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LEVEL III

AIRCRAFT CRASH SURVIVAL DESIGN GUIDE

VOLUME V - AIRCRAFT POSTCRASH SURVIVAL

SIMULA INC.
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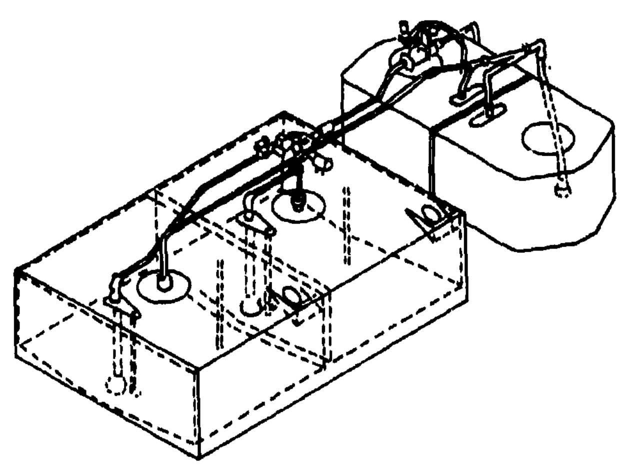
FINAL REPORT



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APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

This revised edition of the Crash Survival Design Guide was prepared for the Applied Technology Laboratory by Simula, Inc., under the terms of Contract DAAJ02-77-C-0021. The original Crash Survival Design Guide was published in 1967 as USAAVLABS Technical Report 67-22. Subsequent revisions were published as USAAVLABS Technical Report 70-22 and USAAMRDL Technical Report 71-22. This current edition consists of a consolidation of design criteria, concepts, and analytical techniques developed through research programs sponsored by this Laboratory over the past 20 years into one report suitable for use as a designer's guide by aircraft design engineers and other interested personnel.

This document has been coordinated with USAAVRADCOM, the U. S. Army Safety Center, the U. S. Army Aeromedical Research Laboratory, and several other Government agencies active in aircraft crashworthiness research and development.

The technical monitors for this program were Messrs. G. T. Singley III, R. E. Bywaters, W. J. Nolan, and H. W. Holland of the Safety and Survivability Technical Area, Aeronautical Systems Division, Applied Technology Laboratory.

Comments or suggestions pertaining to this Design Guide will be welcomed by this Laboratory.

NOTE: This is a revised edition of the Crash Survival Design Guide (formerly USAAMRDL Technical Report 71-22). All previous editions are obsolete and should be destroyed.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This five-volume document has been assembled to assist design engineers in understanding the problems associated with the development of crashworthy U. S. Army aircraft. Contained herein are not only a collection of available information and data pertinent to aircraft crashworthiness but suggested design conditions and criteria as well. The five volumes of the Aircraft Crash Survival Design Guide cover the following topics: -

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Volume I - Design Criteria and Checklists
Volume II - Aircraft Crash Environment and Human Tolerance
Volume III - Aircraft Structural Crashworthiness
Volume IV - Aircraft Seats, Restraints, and Litters
Volume V - Aircraft Postcrash Survival

This volume (Volume V) contains information on the aircraft postcrash environment and design techniques that can be used to reduce postcrash hazards. Topics include the postcrash fire environment, crashworthy fuel systems, ignition source control, fire behavior of interior materials, ditching survival, emergency escape, and crash locator beacons.

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PREFACE

This report was prepared for the Safety and Survivability Technical Area of the Applied Technology Laboratory, U. S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, by Simula Inc. under Contract DAAJ02-77-C-0021, initiated in September 1977. The Department of the Army Project Number is 1L162209AH76. This guide is a revision of USAAMRDL Technical Report 71-22, Crash Survival Design Guide, published October 1971.

A major portion of the data contained herein was taken from U. S. Army-sponsored research in aircraft crashworthiness conducted from 1960 to 1979. Acknowledgment is extended to the U. S. Air Force, Federal Aviation Administration, NASA, and U. S. Navy for their research in crash survival. Appreciation is extended to the following organizations for providing accident case histories leading to the establishment of the impact conditions in aircraft accidents:

- U. S. Army Safety Center, Fort Rucker, Alabama.
- Civil Aeronautics Board, Washington, D. C.
- U. S. Naval Safety Center, Norfolk, Virginia.
- U. S. Air Force Inspection and Safety Center, Norton Air Force Base, California.

Additional credit is due the many authors, individual companies, and organizations listed in the bibliographies for their contributions to the field. The contributions of the following authors to previous editions of the Crash Survival Design Guide are most noteworthy:

> D. F. Carroll, R. L. Cook, S. P. Desjardins, J. K. Drummond, J. H. Haley, Jr., A. D. Harper, H. G. C. Henneberger, N. B. Johnson, G. Kourouklis, W. H. Reed, S. H. Robertson, L. M. Shaw, J. W. Turnbow, and L. W. T. Weinberg.

Volume V has been coauthored by N. B. Johnson and S. H. Robertson. Dr. J. W. Turnbow provided valuable assistance with his recommendations and review of this document. Appreciation is also extended to the staff members of Simula Inc. and the Crash Research Institute for their contributions.

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INTRODUCTION

For many years, emphasis in aircraft accident investigation was placed on finding the cause of the accident. Very little effort was expended in the crash survival aspects of aviation safety. However, it became apparent through detailed studies of accident investigation reports that large improvements in crash survival could be made if consideration were given in the initial aircraft design to the following general survivability factors:

1. Crashworthiness of Aircraft Structure - The ability of the aircraft structure to maintain living space for occupants throughout a crash.

2. Tiedown Chain Strength - The strength of the linkage preventing occupant, cargo, or equipment from becoming missiles during a crash sequence.

3. Occupant Acceleration Environment - The intensity and duration of accelerations experienced by occupants (with tiedown assumed intact) during a crash.

4. Occupant Environment Hazards - Barriers, projections, and loose equipment in the immediate vicinity of the occupant that may cause contact injuries.

5. Postcrash Hazards - The threat to occupant survival posed by fire, drowning, exposure, etc., following the impact sequence.

Early in 1960, the U. S. Army Transportation Research Command* initiated a long-range program to study all aspects of aircraft safety and survivability. Through a series of contracts with the Aviation Safety Engineering and Research Division (AvSER) of the Flight Safety Foundation, the problems associated with occupant survival in aircraft crashes were studied to determine specific relationships between crash forces, structural failures, crash fires, and injuries. A series of reports covering this effort was prepared and distributed by the U. S. Army, beginning in 1959. In October 1965, a special project initiated by the U. S. Army consolidated the design criteria presented in these reports into one technical document suitable for use as a designer's guide by aircraft design

*Now the Applied Technology Laboratory, Research and Technology Laboratories of the U. S. Army Aviation Research and Development Command (AVRADCOM).

engineers and other interested personnel. The document was to be a summary of the current state of the art in crash survival design, using not only data generated under Army contracts, but also information collected from other agencies and organizations. The Crash Survival Design Guide, first published in 1967, realized this goal.

Since its initial publication, the Design Guide has been revised several times to incorporate the results of continuing research in crashworthiness technology. The last revision of TR 71-22 was the basis for the criteria contained in the Army's crashworthiness military standard, MIL-STD-1290(AV), "Light Fixed- and Rotary-Wing Aircraft Crashworthiness" (Reference 1). This current revision, the fourth, contains the most comprehensive treatment of all aspects of aircraft crash survival now documented. It can be used as a general text to establish a basic understanding of the crash environment and the techniques that can be employed to improve chances for survival. It also contains design criteria and checklists on many aspects of crash survival and thus can be used as a source of design requirements.

The current edition of the Aircraft Crash Survival Design Guide is published in five volumes. Volume titles and general subjects included in each volume are as follows:

Volume I - Design Criteria and Checklists

> Pertinent criteria extracted from Volumes II through V, presented in the same order in which they appear in those volumes.

Volume II - Aircraft Crash Environment and Human Tolerance

> Crash environment, human tolerance to impact, military anthropometric data, occupant environment, test dummies, accident information retrieval.

Volume III - Aircraft Structural Crashworthiness

> Crash load estimation, structural response, fuselage and landing gear requirements, rotor requirements, ancillary equipment, cargo restraints, structural modeling.

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1. Military Standard, MIL-STD-1290(AV), LIGHT FIXED- AND ROTARY-WING AIRCRAFT CRASHWORTHINESS, Department of Defense, Washington, D. C., 25 January 1972.

Volume IV - Aircraft Seats, Restraints, and Litters

Operational and crash environment, energy attenuation, seat design, litter requirements, restraint system design, occupant/restraint system/seat modeling.

Volume V - Aircraft Postcrash Survival

Postcrash fire, ditching, emergency escape, crash locator beacons.

This volume (Volume V) contains information on the aircraft postcrash environment and design techniques that can be used to reduce postcrash hazards. It contains a great deal of background information, including data from such sources as full-scale aircraft burn tests, laboratory materials testing, and research and development programs in aircraft fuel systems.

Chapter 1 presents a general discussion of designing for crashworthiness. Chapter 2 contains definitions of terms pertinent to the volume. Chapter 3 describes the postcrash fire environment and relates this environment to human tolerance data in the areas of heat, smoke, and toxic gases. Chapter 4 discusses methods of preventing postcrash fires by containing flammable fluids in crashworthy fuel, oil, and hydraulic systems, modifying fuel properties to reduce crash-induced fuel misting, and controlling potential ignition sources. Chapter 5 discusses the fire behavior of interior materials and presents data on material flammability tests and selected material properties. Chapter 6 describes the ditching environment and provisions that can be incorporated into the aircraft design to increase ditching survival. Chapter 7 presents design requirements for emergency escape exits and emergency lighting, and Chapter 8 discusses crash locator beacons.

1. BACKGROUND DISCUSSION

This volume specifically addresses the hazards that exist in the postcrash phase of U. S. Army aircraft accidents and presents aircraft design criteria that will, if followed, eliminate or reduce the serious consequences of these hazards. Designing for postcrash safety is only a part of the larger effort of designing the entire aircraft for crashworthiness.

The overall objective of designing for crashworthiness is to eliminate unnecessary injuries and fatalities in relatively mild impacts. Results from analyses and research during the past several years have shown that the relatively small cost in dollars and weight of including crashworthy features is an extremely wise investment. The outstanding success of the crashworthy fuel systems in almost entirely eliminating thermal fatalities and injuries in U. S. Army helicopter accidents provides a concrete example of the benefits that can be obtained through crashworthy design. Consequently, new generation aircraft are being procured to rather stringent crashworthy requirements.

The original edition of this design guide dealt primarily with modifications that could be made to existing aircraft to increase their crashworthiness. Now, two approaches to improving aircraft crashworthiness are open. The first approach is to influence the design of new aircraft, and the second is to improve the crashworthiness of existing aircraft. Obviously, much higher levels of crashworthiness can be achieved in the design and development of new aircraft if crashworthiness is considered from the beginning. This is being accomplished at the present time through the use of procurement packages that include pertinent specifications that require certain levels of crashworthiness for various subsystems as well as for the entire aircraft. However, some of the available potential is still being lost due to the historical approach used in designing aircraft. That is, the basic aircraft is designed leaving space and providing attachment provisions for subsystems. Later, when the subsystems are designed, their designs are limited by the previously established, somewhat arbitrary boundary conditions. The boundary conditions may unnecessarily limit the performance of the subsystems. The better approach is to design all systems and subsystems at the same time, at least preliminarily. This enables subsystem considerations to affect the larger systems. This systems approach will produce a more nearly optimum vehicle.

The same principles for improving crashworthiness can be applied to the retrofit of existing aircraft; however, the "cast-in-concrete" status of existing production structure is a more costly and difficult obstacle to overcome. When crashworthiness features must be included through retrofit, the level that can be achieved is usually reduced. Even in retrofit situations, however, the overall objective can be met; i.e., occupant protection can be maximized to eliminate unnecessary injuries.

In earlier editions of the Design Guide, the requirements to provide occupant protection in crashes up to and including the severity of the 95th-percentile survivable crash pulse was expressed. With the deployment of aircraft designed for crash safety, the link to the 95th-percentile survivable crash pulse has been dropped, and the recommended design environment is simply presented as the design pulse. Obviously, the severity of a 95th-percentile survivable crash pulse will be much greater for the new aircraft than for aircraft having no crashworthy requirements placed upon them during their development. The extent of the crash protection provided to the occupant cannot indefinitely continue to be linked to the survivability of the crash as improved crashworthiness increases the severity of the survivable crash, producing a never-ending increase in the level of crashworthiness at the expense of aircraft performance. The crashworthiness levels recommended herein are felt to be a near optimum mix of requirements including considerations of cost, weight, and performance. The crash environments selected for design purposes in this volume are identical to the historical 95th-percentile survivable crash pulses.

Also, in earlier editions of the Design Guide, information was provided on design of fixed-wing transport aircraft. However, considering the volume of new information on crashworthy design and the need to ensure that the size of this document remains within reasonable limits, only the primary aircraft in the Army inventory are considered. Therefore, information given herein is intended to apply to rotary-wing aircraft and light fixed-wing aircraft, defined by a mission gross weight of 12,500 lb or less.

2. DEFINITIONS

2.1 GENERAL TERMS

- The Term "G"

  The ratio of a particular acceleration (a negative acceleration may be referred to as a deceleration) to the acceleration due to gravitational attraction at seat level (32.2 ft/sec^2). With respect to the crash environment, unless otherwise specified, all acceleration values (G) are those at a point approximately at the center of the floor of the fuselage. In accordance with common practice, this report will refer to accelerations measured in "G." To illustrate, it is customarily understood that 5 G represents an acceleration of 5 x 32.2, or 161 ft/sec^2. As a result, crash forces can be thought of in terms of multiples of the weight of objects being accelerated. Therefore, in keeping with common practice, the term G is used in this document to define accelerations or forces.

- Static Strength

  The maximum static load that can be sustained by a structure, often expressed in terms of acceleration (G) of a given mass or, in other words, a load factor.

- Load Factor

  A factor that when multiplied by a weight produces a force used to establish static strength. Load factor is expressed in units of G.

- Forward Load

  Loading in a direction toward the nose of the aircraft parallel to the aircraft longitudinal (roll) axis.

- Aftward Load

  Loading in a direction toward the tail of the aircraft parallel to the aircraft longitudinal (roll) axis.

- Lateral Load

  Loading in a direction parallel to the lateral (pitch) axis of the aircraft.

- Downward Load

  Loading in a downward direction parallel to the vertical (yaw) axis of the aircraft.

- Upward Load

  Loading in an upward direction parallel to the vertical (yaw) axis of the aircraft.

- Velocity Change (ΔV)

  The decrease in velocity of the airframe during the major impact, expressed in feet per second. The major impact is the one in which the highest forces are incurred, not necessarily the initial impact.

2.2 FUEL, OIL, AND HYDRAULIC SYSTEM TERMS

- Boost Pump

  A fuel pump installed in the tank of an aircraft to supply the main (usually engine-driven) fuel pump with sufficiently high inlet pressure to meet net positive suction head (NPSH) requirements under all flight conditions.

- Frangible Attachment

  An attachment possessing a part that is constructed to fail at a predetermined location and/or load.

- Fuel Valve

  Any valve, other than a self-sealing breakaway valve, contained in the fuel supply system, such as fuel shutoff valves, check valves, etc.

- Self-sealing Breakaway Valve

  A fluid-carrying line or tank connection that will separate at a predetermined load and seal at both ends so that an absolute minimum of fluid is lost.

2.3 IGNITION SOURCE CONTROL TERMS

- Fire Curtain

  A baffle made of fire-resistant material that is used to prevent spilled flammable fluids and/or flames from reaching ignition sources or occupiable areas.

- Fire-Resistant Material

  Material able to resist flame penetration for 5 min when subjected to 2000°F flame and still be able to meet its intended function.

- Firewall

  A partition capable of withstanding 2000°F flame over an area of 5 sq in. for a period of 15 min without flame penetration.

- Flammable Fluid

  Any fluid that ignites readily in air, such as hydrocarbon fuels and lubricants.

- Flow Diverter

  A physical barrier that interrupts or diverts the flow of a liquid.

- Ignition Temperature

  The lowest temperature at which a flammable mixture will ignite when introduced into a specific set of circumstances.

- Inerting

  The rendering of an aircraft system or the atmosphere surrounding the system incapable of supporting combustion.

2.4 INTERIOR MATERIALS SELECTION TERMS

- Autoignition Temperature

  The lowest temperature at which a flammable substance will ignite without the application of an outside ignition source, such as flames or sparks.

- Flame Propagation Index (I_s)

  A number calculated by combining two factors derived from the ASTM E 162 radiant panel test for material flammability. One factor is derived from the rate of progress of the flame front and the other is derived from the rate of heat liberated by the material under test.

- **Flame Resistant**

  Material that is self-extinguishing after removal of a flame.

- **Flashover**

  The sudden spread of flame throughout an area due to ignition of combustible vapors that are heated to their flash point.

- **Flash Point**

  The lowest temperature at which vapors above a combustible substance ignite in air when exposed to flame.

- **Intumescent Paint**

  A paint that swells and chars when exposed to flames.

- **Optical Density (D_s)**

  The optical density is defined by the relationship

  $$D_s = \log \frac{100}{T}$$

  where T is the percent of light transmission through a medium (e.g., air, smoke, etc.).

2.5 DITCHING AND EMERGENCY ESCAPE TERMS

- **Brightness**

  The luminous flux emitted per unit of emissive area as projected on a plane normal to the line of sight. Measured in foot-lamberts.

- **Candela (cd)**

  A unit of luminous intensity equal to 1/60 of the luminous intensity of 1 square centimeter of a blackbody surface at the solidification temperature of platinum. Also called candle or new candle.

- Class A Exit

  A door, hatch, canopy, or other exit closure intended primarily for normal entry and exit.

- Class B Exit

  A door, hatch, or other exit closure intended primarily for service or logistic purposes (e.g., cargo hatches and rear loading ramps or clamshell doors).

- Class C Exit

  A window, door, hatch, or other exit closure intended primarily for emergency evacuation.

- Cockpit Enclosure

  That portion of the airframe that encloses the pilot, copilot, or other flight crew members. An aircraft may have multiple cockpits, or the cockpit may be physically integrated with the troop/passenger section.

- Ditching

  The landing of an aircraft on water with the intention of abandoning it.

- Emergency Lighting

  Illumination required for emergency evacuation and rescue when normal illumination is not available.

- Exit Closure

  A window, door, hatch, canopy, or other device used to close, fill, or occupy an exit opening.

- Exit Opening

  An opening provided in aircraft structure to facilitate either normal or emergency exit and entry.

- Exit Release Handle

  The primary handle, lever, or latch used to open or jettison the exit closure from the fuselage to permit emergency evacuation.

- <u>Foot-candle (fc)</u>

  A unit of illuminance on a surface that is everywhere 1 foot from a uniform point source of light of 1 candela.

- <u>Foot-lambert (fL)</u>

  A unit of photometric brightness or luminous intensity per unit emissive area of a surface in a given direction. One foot-lambert is equal to $1/\pi$ candela per square foot.

- <u>Illumination</u>

  The luminous flux per unit area on an intercepting surface at any given point. Measured in foot-candles.

3. POSTCRASH FIRE

3.1 INTRODUCTION

Historically, studies of accident records have indicated that a high percentage of fatalities occur in accidents involving postcrash fire. During the past 10 years, however, the pattern has changed dramatically. The accident records now indicate two distinct patterns. For aircraft not equipped with crashworthy fuel systems, the statistical records remain essentially unchanged. For aircraft containing crashworthy fuel systems, the fire death and injury rates have been reduced to nearly zero.

The postcrash fire environment associated with the aircraft not containing a crashworthy fuel system consists of a combination of many interacting hazards. The total fire threat to the occupant depends upon the magnitude of these hazards combined with the human tolerance limits to each hazard. This chapter describes the postcrash fire environment and discusses human tolerance to heat, toxic gases, and other hazards that greatly affect human survival in a postcrash fire.

3.2 POSTCRASH FIRE ENVIRONMENT

The postcrash fire environment has been extensively studied in test programs as well as in actual crashes by various research organizations including NACA (prior to becoming NASA), NASA, FAA, AvSER, the U. S. Air Force, and the U. S. Army. During some of the test programs, aircraft were crashed and allowed to burn, with data being accumulated during the entire sequence. In other test programs, previously crashed aircraft were instrumented and burned. In addition to full-scale tests, many studies have been performed with various components and mock-ups, and researchers have studied actual aircraft crashes in which occupants were exposed to the postcrash fire environment. From these overall studies, the most significant factors influencing survivability in postcrash fires have emerged.

