSD or generator malfunction. The disconnect unit must be manually re-engaged on the ground. The shaft between the disconnect and the N_2 drive has an undercut area designed to shear to prevent any torque loading back into the N_2 accessory drive.

Generator Drive Disconnect Switch

The four CSD DISCONNECT switches are springloaded and guarded in the normal (OFF) position. Positioning the switch to DISCONNECT will disconnect the respective CSD from the engine drive. The disconnect unit cannot be re-engaged in flight.

Figure C V-27.

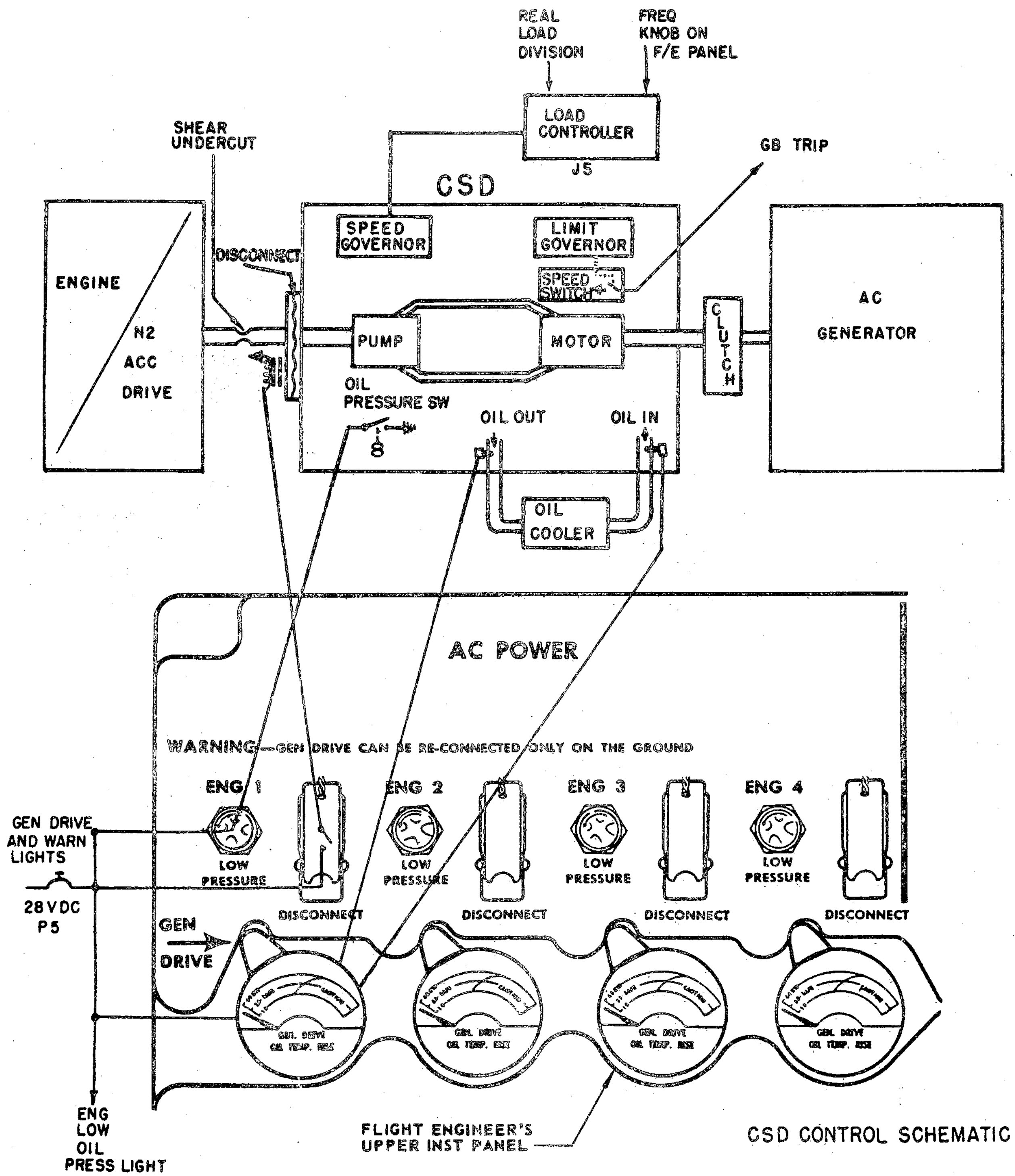

REAL LOAD DIVISION
FREQ KNOB ON F/E PANEL
LOAD CONTROLLER
J5
SHEAR UNDERCUT
GB TRIP
CSD
SPEED GOVERNOR
LIMIT GOVERNOR
SPEED SWITCH
DISCONNECT
ENGINE
N2 ACC DRIVE
PUMP
MOTOR
CLUTCH
AC GENERATOR
OIL PRESSURE SW
OIL OUT
OIL IN
OIL COOLER
AC POWER
WARNING—GEN DRIVE CAN BE RE-CONNECTED ONLY ON THE GROUND
ENG 1
ENG 2
ENG 3
ENG 4
LOW PRESSURE
DISCONNECT
GEN DRIVE AND WARN LIGHTS
28 VDC P5
GEN DRIVE
ENG LOW OIL PRESS LIGHT
FLIGHT ENGINEER'S UPPER INST PANEL
CSD CONTROL SCHEMATIC

Figure C V-28.

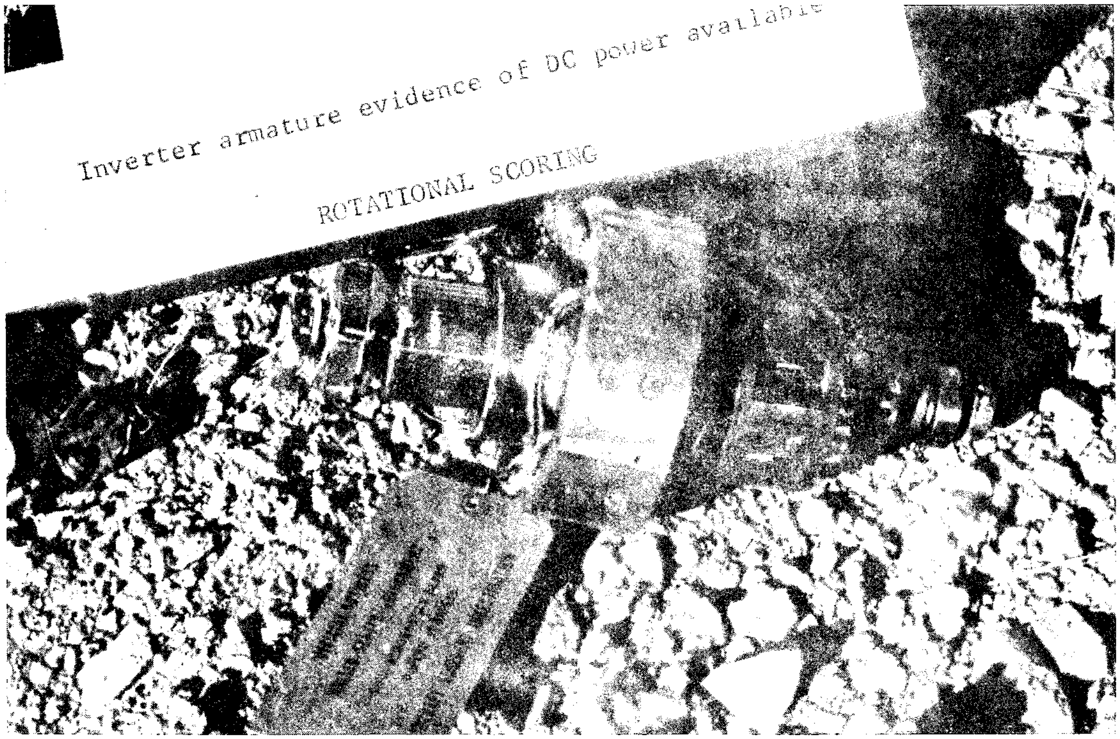


Figure C V-29.

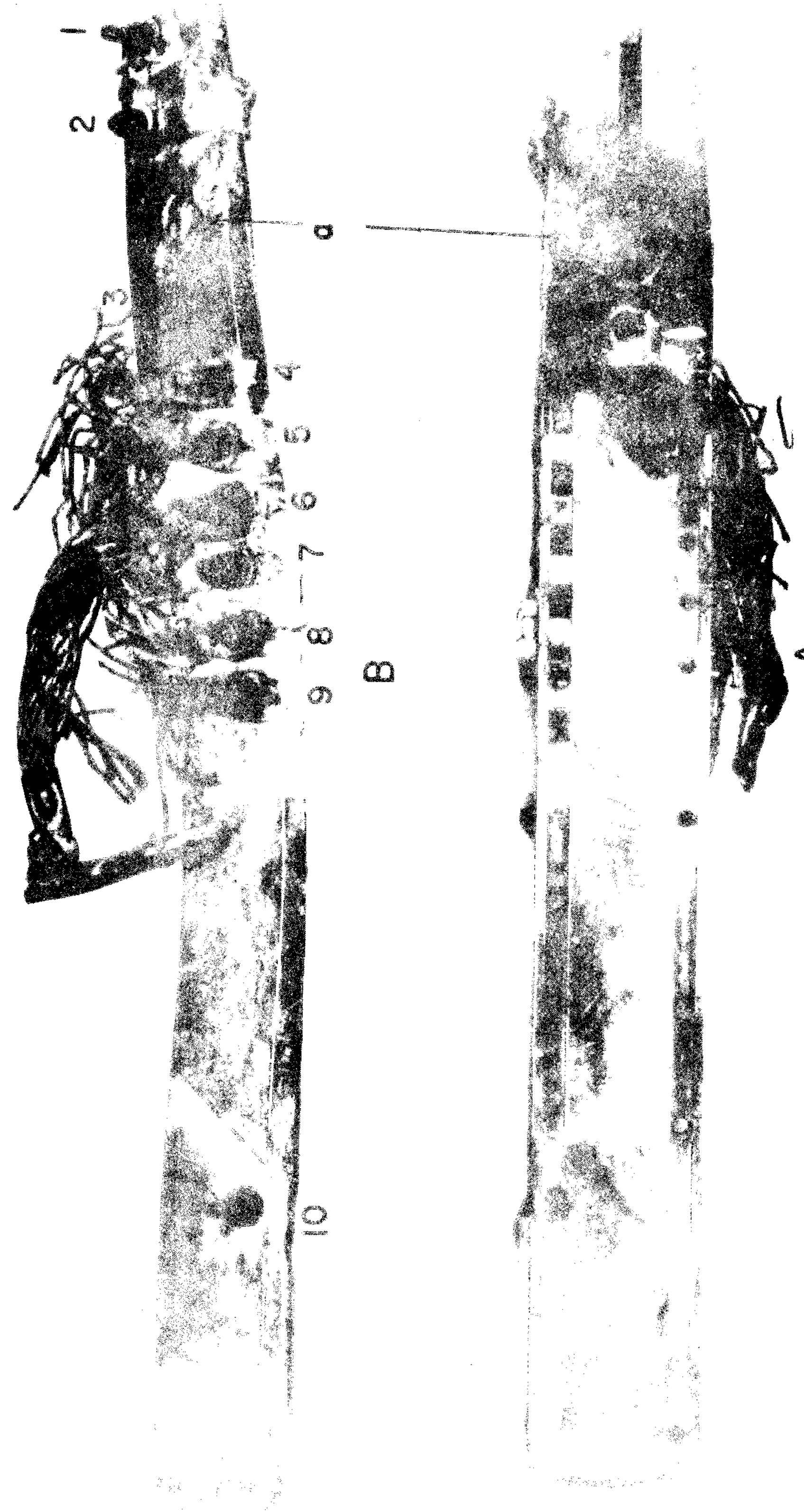
to make this determination. **Figure C V-29** shows characteristic rotational scoring on the DC commutator of an inverter armature indicating rotation at impact, giving evidence of DC power available.

Units exposed to ground fire found loose in the wreckage should be examined internally for evidence of electrical fire. Wiring exposed to external fire will have burned or charred insulation but the wire may be bright, or show surface discoloration only. Wiring burned by electrical fire will evidence overheating throughout. However, take all the circumstances into account.

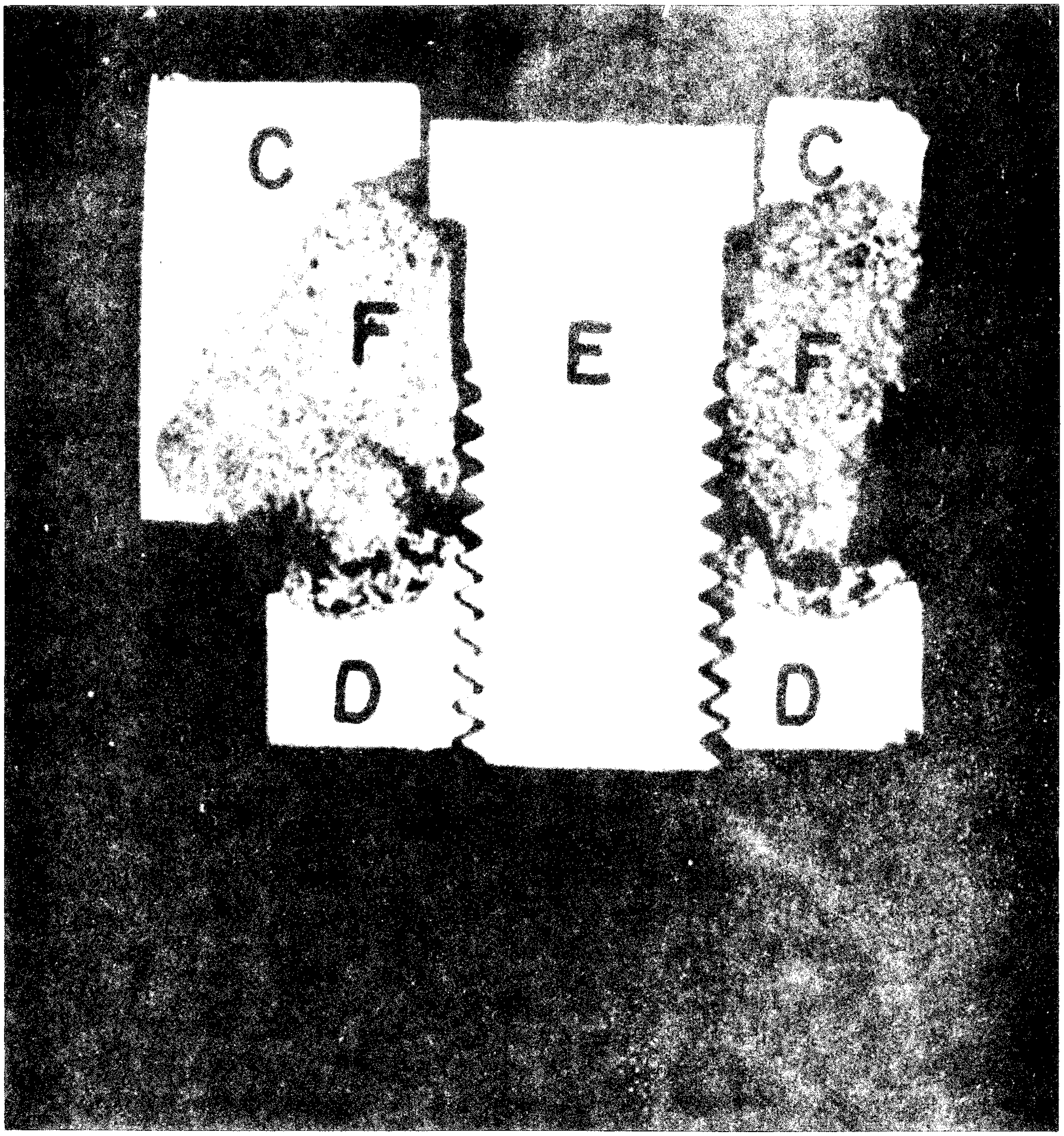
3.2. Electrical Power Distribution Centers — Bus Bars and Terminal Strips

The large bus bars (which are powered through large generator feed cables) distribute power to the various areas of the aircraft. Loose connectors and terminals cause an improper or poor conduction of current with attendant high resistance resulting in elevated temperatures. When the temperatures are sufficiently high the bus bar will begin to melt away with a resulting arcing and erosion of the terminal stud – **Fig. C V-29a** is a typical example. In addition, the evidence of corrosion between the stud (usually steel) and the bus bar (aluminum) is an indication of looseness and an imperfect connection which could lead to trouble. **Figure C V-29b** is a cross section through stud number 5 in **Fig. C V-29a**, showing details of bus bar and stud deterioration.

Check all connections of the circuit protectors attached to the bus bar, and all wiring of the bus bars and terminal strips for looseness. Evidence of arcing between terminals, due to accumulated dust or lint and moisture, may be found when terminal strips are exposed. Some foreign object may have dropped across terminals to cause a short. There can be a variety of causes. It is necessary to look for signs – signs that may be related to some malfunction having a bearing on the cause of the accident.



Bus Bar – Stud Arcing
Figure C V-29a.



Cross section of stud and bus bar arcing and erosion.

Figure C V-29b.

Frequency Regulator Carbon Pile NESA Inverter
Figure C V-30.

3.2.1. Voltage and Frequency Control Devices

Voltage and frequency control devices are the regulators for the generators and inverters. They may be found scattered in the wreckage, in the electrical load centers, or electrical equipment racks. Determine locations from the maintenance manual or schematic diagrams. Setting information cannot be determined by external examination of these devices; they must be tested. Identify them, document their physical condition (such as wiring connections, security of resistors, etc.) and whether or not they show indications of electrical burning or overheat.

You are familiar with the carbon pile regulators, and how pressure is used on the carbon disc stack to increase or decrease voltage and frequency. The regulator, usually found in the control box atop the inverters, may be opened for examination. Failure of this type of regulator is manifested by deterioration (powdering) of the carbon discs, a condition not caused by impact alone. **Figure C V-30** is an example of deteriorated carbon discs from an inoperative frequency regulator for a single-phase inverter.

Some regulators are electronic and are sealed units. Only tests can provide information as to their capabilities, but they should be accounted for and their condition noted.

Other controlling devices, such as the reverse current relay, generator field control relay, etc., may be reset in the air should they trip out. These should be identified and examined for tripping or proper setting. Other relays should have contacts examined for arcing, burning, or welding which may indicate electrical distress somewhere in the power systems.

The items mentioned should be put into protective storage until examination of the system is complete, in case some condition is uncovered which would dictate further examination.

3.2.2. Circuit Protectors and Wiring

Electrical system cables and wiring are protected from effects of short circuits and overloads by means of circuit breakers, fuses, and current limiters. These devices sense rise in temperature from excessive current and act to open the affected circuit, stopping the current flow. Understand that such protective devices are designed primarily to protect the cables or wiring rather than the connected equipment, which can be separately protected by integral devices such as thermal overload switches. It is possible to find an item overheated due to overload, without actuation of the protective device.

The fuse or circuit breaker should react before the cables or wiring insulation smokes. Circuit breakers have failed to react under currents high enough to produce smoking of insulation. It is therefore advisable to inspect visually all cables and wiring, regardless of circuit breaker condition.

The current limiter is essentially a high-amperage fuse installed in main feeder cables to isolate a feeder which suffers a direct short circuit or positive fault to structure. It is known that short circuits to structure occur wherein arcing persists at the point of contact, as in the case of contact with skin material which continues to melt away and sustain an arc. In such cases, the resistance of the arc can limit the current to a value which is insufficient to melt the fusible element of the limiter, and the arc persists. This is a serious hazard, and it can be the ignition source for combustible vapors. The absence of melted limiters should not preclude a close inspection of cables and wiring protected by limiters.

When damage to the aircraft and the electrical system is severe, examination of wire bundles and electrical system units must be accomplished separately. Generally, the individual units of the system, particularly those of greater mass, are scattered over a wide area completely disconnected from their electrical cables or wiring. However, the major wire bundles will be relatively intact and can be identified as to their location in the aircraft.

If fire followed impact, identification of wire bundles becomes dfficult. At first sight, the bundles may appear hopelessly snarled, but a little effort will enable the investigator to unravel the mass sufficiently to identify particular sections. **Figure C V-31** shows a portion of structure and wiring recovered during a transport accident investigation. Positive identification of cables and wiring is primarily

Portion of structure and wiring recovered during transport accident investigation.

Figure C V-31.

by wire numbers found at each end of a wire run or at regular intervals along the length of the wire. If burning has been severe, this method of identification may be useless and identification must be made through comparison of the gauges of the wires in a particular bundle with the gauges of wires known to be bundled together at specific points in the system. Clamps which support wire bundles to the aircraft structure are often found attached to the bundle, torn away from the structure, or still attached to portions of structure. These clamped positions and portions of structure assist in fixing the original location of the wire bundle. Also helpful in this respect are terminals or the remains of switches and circuit breakers still attached to the ends of individual cables or wires.

After identifying and accounting for as much as possible of the major wiring bundles, examine each cable for short circuits to structure and for direct contact with adjacent cables. The point of contact is determined by evidence of arcing and high local heating. Particularly, check support clamp positions and points where cables or wires pass through bulkheads or firewalls. Examine terminal connections for signs of local heating (due to loose terminal connections) or shorted terminals and connections. Inspect fuses and current limiters for melted fusible elements, and relay contact points for signs of arcing.

3.2.3. Electrically Operated Components

Many electrically operated components are installed in every aircraft, pumps, motors, fans, certain selector and control valves, and some actuators, operated by DC and by AC current. Therefore, it is necessary to recover as many as possible of these components for detailed examination and testing. Units unfit for testing, or for which there is no need for testing, may be disassembled for examination. Prior to any testing or internal examination fully document physical condition and any indication of component position prior to the accident.

Consider for a moment the generators. A visual examination, as referred to earlier, may detect signs of excessive output. When a generator operates under excessive load for more than a relatively short period of time, the increase in heat will discolor the commutator bars or melt solder. In severe cases the commutator bars may rise and interfere with the brushes, the brushes will be broken off and carried away. If such a condition is found, it is apparent that the output of the generator was passing through a path of lower resistance than intended, which strongly suggests a short circuit somewhere in the electrical system.

Occasionally a dangerous malfunction known as *over-voltage condition* occurs from the application of generator output to the field circuit of the generator without control. This usually results from a direct short circuit within the generator from armature output to the generator field terminal. An uncontrollable high voltage output and excessive current follows with instantaneous burning out of filaments in light bulbs and radio tubes. To disconnect the offending generator from the circuit the generator switch is turned off, causing the generator reverse current relay to open. However, if the output of the generator is too high, the reverse current relay may not be able to interrupt the current flow and the current will continue to flow in the form of an arc across the relay contacts. This results in the melting of the contact points and possible destruction of the entire relay assembly. If the condition is prolonged, material in the same general location may be ignited.

The purpose of a detailed examination of the various pumps, motors, fans, etc., is to detect evidence of operation or to determine certain positions at impact which will aid in verifying the availability of proper electrical power at the time of the accident. For example, if the basic power system is DC, and examination shows that an AC-operated motor was running at the time of impact, then it is reasonably certain that the DC power system was in operation, since the inverters which supply AC are operated by a DC motor. The reverse is true if the basic power system is AC.

First consideration when dealing with components lying loose in the wreckage is to separate the recovered components into two groups, those operated by DC, and AC power. Examine each unit visually and document the physical condition; completely identify the unit with information from the data plate. If the data plate is missing, make the identification by comparison with like units, by the use of a parts catalog or the wiring diagrams.

Components which drive fans, such as the radio rack or electrical equipment rack cooling fans, recirculating fans, heater blowers, etc., can yield evidence of rotation through the fan blades and their housings. The blade tips may have scraped the housing, leaving marks, and the blades of lighter fans may be bent or otherwise distorted in a direction counter to normal rotation. The heavier type fan blades may actually be broken off. These are evidence of fan operation at impact. The absence of rotational evidence on any rotating component is not positive evidence that it was inoperative. A general knowledge of the system in which the component is installed is necessary to determine whether the component should be operating as a normal course, or whether it should be operating only under specific conditions. The physical damage the unit has sustained may not be sufficient to cause binding or deformation necessary to produce rotational evidence.

Evidence of rotational damage is not positive proof that electrical power was being applied at the instant of impact. Some rotating components are of such mass that a period of coasting follows power shutoff. Therefore, the only certainty is that they were rotating at the time of impact. This data must be combined with information developed in the investigation of the particular system involved and analyzed before determination can be made that electrical power was being applied.