Briefly, it has been observed that many variables can influence the magnitude and threat of a postcrash fire. Some of the more pertinent ones include the relative wind, the type of terrain onto which the flammable fluid has drained, the fuel distribution, the location of the fluid spillage within the aircraft, the number of structural openings (designed or crash produced) that meter the inflowing air available for an internal fire, and the amount of fuel available to spill (Reference 2).

It was noted that using fuels of lower volatility (i.e., Jet A rather than Jet B) makes little difference in the overall fire threat once a postcrash fire has started (Reference 3). However, if the fuel is spilled in liquid form and kept in that state, rather than being formed into a mist, the likelihood of the less volatile fuel catching on fire is measurably reduced. In other words, if the aircraft crashes and comes to a stop with no fire, the chances of a fire then starting are generally less with fuels of lower volatility.

However, the factors that best describe the postcrash fire situation in terms of human survival are the heat, toxic gases, and smoke existing in or near the occupiable area.

3.2.1 Heat

A typical ambient and radiant temperature curve for large cargo/passenger-carrying aircraft tested by NACA is presented in Figure 1. As can be seen on the chart, little temperature increase occurred until 80 sec after impact. One of the main reasons for the delay in temperature rise was the protective shield afforded by the fuselage. Skin burn-through averaged about 80 sec, although some burn-through times occurred before 40 sec and some occurred later. Calculated escape times based on human tolerance to heat varied from 53 to 220 sec, with the average escape time equal to 135 sec. (See Section 3.3.1 for a discussion of the effect of heat on escape times.)

An ambient temperature range typical for the burning passenger/cargo-carrying helicopters tested by AvSER and the U. S. Army Aeromedical Research Laboratory is presented in Figure 2. This chart shows that the temperature started to increase almost immediately after the crash. The early temperature rise was due mainly to two factors. One was that extensive structural breakup occurred upon impact, causing openings that allowed air to be drawn in, providing oxygen for internal fires.

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2. Johnson, N. B., et al., AN APPRAISAL OF THE POSTCRASH FIRE ENVIRONMENT, Dynamic Science (The AvSER Facility); USANLABS Technical Report 70-22-CE, U. S. Army Natick Laboratories, Natick, Massachusetts, September 1969, AD 699826.

3. CRC-Aviation Fuel Safety Report Task Force, AVIATION FUEL SAFETY - 1975, CRC Project No. CA-52-74, Coordinating Research Council, Inc., New York, November 1975.

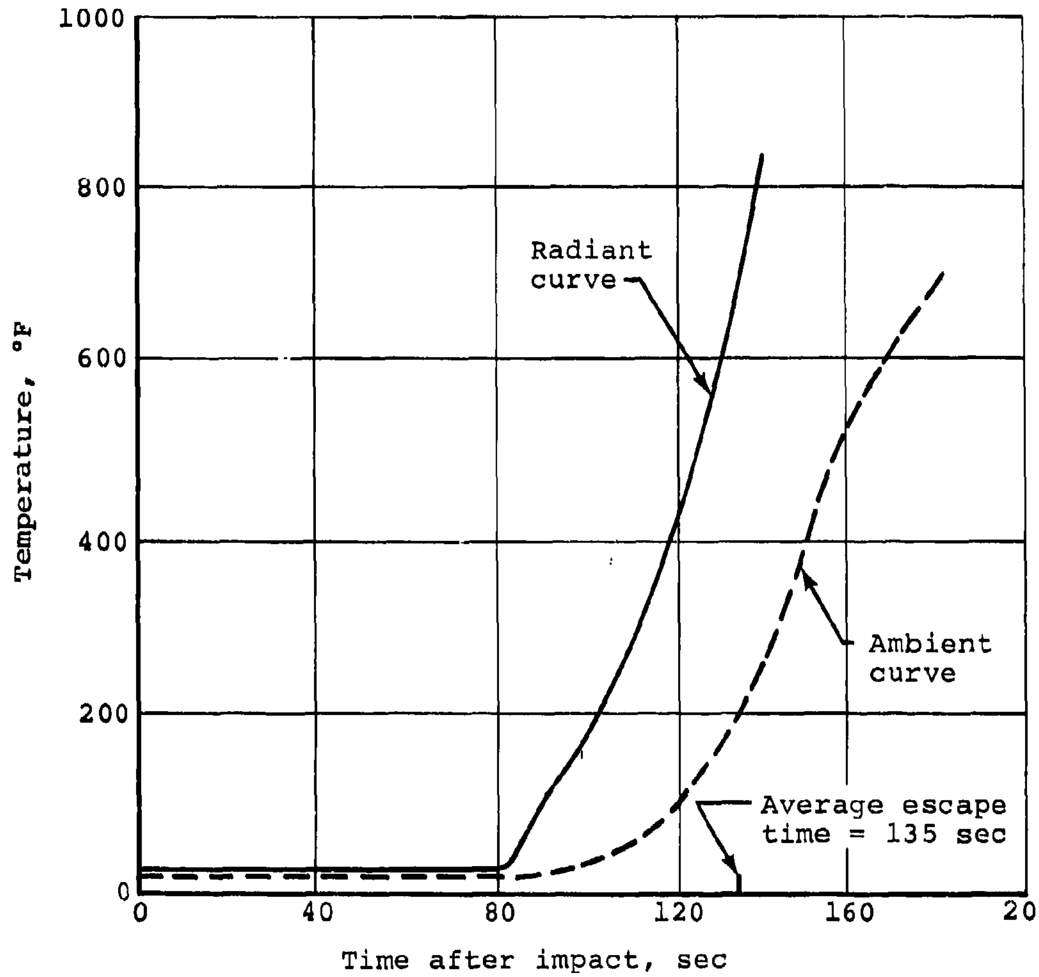

Figure 1. Average recorded ambient and radiant temperatures in large, crashed, burning, passenger/cargo-carrying, fixed-wing aircraft.

The second factor was that, in the normal configuration, the fuselage and the fuel were located in close proximity to one another. As a result, the fire and the occupiable area were nearly superimposed from the start. Reference to Figure 2 shows that the average escape time for these helicopters was in the range ot 7 to 16 sec.

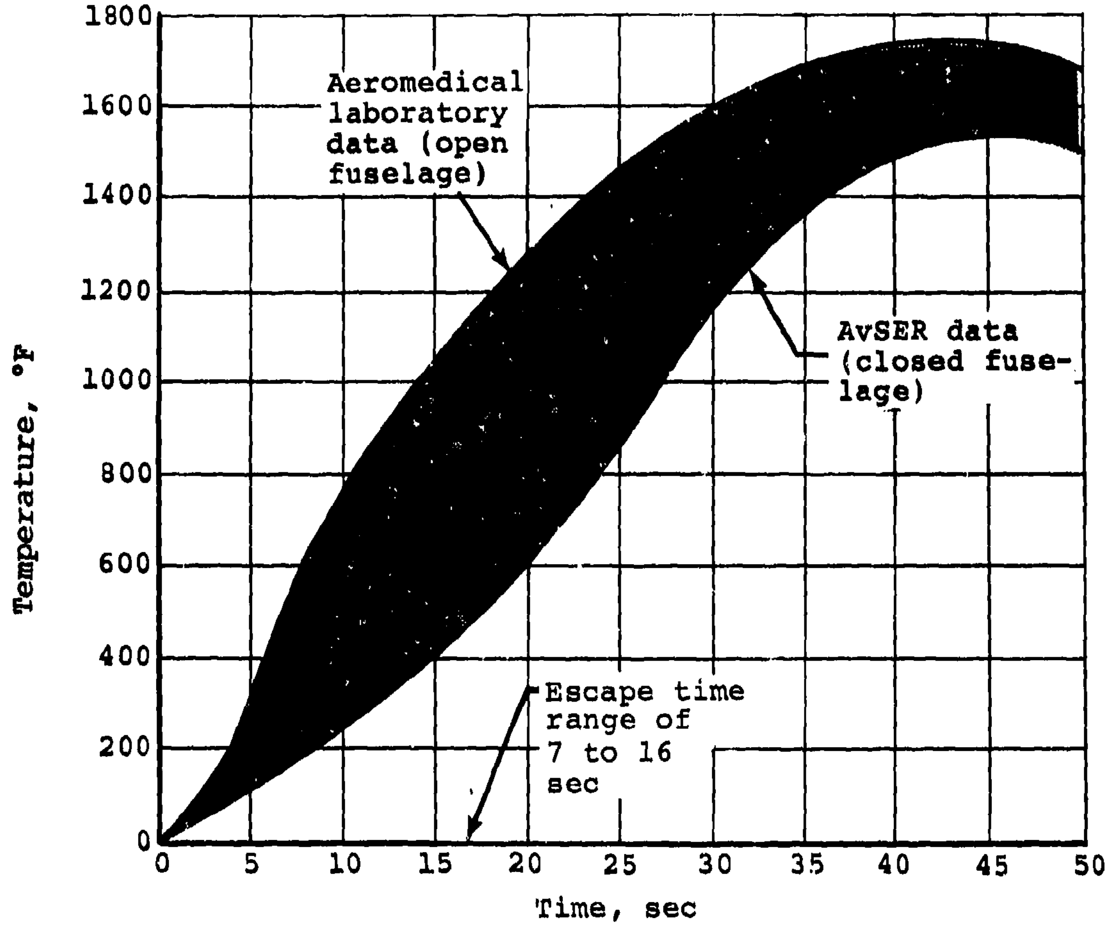

Figure 2. Recorded ambient temperature range in the cabin areas of large, burning, passenger/cargo-carrying helicopters.

Full-scale fire tests on standard aluminum aircraft skin panels show that, for a fuel fire of maximum severity and minimum skin thickness, burn-through may occur in as little as 10 sec. Larger aircraft, which possess thicker skin panels, have burned through in 30 to 40 sec. Figure 3 shows minimum skin melting times based on aircraft gross weight. Escape times are obviously shorter for faster burn-through times. Thus, the very short escape times in light aircraft are due not only to the proximity of the fuel to the occupant but also to the faster burn-through times of the thinner fuselage skins.

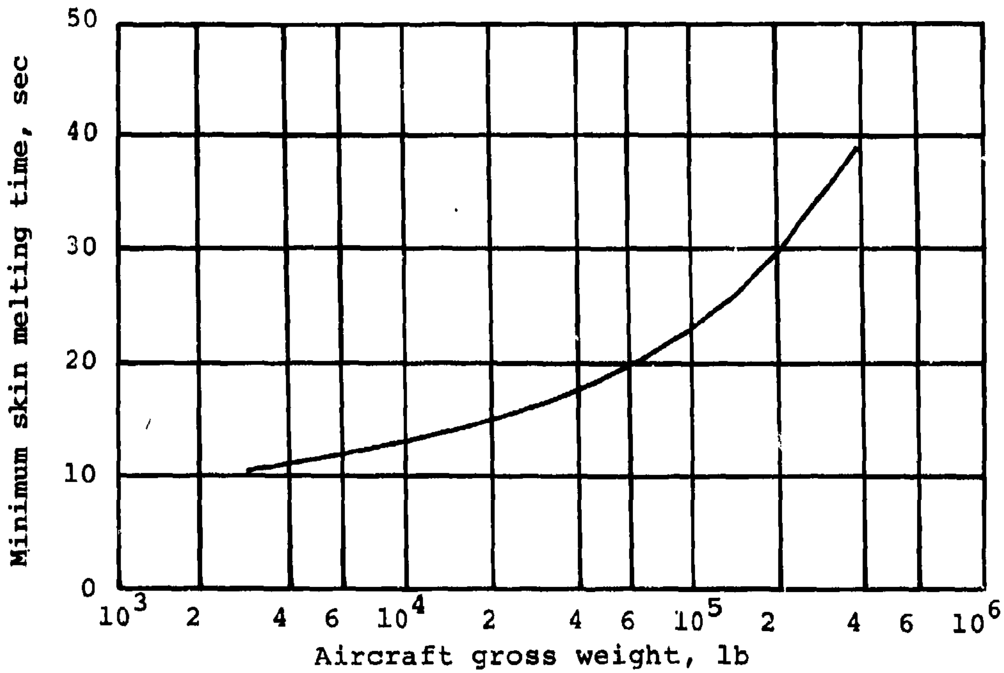

Figure 3. Aircraft skin melting time based on gross weight.

Another factor that can influence burn-through time is insulation. When an aircraft skin is heated externally by a fire, the metal skin attempts to radiate heat internally. When this radiation is prevented or retarded by insulation, skin burn-through occurs more rapidly.

3.2.2 Smoke and Toxic Gases

Aircraft crash fires generate large quantities of dense smoke consisting of unburned carbon particles, ashes, and gaseous combustion products. The hazards of smoke may be both physical (blocking vision) and physiological (irritation of eyes and respiratory tract, toxicity).

The rapid obscuration of vision by smoke has been reported by many survivors of aircraft postcrash fires. In addition, several test programs have documented the generation of large quantities of smoke during burn tests of transport cabin mock-ups used to evaluate aircraft interior materials. However, no quantitative data on smoke obscuration are available from full-scale crash tests.

Quantitative crash test data are available for carbon monoxide (CO), the predominant toxic gas generated during crash fires. The history of carbon monoxide levels typical of NACA's passenger-carrying aircraft experiments is presented in Figure 4. It can be observed that the CO concentrations remained below the 0.8 percent level for about 250 sec, at which time they rapidly increased to 4 percent. This slow-to-develop situation was due to the distribution of the fuel spillage and the protective shield afforded the occupants by the fuselage. Also plotted in Figure 4 is the cumulative carboxyhemoglobin (COHb) level that would be present in an individual exposed to this atmosphere. The escape-limiting 35 percent COHb would be reached in approximately 3 minutes. (See Section 3.3.2 for a discussion of the effect of carbon monoxide on escape times.)

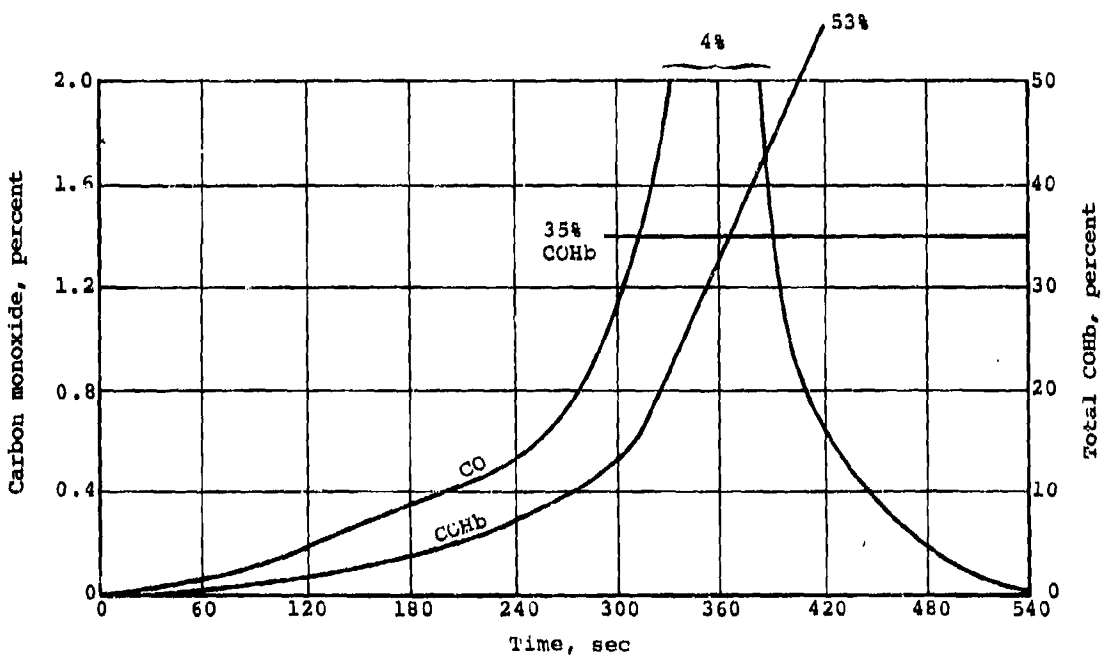

Figure 4. Average recorded CO concentrations and calculated COHb levels in large, crashed, burning, passenger/cargo-carrying fixed-wing aircraft.

The CO levels typical of burning, large, passenger/cargo-carrying helicopters are presented in Figure 5. It can be seen that measurable CO levels started at about 20 sec and that within 45 sec the CO levels had increased to 3 percent. As with the temperatures, the rapid increase of CO was due to the fuel distribution and to the structural breakup. The CO concentration dissipated after 45 sec due to two factors. First, the helicopter fuselages were nearly consumed by fire in 45 sec; thus, they could no longer act as shells to hold the gases in the area. Second, only small quantities (28 gal and 56 gal) of fuel were used during the tests. The rapid dissipation of the CO would preclude the buildup of dangerous COHb levels in an individual exposed to these conditions.

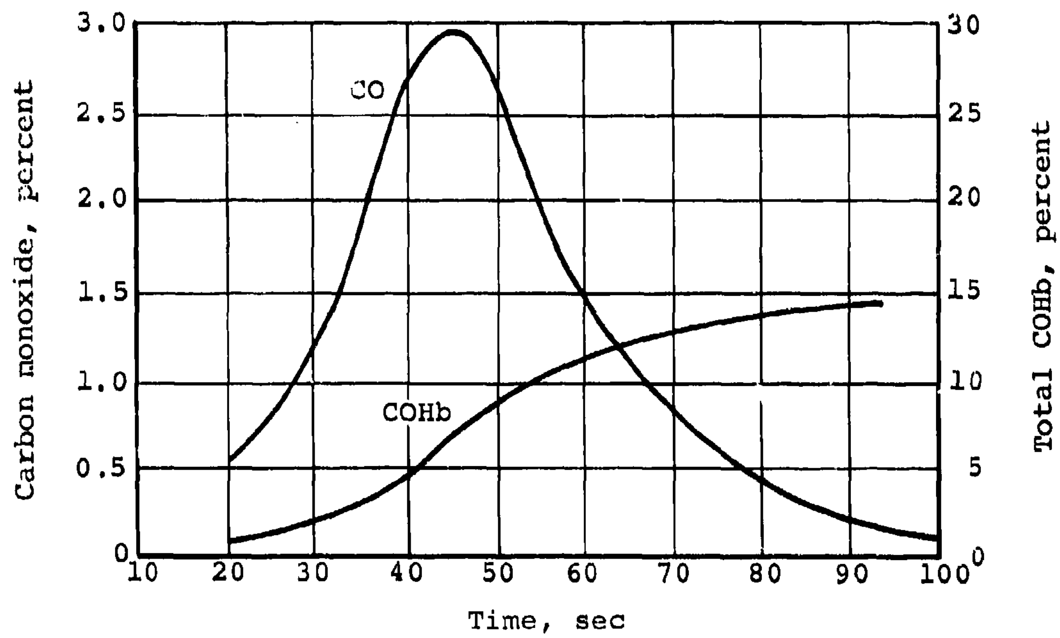

Figure 5. Average recorded CO concentrations and calculated COHb levels in large, crashed, burning, passenger/cargo-carrying helicopters.

Although carbon monoxide is produced in larger amounts than any other toxic gas, large-scale burn tests show that many other gases also are generated, including significant amounts of hydrogen chloride (HCl) and hydrogen cyanide (HCN) (References 4 and 5). Accurate time-concentrations of these in burning aircraft fuselages are not known at this time, but the large quantities present after 3 to 5 minutes raise the possibility that a gas other than carbon monoxide might, at times, be the escape-limiting factor. In fact, the recognition that HCN was a combustion product of many aircraft materials prompted Civil Aeromedical Institute (CAMI) scientists to determine the HCN levels in blood specimens from victims of aircraft accidents involving postcrash fire (Reference 6). It was definitely established that HCN was present at levels greater than normal in the blood of several victims.

3.3 HUMAN SURVIVAL AND ESCAPE

One's ability to perform a self-initiated escape from a burning aircraft becomes hampered when one is unable to think and act as a normal human being. The point at which the incapacitating effect occurs is called the escape limit. An occupant's escape limit is governed by what the person feels (temperature), breathes (toxic gases), and sees, or, in case of smoke, does not see (escape routes, blocked exits, etc.). Human tolerance limits define human body reaction to these factors.

3.3.1 Human Tolerance To Heat

The literature dealing with the subject of human tolerance to heat exposure is rather extensive, but somewhat confusing and misleading. (For the purpose of this discussion, human tolerance to heat is considered for short-term exposures, up to 15 min, rather than heat prostration-type injuries that require a considerably longer exposure time.) Although heat tolerance has been reliably investigated by many researchers, their reports are not always clear, especially in regard to protective measures taken during exposures to extreme heat. The reports by Johnson and Pesman are considered to be the best application

4. Marcy, J. F., A STUDY OF AIR TRANSPORT PASSENGER CABIN FIRES AND MATERIALS, Federal Aviation Agency; Technical Report FAA-ADS-44, National Aviation Agency, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, December 1965, AD 654542.

5. Heine, D., and Brenneman, J., THE FIRE TEST RESULTS, The Airline Pilot, Vol. 35, No. 10, October 1966, pp. 8-11, 18-19.

6. Mohler, S. R., AIR CRASH SURVIVAL: INJURIES AND EVACUATION TOXIC HAZARDS, Aviation, Space, and Environmental Medicine, January 1975, pp. 86-88.

of scientific knowledge to the subject of human thermal tolerance during the aircraft crash-fire environment (References 2 and 7). Therefore, much of the material in this section has been based upon those reports.

Thermal injuries occurring in aircraft crash fires can be divided into two general types: skin injury and respiratory injury.

3.3.1.1 Skin Injury: When exposed to heat, two main factors govern a person's survival ability. They are tolerance to pain and the thermal level at which the exposed skin will experience second-degree burning. References 8 and 9 state that the pain threshold is exceeded when the human skin is heated to a temperature between 108°F and 113°F, with normal human beings experiencing unbearable pain at skin temperatures of 124°F. Moreover, when the skin surface temperature is raised above 111°F, the rate of cellular destruction is more rapid than cellular repair; consequently, an accumulative injury occurs. Obviously, the extent of the injury is dependent on the heat transferred during the exposure time.

Since the temperature values required to produce pain and skin injury are similar, pain is a good indication that injury will occur if the application of heat continues. Therefore, approximate escape limits can be based on extreme pain and, thus, the occurrence of radiative second-degree burns.

To approximate the occupant escape limit as fixed by radiant temperature, one additional factor must be considered; i.e., the radiating surface visible to the exposed area. A hemisphere is considered to be the maximum possible radiating space angle (Figure 6). Figure 7 shows pain threshold times as determined by temperature of the radiative source for several angles of radiation. If, for example, the entire hemispheric

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7. Pesman, G. J., APPRAISAL OF HAZARDS TO HUMAN SURVIVAL IN AIRPLANE CRASH FIRES, NACA Technical Note 2996, Lewis Flight Propulsion Laboratory, National Advisory Committee for Aeronautics, Cleveland, Ohio, September 1953.

8. Buettner, K., Ph.D., EFFECTS OF EXTREME HEAT ON MAN, Journal of the American Medical Association, Vol. 144, October 1950, pp. 732-740.

9. Moritz, A. R., M.D., et al., AN EXPLORATION OF THE CASUALTY PRODUCING ATTRIBUTES OF CONFLAGRATIONS: LOCAL AND SYSTEMIC EFFECTS OF GENERAL CUTANEOUS EXPOSURE TO EXCESSIVE HEAT OF VARYING DURATION AND INTENSITY, Archives of Pathology, Vol. 43, 1947, pp. 466-502.

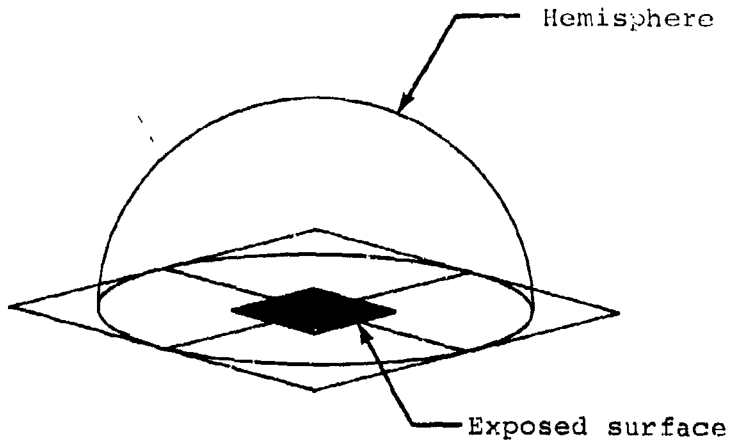

Figure 6. The hemisphere of radiant heat concept.

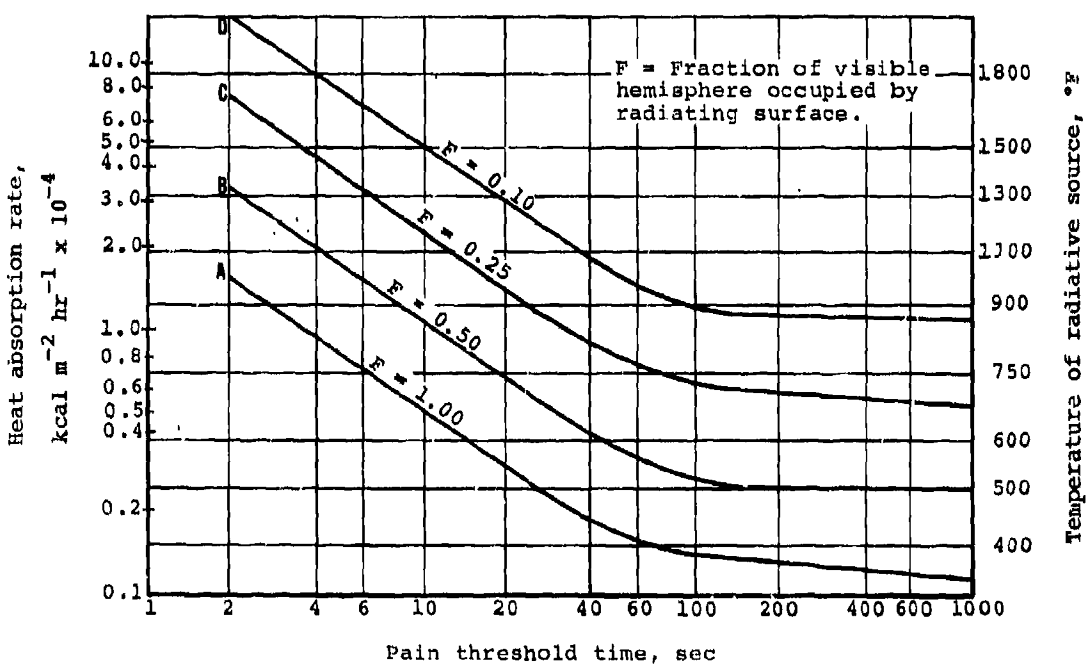

Figure 7. Pain threshold time as a function of temperature of radiant heat source. (Method of deriving curves is contained in Appendix A)

surface were at an elevated temperature, Curve A (F = 1.00) would apply. If only 50 percent of the hemispheric surface were at such temperature, Curve B (F = 0.50) would apply. The escape limit is independent of the distance between the individual and the radiant heat source. As an example of radiant curve usage, assume that an individual is sitting in a crashed aircraft that is engulfed in a fireball. For all practical purposes, the imaginary hemisphere would be 100 percent heated; thus, Curve A would apply. Figure 7 shows that a 20-sec escape time will be reached when the interior aircraft walls reach a radiant temperature of only 550°F.

Experimental data on human body tolerance to convective heat (from hot ambient air) are much more limited than data on tolerance to radiant heat. Convective heat is the primary source of caloric uptake at low temperatures, and severe physiological disturbances may occur at temperatures below those required for second-degree burning of the skin. Thus, extreme pain alone is not sufficient to determine tolerance times to heated ambient air, and the radiative burn curves in Figure 7 cannot be used with ambient air temperatures.

Figure 8 (from Reference 10) is a composite of the experimental work conducted to date on human tolerance to heated ambient air. This curve shows that the available escape time at 400°F would be about 20 sec. This temperature is comparable to the respiratory level temperature of 390°F selected by NACA as discussed in the following section.

3.3.1.2 Respiratory Injury: Since occupants of burning aircraft may inhale hot gases that can inflict respiratory system injuries, a tolerance criterion is needed. However, a thorough knowledge of rapid incapacitation from respiratory system injury is lacking. In fact, the general knowledge concerning this aspect of human tolerance is so limited that, for all practical purposes, there are not enough data available to establish an escape limit threshold.

A temperature of 390°F was chosen by NACA as a threshold value to permit a gross comparison of the relative hazards of respiratory and skin injury levels (Reference 7). The 390°F was chosen since it is the highest known temperature to which a human respiratory system has been exposed without damage.

10. Pryer, A. J., and Yuill, C. H., MASS FIRE LIFE HAZARD, Southwest Research Institute, San Antonio, Texas, September 1966.

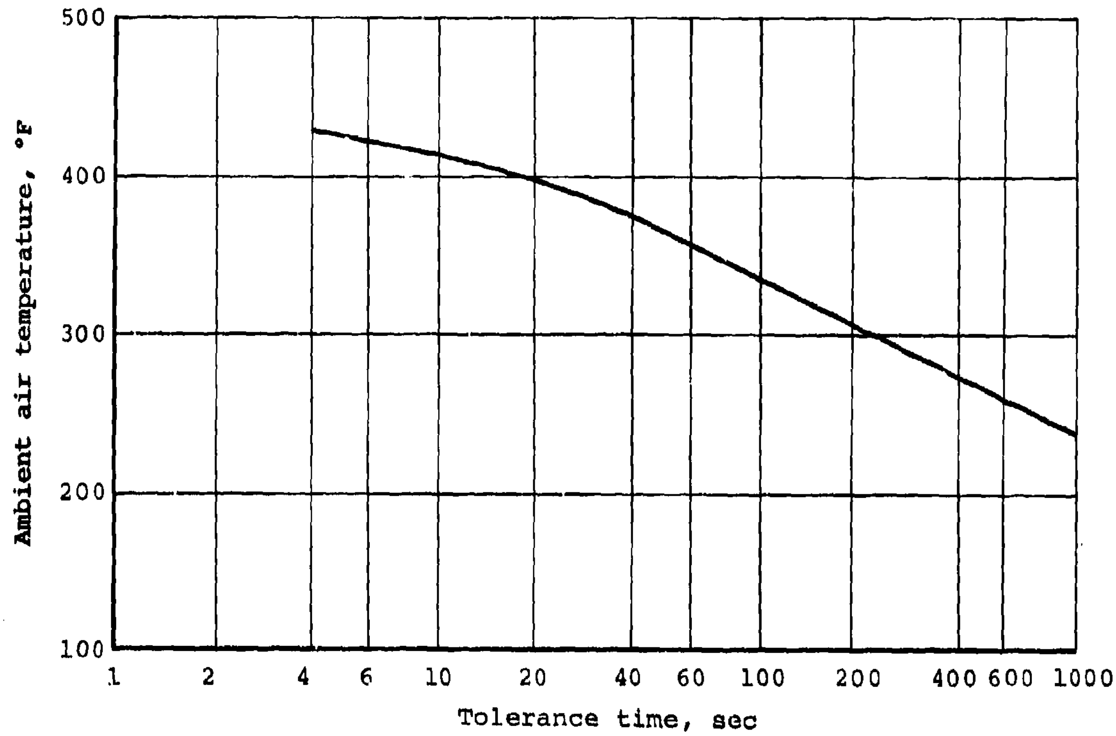

Figure 8. Human tolerance to ambient air temperatures. (From Reference 10)

3.3.2 Human Tolerance to Toxic Gases

The ability to escape successfully from a burning aircraft also depends on a person's tolerance to the many toxic gases present during a crash fire. Of these, carbon monoxide (CO) is generally the most prevalent.

The physiological effects of various carboxyhemoglobin (COHb) levels are shown in Figure 9. During a detailed study, NACA established that when an aircraft occupant breathes enough CO to cause a COHb level (the percent of CO saturation in the blood) of 35 percent, the individual's judgment becomes impaired (Reference 11). Consequently, when a COHb level of 35 percent is reached, the occupant's self-initiated escape capability becomes limited.

11. Forbes, W. H., Sargent, F., and Roughton, F. J. W., THE RATE OF CARBON MONOXIDE UPTAKE BY NORMAL MEN, American Journal of Physiology, Vol. 143, April 1945.

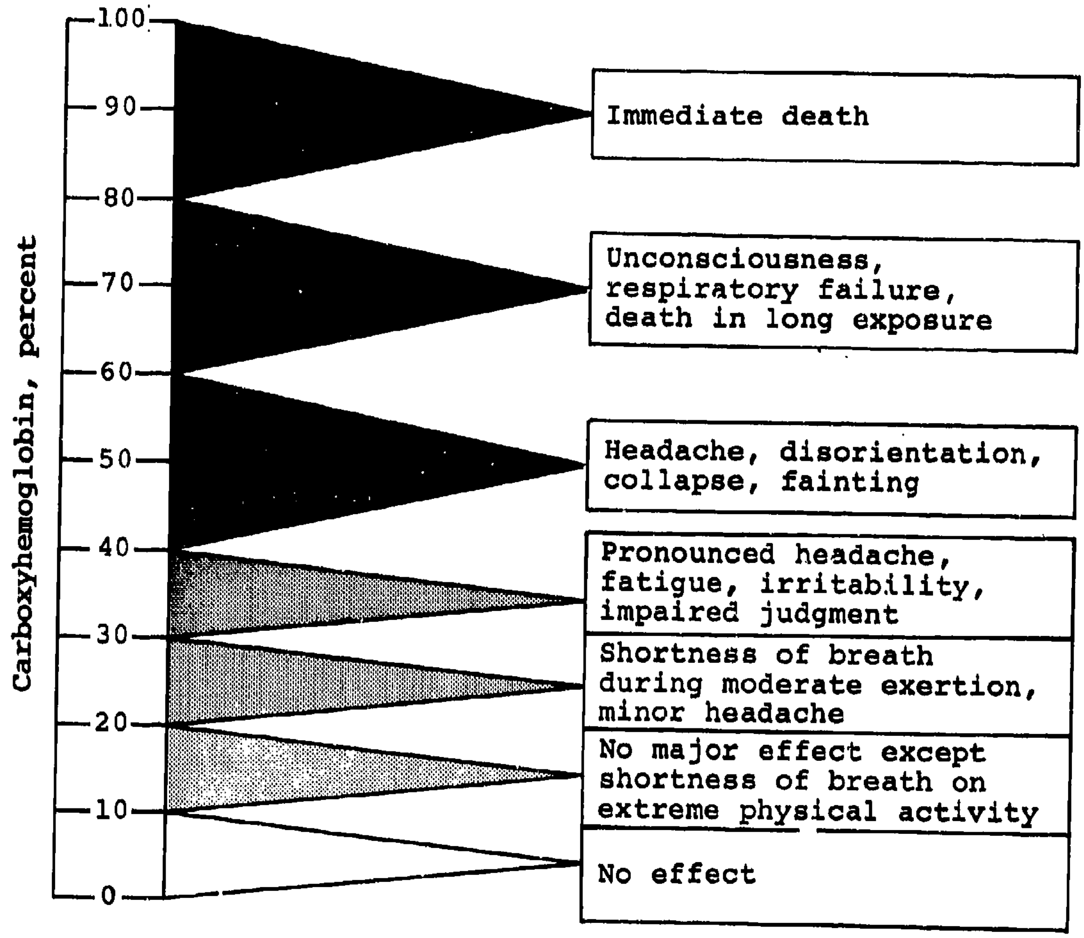

Figure 9. Physiological effects of various carboxyhemoglobin percentages.

The rate of CO uptake in humans exposed to various CO concentrations in the air is shown in Figure 10. These curves were derived by using the following equation from Reference 10:

$$COHb = K \times CO \times t \qquad (1)$$

where COHb = percentage of carboxyhemoglobin formed

CO = percentage of CO in the air

t = exposure time, minutes

K = absorption constant.

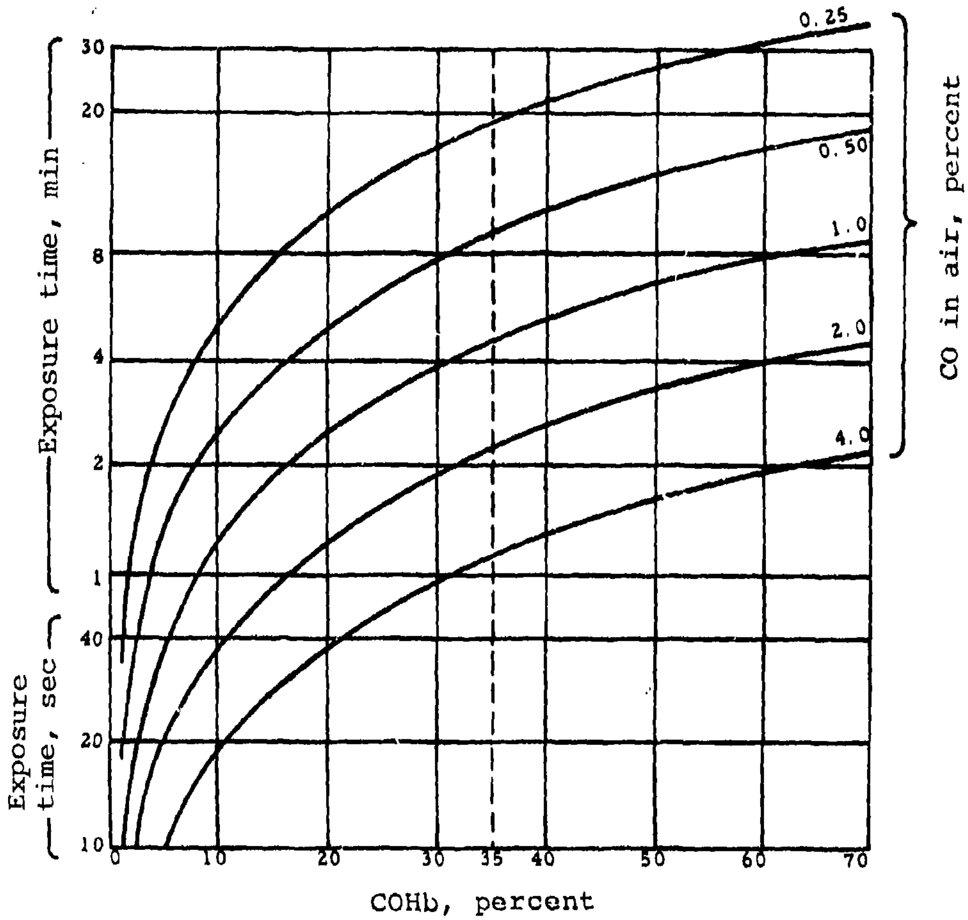

Figure 10. Probable CO uptake in humans exposed to various air concentrations of COHb.

The absorption constant, K, depends upon the ventilation rate (volume of air inhaled per minute) of the exposed person. Since the ventilation rate depends upon the type of work being done, the constant K is equal to 3 for persons at rest, 5 for light activity, 8 for light work, and 11 for heavy work. NACA has chosen a ventilation rate equal to that of persons engaged in light work as approximately that which would be encountered in persons attempting to escape from a burning aircraft (Reference 7). Therefore, a value of 8 was used for the absorption constant in deriving the curves. Figure 10 shows that the escape-limiting 35 percent COHb level is reached when the individual breathes 3.0 percent CO for 90 sec.

Although the escape times as limited by CO inhalation are generally longer than those limited by thermal skin injury, CO cannot be disregarded as a serious hazard. The relationship between ambient temperature and CO concentration will be dependent upon the type of crash, position of the occupant in the aircraft, slope of the impacted terrain, direction of the wind, and availability of fire-fighting equipment. It is entirely possible, especially in larger aircraft, that an individual's escape time could be limited by the CO concentration in the air rather than by thermal injuries.

Other gases that may limit escape times from burning aircraft include hydrogen cyanide (HCN), hydrogen chloride (HCl), nitrogen dioxide (NO_2), and many others. Approximate human tolerance limits to the most commonly expected aircraft fire gases are given in parts per million (p/m) in Table 1 (Reference 12).

TABLE 1. TOLERANCE TO SELECTED COMBUSTION GASES

	Hazardous levels (p/m) for times indicated			
Combustion Gas	Minutes	1/2 hr	1-2 hr	8 hr
Carbon dioxide	50,000	40,000	35,000	32,000
Carbon monoxide	3,000	1,600	800	100
Sulphur dioxide	400	150	50	8
Nitrogen dioxide	240	100	50	30
Hydrogen chloride	1,000	1,000	40	7
Hydrogen cyanide	200	100	50	2

Although there is considerable variation among researchers as to what level of a particular gas does constitute a life hazard, the limits given in Table 1 are typical of the ranges found.