Examination for evidence of rotation can be made of all motors except those which operate intermittently, small motors which open and close valves in units such as hydraulic pressure modules and the pneumatic system valves, which would be covered during investigation of the pertinent systems.

3.3. Functional Check or Testing

Consider an accident wherein the electrical system is relatively intact. It may be possible to conduct functional tests using an external source of power, but the integrity of the system should first be checked with an ohmmeter. The ohmmeter check is desirable before applying full system voltage since a short circuit condition might exist which could produce excessive heating, sparking, or fire under full system voltage. Be sure to remove the aircraft batteries prior to undertaking the ohmmeter check.

Once the integrity of the system has been established a source of power may be connected to the system for an operational check. Always be alert to the possibility of combustible vapors in the vicinity of the aircraft after an accident and insure that no fuel fumes or oil deposits are in the immediate area of the functional check.

Occasionally fuses and circuit breakers of greater capacity than specified are installed in error or are substituted for the proper device. It is advisable to check the fuses and circuit breakers for the proper rating as indicated in the wiring diagrams.

If the behavior of the system during the operational check suggests a short circuit, the fuses and circuit breakers should also be checked. It may be necessary to reset the circuit breaker, or replace the fuse and check for a short circuit with an ohmmeter. If a short circuit is not disclosed, it could be that a short circuit existed but subsequently was removed. This is apt to be the case when a conductor momentarily shorts because of vibration, or when the short has been removed by crash impact. Visual inspection of all wiring for signs of a previous short in the circuit will be necessary. If this method fails to locate the difficulty, the connected equipment such as motors, relays, solenoids, etc., must be removed and given functional tests.

If the system behavior suggests that a circuit is open, the ohmmeter may verify the location of the fault, or it may be found by application of system voltage and use of a voltmeter to check the circuit.

Testing of generators, inverters, or alternators should be accomplished at a facility with the proper equipment and personnel to cover the full range of operation of the component design. Procure the component specifications which will include any tolerances applicable to proper operation. Results of the tests should be entered on forms used by the facility. Discrepancies must be investigated for their cause and effect on system operation.

AC generators should be tested with their respective constant speed drive units for valid results. If the voltage and frequency control devices for the power generating components are recovered intact, it is wise to wire these units into the test equipment for a functional check.

The various electrically-operated components which might be helpful in determining the integrity, or lack thereof, of a system should be given a functional check, the results documented, and discrepancies investigated.

Components unfit for testing should be disassembled and examined. Generator, inverter, alternator and motor armatures, and field windings should be checked for shorts and open circuits with an ohmmeter. The ohmmeter can also verify continuity of relays, switches, and solenoids for various valves.

The ohmmeter and the voltmeter are very helpful devices used to great advantage in the field. Some devices incorporate multiple functions of test capabilities known as volt-ohm-milliammeters (VOM) or multimeters. These are usually lightweight, compact, and portable. They can be purchased at reasonable cost from any electronic equipment store and their use in the field can save a lot of legwork. Obtain one of these units as soon as possible after arrival at the accident scene.

3.4. Examination of Electronic Equipment — Flight Recorder — Voice Recorder

The flight data recorder is required equipment on all turbine-powered aircraft in commercial operations. The recording medium may be aluminum or inconel foil, or Mylar recording tape. All voice recorders use Mylar tape. Recorders will be mounted in the radio or electrical equipment racks, in the tail, or near the aircraft center of gravity.

The main concern with these components is that they are recognized, recovered from the wreckage at the earliest possible moment, and shipped intact to the Bureau of Aviation Safety in Washington, D. C. Under no circumstance are they to be opened or meddled with! They should be photographed as they are found in the wreckage. Their external condition as found should be documented. This includes the condition of the recorder mounting, electrical wiring and tubing attached, and any evidence of exposure to ground or inflight fire. This evidence should also be related to the condition of the wreckage where the recorder is found. Photographs and the written documentation on the voice and flight recorders are to be turned over to the ATC Group Chairman for his notes and report.

3.4.1. Autopilot

The automatic pilot is a device which ranges from the single-axis unit installed in small aircraft to the complicated three-axis installation in the large turbine-powered aircraft. Procure a schematic which shows all of the autopilot components and their locations within the airframe, and a set of the wiring diagrams for the type of system involved. Recover and identify as many of the components as possible.

The majority of the automatic pilot components yield information only by shop testing since they are basically electronic in nature, with few moving parts. The circumstances of the accident and the condition of the components will dictate the type and amount of testing.

Determine whether or not the autopilot was in use by examination of the autopilot control panel in the cockpit. Examine the switches and levers for position and evidence of damage by other objects, the direction of any breaks, etc. With the ohmmeter, check the internal condition of switches for continuity.

Some autopilot servo motors are tied directly into the control cables to operate only when the autopilot is activated. Others have mechanical and magnetic clutches which engage the servo motors. Seldom will the servo be found engaged even if the autopilot is in use, since the clutch will disengage as soon as electrical power is removed. It may be possible for the cable-operated clutch to become engaged by the pulling of the cable during aircraft breakup.

Remaining autopilot system components can be examined only visually for evidence of electrical problems, and by testing where feasible. Associated relays should be examined as suggested earlier for evidence of arcing or burned contacts. Autopilot wiring and circuit protectors should be examined.

3.5. Miscellaneous Items for Examination — Light Bulbs and Radio Tubes

The common light bulb is a good source of information, and it is helpful if it can be determined that a warning light was illuminated at the time of impact. Other bulbs, those from the cabin lights or exterior lights, assist in determining electrical power at impact.

Ground witness statements that lights were visible prior to the crash or statements from survivors are good sources of information; however, the definite indications of light bulbs and radio tubes will be discussed. Account for all exterior lights (landing lights, position or navigation lights, wing illumination or ice lights, and rotating anti-collision lights), and correctly label the function of each. Collect all available warning light bulbs from the cockpit, and label. The cabin overhead and cove lights should be collected. If a bulb envelope is broken, protect the delicate filament remains.

Use a glass (8-10x) for examination of bulb filaments, and look particularly for the following conditions. When a bulb which is not illuminated is subjected to a shock load, no appreciable stretching of the filament occurs when the load is sufficient to cause filament failure. **Figure C V-32** shows a "cold" failure of this type. When the bulb is lighted, a shock load causes stretching of the hot filament at loads far below that causing failure. The filaments of most bulbs are tightly wound coils, and in these bulbs the stretching results in opening the coils like those of a tension spring stretched beyond elastic limit. The increased length also distorts the normal loop formed by the filament. **Figure C V-33** is an example of "hot filament" stretch.

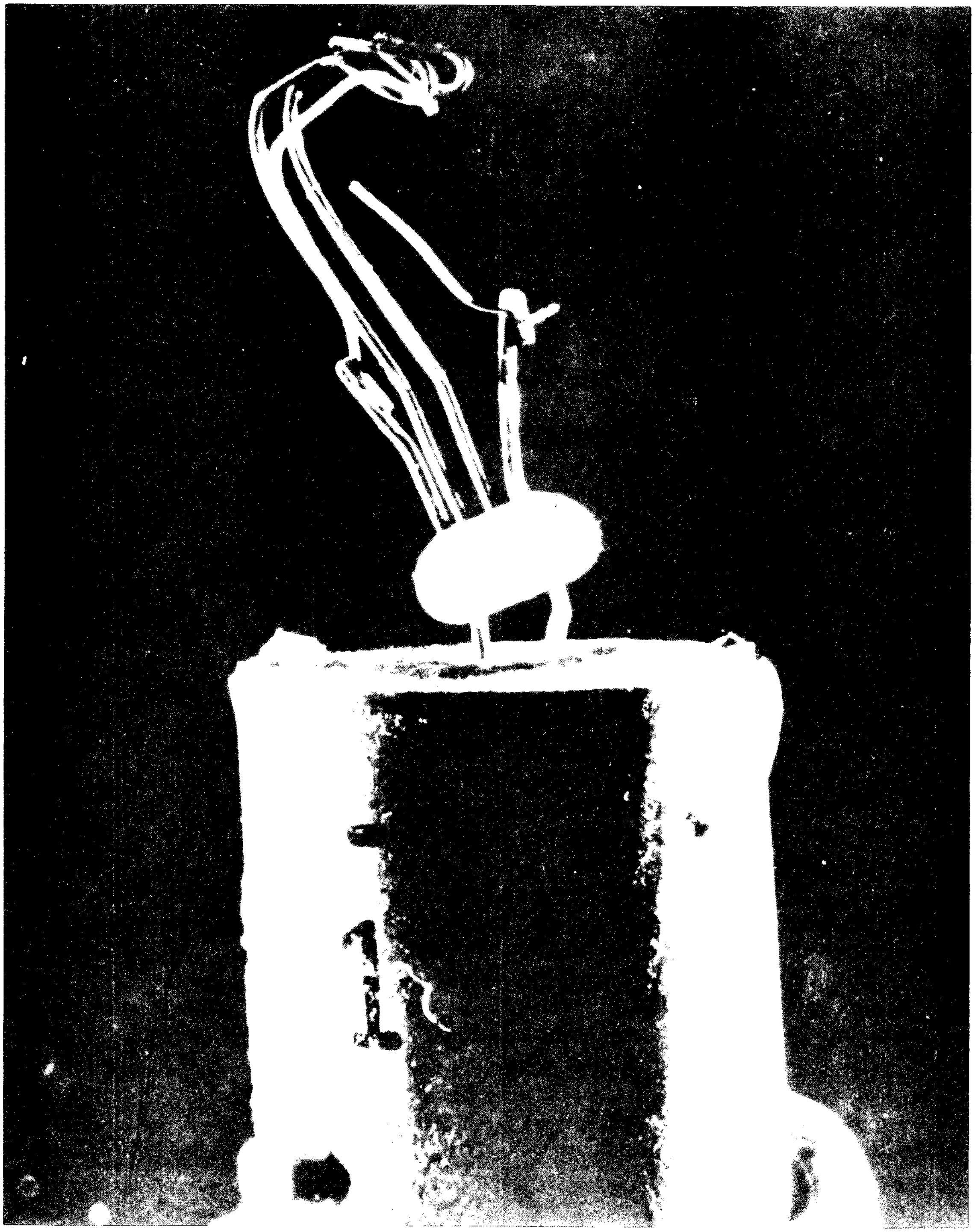
The extent of filament stretch depends upon the load and may be general or localized near the end attachment points or at the bends of the filament over the support hangers, with only a slight opening of the windings. In such cases there is little increase in length and consequently only slight distortion of the loop. Even though the glass envelope is broken and part of the filament is missing, stretching of the filament near its attachment points indicates that the bulb was lighted. A noticeable discoloration of the filament results from oxidation of the tungsten wire, which is not possible when the filament is cold.

If the filament is found broken but clean and bright, and no stretch is noted in the filament coil, it is evident that the failure occurred when the filament was cold. Take a close look at the broken ends of the filament. If the break appears clean, as though snapped off, it is further evidence that the filament was cold. If there is a melted globule on the broken ends and the glass envelope is lightly discolored, it is an indication that the bulb merely burned out, probably prior to impact.

Radio vacuum tubes exhibit similar signs. If a tube is cracked and smoked on the inside of the envelope, the tube was hot at the time of breakage. If the envelope interior appears normal though cracked or broken, no DC power was being supplied to the tube filaments at the time of break.

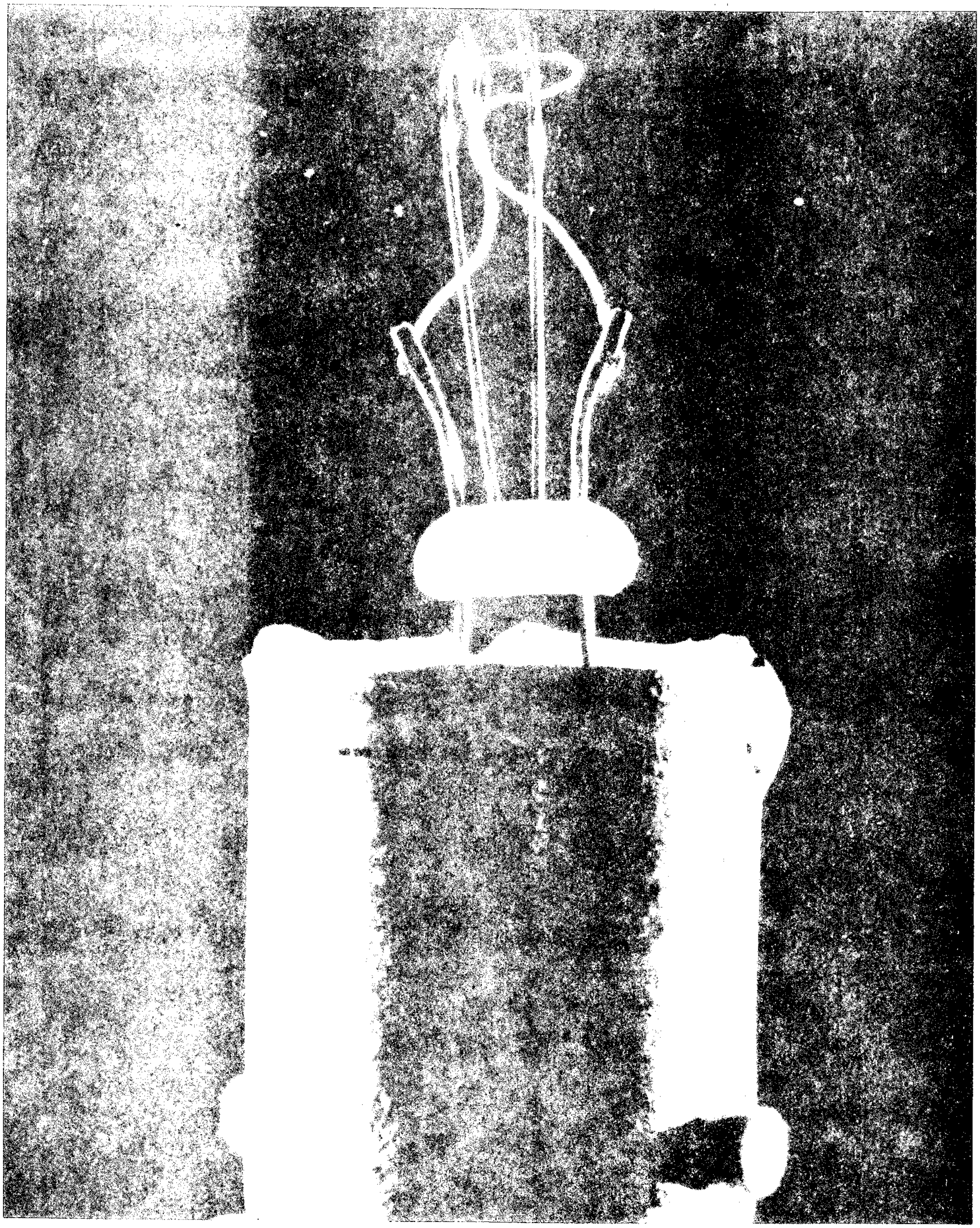
3.5.1 Batteries

Batteries usually will be found disintegrated or badly damaged because of their location in the forward portion of the aircraft. However, if one is found reasonably intact, use the voltmeter to check the amount of voltage. It is advisable to obtain a hydrometer to check the specific gravity of the fluid in any intact cell which



"Cold filament" failure due to shock load.

Figure C V-32.



"Hot filament" distortion due to shock load. Note filament stretch.

Figure C V-33.

will yield information about the state of charge (which is just as important as the voltage). Record the values obtained by these checks.

The condition of batteries could be significant since they furnish power for certain equipment during emergency situations. Should the specific gravity check show a high state of charge with maximum voltage available, it could indicate that the electrical power system had been operating properly. It is no guarantee that electrical power was present at the moment of impact.

A battery found in a low state of charge by the hydrometer check might suggest an emergency which required the battery to be used for a period of time prior to the accident. However, it could indicate that the battery was old and would not sustain a charge. It is advisable to open the cells to examine the condition of the plates and the amount of sediment in the bottom of the case of the lead-acid battery. Great care must be exercised to prevent personal injury from battery acid.

Poor condition of a battery is indicated by partial or extensive disintegration of the positive plates, buckling or warping of the plates, and large amounts of sediment (particles of lead sulphate) in the bottom of the case. Under certain conditions sediment may build up to contact the plates, resulting in shorting of the cell. If these conditions are found, suspect a discrepancy in the electrical system which would result in overcharging the battery. A battery in such condition is unfit for use in an emergency situation. Analysis of the accident circumstances will indicate whether this factor is pertinent.

The popular nickel-cadmium (alkaline) battery presents a different picture. The state of charge for this type of battery cannot be determined merely by checking the voltage and electrolytic specific gravity. One disadvantage is that the individual cells may not charge evenly because each cell may take a different load when charging. This imbalance between cell charge, even though the battery may feel the full charge of 12 or 24 volts, may suggest a dead battery. This results because the weakest cell discharges first, therefore, to determine the true condition of these batteries, check each cell. As in lead-acid batteries, a dead cell in the ni-cad battery will prevent proper performance during emergency situations.

Even though examination of small items such as batteries yields only negative data, the thoroughness of investigation is emphasized. No item is really unimportant when seeking the cause of an accident.

4. Instrumentation — General

Many types of instruments in use today range from simple, self-contained indicators to complex, electrically-operated instruments with three or four integral displays. Some are operated by suction and some by air pressure ratios, such as those tied into the pitot and static pressure systems. Others are gear driven by electric motors and some are synchronous, using small permanent magnets which repeat the positions of various transmitters and transducers. Investigation of the instrumentation systems must necessarily be painstaking to derive valid and meaningful information.

Cockpit documentation has been discussed. During that activity detailed readings and photographs of every instrument, including all meters available in the cockpit, should have been taken. If there is no cockpit, then search the wreckage to recover every instrument. Caution those who work with you to watch where they walk. Instruments are very susceptible to being pushed into the mud by careless feet.

Cockpit instrument arrangements, especially those on the pilot's flight instrument panel, vary according to the desire of the operator. The engine instruments on pilot panels and the flight engineer panels are usually standard in arrangement. Obtain pictures or manuals from the owner/operator showing the particular arrangement at hand. Obtain schematics of the applicable pitot and static pressure systems, the integrated flight and navigational instrument system, and the compass system. It is very important that these schematics be applicable since inestimable confusion and misinterpretation would result if the schematics and the aircraft installations did not agree.

Caution: Systems may vary in a fleet of aircraft of the same type or model.

During recovery of loose instruments or other small indicators provide suitable protective containers. Plastic bags have several advantages: They are convenient to carry and to use; they are clear, and will permit viewing of the indicator face without possibility of movement of the pointers by careless handling; a small tag or card with accident identification, component identification, and the reading as found, can be inserted. For example: *Read 080 degrees when found.* Such documentation can be very important because indication may change during handling if the indicating means is movable. Record the original reading immediately, precluding questions and confusion. Exercise every precaution in handling aircraft clocks, for they are very important in establishing time.

As the various instruments and their panels are recovered, mark them on a schematic or picture to show progress of recovery and the instruments yet to be recovered. This is a desirable practice when accounting for components of any system – it makes a good check-list. It is advisable to identify portions of the instrument, switch, or circuit breaker panels when found. This is easily accomplished by shading on the pictures, highlighting the missing items. This is a method of documentation which can save much writing in notes and reports.

As visual examination proceeds, determine which instruments are fit for testing. Instruments requiring more detailed examination than can be accomplished at the scene should be provided protective storage. The remaining instruments should be boxed, labeled, and stored with other wreckage removed from the scene.

Generally, the wreckage will remain in the custody of the Board until after the Public Hearing or Deposition proceeding, in case evidence points to a need for more investigation. It is wise, therefore, to plainly mark contents when boxing a group of small parts. The parts should also be listed on an inventory sheet for quick reference.

4.1. Visual Inspection and Documentation – Flight and Navigational Instruments

Cockpit instruments are usually damaged to the extent of complete destruction because of their location in the aircraft nose section which usually impacts first. However, some may retain their dial faces and indicating pointers with some kind of indication which must be regarded with suspicion until all facts are known.

The only time the controversial method of determining prior indicator positions with the use of black light can be beneficial is when instruments are present which have dipped pointers painted with fluorescent materials. The practice of coating indicator pointers and incremental markings on dials has been discontinued. Currently, pointers and dial markings are being painted either with white or colored matte-type paint, none of which fluoresce.

An exhaustive study was conducted by the U. S. Naval Air Development Center at Johnsville, Pennsylvania, on the possible use of a thin coating of paint on the backs of instrument pointers as a reliable accident investigation aid. The original premise was that the instrument pointers would on impact momentarily strike the face of the instrument. The study revealed that in the majority of crashes most instrument pointers would not touch the dial face, so for technical and economic reasons the proposal was unfeasible.

The report also stated in part that . . . *only an accident in which the plane is moving extremely rapidly (400 mph or over), and stops in less than ten feet, will result in any appreciable deflection of the needles down to the dial faces. An accident of this nature will rarely occur, and then will be of such force as to destroy the plane completely.*

There will be exceptions when some object strikes the face of an instrument during the impact deceleration and jams the pointer into a fixed position on the dial face. Three exceptions follow:

– A rate-of-climb indicator was recovered: It had been exposed to fire; the instrument cover glass was missing, and the aluminum pointer was reduced almost to white powder. When the instrument was found, the pointer was indicating a 900 f.p.m. rate of descent. Despite careful handling the pointer remains fell off, and a sharp, dark shadow appeared where the dial had been protected by the pointer. This finding matched the rate of descent at impact, computed from the last seconds of the flight data recorder altitude trace.

— A hydraulic system pressure gauge was recovered from salt water and checked with black light. The instrument dial fluoresced. The dial was carefully washed off and the light reapplied. The pointer was distinctly outlined where it had been driven against the dial. The paint was scratched, and the salt water corroded the scratch.

— Another rate-of-climb indicator was removed from the wreckage of a turboprop aircraft. The instrument cover glass revealed an impression in dust made when the pointer actually came off its staff and was impressed against the glass. Under black light the pointer position indicated a zero rate of descent.

These are examples of the necessity for full examination of each instrument regardless of information forthcoming from other sources.

Some flight instruments provide unreliable indications following an accident, but it is imperative that these instruments be fully documented. These are the airspeed indicator, vertical speed or rate-of-climb indicator, mach meter, turn and slip indicator, and the standard old gyro horizon indicator.

The airspeed and rate-of-climb indicators operate directly from the pitot and static pressure systems; the mach meter may be operated from the air data systems, which in turn are basically operated by pitot and static pressures, and the two attitude indicators may be powered by the suction air or electrical systems.

The design of this type of instrument which utilizes a diaphragm, gear, and linkage mechanism, may render questionable indications after an accident because of rapid change from a heavy blow, or disruption of the systems supplying air. Even though these instrument dials may be struck by a foreign object, indications jammed at impact cannot be construed as valid positions. This is due to the possible time interval between the time of initial impact and instrument damage. The interval may be seconds or fractions thereof, but time sufficient for the indications to change from what they were at initial impact. Until all the facts are developed and analyzed the readings should be regarded as questionable.

The following flight and navigational instruments should be examined carefully and given a functional test if practicable or feasible.

4.2. Altimeters

Types of altimeters in use today are the two-pointer, three-pointer, and drum-pointer. The drum-pointer is the latest design used in commercial aviation and will be found in general use on turbine-powered airline aircraft. The three-pointer type is also used by some carriers in the turbine-powered aircraft. Another type, vertical scale instrument, has been developed which will be found on some new aircraft.

Record the altitude reading, if any, indicated by the pointers or the pointer and drum, and the position of the two peripheral indices. The most important feature is the barometric scale position. This scale is gear driven, and is positioned by an adjusting knob, therefore, the position will not vary.