12. FIRE SAFETY ASPECTS OF POLYMERIC MATERIALS, VOLUME 6 - AIRCRAFT: CIVIL AND MILITARY, Publication NAMB 3186, National Materials Advisory Board, National Academy of Sciences, Washington, D. C., 1977, p. 184.

Perhaps more importantly, these data illustrate the relative lethality of the various gases. Animal experiments have confirmed that the toxicity rankings of three of the most common gases are, in decreasing order, HCN, NO_2, and HCl (Reference 13).

The escape times based on CO inhalation must be considered to be the maximum escape times since other toxic gases present in the crash-fire environment are much more toxic than CO. In addition, the synergistic effects of combined gases and heat on toxicity are not well defined, although it has been established that heated gases or combinations of gases can be more lethal to the human than a single cool gas. Until these synergistic effects are studied in more detail, the lethal effects of each gas in a combination must be considered to be additive to the lethal effects of the other gases.

3.3.3 Human Tolerance to Miscellaneous Fire Factors

Discussions with survivors of actual aircraft accidents have indicated that there are many other factors associated with the crash-fire situation that can affect one's ability to escape. Included are visual obstructions, eye and throat irritants, fire-blocked exits, panic, and the heat factor associated with blowing hot air.

Once openings appear in the fuselage shell surrounding the occupants during a crash fire, rapid air flow through the occupiable area can begin. (It was noted during some of the full-scale aircraft burn tests conducted by AvSER that air flow through the fuselage reached speeds as high as 35 mph.) This air flow is usually hot, turbulent, and laden with toxic gases and debris. It can create a high startle factor in the occupants because it affects their breathing and causes them to lose sight of the surrounding area. Particulate matter in the smoke either blocks their vision or gets into their eyes, causing the individual to close them. Further, the smoke enters the respiratory tract, causing severe coughing and choking. Panic often results.

In view of the above hazards, the question of whether it is safer to stand up or crawl out of the aircraft is often asked.

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13. Higgins, E. A., Ph.D., et al., THE ACUTE TOXICITY OF BRIEF EXPOSURES TO HF, HCl, NO_2, AND HCN SINGLY AND IN COMBINATION WITH CO, FAA Civil Aeromedical Institute; Report No. FAA-AM-71-41, Department of Transportation, Federal Aviation Administration, Office of Aviation Medicine, Washingtion, D. C., November 1971, AD 735160.

In the opinion of the authors, there is no safe way. The turbulence in the air flow due to seats, occupants, and other vortex generators is so great that no safe zone exists. Low-flammability clothing, a sound knowledge of evacuation procedures, and the ability and knowledge to hold one's breath while exiting the aircraft are the occupant's primary assets for survival. Once a fire has started, the only aircraft-related evacuation advantages an occupant can have are properly designed and located exits, escape aisles, and emergency lighting.

4. POSTCRASH FIRE PROTECTION

4.1 INTRODUCTION

Postcrash fire research and accident experience with the crashworthy fuel systems have shown that: (1) improvements in ground fire-fighting systems will provide little improvement for chances of survival in accidents where a postcrash fire is present; (2) a reduction of fuel spillage and ignition sources following a crash will reduce the probability of postcrash fire; and (3) greater emphasis on "built-in" postcrash fire protection during the aircraft design stage will improve overall postcrash fire resistance.

This chapter presents basic design guidelines for Army aircraft systems that will inherently resist flammable fluid spillage and ignition during survivable accidents. The fuel containment approach is discussed first, followed by a brief summary of fuel modification. Ignition source control, also presented, is applicable with all forms of spillage. However, it has not, by itself, proved to be a practical solution to the postcrash fire problem.

When designing aircraft fuel, hydraulic, electrical, structural, and other systems, two basic requirements must be met: (1) each system must be highly functional from the standpoint of operational and maintenance needs; and (2) the combined system must exhibit resistance to crash fire. These requirements can be achieved only through a design based on careful integration of the various systems, with full consideration being given to operational and crashworthiness requirements.

The mating of the systems that offer a fire reduction potential as well as the required operational capabilities does not necessarily imply an overall weight or cost increase. Simplicity in the fuel system, which is desirable from the standpoint of requiring minimum attention from the crew, may well lead to a more crashworthy system. By following the design suggestions contained herein, and by thoroughly understanding the fire problem as discussed in Chapter 3, crashworthy systems that will be practical from the standpoint of both weight and cost can be designed.

4.2 FUEL CONTAINMENT

The design philosophy for crash-resistant fuel systems in aircraft is based upon the need to control postcrash fire in otherwise survivable accidents. In examining the basic elements contributing to postcrash fire, three factors emerge: an oxidizer, a combustible agent, and an ignition source.

Since it is not feasible to completely control the supply of oxygen immediately surrounding the aircraft, control is best exercised over the remaining two elements: the fuel and the ignition source.

The ideal fuel system is one that completely contains its flammable fluid both during and after the accident. To accomplish this, all components of the system must resist rupture regardless of the degree of failure of the surrounding structure. Success of such a system depends on proper selection of materials and design techniques in each of the following areas:

- Fuel tanks.
- Fuel lines.
- Supportive components and subsystems.

There is no single, universally adaptable fuel system for aircraft. Each aircraft manufacturer must design his own crashworthy system based on the criteria presented in the following sections. (A rating method to help the designer select a crashworthy fuel system design for his particular aircraft is presented in Reference 14.) Although the criteria given below are specifically applicable to new aircraft design, it also is possible to modify existing aircraft to include most of the crashworthy fuel system principles and components.

4.2.1 Fuel Tanks

4.2.1.1 Tank Location: The location of the flammable fluid-carrying tank in an aircraft is of considerable importance in minimizing the postcrash fire hazard from a tank installation. The location must be considered with respect to occupants, ignition sources, and probable impact areas.

Greater distance between occupants and fuel supply tends to increase escape time in the event of a fire because it reduces the likelihood of fuel entering the occupied area. Also, the tank should be kept away from probable ignition sources. While this is not always feasible, tanks should not be installed in or over the engine compartment, the battery, or other primary

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14. Robertson, S. H., and Turnbow, J. W., A METHOD FOR SELECTING A CRASHWORTHY FUEL SYSTEM DESIGN, paper presented at NATO/AGARD Operational Helicopter Aviation Medicine Symposium at Fort Rucker, Alabama in May 1978, North Atlantic Treaty Organization, Advisory Group for Aerospace and Development, Neuilly-sur-Seine, France.

ignition sources. Another important consideration is the location of tanks with respect to probable impact damage. Accident histories show repeated tank ruptures and consequent fires as a result of landing gear failures, indicating the tank's high degree of vulnerability to damage from surrounding structures.

Locating fuel tanks under a helicopter floor poses a serious threat because of the propensity toward accidents in near-level flight attitude at high sinking speeds. It is obvious that fuel tanks mounted low on the fuselage will contact the ground early in the crash sequence and will be exposed to possible penetrations from rocks, stumps, and other ground irregularities. Thus, a good design technique is to locate fuel tanks higher in the structure. As much aircraft structure as possible should be allowed to crush before the tanks themselves are exposed to direct contact with obstructions.

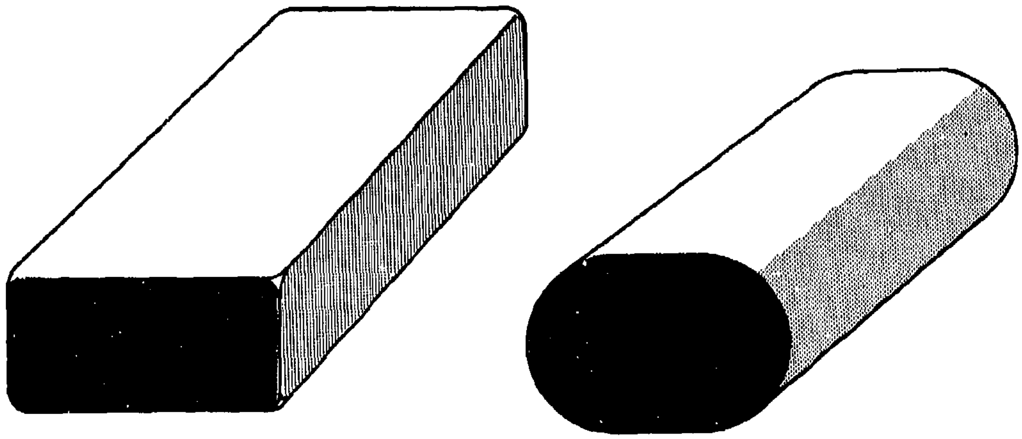
Fuel tanks in the wings should be located as far outboard as possible, but not at the tips. Accident investigations have shown that placing the tanks outboard of the engine nacelles in multiengine aircraft is preferred to locating them inboard of the engines. Placing the tanks in the wing tips should be avoided because these areas are anticipated impact points.

Reduction of fuel tank volume must be considered also. If the fuel tank is nearly full and located in an area where considerable structural collapse occurs, the tank may be subjected to pressures that exceed its design limit. It also may be exposed to puncture by torn and jagged metal. Therefore, if it can be predicted that the structure surrounding the tank may collapse due to compressive loads during a crash, expansion areas into which the tank and its contents may displace should be provided.

Another factor that can govern whether or not a fuel tank will survive a given impact is the method of failure experienced by the aircraft structure surrounding the tank. Care should be taken to ensure that when structural failure occurs in the area of the tank, sharp cutting surfaces, penetrating spars and longerons, and other injurious structures are avoided or controlled.

4.2.1.2 <u>Tank Shape</u>: The ability of the tank to displace easily and without snagging is largely dependent on its shape. Cylindrical or rectangular shapes appear to be best, whereas tanks with protuberances or tanks composed of several interconnecting cells (see Figure 11) are most vulnerable to rupture. Where tanks deviate greatly from the regular cylindrical or parallelepiped shapes, consideration should be given to the use of separate tanks with interconnecting, self-sealing fittings. To minimize snagging and excessive concentration of

Preferred smooth shapes
and flatwise orientation



Avoid irregular shapes
and inside angles

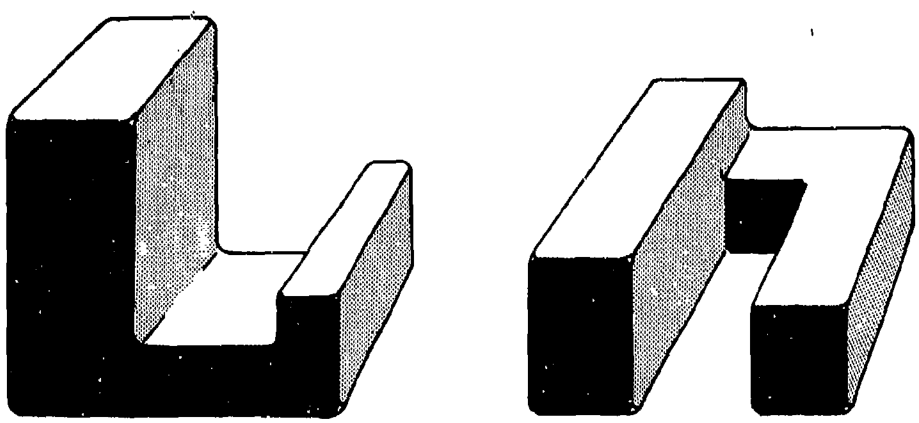


Figure 11. Tank shapes.

stresses, inside angles should be avoided if at all possible, especially in the lower portions of the cell. All outside angles should have a radius of at least 1 in. If possible, the tank should be oriented so that the side with the greatest surface area is facing the direction of probable impact.

4.2.1.3 Tank Materials: The concept of fluid containment requires materials and fabrication techniques that will maximize the energy-absorbing ability of the fuel system. Tanks constructed in accordance with earlier military specifications for crash resistance lacked such qualities and therefore failed under minimal severity of crash conditions. Crash-resistant fuel system research has shown, however, that fuel tanks constructed of materials possessing a high degree of cut and tear resistance, as well as a moderate degree of elongation, can accommodate very high impact levels without loss of fuel. These research programs resulted in Revision B to MIL-T-27422 for crash-resistant tanks (Reference 15).

Tanks made to the specifications of MIL-T-27422B have demonstrated an ability to hold their contents safely during the upper-limit survivable crash. However, these demonstrations have been conducted with fuel tanks containing less than 500 gal installed in small-to-moderate-sized airplanes and helicopters. Additional research in all aspects of fuel tank crashworthiness should be conducted before tanks with capacities exceeding 500 to 600 gal are used.

In order to provide the reader with a better understanding of the properties of crashworthy fuel tank materials, the following general discussion is presented.

Elongation can be obtained by tank deformation or material stretch. The amount of fuel tank elongation actually required is unknown. It is known, however, that fuel tanks lacking the ability to elongate are either fairly strong (heavy) or brittle. Both types are easily ruptured in moderate crashes. On the other hand, crash-resistant fuel tank studies have shown that light tanks that can readily rearrange their shape (deform/elongate), at the same time exhibiting a high degree of cut and tear resistance, can hold their contents during upper-limit survivable crashes.

The amount of tensile strength a tank material should possess also is debatable. Early attempts to define a tank material

15. Military Specification, MIL-T-27422B, TANK, FUEL, CRASH-RESISTANT, AIRCRAFT, Department of Defense, Washington, D. C., 24 February 1970.

property in terms of tensile strength proved unsuccessful. In fact, crash-resistant fuel system studies showed that tanks with lower tensile strengths were more difficult to rupture than ones with higher tensile values, providing, of course, that the tanks still exhibited a high degree of cut and tear resistance (Reference 16).

At the time of this writing, the only reason known for a minimum tensile strength requirement is to provide enough load-carrying capability between the tank wall and the tank fitting to cause the fitting to pull free of the airframe structure rather than out of the tank. This usually requires the failing of some sort of frangible fastener between the tank fitting and the airframe.

What, then, defines whether or not a tank is crashworthy? The overall results of extensive U. S. Army-funded crashworthy fuel system studies indicated that cut, tear, and impact resistance were the key issues. However, tank shape, flexural modulus of the material, reinforcement orientation, and loading rate sensitivity were all involved. The B revision of MIL-T-27422 was prepared as a result of the U. S. Army tests, and is the best source to date to define fuel tank crashworthiness.

The cut- and tear-resistance tests, defined in MIL-T-27422B, are self-explanatory. The values specified have proven to be effective in actual crashes.

The importance of a material's tear resistance is illustrated in Figure 12. These load-deflection curves were obtained from tear tests of 3 x 7-in. specimens containing an initial 3-in.-long slit (see Figure 13). Figure 12 shows the load required to propagate the initial slit as a function of the displacement of the pull jaws of the test device. The area under the curve is a measure of the energy required to completely fail the specimen. The energy required to fail the MIL-T-27422B material is almost six times that required for the 0.063-in. aluminum, although the nylon/rubber composite is lighter in weight than the aluminum. The composite material, though somewhat heavier due to the nature of its construction, far surpasses MIL-T-27422A material, both in the load necessary to propagate the tear and in the energy required to completely

16. Robertson, S. H., and Turnbow, J. W., AIRCRAFT FUEL TANK DESIGN CRITERIA, Aviation Safety Engineering and Research of Flight Safety Foundation; USAAVLABS Technical Report 66-24, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, March 1966, AD 631610.

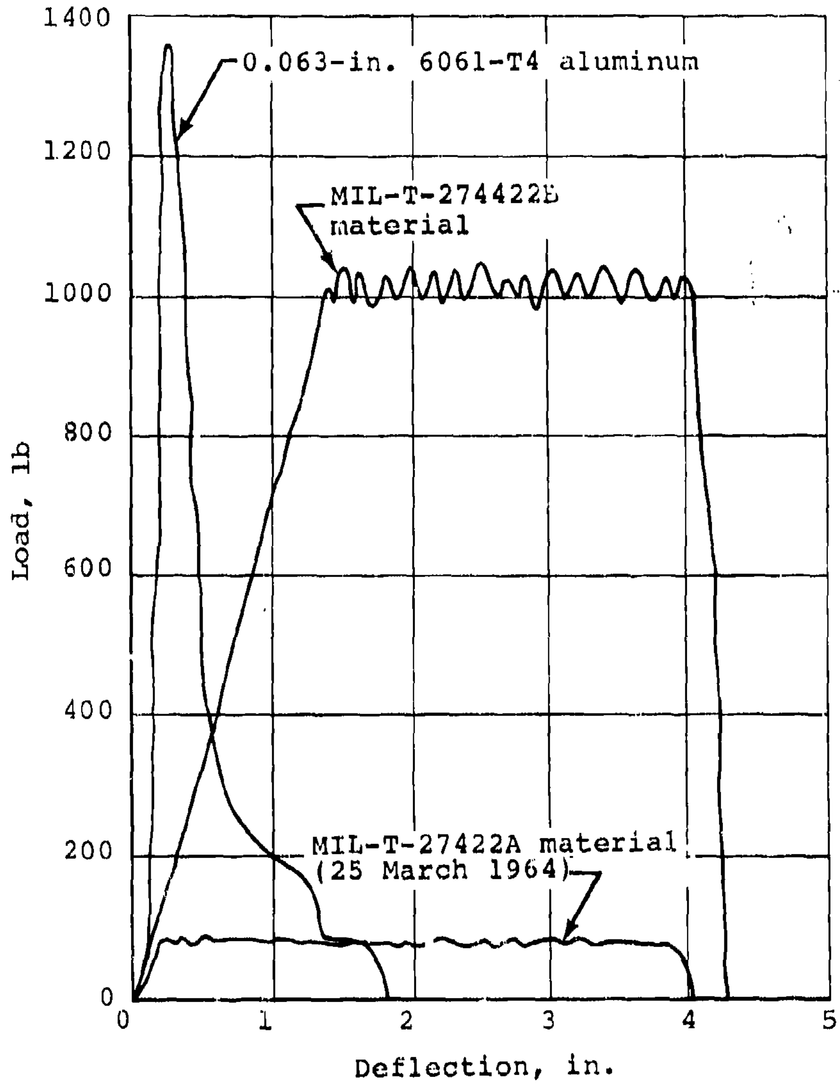
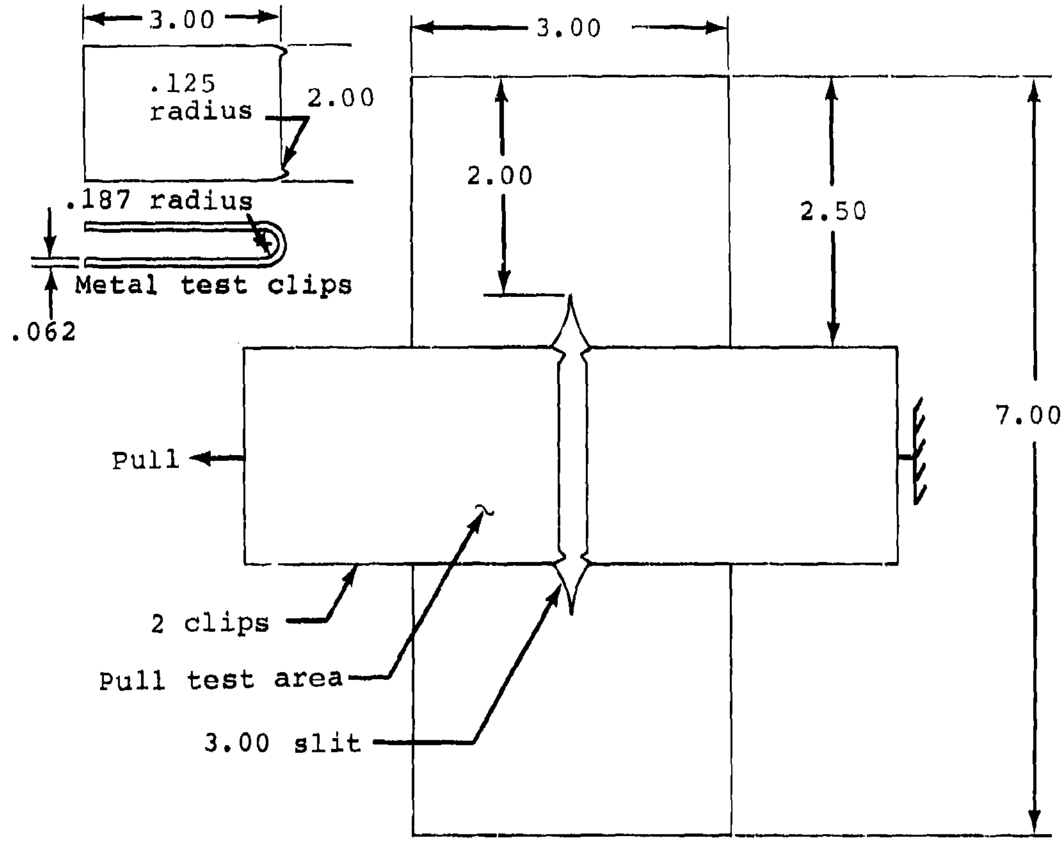


Figure 12. Resistance of materials to tearing.



NOTE: Dimensions in inches unless otherwise specified. Tolerances: ±.032

Figure 13. Slit test for obtaining resistance to tear propagation.

fail the material. Further data on these materials are available in References 16 and 17.

In order to assure that proposed tank designs have seam continuity, proper fitting installation and placement, and other overall crash impact resistance, a drop test requirement was

17. Cook, R. L., et al., IMPROVED CRASH-RESISTANT FUEL CELL MATERIAL, Goodyear Aerospace; USAAVLABS Technical Report 67-6, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, April 1967, AD 813165L.

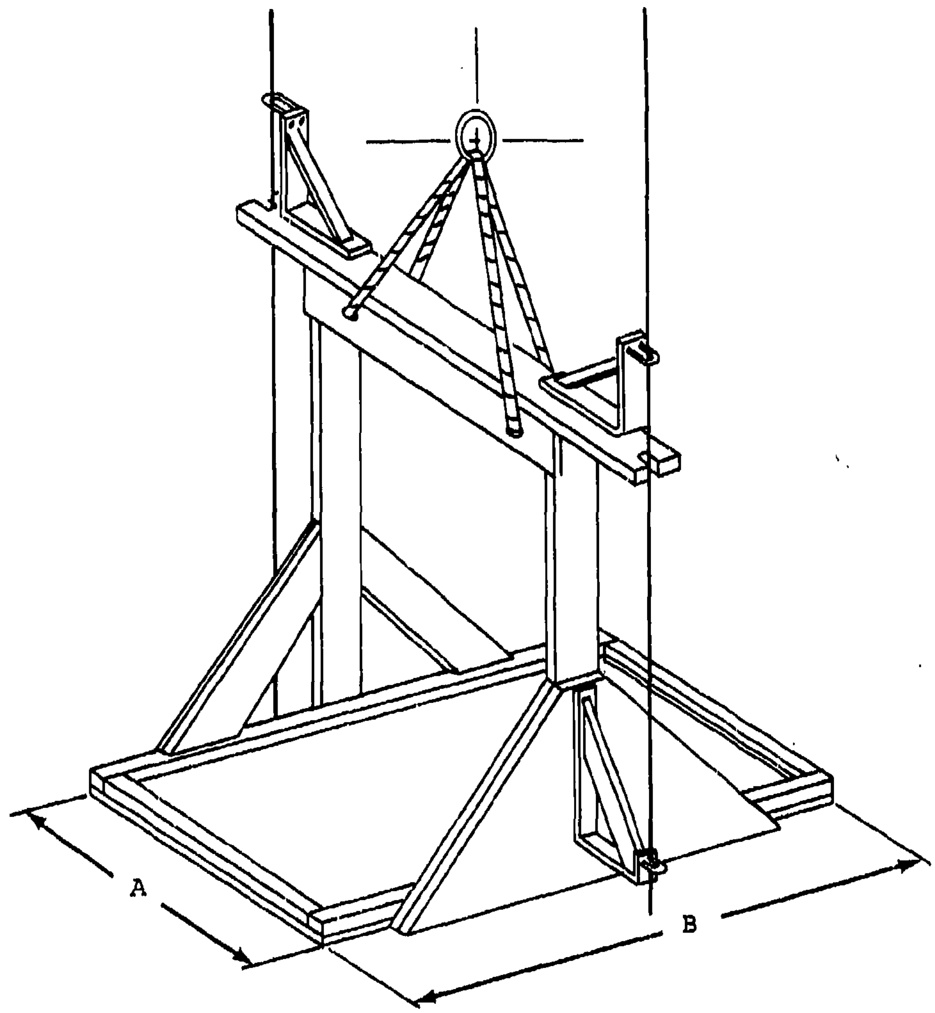
included in the MIL-T-27422B revision. Preproduction tanks in both the standard 30-in. cubes and the look-alike configurations, with all openings suitably closed, are filled with water to normal capacity (air removed) and mounted on a platform of the design shown in Figure 14. Lightweight cord is used to maintain the cell in normal flight attitude. The platform is raised to a height of 65 ft, released, and allowed to drop freely onto a nondeforming surface with the platform horizontal (±10 degrees) at impact for rotary-wing aircraft and at an angle of 20 ± 10 degrees with the horizontal for fixed-wing aircraft. No liquid leakage is allowable following the test.

The 65-ft drop height results in a severe impact test of the fuel tank. This provides an adequate safety margin should an aircraft crash into rough terrain (e.g., rocks and stumps), thereby placing localized loads on the tank. Furthermore, aircraft structures surrounding the fuel tank sometimes fail in a manner that creates additional hazards to the tank. This factor also is accommodated in the safety margin provided by the 65-ft drop test.

Review of recent crashes reveals that fuel tanks that have been designed to the existing criteria, including the 65-ft drop test, are failing and releasing their contents, with fires resulting, in accidents at and slightly above the human survival range. This verifies the validity of the design criteria. No reduction in drop height, or of cut- and tear-resistance values, should be allowed without first conducting a major, long-term test program to measure and define any other requirements necessary to maintain the integrity of crashworthy fuel tanks.

4.2.1.4 Tank Fittings: A fuel cell failure often is caused by physical displacement of the aircraft structure in relation to the tank. This places stress concentrations at tank attachment points such as filler necks/caps, tank outlets, boost pumps, and drains. The tank fitting can be pulled from the tank, tearing the tank wall. Often, if the energy levels are sufficiently high, this tear will circumscribe the entire tank. Until MIL-T-27422B became effective, fuel cell fittings could be torn from standard .30 caliber self-sealing fuel tanks at loads corresponding to about one-third the strength of the tank wall. The new specification requires high-strength insert-retention methods in keeping with the high strength of the new fuel cell materials.

MIL-T-27422B specifies that all fuel tank fittings shall have a pullout strength of at least 80 percent of the fuel cell wall strength. The strength of the cell material is determined by measuring the force required to drive the end of a 4-in. diameter rod through a 13-5/8-in. diaphragm specimen of the cell



NOTE: Dimensions A and B shall not exceed cell dimensions (when the loaded cell is in place for test) by more than 12 in. in either direction.

Figure 14. Drop test fixture.

material that is supported around the perimeter. The rod has a 1/8-in. radius which forces the end into the sides. The rod is driven at a rate of 20 in./min.

A typical method for measuring the fitting pullout strength is shown in Figures 15 and 16. A test sample containing a 4-in.

Figure 15. Typical setup for dynamic testing of fuel cell fitting pullout.

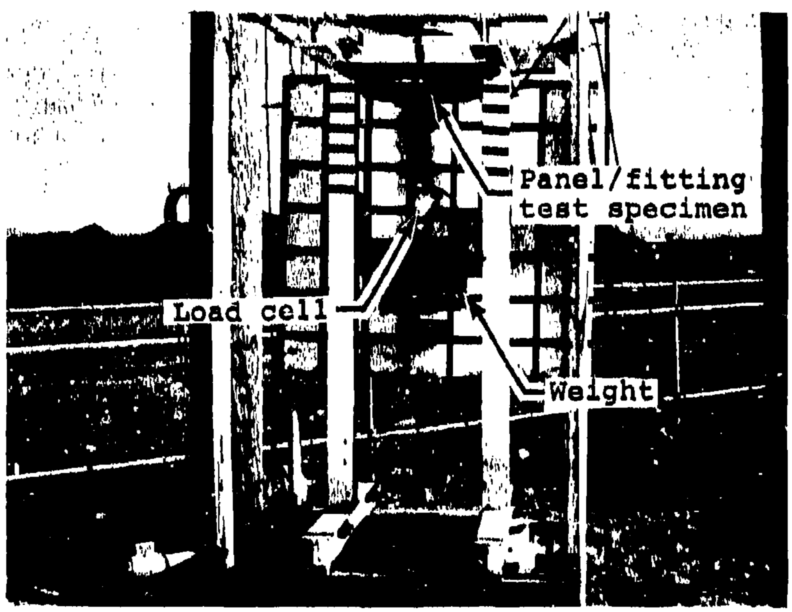


Figure 16. Typical fuel cell fitting pullout following dynamic test.

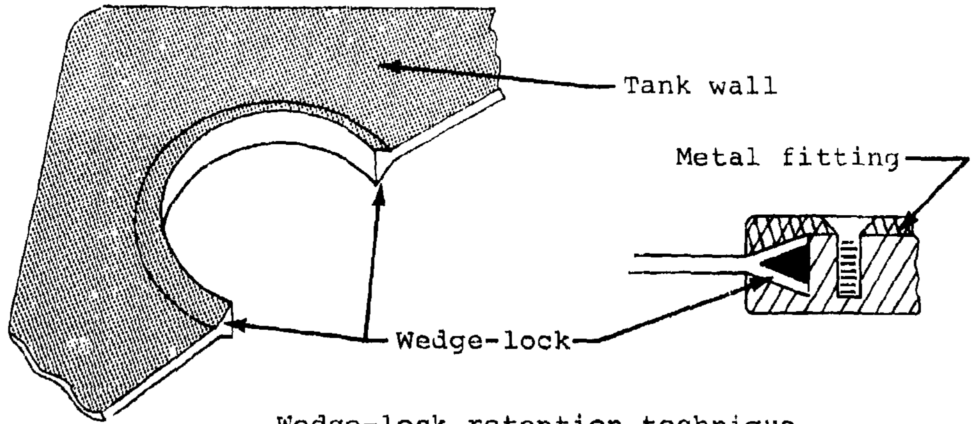
outside diameter fitting is fabricated of the tank material using the same fitting material and attaching methods used on full-size production tanks. A 200-lb weight is attached to the fitting as shown in Figure 15. A force transducer is located between the fitting and the weight, as close to the fitting as possible. The test sample is attached to a rigid drop cage, dropped from a height of 20 ft, and decelerated in a distance of 9 in. or less. There must be sufficient distance between the bottom of the weight and the cage to prevent bottoming prior to fitting pullout. The peak reading from the force transducer is the fitting pullout strength, which must be in excess of 80 percent of the failure load of the tank material but need not exceed 30,000 lb.

It is desirable, as a goal, for the fuel tank fitting to have a pullout strength equal to that of the tank wall. However, tank manufacturers have experienced great difficulty in meeting the 80 percent retention requirement currently specified in MIL-T-27422B. Consequently, the 80 percent value is an obvious compromise.

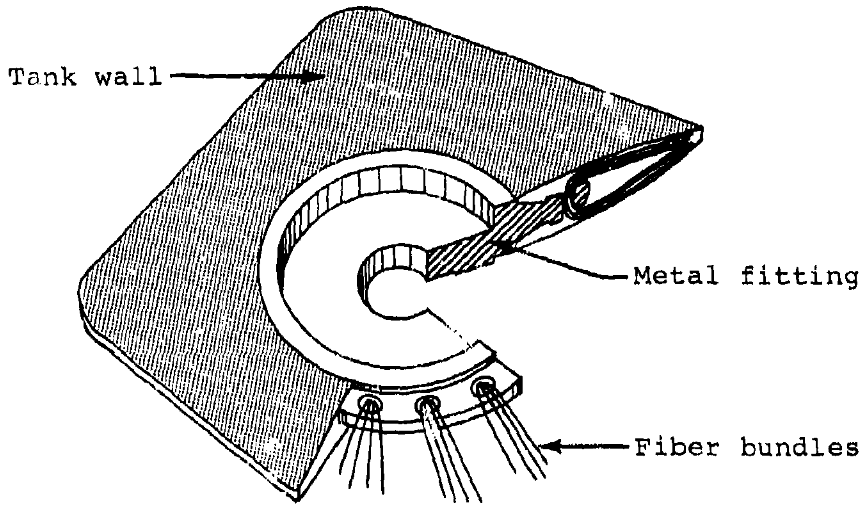
Two high-strength fitting designs that have met the 80-percent retention requirement are shown in Figure 17. Proprietary designs exhibiting strengths close to or even slightly in excess of the fuel cell wall strength have been demonstrated recently. Thus, it may be feasible to delete this compromise in the future.

4.2.1.5 Tank Attachments: To be crashworthy, the fuel tank must be secured to the airframe and connecting plumbing in a way that allows the tank to pull free of the attachments without rupturing when structural displacement occurs in a crash. Frangible brackets or bolts can be incorporated in the attachment technique to ensure their separation at specified loads. Frangible attachments may be designed to fail either the material itself (e.g., thin-walled hollow bolts that will fail during crash impact) or some facet of the design (e.g., protruding flanges that bend on exposure to crash forces). Several concepts, along with their applications, are illustrated in Section 4.2.3. The frangible attachment must be strong enough to meet all operational and service loads of the aircraft within a reasonable margin,* but should fail at 25 to 50

*A factor of 10 is a desirable goal to ensure that inadvertent actuation under normal operation is impossible. It is realized that this goal may not always be compatible with the 50-percent-attachment failure load criterion; however, the service load margin should be as high as possible.



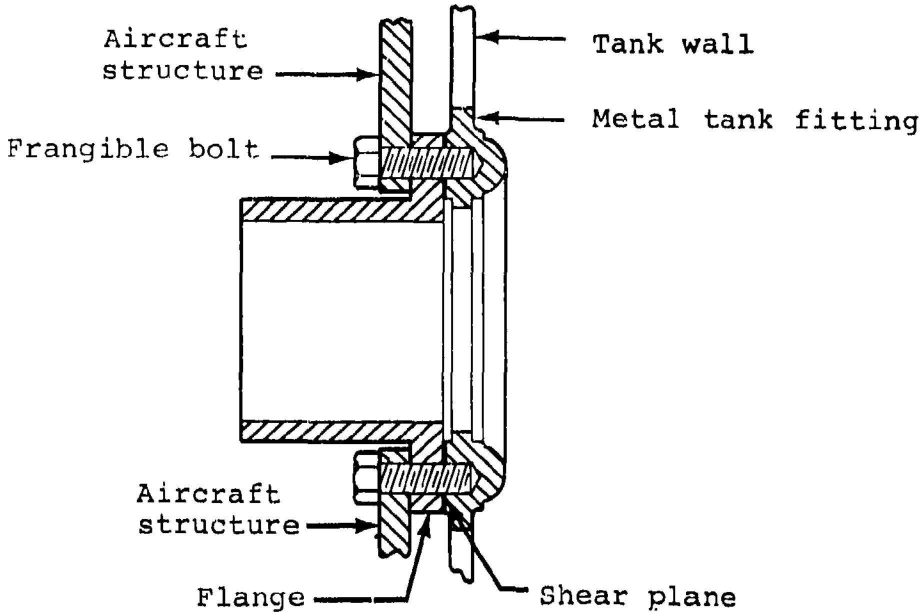
Wedge-lock retention technique



Fiber-lock retention technique

Figure 17. High-strength fitting retention techniques.

percent of the minimum load required to fail the attached system or component. This requires careful analysis of the various components in the fuel system for probable failure loads, load paths, and degrees of deformation. A sample breakaway load calculation is shown in Figure 18.



ITEM	LOWEST FAILURE LOAD (LB)*	FAILURE MODE
Aircraft structure	4000	Shear
Tank fitting	3000	Pull out of tank
Flange	5000	Shear
Frangible bolt	Not more than $\frac{3000}{2} = 1500$ Not less than $\frac{3000}{4} = 750$	Break (tension-shear)

*Loads may or may not be representative; values are for explanatory purposes only.

Figure 18. Sample frangible attachment separation load calculation.

The frangible attachments should be designed to separate efficiently in the direction of force most likely to occur during crash impact. Crash loads, whether tension, shear, compression, or combinations thereof, must be determined for each attachment by analyzing the surrounding aircraft structure and probable impact forces and directions.

4.2.2 Fuel Lines

4.2.2.1 Line Construction: Damaged fuel lines frequently cause spillage in aircraft accidents. Lines often are cut by surrounding structure or worn through by rubbing against rough surfaces. The use of flexible hose armored with a steel-braided harness is strongly suggested in areas of anticipated dragging or structural impingement. In systems where breakaway valves are not provided, hoses 20 to 30 percent longer than the minimum required hose length are desirable. This will allow the hose to shift and displace with collapsing structure, rather than be forced to carry tensile loads. For this reason, it is equally important that couplings and fittings be used sparingly because of their propensity to snag and restrict the natural ability of the hose to shift.

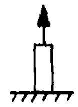
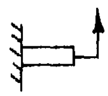
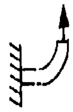
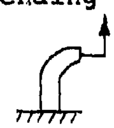
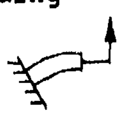
All fittings used in the fuel system should meet the strength requirements of Tables 2, 3, or 4 when tested in the modes shown. The loads are always applied through the hose with freedom allowed for the hose to form the bend radius. Thus, the effective moment arm for the bending tests changes primarily with the line size and secondarily as the applied load produces changes in the bend radius. This test procedure is much easier to mechanize than one requiring a constant moment arm.

All fuel lines should be secured with breakaway (frangible) attachment clips in areas where structural deformation is anticipated. When fuel lines pass through areas where extensive displacement or complete separation is anticipated, self-sealing breakaway valves should be used. The valves may be specifically designed for this purpose (Figure 19), or quick-disconnect valves may be modified for use (Figure 20). (See Section 4.2.3.1 for a more complete discussion of self-sealing breakaway valves.) These valves must meet all operational and service loads of the aircraft within a reasonable margin, but they should separate at between 25 and 50 percent of the minimum failure load for the weakest component in the fluid-carrying system. A sample breakaway load calculation is shown in Figure 21.

In designing a system using line-to-line breakaway valves, one should consider potential hazards to cross-axis shear loading on the valve halves. While omnidirectional separation is not an absolute requisite for most line-to-line valves, it is highly desirable, and every attempt should be made to procure omnidirectional valves if there is any possibility of cross-axis shear loading.

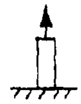
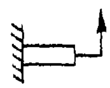
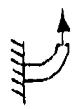
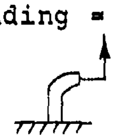
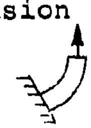
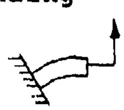
Figures 22 and 23 will assist the designer in determining the lever arms and bending moments imposed on frangible valves or

TABLE 2. REQUIRED MINIMUM INDIVIDUAL LOADS FOR STANDARD HOSE AND HOSE-END FITTING COMBINATIONS

Hose end fitting type	Fitting size*	Minimum tensile load (lb)	Minimum bending load (lb)
STRAIGHT	-4	575	450
Tension =	-6	600	450
	-8	900	700
	-10	1250	950
	-12	1900	1050
Bending =	-16	1950	1450
	-20	2300	1600
	-24	2350	2750
	-32	3500	4000
90° ELBOW	-4**	575	800
Tension =	-6**	600	850
	-8**	900	1250
	-10	1250	575
	-12	1900	675
Bending =	-16	1950	1200
	-20	2300	1250
	-24	2350	2025
	-32	3500	3500
45° ELBOW	-4**	575	
Tension =	-6**	600	425
	-8**	900	425
	-10	1250	425
	-12	1900	600
Bending =	-16	1950	1000
	-20	2300	1600
	-24	2350	2400
	-32	3500	3700

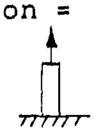
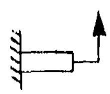
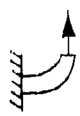
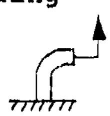
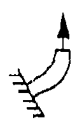
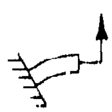
*Fitting size given in 1/16 in. units, i.e., -4 = 4/16 or 1/4 in.
**Elbow material is steel.

TABLE 3. REQUIRED MINIMUM INDIVIDUAL LOADS FOR SELF-SEALING HOSE AND HOSE-END FITTING COMBINATIONS

Hose end fitting type	Fitting size*	Minimum tensile load (lb)	Minimum bending load (lb)
STRAIGHT	-10	2000	-
Tension =	-12	3120	1050
	-16	2850	1650
	-20	2650	1700
Bending =	-24	3850	2500
	-32	2700	-
90° ELBOW	-10	1950	700
Tension =	-12	3400	3700
	-16	3100	4300
	-20	2500	2500
Bending =	-24	3800	2500
			
45° ELBOW	-10	1200	450
Tension =	-12	3000	800
	-16	3200	1800
	-20	2900	1700
Bending =	-24	3850	2500
			

*Fitting size given in 1/16 in. units, i.e., -10 = 10/16 or 5/8 in.