Least reliable is the altitude indication of the pointers, as any damage to the delicate internal mechanism can drive the pointers to any altitude if the rocking shaft pivots break and release the tension spring. The pressure altitude peripheral indices, however, are directly related to positioning of the barometric scale as they are also gear driven and set by an adjusting knob. In the light of the last altimeter setting provided to the flight crew by ATC or the tower, examine the positions of the pointers and the barometric scale and verify the information by transcripts of radio communications between the aircraft and ground stations, and the cockpit voice recorder, if any. Comparison will verify whether or not the altimeters were properly set at the time of the accident. **Figure C V-34** shows damaged altimeter from wreckage of a jet transport compared with a

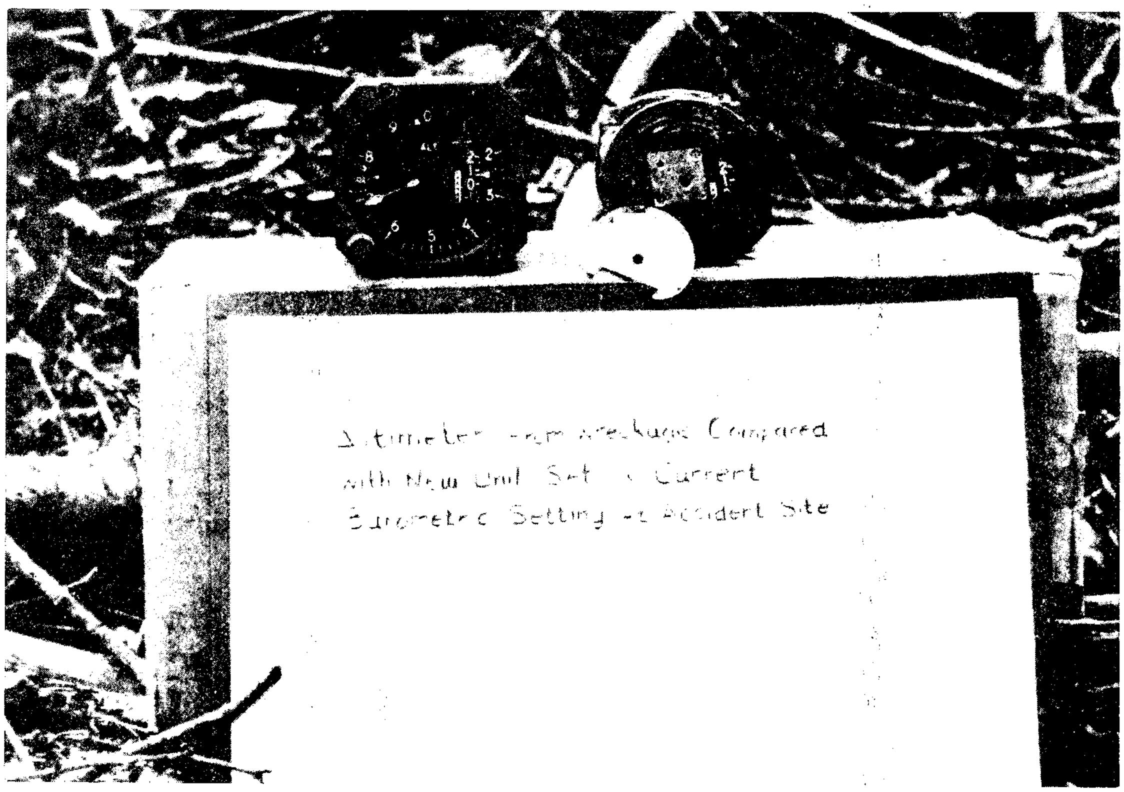


Figure C V-34.

new unit set to current barometric setting at the accident site.

If the barometric scale cannot be read because of damage by fire or other physical disturbance, have the manufacturer interpret the reading.

4.2.1. Magnetic Heading Instruments

Heading indicators in today's aircraft include the liquid-filled standby magnetic compass; the suction air operated, nonmagnetic directional gyro; the magnetic fluxgate or gyrosyn compass; the radio magnetic direction indicator (RMDI), and the rotating compass card found on the new course director instruments (CDI). All these provide continuous heading information to the pilot. With the exception of the standby compass and the suction air operated directional gyro, indicators are electrically powered.

A heading found on the suction air operated directional gyro may be questionable since its card is mounted on the easily moved gimbals. Record the reading as found, document the total instrument condition, and open the case to examine the gyro for evidence of rotating damage from contact of the rotor with the air nozzle. Examine the rotor gimbals for freedom of movement or evidence of cracking and breaking.

Headings of the electrically operated heading indicators should be recorded. The rotating compass cards are driven by small servo motors through a high gearing ratio so that when electrical power is removed the card should remain on the heading of that moment with no coasting or overtravel. Heading indications from these instruments are reliable and may supply evidence of the time of loss of electrical power. Some of the new course director indicators have windows for selecting a course and for displaying mileage from the distance measuring equipment. Record any numbers appearing in these windows, and document the overall condition of these instruments.

The RMDI contains two pointers in addition to the rotating compass card, generally referred to as the single and double pointers. These will refer to the No. 1 and No. 2 receivers, respectively, of either ADF or VOR, and provide bearing information. Record the bearings indicated. In cases where only one instrument of this type is mounted in each pilot's panel the switches will be mounted just below for selecting ADF or VOR. Record the switch positions so that the bearings have meaning. If the switches are broken off, examine the stub ends for the direction of the breaks. These instruments will appear again in the areas of radio and distance measuring equipment.

4.2.2. Gyro Operated Instruments

Turn and bank indicators and artificial horizon instruments contain gyro rotors and can be suction air operated or powered electrically. The gyro rotors in these instruments may be encased with a shroud, depending on the type.

The suction air operated components should be examined for signs of scoring from contact with the air nozzle or gimbal frame, and the others for scoring produced by the contact with their cases. Scoring may be light, heavy, or nonexistent, depending on the force of contact and speed of the rotor. A spinning mass (the rotor) shears its pivots upon impact force, depending upon the angular contact. These rotating masses have an appreciable coasting period of up to several minutes after their driving force is removed. The only definite conclusion is that the driving force was present at some time immediately preceding impact if scoring is found, but not necessarily at the moment of impact, and the lack of scoring evidence does not prove that the gyro was not operating at impact. **Figure C V-35** shows vacuum turn and slip indicator gyro rotor scored by contact with air nozzle at impact.

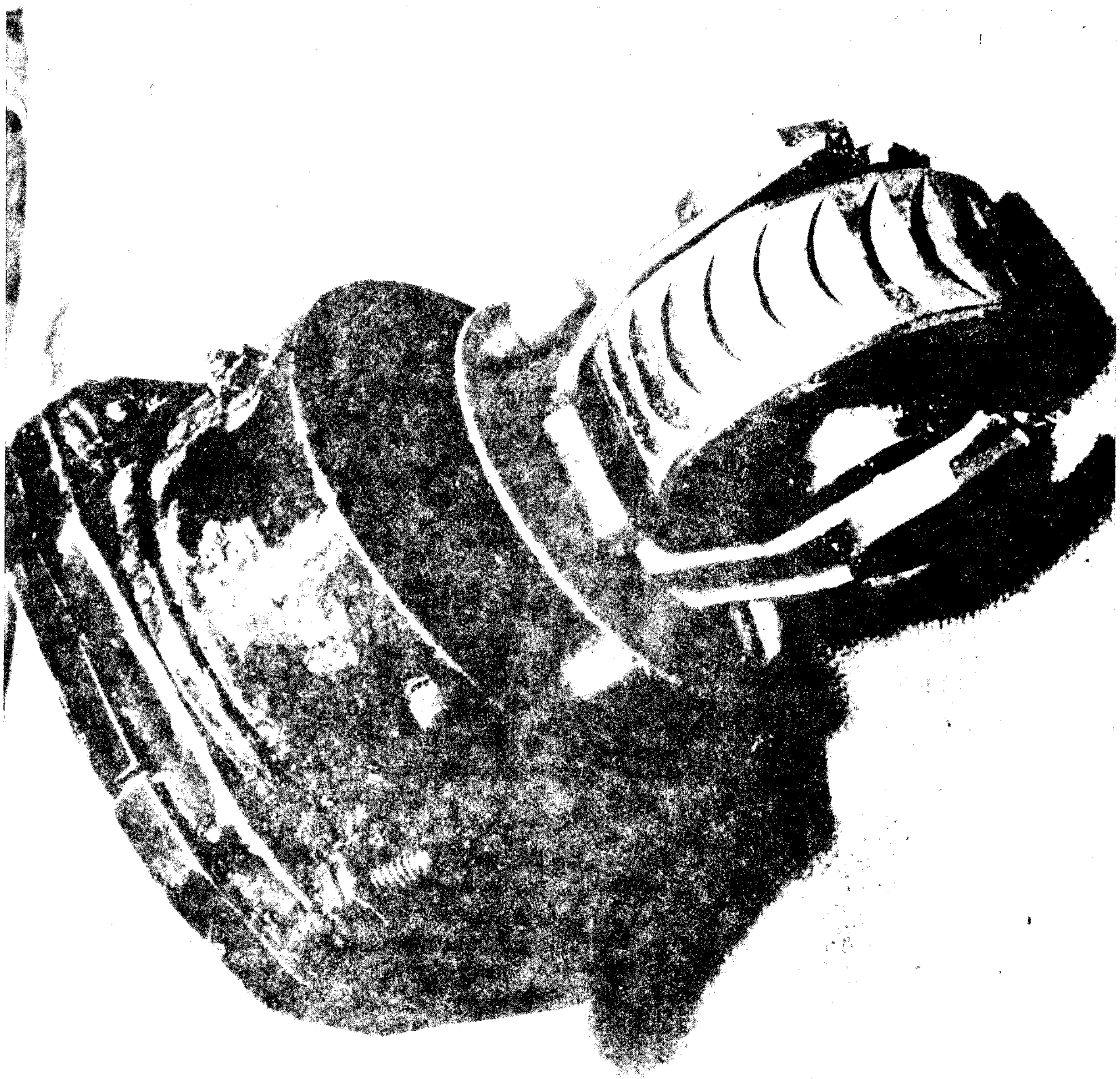
4.2.3. Other Instruments

Other instruments, such as the omnibearing indicator, the omnibearing selector, the ILS indicator, and the distance measuring equipment mileage indicator, can yield meaningful information. With the exception of the omnibearing indicator mounted on the front of the VOR instrumentation unit in the radio rack, all of these instruments display a digital indication of navigational radio bearings. All readings and conditions should be documented. The omnibearing indicator has much the same appearance as the magnesyn compass, with the bearing indication to the selected station shown at the top. The omnibearing selector and the ILS indicator have small windows in which the words **TO** and **FROM** appear; note the one shown and log for later reference.

4.3. Pitot and Static Pressure Systems

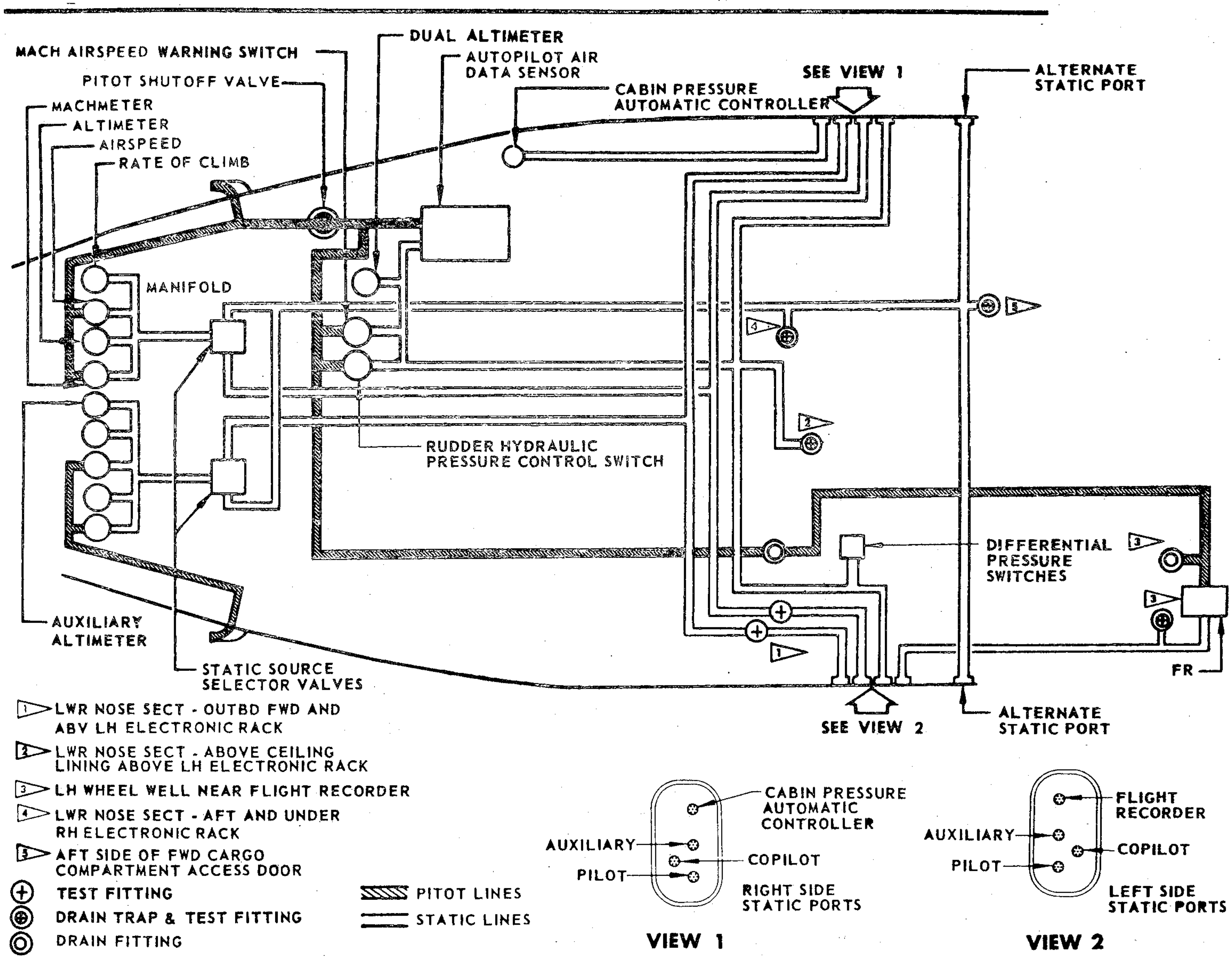
Pitot and static systems are installed in every aircraft to provide the necessary pressures for the measurement of altitude and airspeed. The system complexity will vary according to aircraft type and the instrumentation installed. Some aircraft have both the pitot and static pressure openings in one head unit while modern aircraft have separate pitot head and static ports flush mounted on the fuselage. Some pitot tubes and static port panels contain electric heating elements to prevent freezing of moisture. Heated pitot tubes are required in all transport aircraft. **Figure C V-36** is representative of a typical jet transport pitot static system.

Static pressure openings are installed in areas least likely to be affected by changes in airflow. The separate static system has two parallel openings, one on each side of the aircraft, which provide a balanced system so that yawing or unusual attitudes will not cause erratic instrument operation. The pitot tube is always located to sense the forward ram air. Transport aircraft have dual pitot and static pressure systems, one for each pilot's flight instruments. Aircraft using separate pitot and static systems have an alternate static pressure source controlled by a selector switch on each pilot's in-



Vacuum Turn and Bank Indicator Rotor - Shows Impact Marks at Center of Each Bucket Lip Where Rotor Contacted Air Nozzle.

Figure C V-35.



Typical Jet Transport Pitot Static System.

Figure C V-36.

strument panel. The alternate source is usually common to both pilots' static systems and is selected separately, if the need arises to check for instrument error or malfunction in normal static systems.

Examine the pitot heads for blockage by ice, dirt, insect nests, or other obstructions. The pitot heater element should be checked for continuity at the leads. One clue to actual heater operation at impact is the finding of vegetation or woody material jammed into the pitot tube. The pitot heater provides an intense heat which will dissipate slowly, and if these foreign materials are found to be charred or discolored by heat, certainly the heater was operating at impact. If the pitot head has been exposed to ground fire following the accident, this may not apply.

The static system ports are to be examined for obstructions. Often pieces of masking or cellophane tape have been left covering these ports following the washing or cleaning of the aircraft. Due to static port design and location they are less susceptible than pitot heads to blockage from foreign materials except moisture and ice. If there is a heated static port panel, check the heater element for continuity at the leads.

Recover as much plumbing as possible from both of these systems, check fittings for security, and keep in mind that heat and impact forces can cause looseness. It will be difficult to determine whether plumbing damage preceded or followed the accident.

4.3.1. Suction Air System — Gyroscopic Instruments

The suction air system consists of one or two engine-driven pumps, gauges, regulators, filters, and plumbing. It may consist of an externally-mounted venturi on some light aircraft. This system provides negative airflow pressure to operate gyros in the artificial horizon, directional gyro, and the turn and bank indicators; on aircraft with pneumatic airfoil leading edge de-icing boots it also provides air for the pulsating inflations.

Examine the condition of the drive coupling of the suction air pumps. These usually contain a shear section or cutaway which will fail the coupling under conditions of pump seizure or malfunction. If the coupling is sheared and the fracture surfaces are somewhat polished, it is evidence that the coupling sheared and the engine continued to run. If the fracture is rough, it is probable that sudden engine stoppage caused the coupling to shear. If it is determined that the coupling sheared prior to the accident, look into the system to learn why. It may be that the pump bearings failed, causing seizure, or that the pump failed internally. Determine whether or not this failure was reflected in the instrument operations as a contributing factor to the accident.

The suction pump contains a fusible plug (required by FAA) which will melt at a given temperature, to relieve the pump in case a malfunction occurs and produces excessive heat. Do not be misled. In some cases this plug is found melted, yet the pump can be tested satisfactorily. The fusible plug may melt because of heat from a generator or accessory mounted near the pump. Such an occurrence renders the pump inoperative with loss of instrument operation, under IFR conditions a possible contributing factor to the accident.

If the pumps are to be tested, for the most valid results it is wise to remove the regulator and filters from the wreckage, and plumb them with the pump to check the regulator setting (approximately 4.5 in. hg. or about 10 c.f.m.) and the filter condition.

Instruments associated with the suction air system contain individual filters, or they may be connected to a central air filter. Examine for conditions of clogging which affects efficiency of gyro operation. These filters clog with tobacco tars and lint from the cabin air drawn through the instruments into the air nozzles directing the airflow against the "buckets" in the gyro perimeter.

4.3.2. Compass Systems

The compass system on most aircraft is simple. Already mentioned were the standby magnetic compass and the directional gyro instrument, completely self-contained, the simplest units.

The advanced systems use the flux gate or gyrosyn type transmitter which may consist of a self-contained gyro and flux valve electrically connected to the various heading indicators discussed previously. Later aircraft are using a separate flux valve tied into an electrically operated directional gyro to provide compass heading information. The flux valve transmitters may be located in each wing tip, or both may be located in one wing tip; in some cases they are located in the rear section of the fuselage. The directional gyro unit is usually mounted beneath the cockpit or cabin floor in the equipment racks of transport type aircraft. **Figure C V-37** shows the five major components of a compass system.

The magnetic compass and the directional gyro have limitations. The compass "hunts" with any movement of the aircraft and it lags in a turn. Another disadvantage is its sensitivity to magnetic disturbance from generators and other parts of the aircraft which become magnetized by atmospheric conditions.

The directional gyro instrument is subject to precession which necessitates resetting the heading periodically to the heading indication of the magnetic compass. With no heading information for reference, the directional gyro could be set erroneously if the magnetic compass was faulty.

Two separate compass systems are usually installed in large transport aircraft, the No. 1 captain's system, and the No. 2 first officer's system. Indicators on each pilot's panel utilize both systems. The captain and the first officer may have two RMDI instruments, one operated from his compass system, and one which repeats information from the other's system. Another arrangement is a single instrument with a switch for selecting the No. 1 or No. 2 compass system, which allows each pilot to check one system against the other. Because of these possibilities, properly identify each instrument's location on the two flight instrument panels, and compare the heading indications.

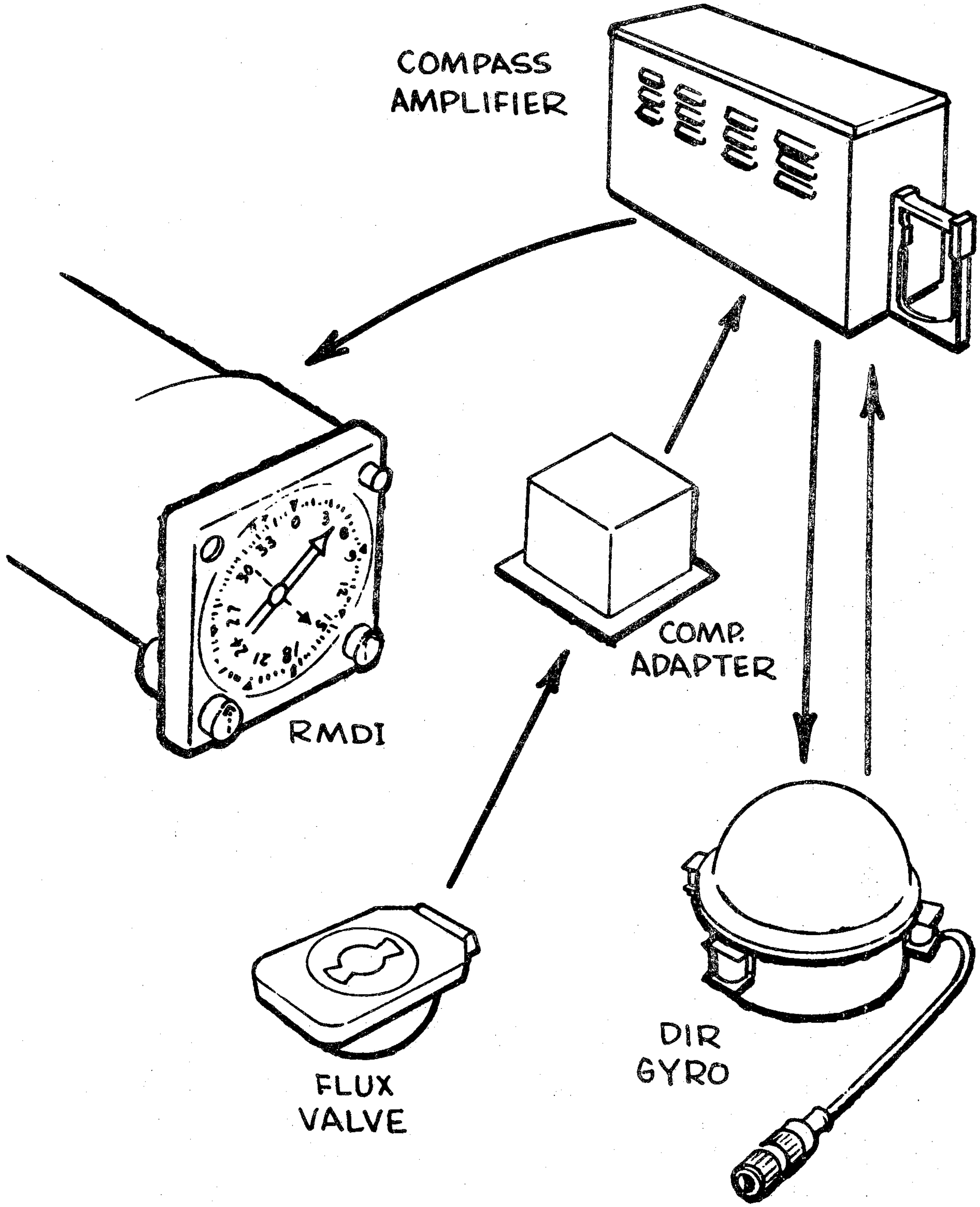
The heading information obtained from the navigational instruments following the accident provides the basic clue to compass system operation. Remember that the rotating compass cards are gear driven by servo motors and may change their last position when electrical power is withdrawn. These indications should be compared with the aircraft heading at impact, when established, and if any appreciable difference exists, look for the cause. The same applies to heading indication differences between instruments. If the instruments on the No. 1 system are in agreement, but the instruments on the No. 2 system show readings that differ at least ten degrees, one compass system may have been operating incorrectly, and it must be identified. This is important, as it could have a definite effect on the contributing cause of the accident.

Compass cards may be distorted so that indications are meaningless, or it may be that the instrument, the compass system, or the electrical system (in whole or in part) malfunctioned or were intentionally deactivated. Sometimes the compass card on a CDI or RMDI may be locked in position at impact. If the aircraft struck the ground in a skid or turned during the ground slide, the heading would be erroneous, but could be the actual heading of the aircraft at the moment of impact. Any questionable readings must be investigated thoroughly to relate the true condition to operation of the compass systems.