TABLE 4. REQUIRED MINIMUM INDIVIDUAL LOADS FOR SELF-SEALING HOSE WITH FLANGED END FITTINGS

Hose end fitting type	Fitting size*	Minimum tensile load (lb)	Minimum bending load (lb)
STRAIGHT	-12	2700	3600
Tension =	-16	2500	1650
	-24	2800	2500
Bending =			
			
90° ELBOW	-12	2400	2950
Tension =	-16	2700	1050
	-24	3900	2500
Bending =			
			
45° ELBOW	-12	3100	1000
Tension =	-16	2100	1350
	-24	3450	2500
Bending =			
			

*Fitting size given in 1/16 in. units, i.e., -12 = 12/16 or 3/4 in.

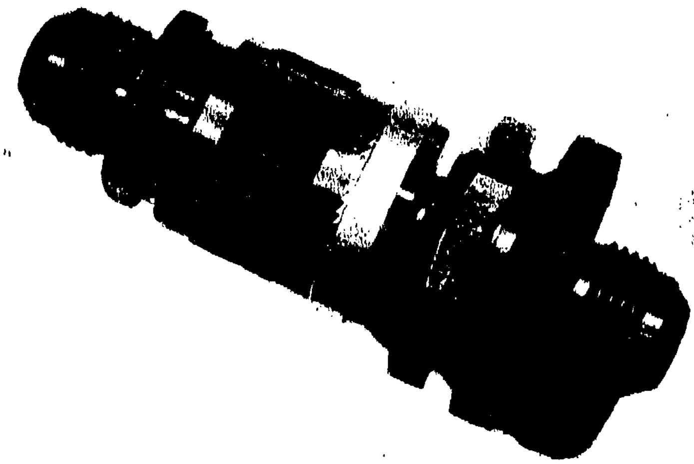


Figure 19. Specifically designed bulkhead (firewall)-to-line breakaway valve. (Tension on attaching hose causes valve separation.)

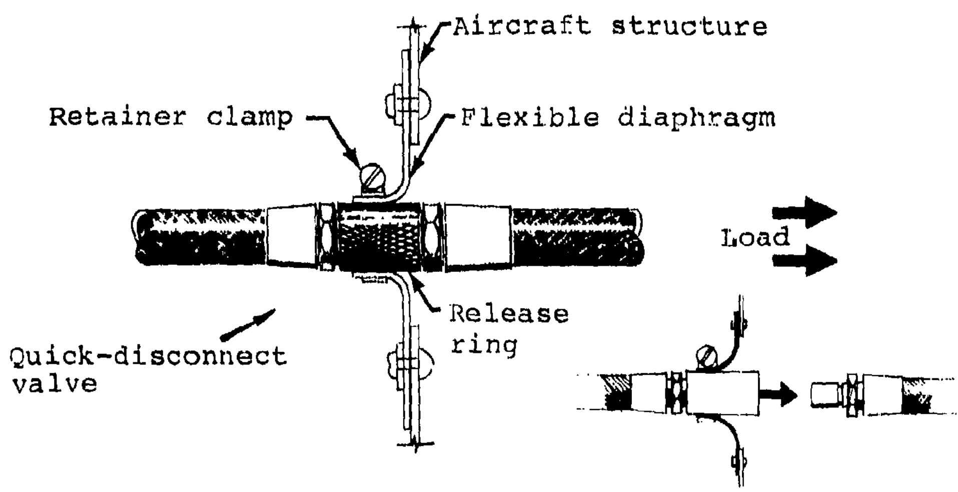
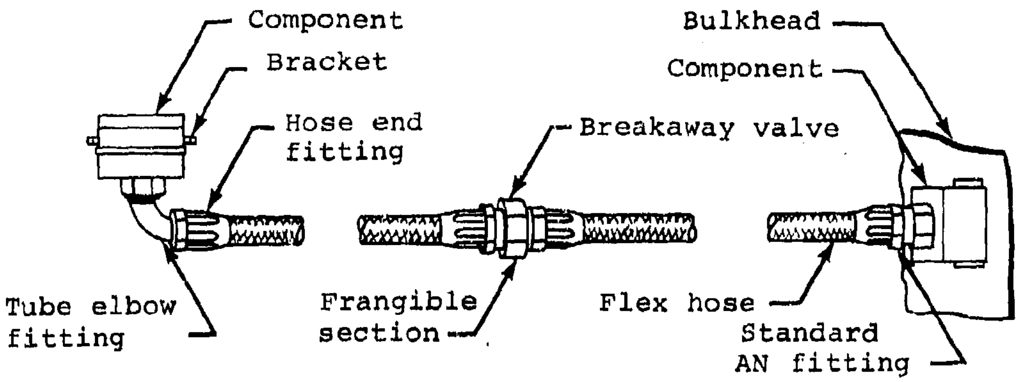


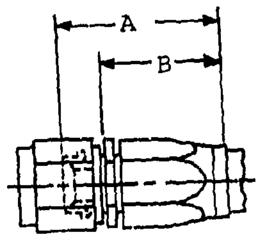
Figure 20. Modified quick-disconnect line-to-line valve. (Pull of designated hose will cause valve separation)



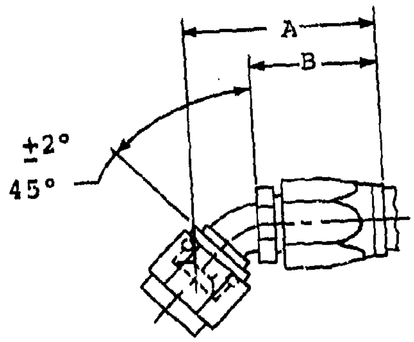
ITEM	LOWEST FAILURE LOAD (LB)*	FAILURE MODE
Flex hose	3000	Tensile breakage
Flex hose	1500	Pull out of end fitting
Hose end fitting	1650	Break (bending)
Standard AN fitting	1700	Break (bending)
Tube elbow fitting	1200	Break (bending)
Component structural attachments	4500	Pull out of structure
Breakaway valve	Not more than $\frac{1200}{2} = 600$ Not less than $\frac{1200}{4} = 300$	Break at frangible section

*Loads may or may not be representative; values are for explanatory purposes only.

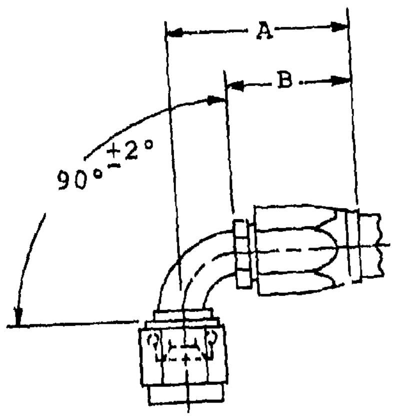
Figure 21. Typical breakaway load calculation for in-line breakaway valve.



Hose Size	A Maximum (in.)	B Maximum (in.)
-4	1.33	1.16
-5	1.38	1.18
-6	1.51	1.29
-8	1.79	1.48
-10	1.94	1.60
-12	2.01	1.70
-16	2.36	1.94
-20	2.64	2.13
-24	2.79	2.18
-32	3.16	2.45

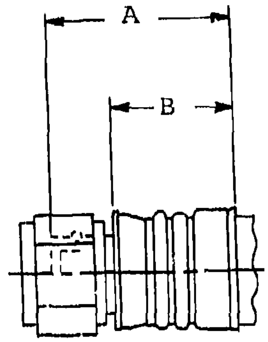


Hose Size	A Maximum (in.)	B Maximum (in.)
-4	1.72	1.16
-5	1.83	1.18
-6	2.00	1.29
-8	2.17	1.48
-10	2.42	1.60
-12	2.79	1.70
-16	3.06	1.94
-20	3.45	2.13
-24	3.65	2.18
-32	4.26	2.45

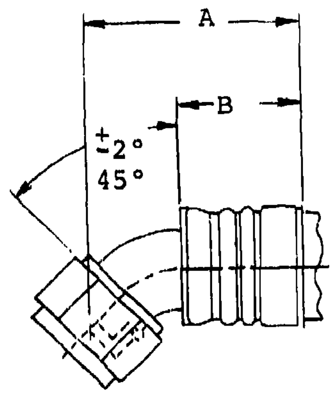


Hose Size	A Maximum (in.)	B Maximum (in.)
-4	1.59	1.16
-5	1.68	1.18
-6	1.85	1.29
-8	2.01	1.48
-10	2.25	1.60
-12	2.66	1.70
-16	2.97	1.94
-20	3.38	2.13
-24	3.59	2.18
-32	4.22	2.45

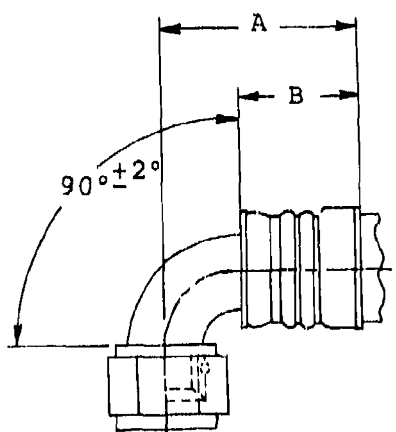
Figure 22. Standard hose fitting dimensions.



Hose Size	A Maximum (in.)	B Maximum (in.)
-10	3.66	3.16
-12	3.54	3.06
-16	3.62	3.06
-20	3.77	3.16
-24	3.76	3.06



Hose Size	A Maximum (in.)	B Maximum (in.)
-10	3.99	3.16
-12	4.07	3.06
-16	4.19	3.06
-20	4.50	3.16
-24	4.53	3.06



Hose Size	A Maximum (in.)	B Maximum (in.)
-10	3.52	3.16
-12	3.94	3.06
-16	4.10	3.06
-20	4.38	3.16
-24	4.47	3.06

Figure 23. Self-sealing hose fitting dimensions.

other attaching hardware of a crashworthy fuel system. The dimensions given are for standard hose fittings. When using nonstandard fittings, consult the appropriate drawings.

When applying these dimensions, one hose diameter (nominal) should be added to dimension A or B, whichever is used. One hose diameter is added because it approximately equals the offset of the load line adjacent to the hose socket as the hose collapses when pulled in the bending mode. Dimension A plus the nominal hose size is to be used in lever arm determinations when standard fittings as shown are used. Dimension B plus the nominal hose size is to be used in determining the lever arm when other than standard elbows are used.

For example, the lever arm of a -16 size standard 90-degree hose fitting for self-sealing hose, from Figure 23, is 4.10 in. plus one hose diameter, or 1 in. Therefore, the lever arm length equals 4.10 + 1 = 5.10 in.

The lever arm of a -10 size standard straight hose fitting for self-sealing hose, from the same figure, is 3.66 in. plus one hose diameter, or .63 in. Therefore, the lever arm length equals 3.66 + .63 = 4.29 in.

For a nonstandard fitting using -10 size self-sealing hose, the lever arm would be 3.13 in. plus one hose diameter, .63 in., plus the length contributed by the nonstandard component. Therefore, the lever arm length equals 3.16 + .63 + component length (in inches).

4.2.2.2 Line Routing: Routing of hoses should be carefully considered during the design stage. Fuel lines should be routed along the heavier structural members, since those members are less likely to deform or separate in an accident. Also, it is important that hoses have a space into which they can deform when necessary. For example, when hoses pass through large flat-plate areas, such as bulkheads or firewalls, the hole allowing line passage should be considerably larger than the outside diameter of the line. Hose stabilization as well as liquid-tight, fire-tight seals still can be maintained if a frangible structure, such as shown in Figure 24, is used.

If design requirements limit the use of the protective measures discussed above, full use should be made of self-sealing breakaway couplings located in areas of anticipated failures. Crossover connections, drains, and outlet lines present a special problem since they are usually located in the lower regions of the tank, where they are vulnerable to impact damage. Space and flexibility should be provided at the connections to allow room for the lines to shift with collapsing structure. Utmost

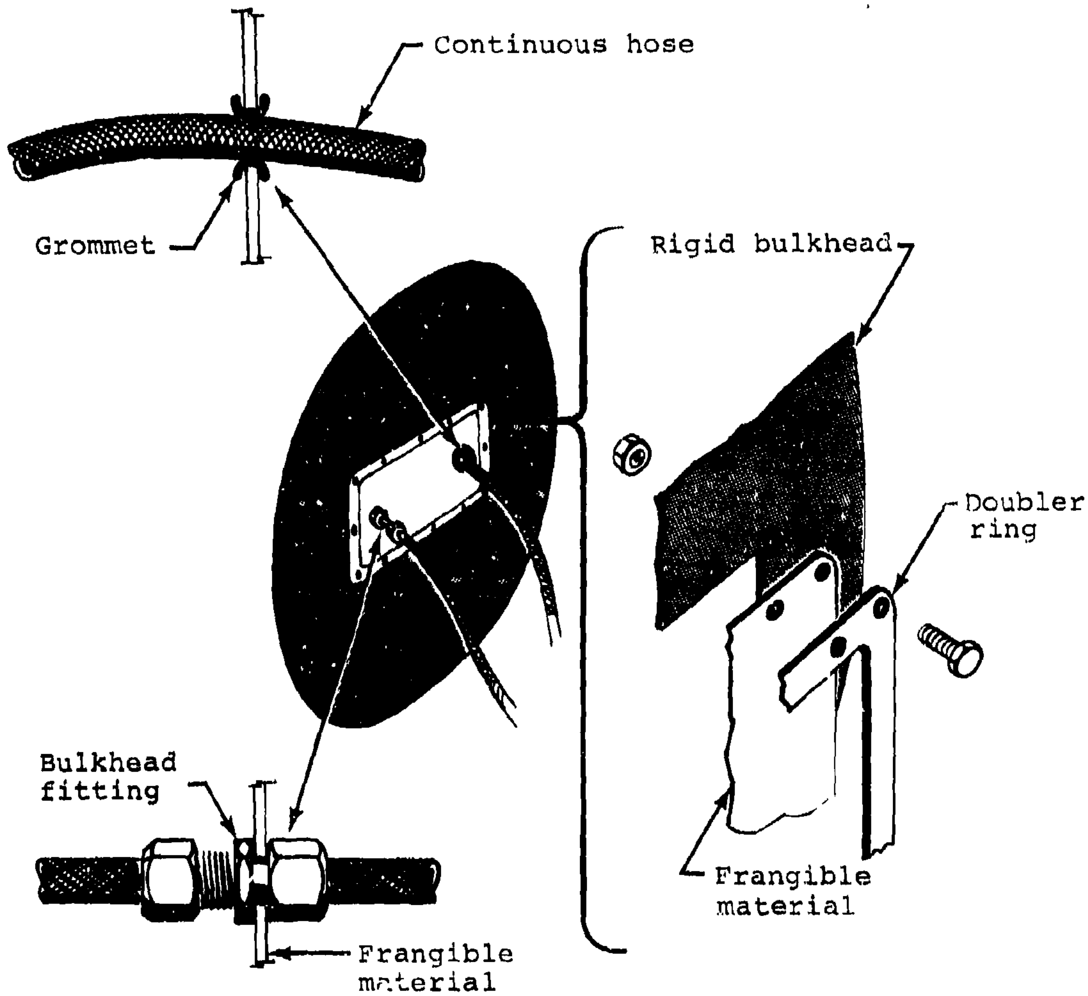


Figure 24. Hose stabilizing with frangible structure.

consideration should be given to using self-sealing breakaway fittings at each line-to-tank attachment point.

4.2.3 Supportive Components

Supportive components play a vital role in crashworthy fuel systems. Aside from providing a solution to specific problems, e.g., a strainer to help clean fuel, they also must be capable of preventing spillage in accidents with resulting forces equal to or better than the tank strength. They must not be the weak link in the system. Care must be taken during the design and testing phase to ensure that the supportive items, some of which are discussed below, will not fail during the crash sequence and allow spillage.

4.2.3.1 <u>Self-Sealing Breakaway Valves</u>: Self-sealing breakaway valves are valves designed to separate into two or more sections and seal the open ends of designated fluid-carrying passages. The openings may be in fuel/oil lines, tanks, pumps, fittings, etc. The valves fall into two general categories: the "one-shot" type, which usually incorporates a frangible portion that breaks upon valve operation (Figure 25), and the quick-disconnect type, which is installed so that it will be triggered (released) during the crash sequence (Figures 20 and 26). Some valves in use today have both these features incorporated into their design. Each specific fuel system design will dictate which of the two types of valves can be used. In either case, the valves must be installed in a manner that precludes inadvertent operation.

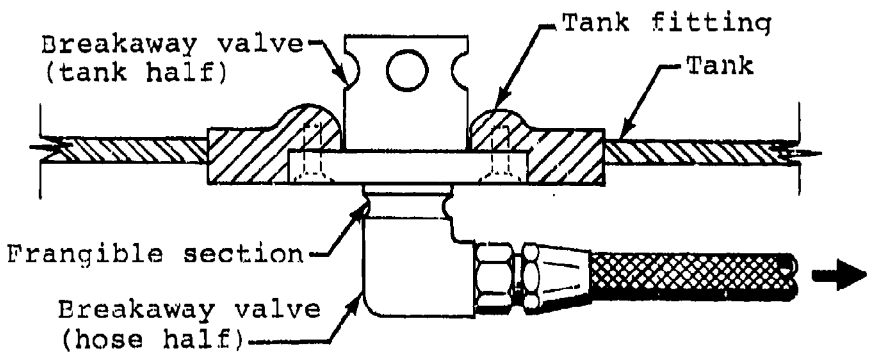


Figure 25. "One-shot" self-sealing valve. (Load on hose or lower valve body causes separation at frangible section.)

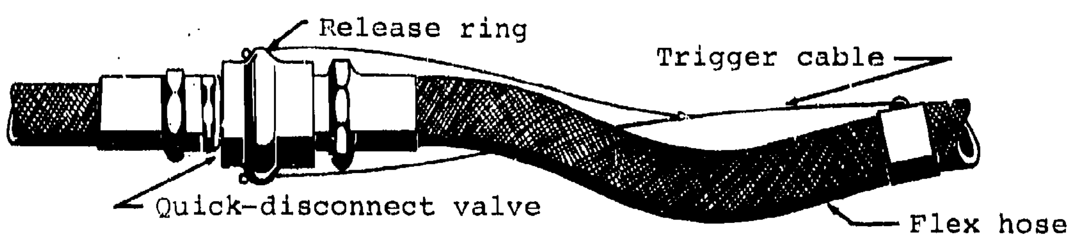


Figure 26. Cable-actuated quick-disconnect valve. (If a triggered cable system is used, its location must be carefully selected to prevent inadvertent valve actuation during normal aircraft operation and maintenance.)

Self-sealing breakaway valves should be located at each fuel-carrying tank outlet and at locations within the fuel line network where extensive displacement is foreseeable, such as wing roots or engine compartments. The purpose of these valves is to prevent rupture of the tank, hoses, or fitting components by placing a "safety fuse" in the load path.

A self-sealing breakaway valve should be used to connect two fuel cells in a direct side-by-side arrangement, especially if there is a high probability that structure failure will occur in the immediate area of the cells. Figure 27 shows a breakaway valve mounted in such a cell-to-cell installation.

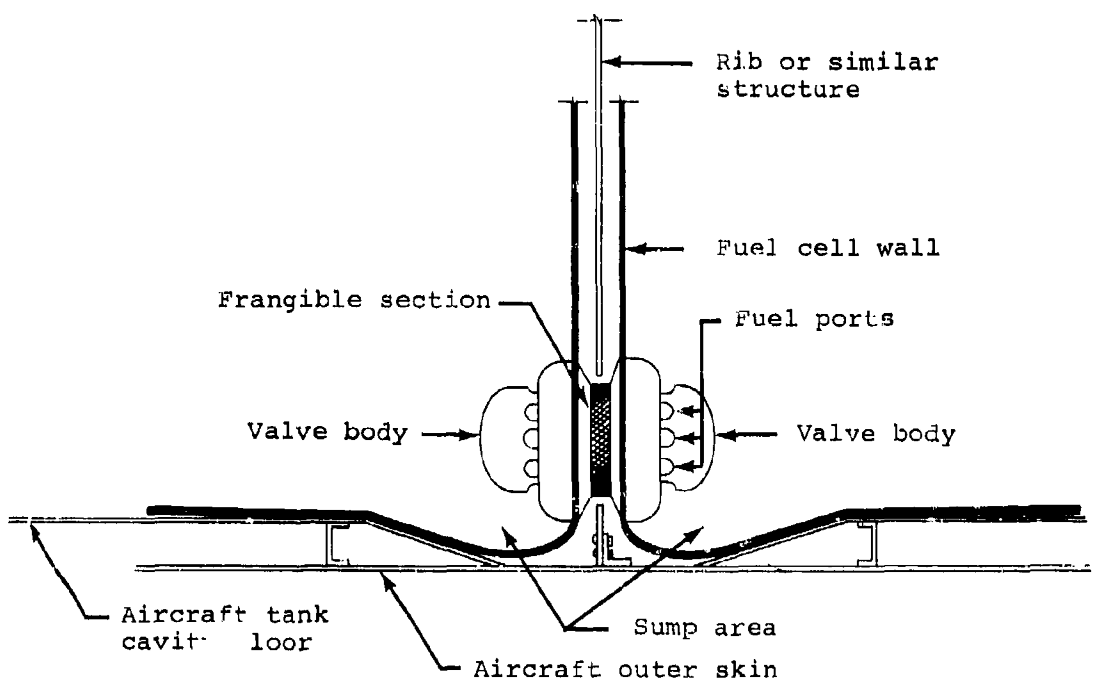
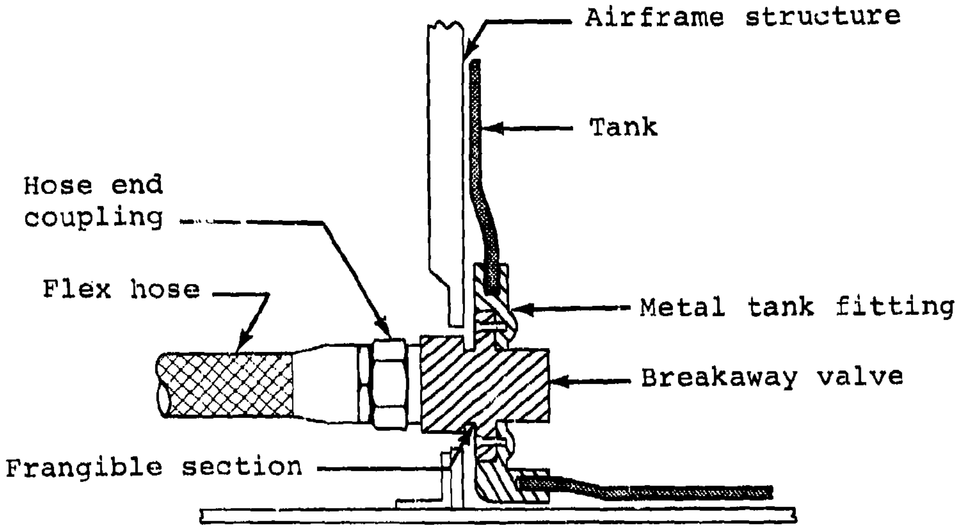


Figure 27. Typical cell-to-cell self-sealing breakaway interconnect valve.

Tank-to-line interconnect valves should be recessed sufficiently into the tank so that the tank half is flush with the tank wall or protrudes only a minimal distance beyond the tank wall after separation. This feature reduces the tendency of the valve to snag on adjacent structure during the crash sequence.

The frangible interconnecting member of each of these valves should be sufficiently strong to meet all operational and service loads of the aircraft within a reasonable margin but should separate at 25 to 50 percent of the minimum failure load for the weakest component in the fluid-carrying line. Figure 28 illustrates a sample breakaway load calculation.



ITEM	LOWEST FAILURE LOAD (LB)*	FAILURE MODE
Flex hose	3000	Tensile breakage
Flex hose	1500	Pull out of end fitting
Tank fitting	7500	Pull out of tank
Hose end coupling	1650	Break (bending)
Breakaway valve	2500	Pull out of tank fitting
Breakaway valve	Not more than $\frac{1500}{2} = 750$ Not less than $\frac{1500}{4} = 375$	Break at frangible section

*Loads may or may not be representative; values are for explanatory purposes only.

Figure 28. Typical method of breakaway load calculation for fuel tank-to-line breakaway valve.

Each valve application should be analyzed to assure that the probable separation load will be exerted in a direction and manner to which the valve is best suited. These loads, whether tension, shear, compression, or combinations thereof, are obtained by analyzing the aircraft for probable impact force and direction and by determining the consequent structural deformation around the valve.

Care should be taken to select valves that allow only a minimum amount of spillage upon separation. The valve should permit no external leakage when partially separated. For this reason, valves with a very short triggering stroke are superior to those with a long stroke.

Operational pressures are dependent on specific applications, but the valve designs can take advantage of the available line pressure to assist in keeping the self-sealing mechanism closed. As in all valve design, light weight and minimal pressure drop are major design objectives, but the resistance of the valve to direct impact or to high compressive loads should not be sacrificed for the sake of weight reduction.

4.2.3.2 Vents: Vent systems become involved in the crash fire episode when the aircraft rolls far enough to one side to allow fuel to drain out of the systems and/or when the vent lines fail.

Vent line failure often occurs at the point of exit from the tank. Failure at this point can be reduced by using short, high-strength fittings between the metal insert in the tank and the vent line. The vent line should be made of wire-covered flexible hose and should be routed in such a manner that it will not obviously become snagged in displacing structure and torn from the tank. Self-sealing breakaway valves also can be placed at the tank-to-line attachment area. This approach becomes mandatory if there is danger of the tank being torn free of the supporting structure.

Vent lines should be routed inside the fuel tank in such a manner that, if rollover occurs, spillage cannot continue. This can be accomplished with siphon breaks and/or U-shaped traps in the line routing.

Recent developments indicate the feasibility of placing vent valves inside the fuel tank. These valves are designed to operate in any attitude and to allow a free flow of air while prohibiting the flow of fuel. They are particularly advantageous during rollover accidents, and can be used in lieu of flexible lines, breakaway valves, and all other alternate considerations. One current type of vent valve is illustrated in Figure 29.

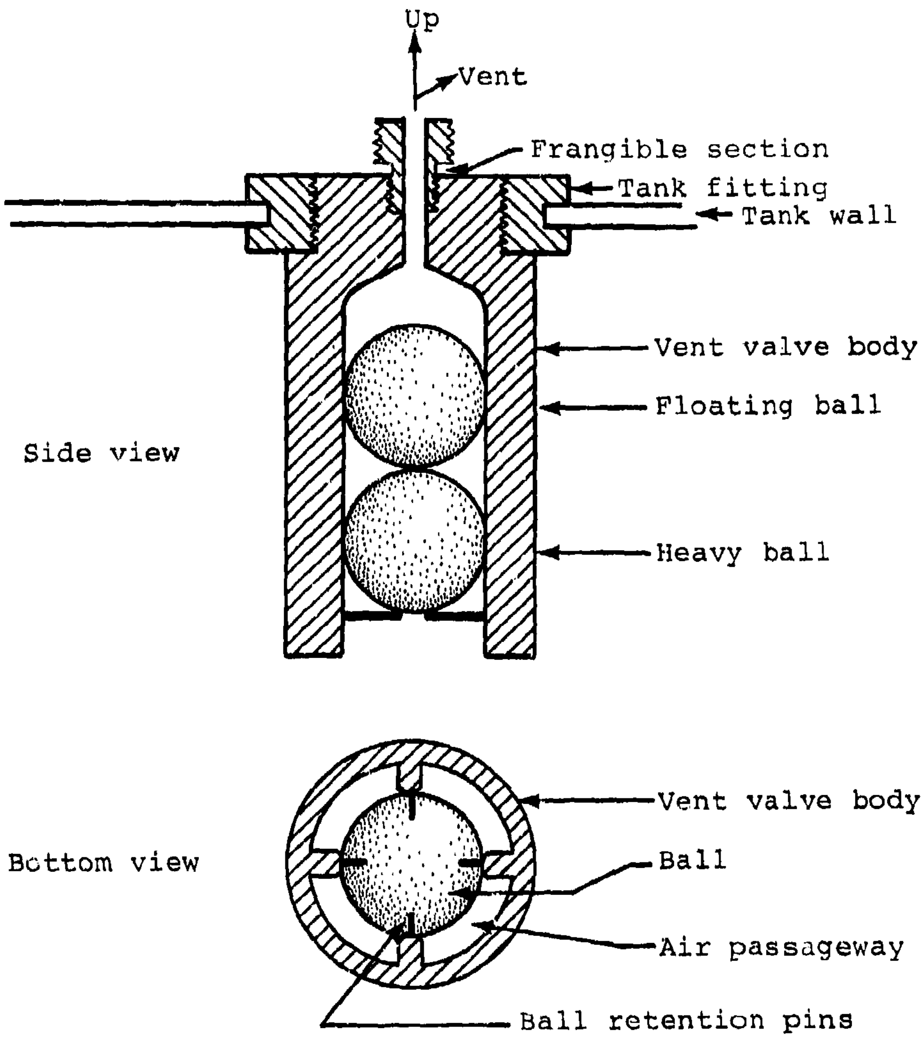


Figure 29. Vent valve.

If the fuel system is to be pressure refueled, it should be noted that a bypass system for tank overpressurization will have to be used. This capability can be built into the vent valve or can be incorporated in a separate unit. In either case, however, care must be taken to ensure that spillage resulting from overpressurization due to tank compression during a crash is released away from aircraft occupants and ignition sources.

4.2.3.3 Boost Pumps: Fuel boost pumps fall into two general categories. There are the tank- or line-mounted types, which pressurize the fuel lines, and the line- or engine-mounted type, which suck fuel from the tank and lines, creating a slight negative pressure in the fuel lines. Suction fuel systems pose a much lower threat in regard to crash fires; however, both systems can pose potential problems. Some boost pumps in use today are installed in the fuel tank and are rigidly bolted to the aircraft structure. Crash damage to the pump can cause fuel spillage and also supply electrical sparks for ignition of fuel.

The state of the art in fuel system design has shown that electrically driven boost pumps can be eliminated. Air-driven boost pumps and engine-mounted suction-type boost pumps now in operation are much less hazardous alternative solutions.

If design requirements dictate that a boost pump be installed in the fuel tank, it is suggested that the pump be air driven and that it be rigidly bolted to the fuel cell only. If the pump must be supported or attached to the aircraft structure, a frangible attachment should be used, as shown in Figure 30.

4.2.3.4 Filler Necks: The filler necks commonly used on present-day aircraft can, and frequently do, cause fuel tank failure. Typical filler neck installations place the cap at one end of the filler tube and the tank at the other end. During periods of structural displacement, the neck can be pulled and torn from the tank, leaving an opening in the tank wall. To prevent fuel spillage, it is imperative that the filler cap remain with the tank. To do so, it must be mounted at, or slightly below, the tank wall surface.

Although the use of filler necks is not recommended, certain aircraft configurations require their use. It is suggested that a frangible type be devised, as shown in Figure 31. Alternatively, a check valve can be placed in the tank filler opening as shown in Figure 32. Another suggestion for filler attachments is the frangible ring concept presented in Figure 33.

4.2.3.5 Quantity Sensors: Accident investigations have shown that quantity sensors cause two types of tank failures. The first type of failure, which is common to most quantity sensor installations, involves the rigid attachment between the sensor entry into the tank and the aircraft structure. This rigid coupling cannot accommodate much structural displacement without inducing a tearing failure in the fuel tank. It is necessary, therefore, that a frangible structure be used for this type of tank attachment (see Figure 34). An alternate approach is to make the probe mounting attachment frangible.

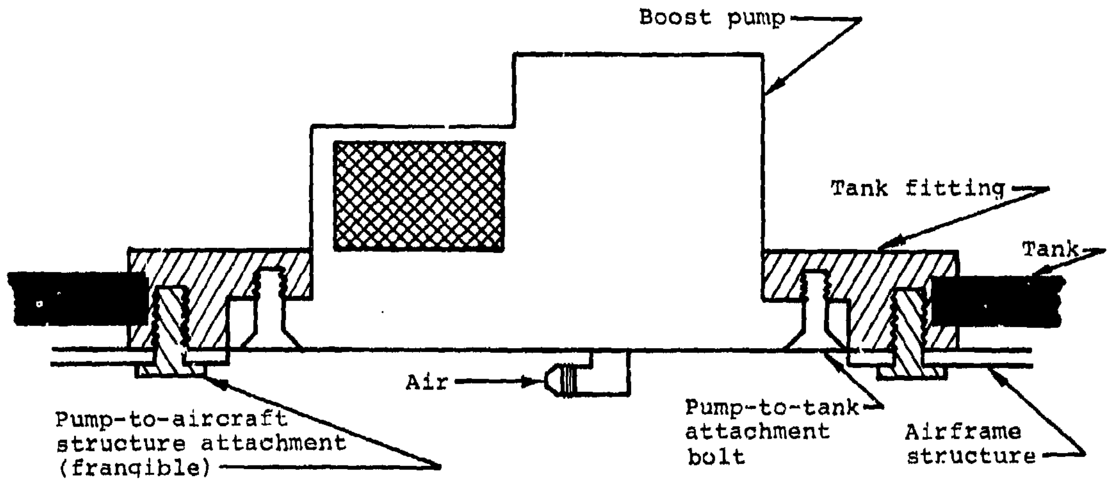


Figure 30. Frangible attachment of air-driven boost pump to airframe structure.

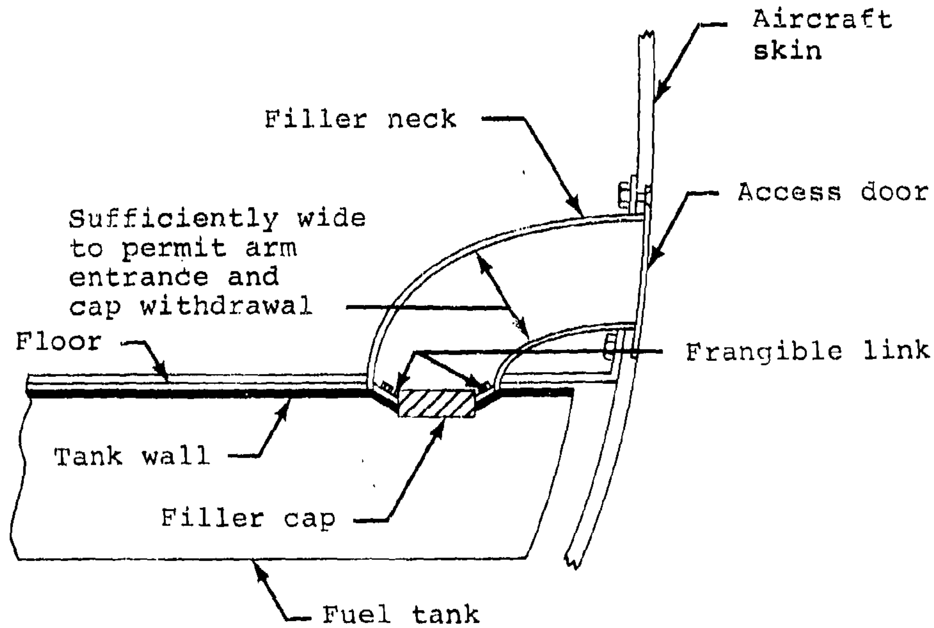


Figure 31. Frangible filler neck installation.

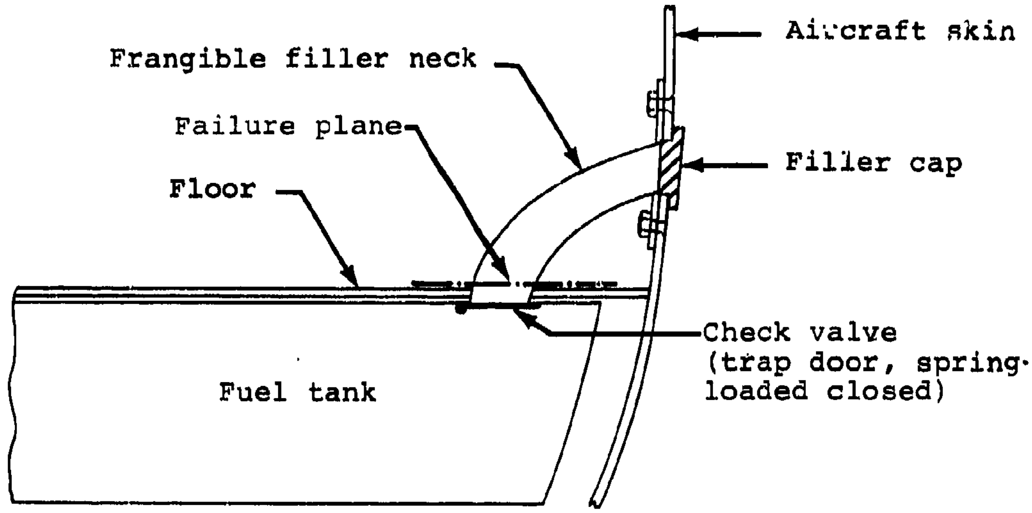


Figure 32. Check valve installed in conjunction with frangible filler neck.

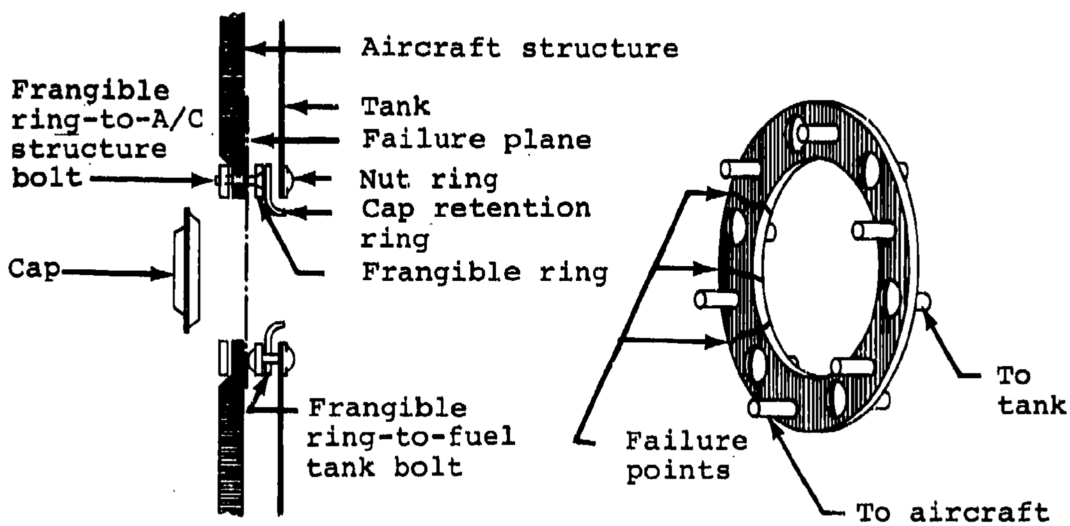


Figure 33. Frangible ring attachment for installing fuel tank fillers.

The second type of sensor-induced tank failure is the puncturing of the tank by the long, rigid, tubular sensing probes in use in many aircraft. Corrective approaches to this problem include mounting the probe at a less hazardous angle or using curved, frangible, low-flexural-rigidity probes, or probes

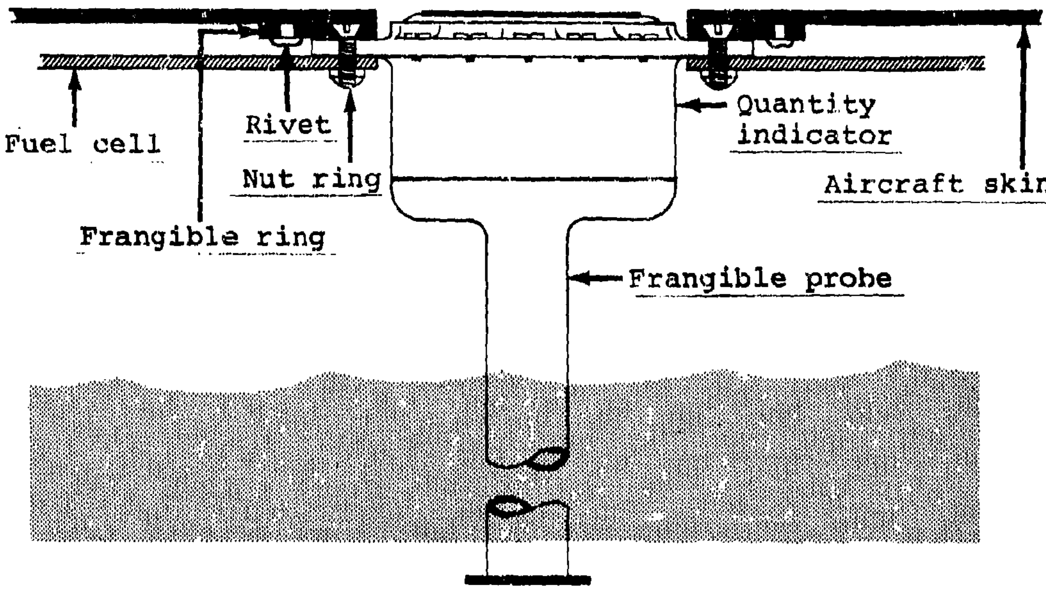


Figure 34. Crashworthy fuel quantity sensor.

equipped with load-spreading shoes, fuel counters, and float-and-arm type sensors. While the new crash-resistant tanks have greatly reduced this problem, it still poses a hazard that should be remedied in the larger cells.