The flux gate transmitter, or the separate directional gyro unit and flux valves, should be tested for operation if feasible. Examine the gyro rotors for evidence of rotational scoring. If the directional gyro is intact, a heading will show in a small window on a self-contained compass card. There is a small window in the flux gate transmitter case, but there is no means of heading presentation and the separate flux valve will yield no definitive information.

4.3.3. Vertical Gyro

A vertical gyro is a separate component used to supply attitude information to the pilot's attitude instruments. It is usually located under the cockpit or cabin floor in the equipment racks. New designs have replaced the self-contained gyro horizon instruments. The signals may be used in conjunction with vari-



Five Major Components of a Compass System.

Figure C V-37.

ous "black boxes" to signal small servos and synchros which position the displays and other attitude instruments. Test these units for operation if undamaged, or examine for signs of rotation if they are unfit for testing. Some shrouds have cutouts through which the gyro rotors are visible, some are encased in a spherical assembly without holes, and others are hermetically sealed. Figure C V-38 is a block diagram of a typical integrated flight instrument system showing how the compass systems, vertical gyros, and navigation systems supply information to the pilot's flight instruments.

4.4. Engine and Other Instruments

Most engine instruments yield invalid indications after an accident. Some are meter movements, delicate and easily damaged; others are synchronous, with small magnets set in a delta winding to repeat the positions of magnets in remote transmitters and transducers. The dials of these instruments may be crushed, with pointers trapped in their last position. All readings should be recorded.

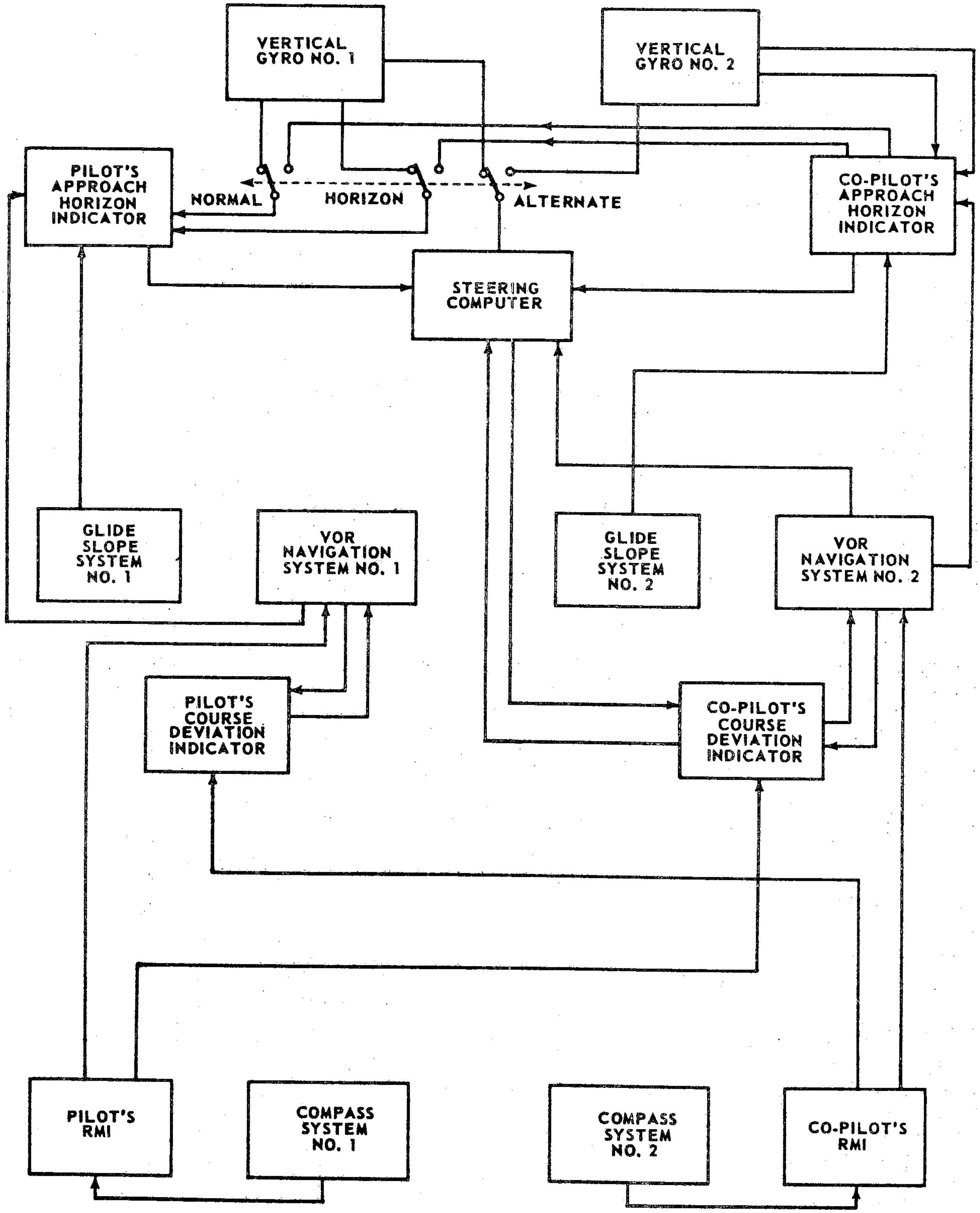
Engine pressure ratio indicators used on jet aircraft engines have counter devices for setting reference values. These instruments display information provided by electronic EPR transmitters, which if undamaged, may be given functional tests. Attempts have failed, because of certain variables, to extract from transmitters evidence of power at impact. The instrument manufacturers claim that these transmitters will not provide sufficient valid information to predict power output, but research is continuing in this area.

Fuel quantity indicators on most aircraft provide no valid clues, even if intact. When the indicators are working properly there is a tolerance of as much as 30 gallons in indication. On some aircraft, specifically the jets, fuel quantity indicators with counters show the fuel remaining in a tank. Some of these aircraft also incorporate a fuel totalizer instrument which indicates total fuel remaining on board. This information may indicate that fuel loading was not symmetrical, which could result in problems of control and trim of the aircraft, or substantiate the amount of fuel on board at the last fueling point, compared with a reasonable burnout for the flight segment involved.

Identify every instrument as to function, position in the panel, and the respective engine, generator, etc., for which it provides indications. Instruments found loose in the wreckage must be examined for the purpose of recording the nomenclature, name of manufacturer, part number, and serial number, for comparison with maintenance records to determine original location. The data plate affixed to the rear of the instrument case supplies this information. The wiring that remains attached to an instrument should be checked for numbers to be traced on a wiring diagram. If an instrument reading cannot be related to a particular engine or system, the reading loses its validity, which is true of components from other systems.

Instruments may be loose or in their panels. If a group of engine instruments is assembled and empty portions of panels are recovered, try to match instrument damage with damage in the panels. Sometimes scratch marks across a panel face will match those found on instrument dials. Instrument cases may be deformed in a certain direction, coinciding with bending damage on a panel. This type of examination will increase your perceptivity and observational accuity. Through practice, eyes and analytical faculties will be trained to spot similarities as a matter of course, and recognition of small details with mental notes as to their meaning will become automatic. Development of good work habits comes only through attention to detail, detail which will sometimes provide the key to the solution of a problem presented in analysis of the facts.

It is not the purpose of this manual to discuss every type of instrument, but to explain why all instruments must be accounted for within the realm of possibility, and why they should be examined. Two types of instruments were explored only to illustrate what can be done with them and to emphasize that every angle must be considered and documented. Notes may be voluminous, but careful documentation will provide intelligent answers to future questions.



Integrated Flight System Instrument Block Diagram.

Figure C V-38.

4.5. Functional Check or Testing of Components or Systems

Because of the delicate mechanisms contained in most instruments, seldom in a total wreckage will one be found fit for testing, however, it may be possible for the instrument manufacturer to extract some information.

Some altimeters have survived crashes and could be tested for operation. One such case occurred when a Constellation slammed down on its belly from an altitude of approximately 200′ and burned. The cockpit was gutted by fire, but the pilot's instrument panels survived; soot from smoke and fire was wiped away, and three of the instruments had only cracked cover glasses.

Both altimeters were indicating approximately field elevation, and the barometric scales were found set to the last reading given to the flight crew. During the week of the investigation both altimeters were observed and readings changed daily with fluctuations in ambient pressure at the airport. Neither altimeter was touched for any reason during this observation. Indications were that both altimeters were operable despite the crash forces and fire. Both altimeters and the rate-of-climb indicators were removed from the panels and taken to an instrument shop.

The four instruments were placed in a pressure chamber along with a third altimeter for comparison; the barometric scale was not changed on either altimeter. Upon evacuation of air from the chamber both altimeters climbed without hesitation, and both rate-of-climb indicators registered the same rate of ascent. At a simulated altitude of 1500′ the chamber panel vibrator was turned on and both altimeters jumped upward approximately 70′ (less than the allowable 100′). The simulated climb was terminated at 3000′, and both altimeters stabilized, with a difference in reading of 10′. Both rate-of-climb indicators immediately returned to zero.

After five minutes, descent was simulated, and the four instruments responded immediately without hesitation at any time; both altimeters returned smoothly to original position. This test proved that the altimeters were in proper working order and could not be considered causal factors in the accident.

This points up attention to detail — the mere noting of a change in the altimeters on the second day suggested checking them on succeeding days, which indicated that a functional check was feasible, an important factor in this particular accident.

Testing of instruments depends upon the circumstances surrounding the accident and whether the instruments are capable of being tested. If testing is advisable, it should be accomplished by an appropriately certificated instrument facility.

4.6. Disassembly and Minute Inspection

As an example of evidence to be found during disassembly, consider the case of an altimeter removed from the wreckage of a Constellation which struck a mountain while maneuvering to land in a snowstorm. It was found that the barometric scale adjustment screw was sufficiently unscrewed to prevent locking with the shoulder on the adjusting shaft. With the screw in this position, it was possible to exert a slight pulling force on the adjusting knob, disengaging the barometric scale from the altimeter pointers and moving the scale independently of the pointers.

The pilot's altimeters had been removed the day preceding the accident and adjustments were made to both of them by a noncertificated mechanic who inspected his own work, used an inspector's stamp, and signed the name of the repair station representative, without authorization.

The altimeters were reinstalled in the aircraft by a mechanic doing this type work for the first time. This mechanic also inspected his own work, closed the instrument panel, and without authorization signed the job as completed. The static system was not checked as required by the maintenance manual, nor was the requirement met to note in the aircraft log book that the static system had been opened, which would advise the next flight crew that the system should be checked.

At the time of the accident the airport at destination showed a pressure of 29.96″ hg.; the pilot's altimeter was recovered with a setting of 29.93″ hg.; the barometric pressure setting at the departure airport was 30.19″ hg. Therefore, a possible error of 280′ could have existed if the pilot turned the barometric scale knob with the pointers disengaged when resetting the altimeter to the pressure provided. This could mean that the pointers were reading 8975′ m.s.l. when actually the aircraft was at 8695′ m.s.l. The aircraft struck the mountain at the 8675′ level. The highest terrain along the aircraft's flight path was 8901′ m.s.l.

In another case, examination of the vertical gyros showed that the nosedown pitch stops were completely broken off when an aircraft broke up in flight. The manufacturer determined that damage was caused by a pitchdown velocity of 50 r.p.m., which is equivalent to a rotation of 300°/second. This information was used to determine the sequence of events when electrical power was lost.

These examples emphasize the importance of thoroughly checking all of the major components in a system, for in this way only can the facts be developed to complete an analysis.

5. Radio Equipment — General

In large aircraft all radio equipment is located in a rack in the cockpit or in racks under the cockpit floor. On smaller aircraft radio equipment could be under or behind the seats, behind the instrument panel, or in the aircraft nose compartment.

Controls for frequency selection, function selection, and volume control are mounted on instrument panels, on the pedestal or console, or on the overhead panels. Circuit protection is located on circuit breaker panels, while equipment protection probably will be included in the individual units.

Antennae for the various pieces of equipment are in different locations on the airframe; some protrude, and others are flush-mounted. Certain protruding antennae can be heated to prevent ice formation, as on the B-727, on which the center engine air intake is installed atop the fuselage just forward of the vertical fin. The No. 1 VHF communication blade antenna is mounted on the forward fuselage and is heated so that ice will not form and break off to be ingested by the center engine.

As in the other systems, procure from the maintenance manual or elsewhere a picture of the location and types of equipment installed and antenna locations. The wiring diagram manual used for the electrical systems will also cover wiring for various radio and related electronic equipment circuits.

The distance measuring equipment (DME) is located with the other radio equipment. This is a relatively new piece of navigational equipment required on all large commercial aircraft. It is available to small aircraft. The DME equipment transmits signals to a VORTAC ground station. The signal is returned to the unit which then compares the time between transmission and reception, and converts this time interval into slant-range mileage information shown to the pilot by an indicator on his panel. Some new course director instruments display this mileage information in a small upper right corner window on the instrument face. Remember that this mileage is slant-range from the aircraft to the ground station, not distance over the ground.

Account for and document each piece of equipment, the various antennae, and lightning arresters.

In addition to pictures and diagrams, obtain a listing of the various aids to navigation in the area of the accident, showing their frequencies and the frequencies assigned to the tower, ground and approach control, departure control, the applicable ATC center frequency, and the company frequency used by the crew in that particular location.

5.1. Visual Inspection and Documentation — Cockpit Controls

Cockpit controls consist of frequency control heads for each type of radio equipment installed. A single control head with dual functions for the VHF units selects the basic frequency and the decimals of the basic frequency. Also mounted on the same head is the volume control knob. The frequency is displayed by a digital counter in megacycles (Mc). A switch for turning on the DME or placing it in standby is the only control for the DME, since it is automatically tuned to the proper channel when the No. 1 VHF NAV receiver is tuned. If two DME units are installed, the second will be tuned automatically by the No. 2 VHF NAV unit. A VORTAC station has an assigned channel as well as VHF frequency. When the proper frequency is selected, the DME unit is automatically tuned to the channel assigned to that station. Not all VOR stations have the DME facility, in which case, the DME will go into the search mode and the DME mileage indicator will be inoperative for the time being.

The LF or ADF frequencies are selected by a "coffee grinder" handle with a separate volume control. A frequency range selector may have three or four positions. Found on some of the later equipment is a fourth range which handles frequencies below 200 kc. Also mounted on this panel is a selector for ADF, loop, or antenna. A small window displays the frequencies of all the ranges. All readings on this unit are in kilocycles (kc.) and when a particular range is selected the others will be masked.

Another type of control panel is the audio selector panel, one provided for each flight crew member, the observer, and navigator. These panels contain the switches and selectors for communication transmission and reception, and the switches for monitoring the navigation equipment signals and identification. These should be recovered and identified as to which position they serve.

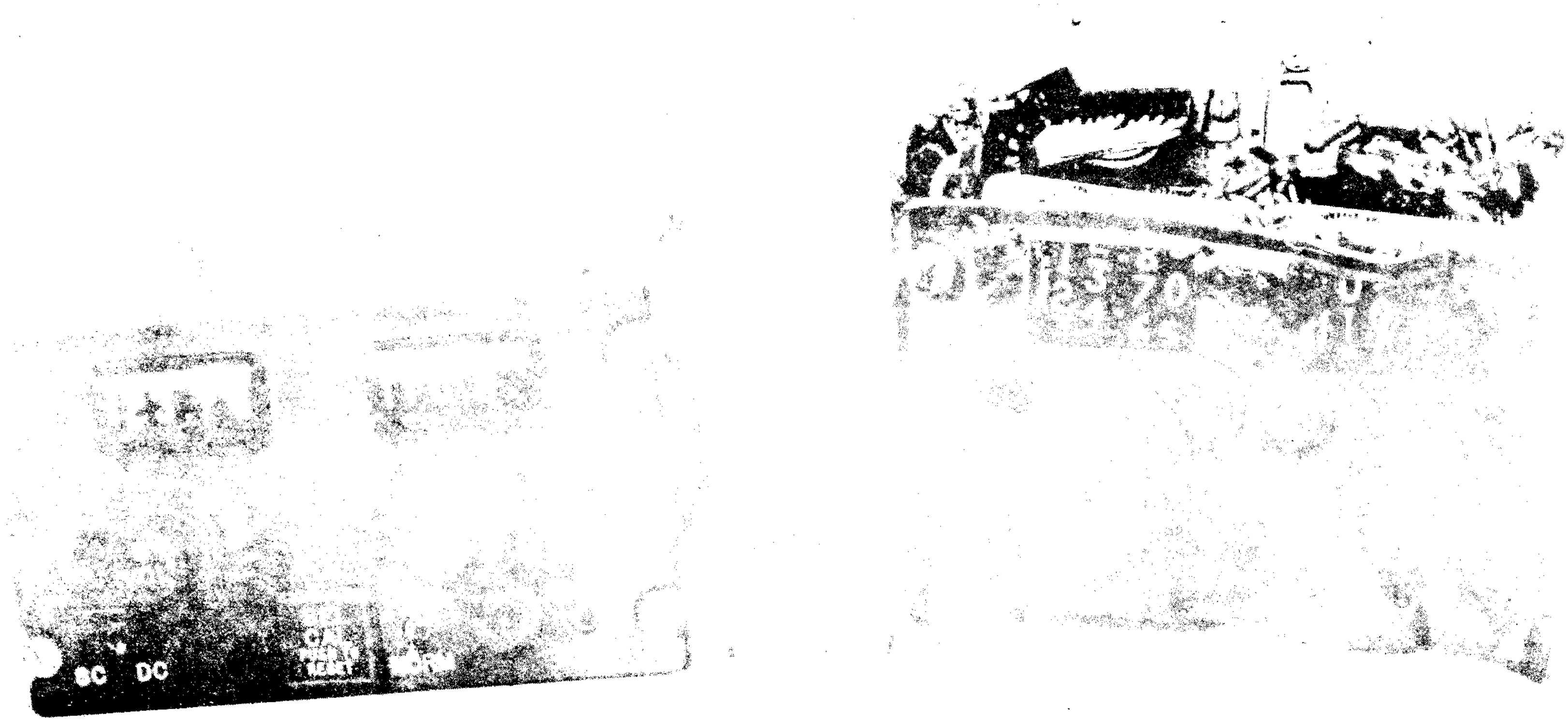
During cockpit documentation, readings, from each frequency selector and the switch positions on control and audio panels should be obtained. Each frequency reading should be related to its proper unit (Example: "No. 1 VHF NAV receiver – 108.9 mc., No. 1 ADF – 200-400 kc. range, frequency 359 kc.," etc.). Sometimes damage to the control head obliterates the numbers, in which case, open the box and relate the visible numbers to the damaged area to interpolate the selected frequencies. Index the position of the volume control knobs as found, turn them from one extreme to the other, and estimate the percentage of travel to figure the volume which was being used. **Figure C V-39** shows a damaged control head compared with an undamaged unit to verify frequencies selected and volume levels.

Log all switch positions by panel and note any damage to the panels which could have changed a switch from its previous position. Reference all frequencies noted on the panels to the list of frequencies suggested earlier. Avoid conclusions as to which radios were tuned to which frequencies until the equipment has been examined. Note which stations should have been selected according to the regime of flight prior to the accident. Note the mileage readings on the DME indicators. The list of frequencies of VHF NAV aids should include the DME channels assigned if they are VORTAC facilities.

5.2. VHF Communications Equipment

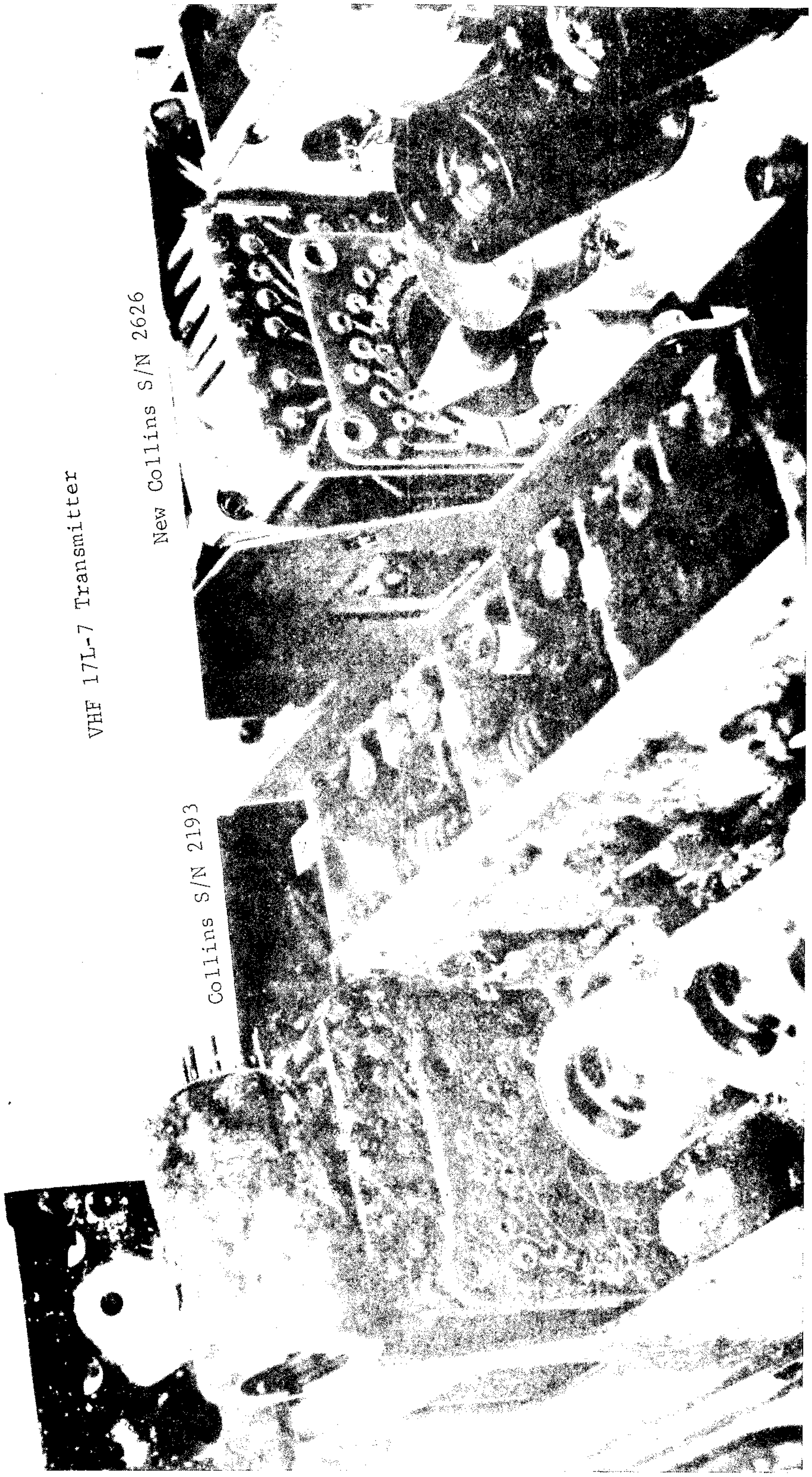
Two VHF transmitters and two receivers, and occasionally a transceiver, are installed on large aircraft. These units usually utilize electro-mechanical switching for frequency selection with two indicators under protective cover mounted on the front of the box. One of these rotary indicators contains the basic frequency numbers and the other contains the decimal frequency numbers. Remove the front cover, copy the numbers visible in the cutouts, and total them for the selected frequency. This selection cannot change through handling because the rotary indicators are chain or gear driven.

It sometimes happens that the front of the box is broken and the indicators are missing. It is then necessary to examine the rotary switch to which the crystals are attached. Each crystal has a specific value in kilocycles, which is multiplied within the unit so that a particular pair of crystals provide a specific frequency. Examination reveals which pair of crystals was selected. Note the crystal frequencies, and contact the equipment manufacturer for a frequency determination. **Figure C V-40** compares a damaged VHF 17L7 transmitter with an undamaged unit to assist in determining



VHF-VOR control heads compared to verify frequency selections and volume level.

Figure C V-39.



Rotary switch indicating selected crystals.

Figure C V-40.

frequency selection by checking rotary switch to find which pair of crystals was in use.

If neither method is feasible, note position of the battery of tuning slugs inside the unit, and take the equipment to the manufacturer for a detailed examination and an approximation of selected frequencies. It is not prima facie evidence that his radio was on the proper frequency because a pilot was talking to the tower or approach control. Verify. **Figure C V-41** compares physical position of tuning slugs in damaged and undamaged VHF receiver units to verify frequency selection.

Each radio unit should be given close visual examination for evidence of electrical burning, arcing, or overheating, by odor and by sight, on units not exposed to fire. Check for broken tubes, and examine heater filaments for discoloration due to oxidation which will occur when a heated filament comes into contact with air. Note wires broken away from switches, capacitors, resistors, etc., which may have disabled the equipment. If the equipment is broken or crushed, it is difficult to detect preexisting damage.

If the unit is damaged by fire, determine whether it was ground or inflight fire. Documentation of this equipment must identify each unit by nomenclature, part number, serial number, make and model, and position in the aircraft.

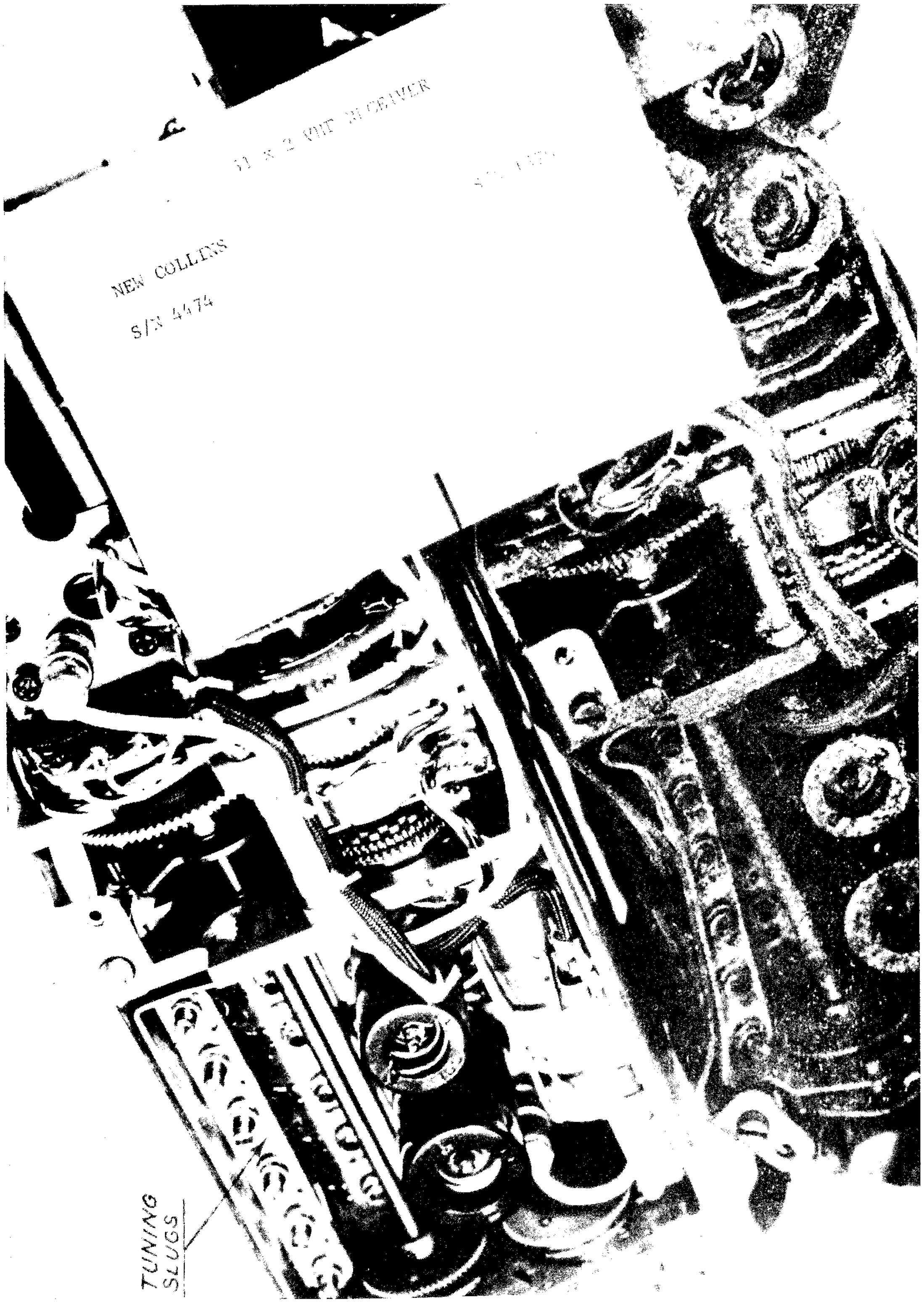
5.3. VHF Navigation Equipment

The VHF navigation equipment usually consists of two receivers, a VOR accessory unit with two omnibearing indicators, and two glide slope receivers which operate only during an ILS approach. The FAR's require only one glide slope receiver. VHF NAV receivers are similar to the communication receivers in that they have many channels, all of which may not be in use. External and internal examination of these units will be the same as in the communication receivers. Some equipment uses purely electronic switching for frequency selection, so these units must be tested to determine selected frequencies.

One VOR instrumentation unit may serve each VHF NAV receiver or one may serve both. One omnibearing indicator is on the single, and two are on the dual unit. The gear-driven OBI remains in its last position when electrical power is removed. Document the readings found on these indicators, for they provide a bearing from the aircraft to the selected station. These OBI's do not operate when the receivers are tuned to an ILS frequency. Figure C V-42 shows omnibearing indicator readings on two Wilcox Model 706A VHF navigation receivers.

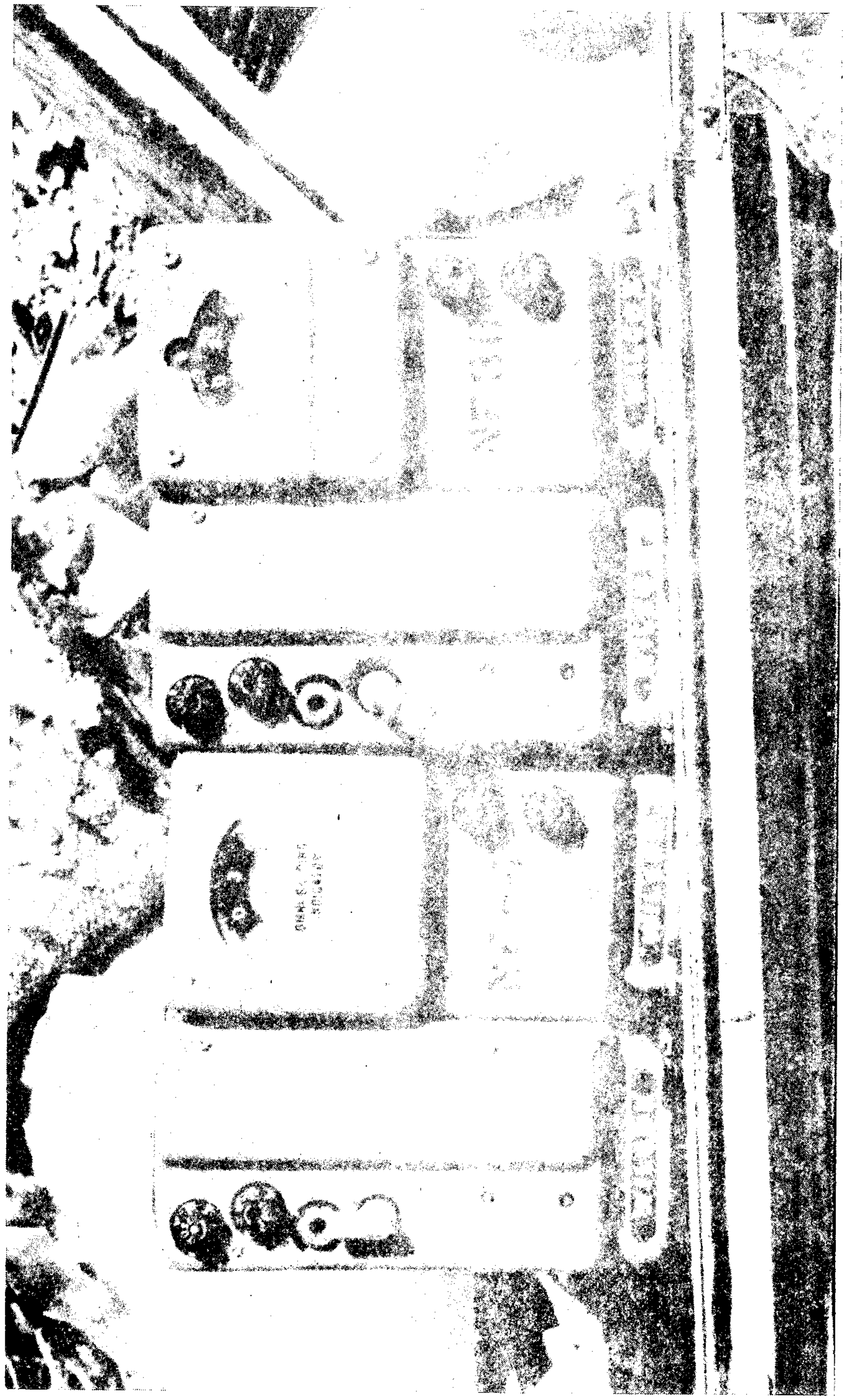
Bearings relating to VOR or ADF stations, indicated by the pointers, of the RMDI's may be very important to the investigation as more facts become known and a probable flight path is established. When the selected frequency is established and the station is located in relation to the wreckage, plot the bearings on the applicable aeronautical chart. If the plotted bearings do not connect the accident scene and the station, plot the reciprocal of the indicator readings from the station to determine their relationship to the probable flight path. This procedure may provide a clue as to when the electrical power to the radios was interrupted.

Another factor to consider when examining the RMDI is whether or not the heading presented is reasonable in relation to the probable heading of the aircraft at impact. If it is, and the RMDI pointer bearing is within reason, then electrical power probably was available to the radios and the compass system right up to impact. If the bearing is not reasonable in relation to impact, then suspect that radio electrical power was lost at some point prior to impact. By drawing the reciprocal of the observed bearing back from the station to the probable flight path, this point should be established. One way to check the RMDI pointer bearing is to compare it with the omnibearing indicator on the VOR instrumentation unit for the particular receiver. If all equipment was performing properly, they should agree; if not, then suspect and seek a failure someplace in the system. The fault could be in the instrument, the instrumentation unit or related circuitry, or the receiver itself. **Figure C V-43** shows a systems investigator checking RMDI



VHF receiver frequency selection verified by tuning slug position.

Figure C V-41.



Wilcox VHF navigation receivers showing omnibearing indicator readings.

Figure C V-42.

pointer bearings at a turboprop aircraft accident scene.

The glide slope receivers operate in the UHF range and their 20 assigned frequencies (from 329.30 to 335.00 mc.) are paired with the 20 assigned ILS localizer frequencies (from 108.1 to 111.9 mc.). Switching is automatic through relays. When a certain ILS frequency is selected on the VHF NAV receiver a relay automatically selects the crystal for the proper glide slope frequency. Once electrical power is removed, the relay relaxes and the glide slope receiver is out of operation. It has no memory.

Interior of these units are examined for evidence of electrical burning, arcing, or overheating. Blowers installed for cooling of rack-mounted components should be examined for evidence of rotation. Some aircraft have several small AC motor-driven fans; others have one large fan. Some of the larger fans are installed in ducting which draws air over both radio and electrical equipment racks. This ducting should be examined internally for evidence of smoke.

5.4. Distance Measuring Equipment

DME equipment consists of a single radio rack-mounted unit, with vacuum tubes or transistors. Usually a small cooling blower is installed in this unit. The tubes should be examined for evidence of normally heated elements at the time of impact, and the cooling fan for evidence of operation. The forward portion of the DME unit contains a device called the *distance mechanism module*. This unit is gear driven by small servo motors and will not coast after electrical power is removed. It may have one or two indicators which show mileage in tens and units through holes in the top and front of the unit. By adding the two readings, determine the nautical miles from a selected station. **Figure C V-44** is a top view of a module reading 9.6 miles. This module may be retained in the unit or it may be found loose in the wreckage -- it must be recovered. On top of the case at the rear two small windows show the selected channel. Log the channel shown, and compare it to the VORTAC stations on the list. Only 126 channels are assigned, but there should be no duplicates in any given area of the country. Some ILS installations are being programmed to provide DME information.

The channel, mileage, whether or not the unit was searching or tracking, and whether the unit was channeling at the time of electrical power failure or if it was locked to a station, can best be determined by the manufacturer. His facility may find evidence of malfunction not apparent to the investigator. Remember that the DME may not have been operating on *any* channel if a VORTAC station was not tuned in.

If the accident occurred en route and the VHF NAV receiver (No. 1 only, if only one DME was installed) is tuned to a VORTAC station, the mileage indication on the distance module in the unit will be the most accurate. Compare this indication to the measured nautical miles from the accident scene to the pertinent VORTAC station. This reveals when electrical power was lost to the DME (but not necessarily to the whole system), if the readings are very far apart.

To illustrate: In an accident involving inflight fire just prior to the crash the DME module read 47 nautical miles. Examination disclosed that the DME was selected to the channel assigned to a VORTAC station just ahead of the flight and that its distance from the accident scene was approximately 42 nautical miles. This indicated that electrical power to the DME must have terminated five miles back, but unfortunately, it did not defintely prove that electrical power was lost at that time, approximately one minute prior to the accident.

Another example concerned an aircraft which struck a hill during an IFR approach in heavy snow. The approach required an operable DME, since there was no ILS facility. The distance mechanism module, found out of the DME unit, was reading 9.6 nautical miles from the VORTAC on the airport, within less than one-half mile of the measured distance from the accident scene to the airport. This indication, along with the fact that the flight data recorder was found to be operating right up to

Systems investigator checking RMDI pointer bearings at a turboprop aircraft accident scene.

Figure C V-43.

the moment of impact, proved that the electrical power system was operating.

5.5. Low Frequency (ADF) Radio Equipment

Two types of ADF equipment may be in the aircraft, the older type capacitor-tuned by a flex shaft driven condenser, or one of the later types, strictly electronically tuned. None of the ADF receivers has frequency indicators like VHF units, so determination of previously-selected frequencies will be more difficult. All work on the receiver and its related components should be performed in a radio shop or certificated repair station where similar equipment is available for comparison.

During cockpit documentation the band and frequency selected on each ADF control panel should be logged. If the reading is obliterated by fire or physical damage, disassemble the control panel and examine the frequency scale. Note the band cutout position on the mask as selected; it should be in the center of the viewing window on the panel, but if the panel has been damaged the band may be off center. The band selector knob may have been displaced from the detent. If so, determine by analyzing the damage, in which direction the knob moved, then position it in the appropriate detent so that the proper portion of the frequency scale is visible beneath the cutout. Scribe the frequency scale by outlining the shape of the cutout and remove for examination. **Figure C V-45** ADF frequency selection determination.

To additional frequency indicators are to be found on ADF control panels. On the first, the frequency dial is displayed on a drum, providing equal dial lengths for the three bands. The band selector actuates a masking drum which surrounds the frequency dial drum, revealing only the band in operation. This is the same arrangement as the masking plates with cutouts. The second type is a digital display controlled by three knobs by which the pilot dials electronically the desired frequency without the use of the flex shaft. These frequency indicators may be examined in the manner discussed earlier. The digital type is the most susceptible to change from contact with foreign objects during aircraft breakup.

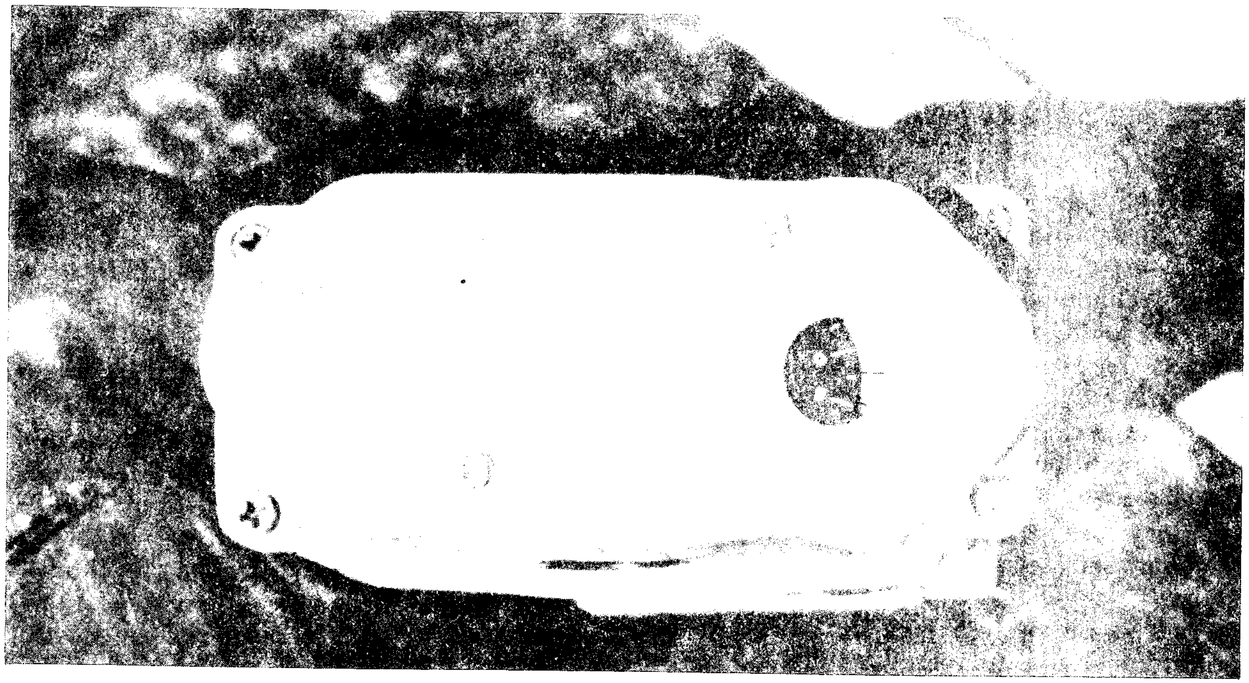
Cockpit documentation covered the RMDI's and it should have been determined whether or not any of the pointer-indicated bearings were related to the ADF radios. If so, then use the same method of plotting ADF bearings as used with the VHF NAV bearings, to confirm that valid information is provided.

ADF, and all radio equipment requiring tests should be taken to a radio shop, along with the ADF control panels and the two ADF loop assemblies. The position of the loop assemblies can be related to the bearing to a selected station. When removing the old type ADF receivers from the wreckage, disconnect the flexible drive shaft to prevent changing the tuned position of the condensers. Using this type of receiver as a start, remove the case or dust cover so that internal examination may be made. If the unit is not severely damaged, it may be possible to perform a functional test; new tubes may be substituted for broken ones. The selected band should be known by the testing personnel; the functional test will verify.

If the receiver is damaged beyond testing, compare with a normal receiver. Hook a flex shaft to the latter, using the same kind of control panel; position its condenser to match that of the damaged receiver. Since the selected band is known, the good receiver can supply the selected frequency.

This method may be used to compare a damaged frequency scale. Match the scribed area with a new scale, and take the center of the area scribed to determine the selected frequency. Relate this frequency to the list of navigation aids to find the selected station. It should be reasonably close to one on the list, unless the aircraft was en route, navigating on VHF alone.

The newer types of ADF receivers have no moving parts for comparison, and they can only be functionally tested. Insure that the radio shop selected for tests has the proper specifications.



Top view of distance mechanism module reading 9.6 miles.

Figure C V-44.

5.6. Miscellaneous Equipment — Marker Beacon Receiver

The marker beacon receiver operates on the sole frequency of 75 mc., therefore, it is a simple receiver without moving parts or memory. Special equipment is required for the only test on this receiver, that is, whether or not it was capable of receiving a signal from a marker. The beacon function of all radio facilities, ILS markers, fan markers, etc., is transmitted on 75 mc.

5.6.1. ATC Transponder

The ATC transponder is used primarily for inflight identification by air route traffic control facilities. It is similar in some respects to the IFF equipment of WW II. A control panel for selecting the proper code is located in the cockpit. When radar identification is initiated, upon request the flight crew *squawks* a certain code. When selected on the control panel, the "ident" button will be pushed and this particular emission from the transponder will expand the image so that the aircraft can be identified by the ground radar controller. The transponder is required to operate at all times during flight. Normally, a transponder-equipped aircraft provides a more conspicuous radar image than "skin paint," and the "ident" function makes the image even more conspicuous. If the transponder ceases to operate, the radar image changes noticeably, which opens a new avenue of exploration with regard to the presence of electrical power. Check with the ATC Group Chairman to learn whether the radar controller handling the flight at the time of the accident noticed indication of transponder failure during his contact with the flight, and at which point. If he did not, consider that electrical power was available at the time of the accident.

This may point to electrical power failure to the transponder only, for an approximation as to where in the flight path the failure occurred.

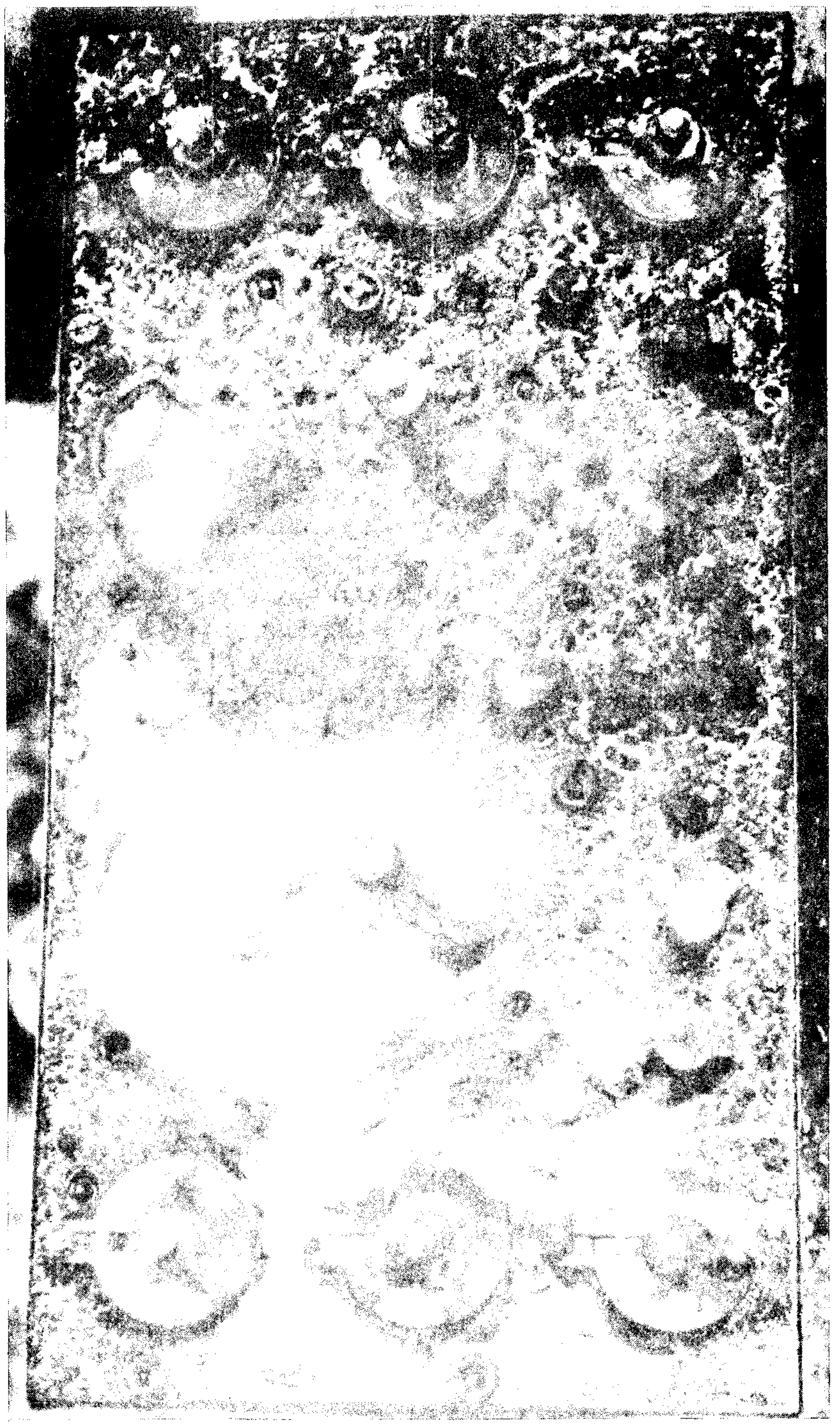


Figure C V-45.
Radio selector panel ADF frequency selection determination.

Its relation to total electrical power failure can be determined only by analyzing facts developed in examination of all electrically-operated components.

5.6.2. Weather Radar

The weather radar in itself can provide very little evidence. However, its AC-operated cooling blower can be examined for indication of operation at the time of impact.

5.7. Functional Testing and Disassembly

Fully document all tests and disassemblings, with notes, forms, charts, photographs, etc. Any evidence of poor design, abnormalty, or discrepancy noted during testing should be recorded, and a recommendation for corrective action should be initiated.

6. Pneumatic Systems

The primary source of pneumatic supply in older aircraft may be ram air, or a combination of ram air and engine-driven superchargers or compressors. Turbine-powered aircraft use bleed air from the engine compressor sections. Air from these sources is used to condition air, to pressurize the aircraft for passenger comfort, and to provide protection against ice formation on the wings, empennage, and cockpit windshields. **Figures C V-46 and C V-47** describe and depict a typical jet transport pneumatic system.

The air conditioning systems may contain gasoline-fired heaters for warmth, and expansion turbines or refrigeration units for cooling. The thermal anti-icing systems may use gasoline-fired heaters or compression-heated bleed air from the turbine engines. The pressurization systems utilize air from the engine-driven superchargers and compressors, or bleed air from the turbine engines.

Many components and ducts in these systems are required for regulation of airflow and temperature – turbo-compressors, air-cycle machines, or cooling turbines, temperature and flow sensors, pressure regulators, valves, heat exchangers, and various fans for recirculation or airflow augmentation. Some turbine-powered aircraft also have air storage bottles to provide for engine starting when ground equipment is unavailable.

The turbine-powered aircraft incorporate the most complex pneumatic system. One basic problem is the identification of various valves located throughout the aircraft. The same part-numbered valve may be used in any number of places, and it may have different functions. An up-to-date listing of these valves, components, and locations within the system is mandatory. Identify the valves and ducts by serial number. The applicable parts catalog can be invaluable in establishing location and manner of installation of the components.

Documentation of the cockpit provided the control arrangement for air conditioning, pressurization, and anti-icing. This should be compared with system operation requirements to provide positions of the various valves and evidence of inflight malfunctions which could have a bearing on the other systems.

An example of what can happen concerns a turboprop transport aircraft in which a portion of the empennage anti-icing air duct failed in the area of the aft lower baggage compartment during ground start operations with the No. 1 and No. 4 engines running. This failure, in which the duct split lengthwise along the welded seam line, caused insulation to be blown into the cabin area and engine bleed air to pressurize the main cabin. Examination of the failed area indicated a type of break unassociated with excessive air pressure; the failure occurred from weakening of the duct seam by severe intergranular oxidation or corrosion probably initiated during manufacture of the duct.

This happened with only two engines running at idle speed. Imagine the results if it should happen during cruise, with four engines operating. The temperatures in the duct approach 150°F, twice the temperature desired in the cabin. Heated electrical wiring could cause erratic operation of other components, and pieces of insulation blown into the cabin area could enter the pressurization outflow valves. The necessity is apparent for examination for indications to be related to operation of other

PNEUMATIC

GENERAL

Air for the pneumatic system is bled from either of two different sections of the engine, and distributed through a pneumatic manifold. This manifold extends from both outboard pylons, spanwise through the wing, and forward in the fuselage to the nose section. In the nose section, the left and right sections of the manifold are joined by a crossfeed valve.

The pneumatic system air supplies the following dependent systems:

Cabin Air Turbocompressor System

Refrigeration System

Airfoil and Wing Leading Edge Slot De-Icing System

Airscoop and Engine Anti-Icing System

Windshield Rain Removal System

Thrust Brake System

Engine Pneumatic Start System

The pneumatic system controls and indications are provided on the systems engineer's panel.

Typical Jet Transport Pneumatic System.

Figure C V-46.

systems or possibly the accident cause. **Figures C V-48** and **C V-49** show the failed pneumatic anti-icing duct and enlarged views of the intergranular damage.

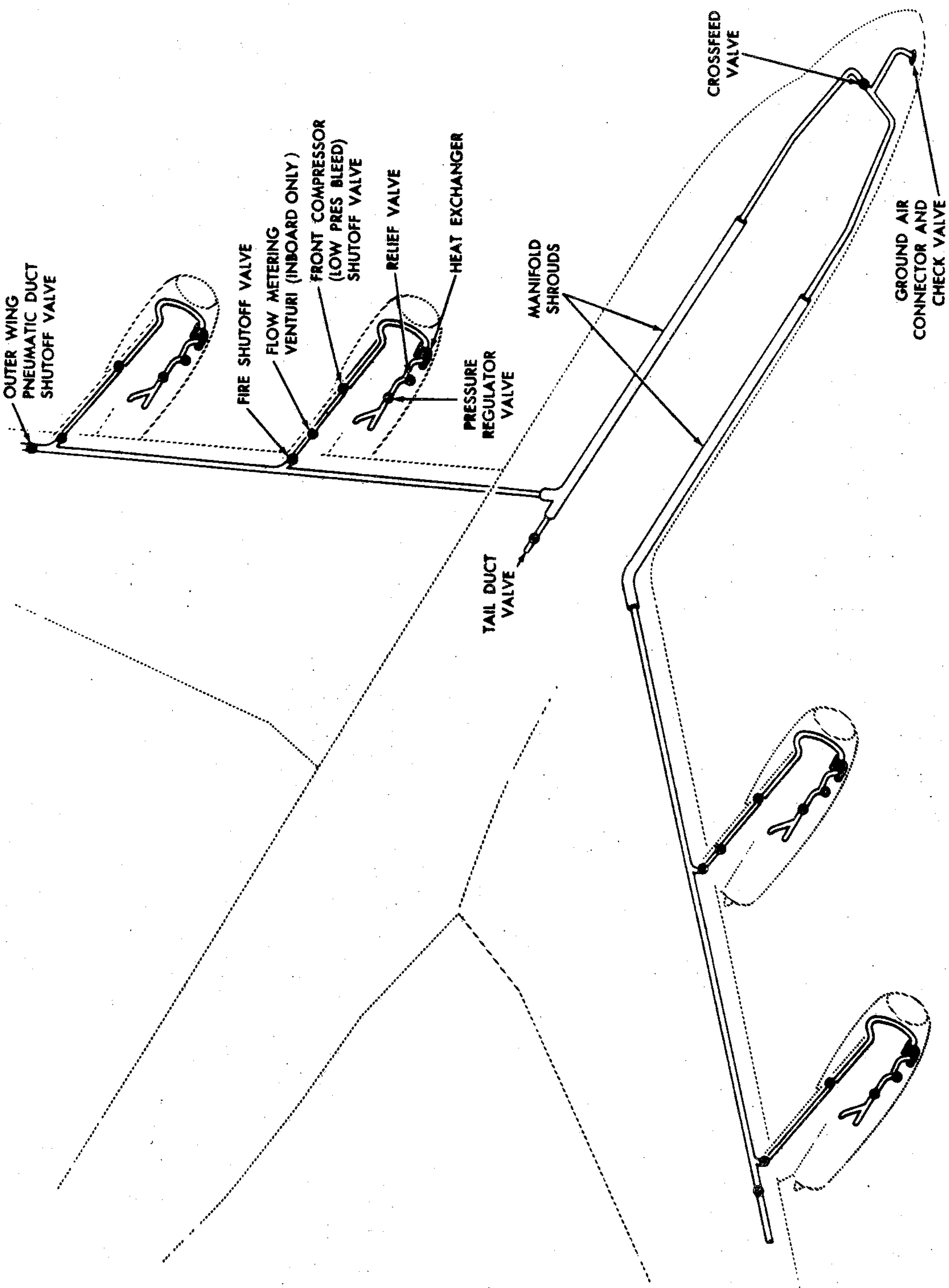
6.1. Visual Inspection and Documentation — Engine-Driven Superchargers

Engine-driven superchargers are found on the DC-6/7 and Constellation aircraft; they are driven by the outboard engine accessory section gearing and provide air for pressurization and air conditioning. They may be declutched or disengaged in flight and re-engaged only by stopping the engine.

The Douglas supercharger incorporates two hydraulic pumps for output control. Both are piston-type positive displacement pumps, one fixed, and the other variable. These pumps control impeller speed, and are disengaged by cables attached to handles in the cockpit mounted on the floor to the right of the first officer. The declutching mechanism on the drive shaft is springloaded to the engaged position, so the disengage handles are latched in the disengaged position. Log the position of these handles for comparison with the condition of the supercharger drive mechanism. In the DC-6B/7 models mounted in the same location, and between the disengage handles, is an emergency depressurization lever (Johnson Bar), and when this lever is pulled up it automatically operates the supercharger disengage handles and other devices. The position of this handle should be logged.

Examine the supercharger impeller for evidence of interior scraping of the scroll or case, which would indicate rotation at impact. Indentations where the scroll or case was crushed against the impeller blades would indicate nonrotation. Examination of the blades helps in any determination.

The two hydraulic pumps should be examined in the same manner as those of the hydraulic system. Yoke position of the variable

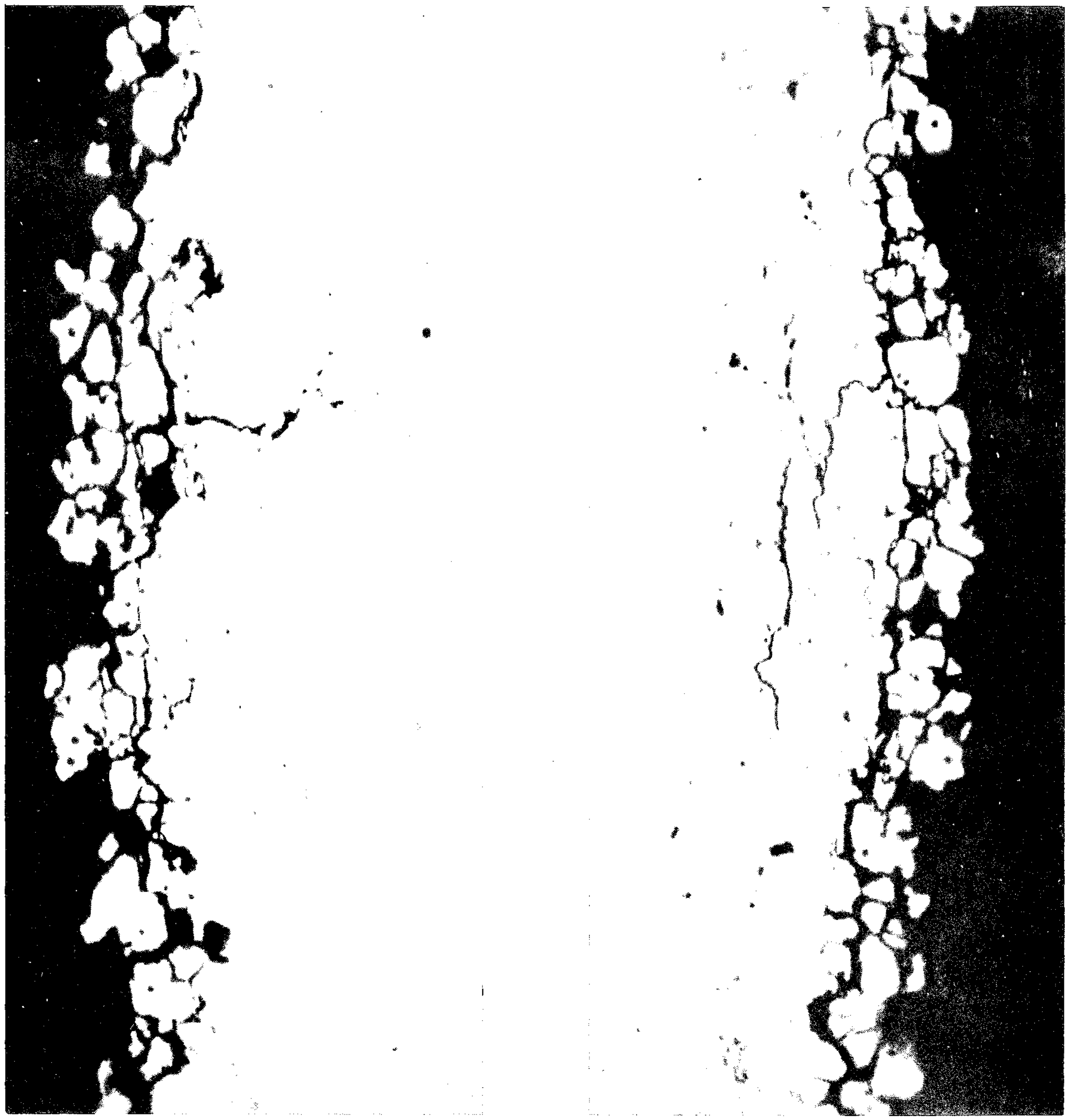


Pneumatic System Plumbing Typical Jet Transport

Figure C V-47.

Failed anti-icing duct, as received X 1/5.

Figure C V-48.



Enlarged views of intergranular damage.

Figure C V-49.

pump may also be revealing since a high yoke angle is associated with low engine speed. The supercharger hydraulic system tries to keep the impeller speed up to air conditioning and pressurization requirements. The system senses a decrease in impeller speed as the engine speed decreases, and the variable pump yoke must move to a higher angle to increase pump output and impeller speed.

The Constellation supercharger is similar, but it uses a single gear-type pump which provides hydraulic control through various valves. Older Constellations have two variable displacement pumps, driven by the drive shaft from the engine, which provide the same control. Two other pumps mounted together on some aircraft provide for lubrication, and scavenging of fluid. Examine the impeller and scroll for rotation indications and the pumps for condition and evidence of operation. Check whether one or both superchargers have been declutched or disconnected from the engine drive by cable. They can be reset only on the ground.

6.1.1. Engine-Driven Compressors

Engine-driven compressors are usually found only on the Convair 240/340/440 models, mounted on the right-hand or No. 2 engine. The compressor is declutched by a solenoid which releases the disconnect rod; the rod contacts the threaded area on the drive shaft, and causes the shaft to retract and disconnect from the engine. It can be reset only on the ground. The solenoid is automatically powered by the left-hand or No. 1 engine autofeathering circuit, or may be operated by a switch on the first officer's console. The two-speed impeller is controlled hydraulically. Examine these units in the same manner as the superchargers.

6.1.2. Engine Bleed Air System

The engine bleed air system will be covered by the Powerplants Group, however, how it fits into the pneumatic systems should be understood. Bleed air is usually taken from the 8th and 13th stages of the engine compressor sections to operate turbocompressors (B707/720, CV880/990, DC-8), or is sent directly through the systems (B727, DC-9). The turbocompressors operated by bleed air take ambient air and route it through the air conditioning systems. Once this bleed air has passed from the engine plumbing the Systems Group is responsible for the investigation of the pneumatic systems.

6.1.3. Air-Conditioning System

Using the schematic diagram for air conditioning, account for all major components shown, and document conditions and positions. Turbocompressors, expansion turbines, and motor-driven blowers and fans should be examined for evidence of operation. A review of the particular system involved indicates which should be running, in the properly operating system; the same applies to the various control valves throughout the system. This is why each valve, compressor, blower, and fan must be identified as to its function in the system.

Ducting will be of combinations of fiberglas, aluminum, and stainless steel. All ducting should be examined internally and externally for evidence of smoke or fire. If such evidence is found, ground fire damage notwithstanding, determine the source. At times, the air cycle machines and other turbines may leak lubricating oil into the ducting. At the temperatures which prevail, especially those systems operated from engine bleed air, this oil will vaporize and enter the habitable portions of the aircraft as smoke or fumes. The examination of turbine or compressor will reveal such leakage. Identify and examine adjacent downstream ducting from these units. If evident in the rotating unit, but not in adjacent downstream ducting, conclude that the leakage resulted from crash damage.

Examination of electrical wiring for evidence of electrical burning or overheating should be expanded to include damage to wiring by heated air escaping from loose duct joints or duct failures. In some cases this heat is sufficient to char the wiring insulation and cause electrically-operated units to become erratic; circuit breakers may trip, and fire or overheat warnings may occur.

These events can occur at inopportune times; they can distract the flight crew when utmost attention is required to navigate and control the aircraft. Heat damage to wiring, with resultant circuit breaker tripping, can result in loss of some navigational instruments and equipment. Because a particular system has no relation to the actual flying of the aircraft, do not assume that it could not have been a causal factor in an accident. So many interrelationships are involved each system must be investigated.

The positions of the various ram air inlet and outlet doors should be logged for comparison with those expected in a normally-operating system. The motor-operated actuators for these doors should remain in the last position following loss of electrical power, and provide the most valid evidence. If the associated doors are in no condition to be examined, compare the actuator with one on another aircraft (a type of comparison to keep in mind for any actuator).

The aircraft heaters may exhibit evidence of fire outside the combustion chamber; fuel plumbing may evidence leakage which could have contributed to fire in flight. If more than one heater is installed (the DC-6/7 aircraft have four identical gasoline-fired heaters,) identify each by serial number or by portions of attached structure.

The following is an example of what can occur. A DC-6B, ferried from Miami to Seattle, received a tail heater fire warning en route. Because of an improperly attached heater exhaust, hot gases escaped and were sensed by the fire detectors mounted in the tail area. Mechanics repositioned the exhaust outlet on the heater and applied a generous amount of sealing compound around the joint. The outgoing flight engineer noted that the work was done, crawled up to the tail compartment access door to check, and noticed a pungent odor which the mechanics attributed to the sealing compound. The flight took off for Anchorage.

Just before reaching the assigned flight altitude of 18,000′, the crew turned on the anti-icing heaters because of cloud conditions. Fifteen minutes later a tail heater fire warning was received. Through a small window in the rear pressure bulkhead the flight engineer saw a blue flame about a foot high coming from the exhaust shroud joint on the heater. He reported this to the pilots (who immediately depressurized the aircraft and turned back for Seattle), and entered the tail compartment to fight the fire with a portable extinguisher. Investigation disclosed that the sealing compound used by the mechanics was contaminated with a solvent; when the heater was turned on in flight, the fumes ignited.

6.1.4. Pressurization System

Pressure in the aircraft is automatically controlled, but at times manual control is required. The cockpit control panel settings should be recorded and related to the positions of such components as the cabin pressure control valve, outflow valves, and emergency depressurization valves.

All pressure regulating valves should be recovered and examined. Cabin insulation occasionally has clogged an outflow valve, causing pressurization to be erratic or impossible. Some of the emergency depressurization valves are cable operated and could be opened by cable pull during aircraft breakup. The cabin pressure control valve on the DC-6/7 aircraft could be tripped closed by cable when the crew was fighting an underfloor compartment fire, or during aircraft breakup. Correlate all the facts and circumstances before making decisions.

6.1.5. Ice and Rain Protection Systems

All of the components in ice and rain protection systems are not of a pneumatic nature, but thermal anti-icing, electrical windshield heat, alcohol, and windshield wipers are related in function. The pneumatic airfoil deicer boots operated by the suction air pumps have been discussed, so consider the hot wing and tail. In some aircraft heated air is provided by gasoline-fired heaters, like the cabin heaters. Hot air leaves the heater at 185°C, is routed through ducts in the leading edges of the wings and empennage, and is exhausted back over the control surfaces. On turbine-powered aircraft heated air is provided by the engine bleed air system.

The wing leading edges also contain electrical wiring from the fuel quantity measuring system, wingtip lights, and the compass system flux valves or flux gate transmitters. They also contain wiring for the various components and relays in the engine nacelles. Broken ducts or loose duct joints allow heated air to impinge on the wiring, causing erratic operation, or depending upon wire damage, complete failure of certain systems. This must be considered, and the wiring must be examined for such evidence.

Some aircraft windshields are kept free from ice by the flow of heated air between the inner and outer panes. Other windshields are heated electrically by the NESA system. A separate single-phase, 115VAC, 400 cycle inverter provides the power, and along with the other power generating components should be examined for evidence of maloperation. Log the position of the cockpit controls for windshield heat and compare them with the valve positions for this system.

Windshield wipers can be important factors in an accident. An example concerns a C-46 with a full load which took off in the rain. The hydraulic windshield wipers were operating. Shortly after landing gear retraction, the crew received a fire warning from the No. 2 engine; the engine was feathered, and fire extinguishing agent was released. The flight continued around the field for an emergency landing, but the landing gear would not extend to the locked position. The hydraulic pressure gauge read zero. Losing altitude, without time for the crew to troubleshoot the system, the aircraft was forced to land in a cornfield.

During the investigation, hydraulic pressure could not be obtained by using the hand pump; it was noted that the windshield wiper speed control valve was in the near full-open position. This valve was closed, and pressure could then be raised in the system to 650 p.s.i. by the hand pump. The windshield wiper valve was opened, and pressure fell to zero immediately, with no movement of the wiper blades. Fluid could be heard flowing through the windshield wiper hydraulic motor as the pressure fell off.

The motor was removed and tested. The results disclosed that at a normal system pressure of 100 p.s.i. the motor was bypassing hydraulic pressure in one direction by 350 p.s.i. This reduced system pressure to 750 p.s.i. which was insufficient to extend and lock the main landing gear. Previous tests of landing gear operation on other C-46 aircraft showed 900 p.s.i. to be the required average pressure to lock the landing gear down. Disassembly of the hydraulic wiper motor disclosed a hardened, misshapen cup seal on one end of the actuator shaft which was allowing pressure to bypass. If the flight crew had recognized that the wiper blades were stopped at the time of attempted landing gear extension, and had closed the speed control valve, sufficient pressure would have been available from the remaining hydraulic pump to lower and lock the landing gear for a normal landing.

6.1.6. Fairchild F-27 Pneumatic System

Cabin pressurization and air conditioning are provided by two engine-driven, positive-displacement blowers forcing air into the cabin through ducting. The flow of air is controlled by spill valves, an outflow valve, duct pressure relief valve, and a cabin pressure safety valve. Air is conditioned by a ram air heat exchanger and an air cycle machine or vapor cycle system (Freon).

The engine-driven Rootes blowers should be examined for evidence of failed drive gears, lubrication, and bearing wear or failure. The spill valves are motor operated and are located at wing station 80. On the ground they permit the full blower output to be dumped overboard automatically, in flight, by switches on the first officer's side panel. These are butterfly type valves, they should be closed in flight, and their positions should be compared to the position of the switches in the cockpit.

The outflow valve is also motor operated and should remain in the last position when electrical power is removed. The cabin pressure safety valve will open automatically when the structural limits of the aircraft are reached.

The valve may be opened to relieve cabin pressure when the first officer operates the cabin pressure manual dump valve on his console. The actual position of these valves must be logged and compared with cockpit control settings.

Examine for evidence, as explained earlier, the recirculating blower, combustion heater, and combustion air blower.

The primary pneumatic system consists of a gearbox - driven (engine - driven) compressor, unloading valve, bleed valve, moisture separator, chemical drier, back pressure valve (right nacelle only) and a filter. In addition, each nacelle contains a shuttle valve, disc-type relief valve, and a ground charging connection to aid in ground maintenance or initial filling. Each power section independently supplies compressed air to the primary and emergency systems. The air in the primary system is stored in two storage bottles and delivers air for normal operation of components as required by directional control valves. A 100-cubic-inch bottle is used for the main wheel brakes and a 750-cubic-inch bottle is used for gear operation, nose wheel steering, propeller brakes, and passenger door retraction.

An isolation valve located in the line between the large primary bottle and the smaller brake bottle permits maintenance to be performed downstream of the valve without discharging the large bottle. It will also cut off the pneumatic supply to the primary system, so be careful to log the position of this valve in case it may have been the cause of an apparent loss of pneumatic pressure. Downstream of the isolation valve is a pressure-reducing valve which is used to reduce system pressure from 3300 to 1000 p.s.i. All components, except the pressure gauges and the power section components, are located in the fuselage pneumatic compartment to the left of the crew compartment entrance door, directly behind the captain's seat.

The gearbox-driven pneumatic compressors are of four-stage radial design, and should be examined for evidence of operation at the time of the accident. The safety disc fitting in the bottom of the engine nacelles should be examined for evidence of failure due to overpressure in the system. The primary air storage bottle and the emergency bottle are located together in the left side of the pneumatic compartment. The emergency bottle is identical to the brake bottle, and provides storage of air for emergency operation of landing gear extension and brakes. These three bottles should be recovered to insure that they did not burst. Most probably, the plumbing would be disrupted and the bottles emptied before pressure could cause them to burst, during aircraft breakup.

The pneumatic actuators for the landing gear should be treated in the same manner as the hydraulic actuators, that is, measure the piston rod extension for position at impact. Bent piston rods give good position information. The pneumatic compartment directly behind the captain should be closely inspected. The lines in this compartment carry pneumatic air at a pressure of 3300 p.s.i., and through reducers in the same panel, 1000 p.s.i. Other reducers further drop the pressure to 100 p.s.i. All lines and fittings in this panel must be examined for evidence of failure in flight, as a failure in these lines could result in catastrophe.

6.2. Testing and Disassembly

Circumstances surrounding the accident determine the work to be done in the area of disassembly and testing. Components such as turbocompressors, air cycle machines, expansion turbines, and recirculating fans can create smoke in the cockpit and cabin, as will electrical or other burning. These may be the result of lubricating oil leakage, bearing failures, and overheated or burned electric motors.

Check the motor-operated valves for continuity with the multitester. If the circumstances indicate problems with air conditioning, pressurization, or the primary pneumatic system, functionally test the blowers, compressors, or the various turbines. Look at each system with a questioning eye for the answer to *how this system could have contributed to the accident*. It will be necessary to test, disassemble, and examine until the question is answered without qualification.

7. Fire Detection and Protection — General

The fire and overheat detection systems vary between aircraft types and between operators of the same aircraft type. All systems depend upon electrical power for operation. (Most systems operate from 28 vdc circuits, but some use 115 vac.) The detectors may be one of three types – thermocouple, thermal switch, or continuous loop. The first two are usually series connected and all three are connected to relay panels where detector signals are turned into fire or overheat warnings in the cockpit. One warning bell is common to all detector systems in the aircraft.

Areas of detection coverage also vary. In the older aircraft, such as the Convair and Douglas models, the detection systems covered the engine and nacelle areas as well as underfloor compartments. Some aircraft had fire protection for all of these areas, and some provided only partial protection. The large turbine-powered transport aircraft, in general, provide detection for the engine and nacelle areas and the main gear wells, but protection for the engines and nacelles only. **Figures C V-50** and **C V-51** cover a typical jet transport fire detection system. The underfloor compartments are designed so that fire cannot be sustained, and protection is not necessary. Some of these compartments, however, may be provided with overheat detection.

FIRE PROTECTION

I. INTRODUCTION

Fire/Overheat protection on the 727-23 airplane consists of the following:

A. Engines

Each engine and engine firewall contains a sensor strip which, in the event of an overheat or fire condition, provides a warning in the cockpit by illuminating a warning light for the respective engine and sounding a warning bell. A freon extinguishant system is provided to extinguish a fire.

B. Wheel Wells

The main and nose wheel wells contain continuous strip detectors which, in the event of an overheat or fire condition, provide a warning in the cockpit by illuminating a warning light and sounding a warning bell. No fire extinguishing system is provided.

C. Overheat Detection

Continuous loop detectors are used for overheat detection in areas adjacent to air conditioning and anti-ice ducts in the No. 1 and No. 3 engine strut areas inboard of the engine firewall, on both sides of the aft entry stairs, in the ceiling above the aft cargo compartment, and in the keel beam area. An overheat condition is indicated by illumination of a warning light in the cockpit. No aural warning is provided. No fire extinguishing system is provided.

D. Auxiliary Power Unit

The APU contains a sensor strip which, in the event of an overheat or fire condition, automatically shuts down the APU and provides a cockpit warning by sounding a bell and illuminating a warning light. An external warning is provided by a flashing red light in the left wheel well and an intermittent horn in the nose wheel well. A freon extinguishant system is provided to extinguish an APU fire.

E. Cargo Compartment

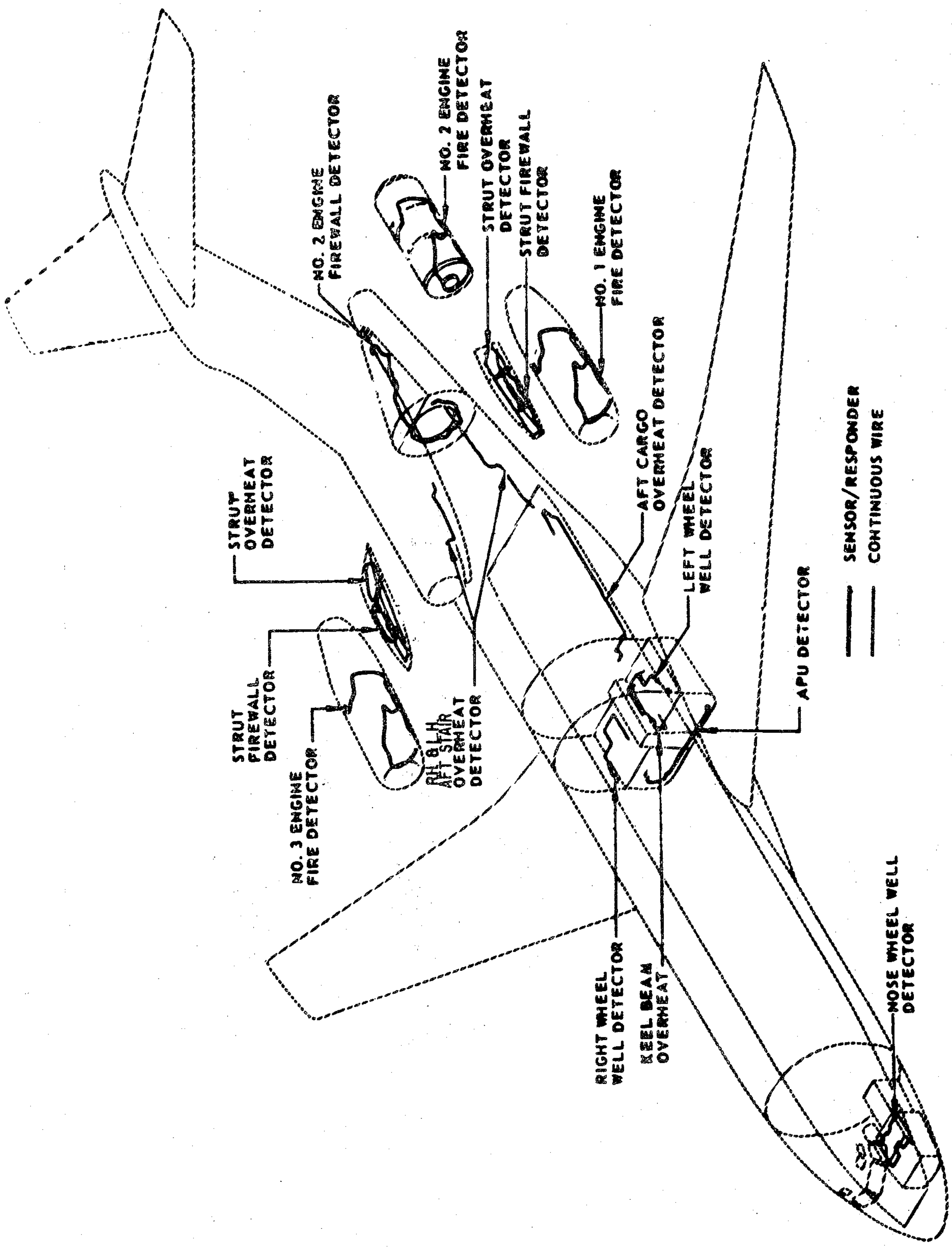
The cargo compartments are sufficiently sealed to meet FAA Class "D" (smother type)) compartment specifications. No fire detection or extinguishing system is provided.

F. Cockpit/Cabin

Portable CO_2 and water fire extinguishers are provided for use on localized cabin or cockpit fires.

Typical Jet Transport Fire Protection System Description.

Figure C V-50.



Typical Jet Transport Fire Detection Sensor and Responder Locations.

Figure C V-51.

Fire protection is provided by containers of CO_2 or agents from the freon family. Some of the latter, while being good fire suppressants, are highly toxic to personnel and should be carefully handled if the container has not been discharged. Obtain a schematic of the appropriate system to account for the main components. All fire extinguishing agent containers and their control heads should be accounted for and examined. The fire control panels in the cockpit must be documented and the indications compared with control head and container conditions. Figures C V-52 and C V-53 describe a typical turboprop transport fire protection system.

Each single-mounted container or each set of simultaneously-operated containers have thermal-discharge indicators mounted near the installation visible from outside the aircraft, usually consisting of small ports covered with a thin red plastic disc. These discs blow out, should an overheat condition discharge the containers. A single indicator shows intentional discharge; the port will be covered with a yellow disc, ruptured by a small plunger in the port body when the crew discharged the fire extinguishing agent.

Examine these discs. If thermal discharge has taken place, the discs will be blown out completely. If intact, and the containers are found empty, the cause was not thermal discharge while the plumbing was intact. If the yellow or system discharge disc is in place, the pilots did not discharge the containers intentionally. Impact forces cannot cause the system discharge indicator plunger to activate against its spring load. This plunger normally returns to the spring-loaded position when pressure dissipates after an intentional discharge. Damage to the port body will sometimes hold the plunger in the extended position if it occurs while the containers are being discharged (which provides some evidence). Examine the containers for this evidence.

7.1. Visual Inspection and Documentation — Fire Extinguisher Bottles

The CO_2 bottles may have a cable-operated cutter head which is punctured to release the agent. Another type, the flood valve, may be cable or solenoid operated to release the agent. One example is the installation on the DC-6 aircraft, two banks of three bottles mounted in the nose wheel well. In each bank one bottle has a solenoid head and two have cable-operated heads connected in series. The three are connected by plumbing in such manner that if the cable-operated bottles are discharged the solenoid-operated flood valve will also open for discharge. The reverse is true if the solenoid-operated bottle is discharged. Any bottle of the three can discharge because of thermal overpressure without affecting the other two.

When examining the control heads and valves on a bottle of this type, remember that they are cable operated and that the forces of aircraft breakup can pull the cables and discharge the CO_2. The solenoid-operated valves are not affected in this manner, but because of the plumbing arrangement this bottle may be found empty if the others are empty. If a bottle is found intact and there is doubt about the contents, it must be weighed. A stencil on the side of the bottle gives both the empty and full weights. The flood valves contain a safety seal which blows under excessive pressure caused by overheat.

The later high-rate-discharge systems use the spherical bottles described. The Water Kidde bottle, for example, has an electrical control head. To discharge the contents, DC electrical power is applied to two powder squibs. These are fired, a slug is blown against a copper seal, shattering it, and releasing the extinguishing agent. A screen located in the control head prevents pieces of the seal from being blown back into the discharge lines. This bottle also contains a fusible plug which melts at 260°F and allows the agent to discharge overboard. Other bottles have a safety disc which will rupture when an overheat condition causes the bottle pressure to rise to the range of 1400-1790 p.s.i. The bottle is normally charged to 600 p.s.i. with nitrogen propellant gas.

The latest bottle of this type (used on the DC-9) has three control heads with explosive cartridges and outlet ports. Two ports are plumbed to the engines, while the third is di-

FIRE PROTECTION

I. INTRODUCTION

A. General

The fire protection system on the Electra consists of an integral engine/nacelle fire detection system with two individual HRD (high rate discharge) fire extinguishing systems. Each nacelle is divided into three separate zones; zone I is the engine combustion and tailpipe area, zone II is the engine compressor and accessories area, and zone III is the lower nacelle and forward wheel well area. All three zones have fire detection and fire protection.

The engine/nacelle fire protection system is actually two completely independent systems, one for the engines/nacelles on the left wing, engine #1 and #2, and the other system on the right wing for engines #3 and #4. There is no method of transferring the extinguishant from the system in one wing to the system in the other wing.

The fuselage underfloor compartments do not incorporate a fire detection or fire extinguishant system. All underfloor compartments are constructed of heat-resistant material. Fire or overheat protection depends on oxygen starvation to smother any fire.

All engine fire warning lights and controls for discharging HRD are mounted on the forward center instrument panel glare shield, easily accessible to all three crewmembers.

Fire extinguishant from the aircraft system may be discharged to any engine/nacelle either in flight or on the ground. Also incorporated in the side of each nacelle is a spring-loaded door for injecting the nozzle of a portable ground CO_2 fire extinguisher.

Portable CO_2 and water type fire extinguishers are installed in the fuselage area and may be used to combat localized fires.

Description Typical Turboprop Transport Fire Protection System.

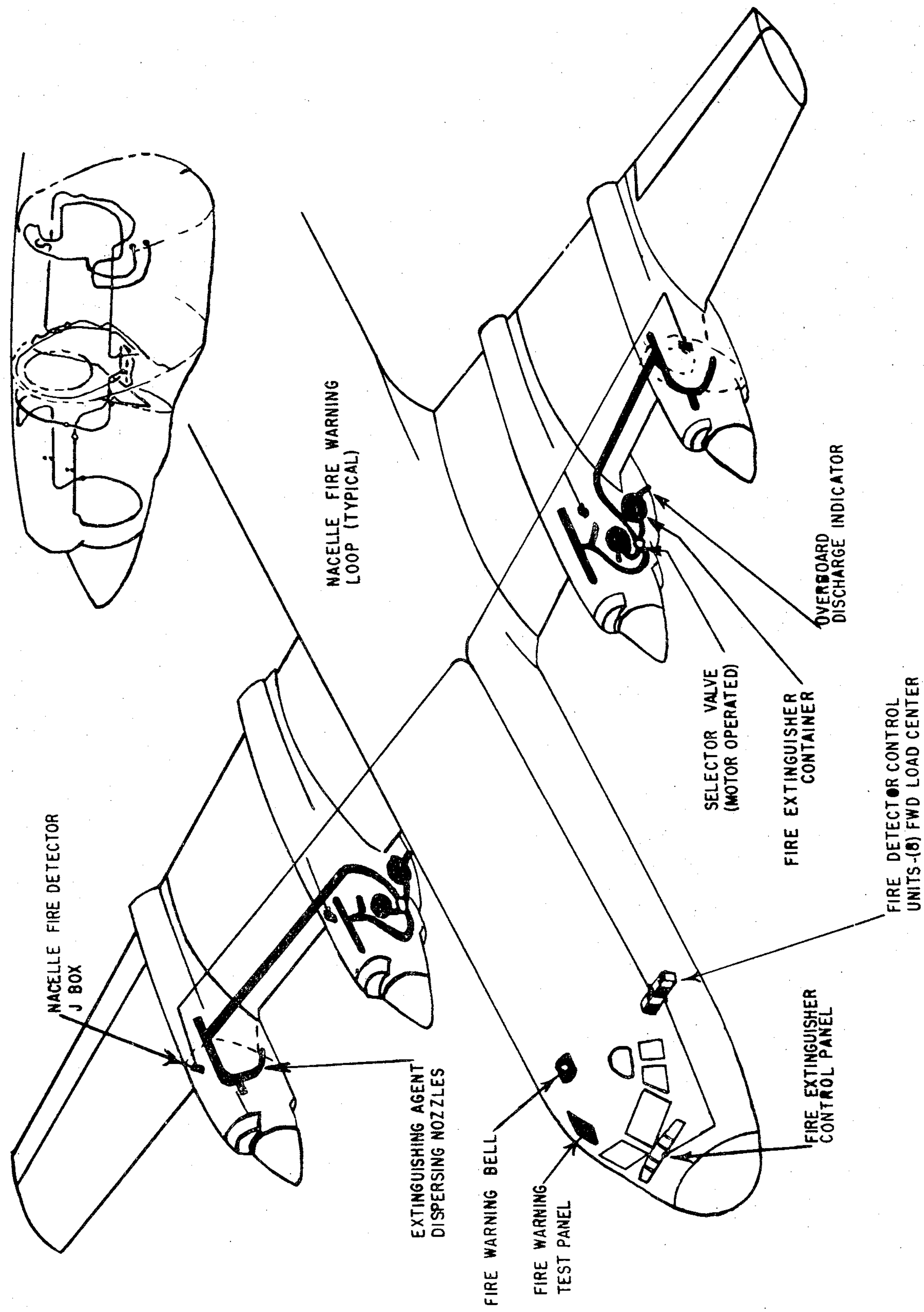
Figure C V-52.

rected to the auxiliary power unit. This arrangement eliminates the need for solenoid-operated direction valves and associated wiring.

If any bottle is recovered intact, check the pressure gauge; it should read zero or nearly 600 p.s.i. No other readings are possible, because once a bottle starts to discharge, it cannot be stopped. If pressure is indicated, the bottle was not used. In such case, arrange to have the bottle discharged as soon as possible as it represents a hazard.

Consider the F-27 which crashed on a mountain during an instrument approach in a snowstorm (discussed briefly in connection with DME equipment). This aircraft was equipped with two high-rate-discharge bottles in each engine nacelle. One bottle, from the right engine nacelle, was found free in the wreckage with the entire control head missing, which was recovered with indicating pin in the retracted position showing that it had not been discharged. Its mate was found free, but completely intact. The indicator pin in its control head was likewise retracted, so it had to be considered fully charged. The two bottles in the left engine nacelle remained intact in their mounting, and both indicator pins were retracted. This meant that none of the bottles had been discharged by the flight crew prior to the accident.

Since the wreckage had to be taken down the mountain, it was decided to discharge the three full bottles where they lay to preclude injury to personnel. The fully-charged auxiliary aircraft battery survived the accident and was recovered from the rear of the left engine nacelle, an indication that the electrical system had been working prior to the accident. Aircraft wire was attached to the two terminals in the control head of one bottle, and its plumbing was covered with blankets, pillows, etc., to absorb as much of the extinguishing agent as possible. The two wires were then connected to the battery, and the bottle discharged normally. After a few minutes the second bottle was connected to the battery and discharged.



Location of Fire Protection and Detection Units.

Figure C V-53.

The one charged free bottle was then stuck between the other two bottles for security, and was discharged in the same manner.

This procedure accomplished several things. A functional test was performed at the scene which showed the integrity of a battery and three of four fire extinguisher bottles. The results indicated that the flight crew had not received a distracting fire warning from the engines, causing them to descend below a safe altitude. The bottles were fully charged, available for use if needed. Many things can be accomplished at the scene if the systems are understood.

Once it is determined that the bottles are empty, disassemble the control heads to examine the copper seals and the thermal plugs or discs. If intact, then the bottle was not discharged intentionally or by heat. It is possible for the explosive charge to be fired by exposure to fire, so take this into consideration if the bottle has been exposed to heat or fire. Just as in the other systems, do not draw any conclusions until all the facts have been gathered. If the bottle contains a pressure blowout disc for thermal relief and it is found broken, then this was first in the sequence of events when both the disc and the copper seal are found destroyed. If the copper seal failed first, there would be no pressure to fail the thermal disc.

7.1.2. Fire Extinguishing System Plumbing

It is advisable to examine for leakage, breaks, or chafing the plumbing from the fire extinguisher to the engines and compartments. A prime example of such evidence is in the Pan American Boeing 707 accident at San Francisco. On takeoff, the No. 4 engine third stage turbine failed with the result that the engine and a 27′ section of the right wing were lost, and a roaring fire was started from a broken fuel line. The first officer discharged both fire extinguisher bottles on that side, and the fire went out about the same time. He did not know whether the extinguishers or the fuel shutoff valve smothered the fire. Investigation proved that the fuel shutoff valve was responsible.

A portion of the No. 4 engine fire extinguisher line from the right wing leading edge compartment was found chafed into a hole 1 1/8″ x 3/16″, about 40″ from the pylon attachment. The line was located so that it was chafing on the wing front spar lower cap; the chafed area was 6½″ in length. A hole of this size would dissipate much of the charge, reducing its effectiveness considerably. The same line to the No. 1 engine was examined, and it was found to be chafing on a wiring loom support bracket.

Pan American inspected their entire fleet of 707/720 aircraft and found 16 cases of chafing in 84 aircraft. One of these aircraft, only two months old, had chafing on both sides (No. 1 and No. 4). One line had a hole at a point of chafing against a thermal anti-icing duct. As a result of these findings, the Board recommended that operators of 707/720 equipment make an immediate inspection of these areas. The line is very difficult to inspect visually because of its location, so it is possible that chafing could be found on any airplane of this type. This is another reason why it is imperative to pay attention to detail.

7.1.3. Portable Fire Extinguishers

Two types of portable extinguishers are in use today, CO_2, and water. The old pyrene bottle is now a thing of the past, however, some may be found in the older aircraft. The emergency equipment storage chart shows the type, number, and location of portable extinguishers, all of which must be accounted for and examined. The CO_2 bottles have copper safety wire on the triggers. If the safety wire is intact, then the extinguisher was not used. If the wire is broken, pull the trigger to determine if any CO_2 remains.

The water extinguishers are pressurized by a CO_2 cartridge when the handle is twisted. This handle is also safetied with copper wire. Because of the design, the handle is not likely to be twisted as the result of aircraft breakup. This type of extinguisher can be checked for operation, or the CO_2 cartridge may be removed for examination.

7.1.4. Fire Detection Relay Boxes or Panels

These boxes or panels should be recovered and identified, and the relays should be examined for condition. Look for evidence of arcing on the contacts, or possibly for contacts which might be welded together. Welded contacts might suggest a fire warning, false or otherwise, prior to the accident. Check associated wiring for loose connections.

7.2. Testing and Disassembly

Testing and disassembly have been discussed to some extent. The need for testing will depend upon the circumstances, and the feasibility of testing will depend upon the condition of the available components. Investigation at the scene will, in all probability, suffice to determine whether or not the fire extinguishing system was used.

8. Oxygen Systems — General

The location and number of oxygen cylinders varies, depending upon aircraft type, but one large oxygen cylinder for the flight crew, from one to four for the passengers, and several small portable cylinders for first aid and other emergencies are usually carried. A schematic of the oxygen systems and the emergency storage indicates the number, type, and location of the cylinders.

8.1. Visual Inspection and Documentation

All cylinders must be accounted for and examined with great care. Though improbable, a cylinder could burst in flight, which must be ascertained. Record the gauge reading. If control valves are found attached to the cylinder or lying loose in the wreckage, check position, if open, check to see how far open.

If oxygen is present in any cylinder, it might be advisable to have the gas checked. In some cases a cylinder has been filled with ether by mistake. Such a finding might not apply to the accident cause, but it could be used to alert the user and the supplier to a dangerous situation.

If it has survived the accident, the oxygen system plumbing should be examined for evidence of burst lines, indicating that oxygen pressure was available at the time of bursting. Any burst area should be examined, since a fire resulting from the reaction of oxygen with oily or greasy substances might have been uncontrollable.

8.2 Testing and Disassembly

Little in the way of testing or disassembly can be accomplished with the oxygen systems. Circumstances surrounding the accident, such as reported smoke in the cockpit prior to the accident, may suggest a test of the oxygen regulators for proper operation and for a determination of smoke mask usage.

PART C — AIRCRAFT AIRWORTHINESS INVESTIGATION

CHAPTER VI

MANUFACTURING, MODIFICATION, AND MAINTENANCE

1. Methods and Procedures — General

All civil aircraft are required to meet certain FAA standards of airworthiness prior to certification. One of the early determinations to be made during each aircraft accident investigation is the FAA Airworthiness Standard under which the subject aircraft was approved and operating. After this decision, the prior condition of the various components can be evaluated, based on the original approval basis and type of operation being conducted. If a possibility of improper or incomplete manufacturing, modification, or maintenance procedures exists, the investigation should be continued at the appropriate facility.

1.1. Manufacturers

Manufacturers of complete airframes, powerplants, or propellers approved by the Federal Aviation Administration for use in civil aviation are issued an approved *Type Certificate*. Each manufacturer maintains a complete file of FAA-approved data which must be followed in the process of fabricating each type-certificated article produced. Accident investigations sometimes reveal that certain manufacturing operations are incorrectly performed or omitted, resulting in a causal or contributory factor. A partial list of items follows:

- Heat treatment – omitted or improper.
- Plating – omitted or improper.
- Dimensions – incorrect.
- Material – does not conform to data.
- Alignment – improper.
- Tolerances and fit – incorrect.
- Rigging – improper.

If manufacturing nonconformity is suspected, the Investigator-in-charge should visit the manufacturer's facility to make a positive determination.

Upon identification, a nonconformity is made the subject of a letter of recommendation from the Safety Board to the FAA in case of a fatal accident, or in a nonfatal accident, from the FAA field investigator to the concerned FAA Engineering Branch.

These recommendations, if well founded, usually result in corrective action by the manufacturer within his facility to prevent a recurrence of the problem. If a number of aircraft in service are affected, and are difficult to locate for rework, an *Airworthiness Directive* is issued by the FAA, distributed by mail to all owners of the particular make and model of aircraft, and to all civil maintenance facilities.

1.2. Modifiers

Individuals or companies making modifications to aircraft, engines, powerplants, or accessories used in civil aviation must obtain approval from the FAA for such modifications.

Each modifier receiving FAA authorization is issued a *Supplemental Type Certificate*, and holds a file of approved data to cover the specific modifications.

If the accident investigation detects improperly performed or incompatible modifications on the aircraft, these should be checked with the STC holder or the facility which accomplished the work. If methods or workmanship are at fault, corrective action should be accomplished by the individual or facility involved. If the STC data is incorrect or incomplete, a Safety Board or FAA recommendation to the STC holder or an *AD* to owners of modified aircraft will be required.

A partial list of modification problems follows:

- Weight and balance – incorrectly computed.
- Parts – do not conform to data.
- Hardware – not aircraft specification.
- Workmanship – not to FAA standards.
- Dimensions – incorrect.
- Alignment – improper.
- Rigging – improper.

1.3. Maintenance

Mechanics or maintenance facilities performing inspections, repairs, alterations, or modifications on U.S. civil aircraft are governed by specific FAA regulations.

If in the course of an accident investigation it becomes apparent that improper or incomplete maintenance procedures have contributed to the causal area, it will be appropriate to continue the investigation at the facility involved.

It should be remembered that if a considerable period of time has elapsed since the work was accomplished, or if the aircraft has been operated rather extensively, it may be more difficult to establish the facts.

A partial list of some maintenance errors or omissions follows:

- Annual inspection not accomplished.
- One hundred hour inspection (if required) not accomplished.
- Airworthiness Directives (AD's) not complied with.
- Unauthorized repairs accomplished.
- Unauthorized alterations accomplished.
- Aircraft not properly released to service.
- Non-aviation parts installed.
- Weight and balance not amended.
- Maintenance records not completed.

Where improper maintenance practices are found to have contributed to the cause of an accident, the information should be documented, and the appropriate FAA facility having jurisdiction over the area where the maintenance was performed should be advised. This permits the necessary corrective action.

2. Compliance with Federal Aviation Regulations — General

Persons involved in manufacturing, modification, or maintenance of U.S. civil aircraft are required to conduct these activities in accordance with specific Federal Aviation Regulations. All cases of noncompliance with FAR's brought to the attention of accident investigators should be made a matter of record in the accident report. When such information is known to FAA personnel, each case of noncompliance will be investigated, and appropriate remedial or corrective action will be initiated.

2.1. Manufacturers

Aircraft engine, propeller, and accessory manufacturers of products for civil aviation are governed by some or all of the following FAR's.

FAR 21 – Certification Procedures for Products and Parts

FAR 23 – Airworthiness Standards: Normal Utility and Acrobatic Category Airplanes

FAR 25 – Airworthiness Standards: Transport Category Airplanes

FAR 27 – Airworthiness Standards: Normal Category Rotorcraft

FAR 29 – Airworthiness Standards: Transport Category Rotorcraft

FAR 33 — Airworthiness Standards. Aircraft Engines

FAR 35 — Airworthiness Standards. Propellers

FAR 37 — Technical Standard Order Authorizations

FAR 39 — Airworthiness Directives

FAR 43 — Maintenance, Preventive Maintenance, Rebuilding and Alteration

2.2. Modifiers

Individuals or facilities involved in the modification of U.S. civil aircraft, engines, propellers, or accessories are governed by Federal Aviation Regulations listed under section 2.1., applicable to the modifications being accomplished.

2.3. Maintenance

Mechanics and maintenance facilities engaged in inspection, repair, alteration, and overhaul of U.S. civil aircraft are governed by the following Federal Aviation Regulations:

FAR 43 — Maintenance, Preventive Maintenance, Rebuilding and Alteration

FAR 91 — General Operating and Flight Rules

FAR 121 — Certification and Operations: Air Carriers and Commercial Operators of Small Aircraft

FAR 127 — Certification and Operations: Scheduled Air Carriers with Helicopters

FAR 133 — Rotorcraft External-Load Operations

FAR 135 — Air Taxi Operators and Commercial Operators of Small Aircraft

FAR 137 — Agricultural Aircraft Operations

FAR 141 — Pilot Schools

FAR 145 — Repair Stations

In addition, any regulation listed in section 2.1., applicable to the type of aircraft or product involved, would be pertinent.

3. Maintenance Records

Records examination following an accident reveals valuable information about the history of the aircraft and its components. In many instances these records have shown discrepancies and omission in maintenance and procedures which resulted in causal factors. This portion of the investigation requires thorough and tedious search of all pertinent records.

3.1. Air Carrier Records

Voluminous air carrier records create quite a problem for the investigator. The airlines usually cooperate in furnishing the records requested by the investigator, but they do not normally volunteer additional information. It is up to the investigator to determine what will be necessary to complete the investigation.

3.1.1. Records Required

Records required to be maintained by the airline are listed in the operating manuals. The investigator can obtain copies of these manuals for use during the investigation. *Mechanical Reliability Reports* and *Mechanical Interruption Summaries* are required for all air carriers, and these can be a source of useful information.

3.1.2. In-house Records

Major air carriers have additional records which are not required by the manuals, and if the investigator is familiar with these, and their use, he may obtain valuable information that may otherwise be lost.

3.1.3. Coordination with the Airline and ACDO Holding the Certificate

The NTSB field office personnel in the area where air carriers are located should be familiar with the operation of the airlines in order to proceed intelligently with an investigation.

Another important source of information and help is the FAA assigned air carrier inspector from the ACDO holding the airline certificate. His continual surveillance of the airline gives him a good understanding of how the airline operates.

3.1.4. Coordination with the Investigator-in-Charge

Very close coordination by the Records Group should be maintained with the Investigator-in-charge at all times. He should be alerted as soon as possible if the Records Group discovers anything that may affect the course of the overall investigation. This includes discrepancies, major repair, replacement of major components, or any other causal factor.

The modern air carrier records are becoming so complex that the trend in records investigation is for the IIC to alert the Records Group to probable causal areas to be investigated. In some instances it is practically impossible to go through and check all the records available to the investigators, therefore, the IIC directs the records investigation to the areas most likely to furnish clues to the probable cause.

3.2. General Aviation Records

General aviation records investigation is quite different from air carrier records investigation. The regulations require that only minimum records be maintained by the owner/operator, therefore, the investigator must use his ingenuity to uncover the history of the aircraft.

3.2.1. Documents Required

Very few documents are required to be maintained for general aviation. These are:

a. Permanent maintenance record of the aircraft and each engine.

b. Flight manual showing operating limitation. (Aircraft under 6,000 lbs. may use placards and markings.)

c. Airworthiness certificate.

d. Registration certificate.

e. Weight and balance information.

All items except (a) are required to be in the aircraft during flight.

3.2.2. Methods of Investigation

Quite frequently it is difficult to obtain records in a general aviation accident. If the aircraft burns, all of the records are usually lost with the aircraft. The only alternative left to the investigator is to obtain all available records, and question individuals who might have information concerning the aircraft. Information usually can be obtained from the owner. If he is not available, persons who knew him or his aircraft (mechanics, repair stations, or shops) may supply information. These people may not have a direct knowledge of the history of the aircraft, but they frequently may furnish clues to aid in the investigation.

4. Quality Control

Manufacturers, airlines, repair stations, modification facilities, and some maintenance shops will have a system of quality control established to assure airworthiness of completed work. A complete quality control inspection system usually includes the following:

a. *Receiving Inspection* – To assure that no materials or parts are accepted until they have been inspected and found to conform to specifications.

b. *Storage and Issuance Control* – Stored materials and parts are protected from deterioration and properly identified as to specification and inspection status.

c. *Process Inspection* – Each process should be inspected for conformity to FAA-approved process specifications or authized technical data.

d. *Process Data Control* – A system of review of all process technical data to assure that new processes and significant changes to previously approved specifications are evaluated and approved or disapproved in accordance with type design data.

e. *Production Control* — A system of flow sheets or cards showing materials to be used, machine operations, processes, inspections, and steps to be followed for all parts and assemblies to assure that physical dimensional, and functional characteristics will comply with type design data.

f. *Drawings and Change Control* — A system to assure that only current drawings or other engineering information are released to production and inspection personnel. Any changes to drawings are properly classified and approved by FAA prior to being released to inspection as a basis for acceptance.

g. *Inspection Tools and Testing Equipment* — A suitable system to provide for original and periodic accuracy checks for all measuring and testing devices used in the manufacture of the product.

h. *Inspection Status* — Adequate procedures for use of stamps, tags, or other means to indicate the inspection status of materials, parts, components, and assemblies.

i. *Inspection Stations* — Sufficient inspection stations with technical data and equipment should be established at appropriate locations for inspection of parts and assemblies at that state of production to assure conformity to data.

j. *Materials Review System* — A method of isolating all materials, parts, components, etc., which are damaged, do not fit properly, or do not conform to type design data. This system includes a Board to review all discrepant items to determine whether they may be used as-is, reworked, repaired, or scrapped.

k. *Assembly Inspection* — Observation of final assembly operations of the product and its major components provides an indication of accuracy, skill, and care used in fabrication of individual parts. All parts should assemble without undue forcing; if they do not, possible inadequate tooling, improper processing, or deficient quality control may be indicated

If an accident investigation reveals evidence of a breakdown in the quality control system of a facility, the basic areas listed above should be checked to determine if the objectives are being met.

PART C — AIRCRAFT AIRWORTHINESS INVESTIGATION

CHAPTER VII

HELICOPTERS

1. General

A helicopter accident will sometimes confront an investigator with what appears to be an impossible task due to extreme disintegration of the aircraft. Considering the amount of potential energy stored in a rotating rotor system, it is easy to see that loss of a blade or tip could cause a vibration of sufficient magnitude to cause an inflight breakup.

(Photo — wreckage after loss of rotor system)

Loss of control by linkage disconnect or even loss of power can result in heavy impact damage.

(Photo — tail rotor drive shaft failure)

Good basic investigative techniques apply to helicopter investigations. A good understanding of aerodynamics, as applied to helicopters and stability and control problems, is required by a helicopter investigative group. It is usually necessary to establish flight control and power/drive train continuity.

The investigation should proceed as outlined in other chapters of this text. Four major requirements for a successful investigation are:

1. A knowledge of helicopter aerodynamics and stability and control problems.
2. A good plan.
3. Patience.
4. Inquisitiveness.

2. Helicopter Aerodynamics

Helicopters, like fixed wing aircraft, are acted upon by four basic forces — lift, drag, thrust and weight. The main difference between these two types of aircraft is the source of lift. One aircraft has fixed wings and the other has rotary wings, but the same principles of lift apply to both.

The length, width, and shape of an airfoil all affect its lifting capacity. Each airfoil has two primary factors affecting the amount of lift it can generate. The relation between these two factors, velocity of airflow and angle of attack, and their effect on lift, can generally be expressed as follows:

For a given angle of attack, the greater the speed, the greater the lift.

For a given speed, the greater the angle of attack (up to the stalling angle), the greater the lift.

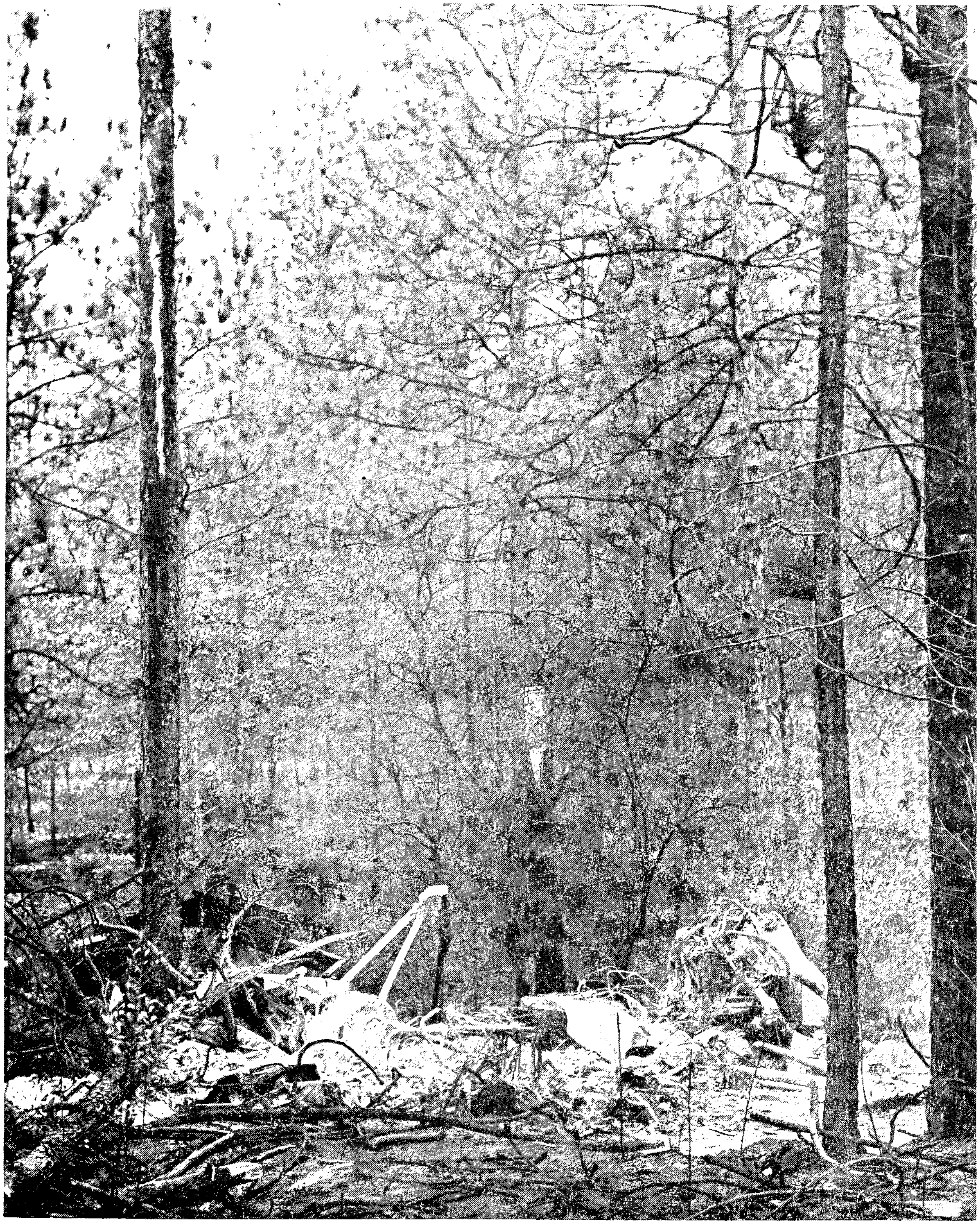
Lift can be varied by varying either of these factors. Furthermore, increasing either speed or angle of attack, or both (up to certain limits), increases lift.

2.1. Velocity of Airflow

Velocity of airflow is required in order to develop sufficient lift to get a helicopter into the air and keep it there. This means that the rotary airfoils (rotorblades) must be moved through the air at a relatively high speed. Since the speed of the rotorblades and the resultant velocity of airflow over them is independent of fuselage speed, a helicopter does not require high forward speeds of the aircraft for takeoff, flight, or landing. It is not limited to forward flight. It can rise vertically, hover, and can be flown forward, backward, or laterally.

2.2. Angle of Attack

Angle of attack is defined elsewhere in this text. The pilot can increase or decrease the angle of attack of a helicopter without chang-



ing the attitude of the fuselage. This is done by changing the pitch of the rotor blades collectively by means of a cockpit control called "collective pitch." Under certain conditions of flight the angle of attack continually changes as the rotor blades turn through 360°. This condition causes the rotor system to tilt and is controlled by cyclic pitch control inputs.

2.3. Angle of Incidence

Angle of incidence of a conventional aircraft is determined by the designer, and is built into the aircraft. This angle may not be changed by the pilot.

The helicopter's angle of incidence can be varied by the pilot. This change is accomplished by the pilot whenever he moves the cyclic pitch control from neutral, or the rotor plane of rotation is changed by the collective pitch control.

2.4. Airfoils

Airfoil sections for a conventional aircraft are selected to meet specific requirements. Most airfoils for the aircraft are unsymmetrical. An unsymmetric airfoil section for use in a rotor blade is unsatisfactory because the center of pressure "walks" forward or rearward as the angle of attack changes. This shifting of the center of pressure cannot be tolerated in a helicopter rotor because of the introduction of pitch-changing forces that would be dangerous. Therefore, the center of pressure travel is controlled by design, and is usually at a point 25% back from the leading edge of the rotor blade. Use of a symmetrical airfoil effectively limits the travel of the center of pressure.

2.5. Thrust and Drag

A helicopter gets both thrust and lift from the main rotor. Thrust acts in the direction of flight, and drag acts opposite the direction of flight. This holds true for vertical flight, forward flight, or rearward flight.

3. Flight Characteristics

The aerodynamics involved in producing lift by use of a rotating airfoil introduces certain stability and control problems. A control input to a rotating airfoil must be inserted at a point where gyroscopic reaction will cause the desired effect. A basic discussion of these characteristics will be covered in this section.

3.1. Torque

As the rotor of a helicopter rotates in one direction, the tendency of the fuselage to rotate in the opposite direction is *torque*. This reaction or torque effect is proportional in magnitude to the power being delivered by the engine. It is important to note that there is no torque when the engine is inoperative. It can then be said that there is no torque effect during an autorotation. The normal method of controlling torque in a single rotor helicopter is by use of a tail rotor. Most single rotor system rotate counterclockwise from the pilot's view The direction of torque would be opposite, tending to rotate the helicopter clockwise, therefore, in order to control this tendency the tail rotor must exert thrust in a direction opposite torque or tend to rotate the helicopter in a counterclockwise direction.

Pilot control of the tail rotor is accomplished by use of rudder pedals. These pedals allow the pilot to increase or decrease tail rotor thrust as required to control torque effect.

Since torque is proportional to engine power output, it must be remembered that a change in power will cause a corresponding change in torque.

Torque cannot be completely compensated for by use of tail rotor thrust only. Complete compensation requires a couple, or the matching of two equal forces acting in opposite directions. This second force is introduced by rigging the helicopter with the tip path plane of the rotor tilted from 1 - 2½ degrees to the left, depending on the helicopter type.

This slight tilt of the rotor tip path plane t the left results in a thrust vector to the left. This force, combined with the thrust control of tail rotor, forms the couple required to completely compensate for torque.

3.2. Tail Rotor Failure

Tail rotor failure of a helicopter in flight is a difficult emergency for the pilot. The first indication of such failure is loss of directional control. Without a tail rotor, the helicopter under discussion will rotate in a clockwise direction. The rotation rate will be proportional to the amount of power driving the rotor. The only means by which the pilot can stop this rotation is by a reduction of power. By removing the engine as the driving power for the rotor, the pilot may remove all engine torque effect. This is accomplished by placing the helicopter into an autorotation. While in autorotation, the helicopter will turn to the left, due to bearing and gear friction in the transmission system. This friction drag, however, is less than the torque effect and can be corrected by maintaining adequate gliding speed and making appropriate correction with the cyclic pitch control.

3.3. Hovering

Hovering is the maintaining of a position above a fixed spot on the ground, usually at an altitude of 6 to 10 feet.

In order to hover, the rotor system must supply lift equal to the helicopter's weight. Lift is controlled by controlling the pitch and r.p.m. of the rotor blades. The pitch and r.p.m. of the rotor blades is controlled by the collective pitch control.

Heading is controlled in a hover by controlling tail rotor thrust with the rudders.

3.4. Ground Effect

Ground Effect is experienced near the ground as a result of the ground or surface deflecting or turning the downwash or induced flow from the rotor of an aircraft hovering, thus reducing induced drag and increasing lift. As a result, less power is required to hover in ground effect than out of ground effect.

3.5. Translational Lift

As a helicopter leaves a hover and moves into forward flight it has a settling tendency. As forward speed is increased, the noticeable increase of lift which becomes available at about 15 miles per hour is called *translational lift*.

3.6. Autorotation

Autorotation is the process of producing lift with airfoils which rotate freely as air passes through the rotor from the bottom. Under power-off conditions, the helicopter will descend, and airflow will be established up through the rotor system. With the engine disengaged from the rotor system, pitch must be reduced by lowering the collective control, and during the descent energy is stored in the rotor system.

An increase of rotor r.p.m. is accomplished by pitch control, and rate of descent. As mentioned before, the pilot must reduce pitch in order to keep the rotor blades turning at sufficient speed to maintain the required centrifugal force. Otherwise, the blades will fold up and the helicopter will tumble out of control. A landing from an autorotation is accomplished by commencing a flare with aft cyclic at an altitude of 50 - 75 feet, with no change in collective pitch. This flare will reduce airspeed and rate of descent, will cause a slight increase in rotor r.p.m., level the helicopter as it settles, and gradually increase collective pitch to control final descent rate and touchdown.

3.7. Gyroscopic Precession

Gyroscopic precession is a quality of all rotating bodies in which an applied force is manifested approximately 90° in the direction of rotation from the point where the force is applied. Thus, if a downward force is applied to the right side of a rotor disc area, gyroscopic precession will cause the disc area to tilt down in front, provided the rotor system is rotating counterclockwise. Pitch changes to the rotor system are applied in this manner and are initiated by use of the cyclic pitch control.

3.8. Weight Limitations

The *weight limitation* on helicopters varies within broad limits, depending upon controlling criteria. The maximum weight of a helicopter becomes a variable quantity to produce optimums under different conditions. The weight limitation charts for each specific type of helicopter should be understood and complied with by a pilot. The heavier a helicopter is loaded, the less shock loading it can sustain before a failure occurs. Rough air will reduce the maximum planning weight for a particular mission.

Many accidents have occurred because pilots did not understand weight limitation and density altitude. Example: A pilot takes off at sea level with the helicopter loaded to maximum. His first intended landing point is ten miles away at an elevation of 3000 feet. With proper knowledge he will know that his helicopter may be too heavy to make a safe landing because of the density altitude. If the approach is continued into the landing site, the pilot may soon discover that he has insufficient power to accomplish a hover landing.

3.9. Power Settling

Power settling is an uncontrolled loss of altitude, aggravated by heavy gross weight, unfavorable density conditions, and low forward speed. This condition can be compared to a stall of a conventional aircraft. In either type of aircraft stalls or power settling near the ground are dangerous. A severe condition of power settling can result in the downwash of the rotor being recirculated – up, around, and back down through the outer rim of the rotor disc. This adverse circulation can become so severe that full up collective control will not produce sufficient lift to control the rate of descent. Recovery from this condition requires an increase in forward speed and a reduction of collective pitch, both of which are hazardous near the ground. An altitude loss of 400 to 700 feet may occur before the condition is recognized and recovery is complete. During an approach to a landing or a takeoff over congested area this condition can best be avoided by a thorough understanding of its cause.

A true condition of power settling is rare. The condition often encountered and described as "power settling" is a high rate of descent as a result of insufficient power to allow a flare or reduction of rate of descent for landing prior to contacting the ground. This condition is usually encountered at high density altitudes with high gross weights. If insufficient power is available to check this rate of descent, recovery would be the same as for true power settling.

3.10. Blade Stall

Blade stall usually occurs at the tip of the retreating blade due to the high angle of attack and slow speed of the retreating blade. The stalled blade sections exist throughout a small portion of the outer disc.

During flight conditions with high values of forward airspeed, gross weight and altitude, the retreating blade has an excessive angle of attack and low speed. These conditions aggravate blade stall. Mild blade stall will cause a roughness in both helicopter and flight controls. Severe blade stall will cause an abrupt pitchup of the nose of the helicopter. Although the retreating blade stalls on the pilot's left side, the resulting loss of lift is manifested 90° later, thereby causing the tip-path plane to tilt downward at the rear. This uncontrolled pitchup will last only a short period of time as full control is automatically restored when the airspeed decreases in the nose high attitude and the excessively high angle of attack no longer exists.

The blade stall is easily corrected by one or a combination of the following:

1. Reduce collective pitch.
2. Increase rotor r.p.m.
3. Decrease severity of maneuver.
4. Gradually decrease airspeed.