4.2.3.6 Sump Drains: Sump drains are a frequent source of fuel spillage because their design dictates that they be located at the lowest point in the tank, in close proximity to the most probable impact area. Figure 35 illustrates some design concepts that permit maximum drainage without the drain protruding beyond the face of the tank.

4.2.3.7 Fuel Strainers and Filters: In-line fuel strainers should not be located in the engine compartment if such a practice can be avoided. Engines are sometimes torn loose during crash impact, and the strainers located in the compartment are susceptible to damage from the displaced engine. Mounting of the strainers directly on the engine is not desirable. The engine location might afford some protection during a crash, but its proximity to the hot engine surfaces creates an additional hazard from ballistic hits. Strainers should have a structural attachment capable of withstanding a 30 G load applied in any direction to minimize the possibility of their being torn loose during crash impact. Self-sealing breakaway couplings should be used to attach fuel lines to the fuel strainers if there is a probability of line damage at this point.

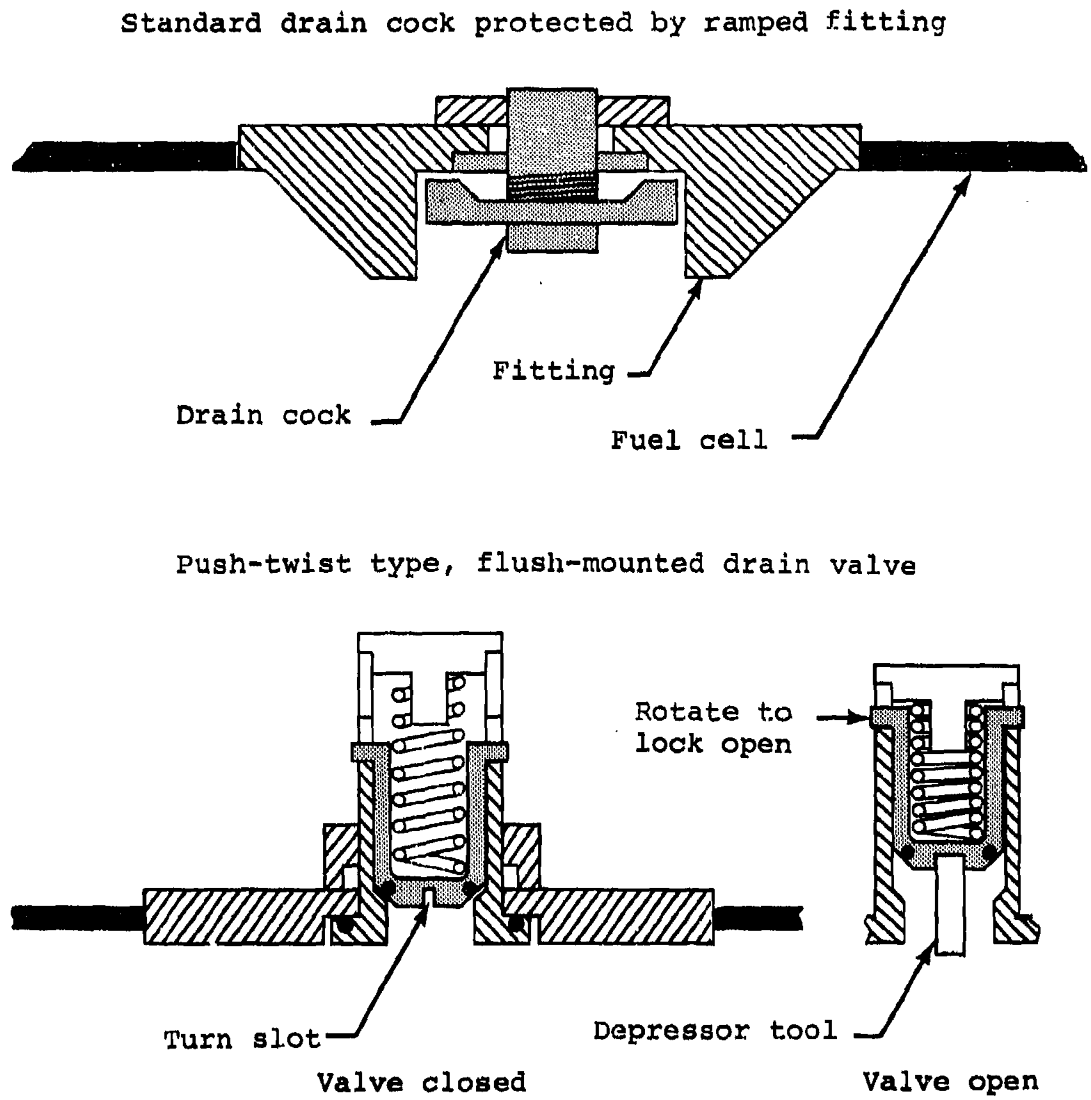


Figure 35. Design concepts for crashworthy fuel drain cocks.

4.2.3.8 <u>Caps and Access Covers</u>: These items play a major role in crashworthy fuel containment. Since they function as seals for tank openings, their failure could be catastrophic. Caps having a minimum rating of 75 psi or greater should be used. Access covers must not be the weak link in the fuel tank. They must be capable of carrying loads equal to or greater than those which the tank can withstand.

4.2.4 Fuel System Full-Scale Crash Test

Consideration should be given to conducting a crash test with the complete crashworthy fuel system in enough of the airframe to create a realistic situation. Since the subject aircraft can crash in a variety of attitudes and speeds, the attitude and impact velocity for the fuel system test should be representative of the attitudes and velocities used in the crashworthy design of the overall aircraft. The recommended design velocity changes are listed in Table 5. The reader is referred to Volume II for a complete discussion of crash design conditions.

TABLE 5. SUMMARY OF DESIGN VELOCITIES FOR ROTARY- AND LIGHT FIXED-WING AIRCRAFT

Impact Direction	Velocity change (ft/sec)
Longitudinal	50
Vertical	42
Lateral*	25
Lateral**	30

*Light fixed-wing.
**Rotary-wing.

4.3 OIL AND HYDRAULIC FLUID CONTAINMENT

Oil and hydraulic fluid spillage often occurs in aircraft accidents. Fortunately, these fluids are carried in much smaller quantities than fuel. However, they are easily ignited; oil is usually carried hot, which makes ignition easier; they are pressurized in places, which converts them into mists when they are released, making ignition easier; and they are often carried near the hot engine, which can readily provide ignition. When oil or hydraulic fluids are ignited, they, by themselves, constitute a low threat to aircraft occupants. But, unfortunately, they function as ignition sources for other combustibles, especially spilled fuel. Further, they migrate throughout the wreckage, carrying with them flames that otherwise would not be present. Oil and hydraulic fluid spillage, therefore, should be prevented at all reasonable cost.

The crashworthy design criteria presented for the fuel tanks, lines, and supportive components all apply to these fluid systems, with one possible exception. Because of their relatively small capacities, properly protected metal tanks may be used. It should be recognized, however, that metal tanks are punctured easily and are not tear resistant. If tank puncture is likely, several alternatives are available: a crashworthy tank, like the fuel tank, can be used; the tank can be relocated to a safer area; or the tank can be shielded.

Experiments have been performed to determine the practicality of shielding a metal oil tank with a 1/2-inch-thick felt cover made of ballistic nylon, as shown in Figure 36 (Reference 18). As an added degree of spillage protection, the outside surface of the felt was coated with a thin layer of polyurethane resin to make it leakproof. Since preliminary experiments proved satisfactory, a similar system was crash-tested in a U. S. Army UH-1 helicopter. The tank sustained severe impact damage, rupturing a tank seam. The spillage leaked out into the felt cover, but did not escape from the felt due to the polyurethane coating. This system is simple, light in weight, easy to install, and relatively low in cost. A similar felt tank cover is now being used with a high degree of success on all Indianapolis-type racing cars.

When metal lines must be used in these systems, they should be designed to incorporate a coil or two of extra line length so that the line can stretch to accommodate some structural distortion. Also, the lines should be attached to the airframe with clamps that will fail and release the fluid lines before the line itself fails, thereby allowing the line to change its routing to help accommodate structural distortion.

Hydraulic fluids that inherently resist burning should be used whenever possible (Reference 19). Most of those fluids, however, have operational and maintenance problems associated with their use. Therefore, designers may wish to consider the trade-off of using conventional hydraulic fluids, as compared with using fire-resistant fluids. It should be noted that, even though the new fluids are fire resistant, most of them

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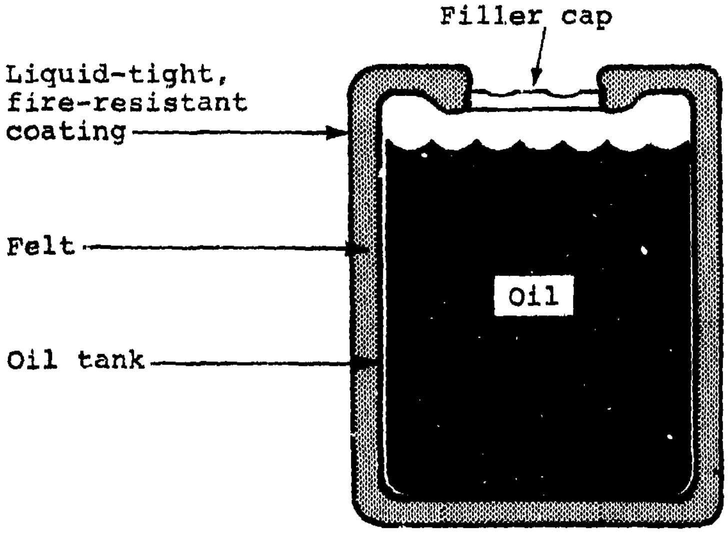
18. Robertson, S. H., DEVELOPMENT OF A CRASH-RESISTANT FLAMMABLE FLUIDS SYSTEM FOR THE UH-1A HELICOPTER, Dynamic Science; USAAVLABS Technical Report 68-82, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, January 1969, AD 688165.

19. Military Specification, MIL-H-83282, HYDRAULIC FLUID, FIRE RESISTANT SYNTHETIC HYDROCARBON BASE, AIRCRAFT, Department of Defense, Washington, D. C., 22 February 1974.



NOTE: The felt is not bonded to the oil tank.

Figure 36. Felt oil tank cover.

will still burn, especially when in a mist state. The characteristics of each fluid must be studied before the final tradeoff decision is made.

4.4 FUEL MODIFICATION

One method for decreasing the postcrash fire potential is to decrease the susceptibility of aircraft fuels to dispersion and atomization, reducing the formation of combustible fuel/air mixtures. This can be done through the use of fuel modification additives. These modifying agents have been classified as either antimist, emulsification, or gelling additives. The blending of these additives into standard aviation fuels provides fuel properties that decrease the tendency to disperse, atomize, and form fuel mists following crash-induced fuel system failures. As a result, retardation of fuel mist fireballing and fire propagation can be achieved.

Several studies have shown the feasibility of providing post-crash fire protection through the use of antimist fuels, emulsified fuels, and gelled fuels (References 20, 21, 22, and 23).

This approach has been fairly successful when used with low volatility fuels such as JP-5, JP-8, and Jet A. However, modification of highly volatile fuels, such as JP-4 and aviation gasoline, has not been effective. Emulsified and gelled fuels have received little attention recently due to their inherent system compatibility problems. Further, consideration of modified fuels has declined since the development and use of crashworthy fuel systems in rotary-wing aircraft. However, the possible use of antimist fuels in fixed-wing aircraft where crashworthy fuel tanks are less feasible has generated recent interest.

Although turbine engine performance is not adversely affected by use of antimist fuel blends, it has been found that these fuels must be degraded before starting and restarting a turbine engine with a standard fuel system. During startup, the characteristics of the antimist fuel suppress the atomization of the fuel through the fuel nozzle, thus starving the initial ignition. To alleviate these problems, processes to reverse the antimist fuel blend so that the blended fuel can be brought back to the neat state prior to introduction into the aircraft fuel feed system are being investigated.

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20. Weatherford, W. D., Jr., and Wright, B. R., STATUS OF RESEARCH OF ANTIMIST AIRCRAFT TURBINE ENGINE FUELS IN THE UNITED STATES, in Aircraft Fire Safety, AGARD Conference Proceedings No. 166, North Atlantic Treaty Organization, Advisory Group for Aerospace Research and Development, Neuilly-sur-Seine, France, October 1975, pp. 2.1-2.12, AD A018180.

21. San Miguel, A., ANTIMISTING FUEL KINEMATICS RELATED TO AIRCRAFT CRASH LANDINGS, Journal of Aircraft, Vol. 15, No. 3, March 1978, pp. 137-142.

22. Shaw, L. M., SAFETY EVALUATION OF EMULSIFIED FUELS, Dynamic Science; USAAMRDL Technical Report 71-29, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, June 1971, AD 729330.

23. Shaw, L. M., SAFETY EVALUATION OF ANTIMIST FUELS, Dynamic Science; Report 9130-73-112, U. S. Army Mobility Equipment Research and Development Center, Aberdeen Proving Ground, Maryland, November 1973, AD 773035.

4.5 IGNITION SOURCE CONTROL

Flammable fluids will ignite throughout a wide range of temperature, pressure, atmospheric composition, and ignition source conditions. Generally, ignition of spilled combustibles during the crash occurs from one or more of the following: electrical sources, flames, hot surfaces, and friction sparks. Components usually involved in the ignition process include the engine, exhaust system, heater, battery, wiring system, and various light bulbs.

4.5.1 Electrical Sources

The aircraft electrical system is a potential crash-fire ignition source because it is distributed extensively throughout the aircraft and because electrical discharges are able to concentrate a high amount of energy into a small volume.

Disruption of a current-carrying electrical circuit can result in fuel ignition by electrical sparks and arcs that are released when exposed wires contact grounded surfaces. Ignition also can be provided by wires that have been heated either by short circuiting or by normal means, as in an incandescent light filament. The common incandescent filament in a landing light is hot enough to ignite fuel 0.75 to 1.50 sec after bulb breakage.

Perhaps the most important aspect of an electrical discharge ignition source is the great amount of energy present compared with the small amount actually required to produce fire ignition under ideal conditions. Approximately 0.15 millijoule (0.11×10^{-3} ft-lb) is the minimum energy for spark ignition under ideal temperature, pressure, and mixture conditions.

The ignition potential of the aircraft's electrical system may be reduced by aircraft modification at the system level and at the component level. The system level approach is concerned with de-energizing electrical generation or storage systems, whereas the component level approach is concerned with component location and environment.

4.5.1.1 System Level Approach: Reduction of crash-fire ignition by the electrical system can be achieved by removing from the electrical circuit all electrical generation or storage systems before or during the early phases of the crash sequence. The de-energizing can be accomplished by opening the electrical circuit at the output terminals of each energy-producing component.

The time required for this de-energizing operation is of utmost importance. Crash-fire data previously reported by NACA, using both aviation-grade gasoline and low-volatility fuel, indicate a minimum time of 0.7 sec between impact and fire ignition with the electrical system as the source. During helicopter crash tests by AvSER, it was observed that fire started to propagate approximately 0.58 sec after ground impact. During tests with simulated fuels, massive fuel spillage was in progress as early as 0.20 sec after impact. Therefore, each de-energizing device must be capable of activation within a maximum time of 0.20 sec.

The primary items to be considered for de-energizing are the batteries, generators, and inverters. Several precautions must be taken. Since the battery can remain a potential ignition source for hours after a crash, ends of wires severed from batteries must be prevented from contacting the structure and thereby providing a new ignition source. The generators and inverters cannot be satisfactorily de-energized by simply opening field circuits. There is a considerable time lag (0.385 sec for a rotating inverter) between DC input cutoff and AC output termination (Reference 24). Therefore, for complete safety, these components must be disconnected from buses on their output sides. NACA also recommended that consideration be given to grounding the armatures of main electrical components close to those components (Reference 25).

Magnetos and igniters are of special interest, since they are high-energy sources of ignition. If these components were de-energized, the fuel in the engine during the crash event would not be ignited. However, raw fuel then would be pumped into the hot exhaust manifold, resulting in a fire. Recent crash-fire research has demonstrated that it is better to turn the fuel off and to leave the ignition system on throughout the crash sequence (Reference 24).

Relays can be used to de-energize components and to activate other inerting elements. In the case of batteries, only non-essential buses should be disconnected initially. Power must

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24. Robertson, S. H., et al., THEORY, DEVELOPMENT, AND TEST OF A CRASH-FIRE INERTING SYSTEM FOR RECIPROCATING ENGINE HELICOPTERS, Aviation Safety Engineering and Research of Flight Safety Foundation; TRECOM Technical Report 63-49, U. S. Army Transportation Research Command, Fort Eustis, Virginia, December 1963.

25. Pinkel, I. I., et al., ORIGIN AND PREVENTION OF CRASH FIRES IN TURBOJET AIRCRAFT, NACA Report 1019, Lewis Flight Propulsion Laboratory, National Advisory Committee for Aeronautics, Cleveland, Ohio, 1958.

be provided to other elements of the crash-fire prevention system until these elements have completed their design functions. A time-delay unit can be used to cut off power to inerting elements and to ground the disconnected buses. An alternative to a relay contact is the explosive cable cutter shown in Figure 37. The electrical system inerters must, in any case, be capable of resetting components in the event of inadvertent operation.

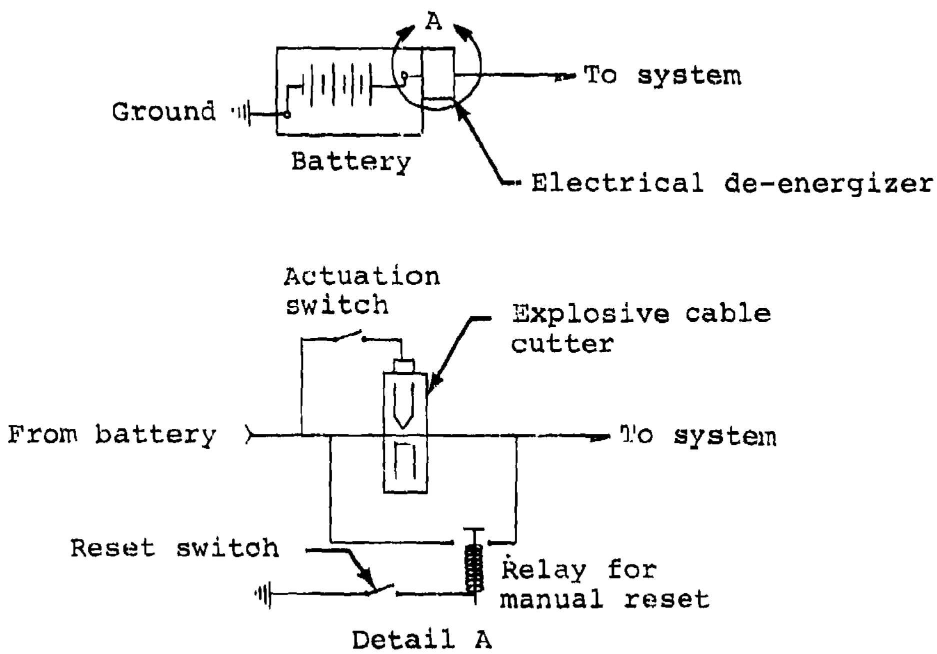

Figure 37. Cable cutter de-energizing method.

4.5.1.2 <u>Component Level Approach</u>: The ignition hazard associated with the electrical system can be reduced at the component level by controlling component location and environment. The following guidelines are applicable to batteries, inverters, generators, alternators, magnetos, igniters, radar, antennas, and lights.

Components should be located above and away from flammable fluid sources. Leaking flammable fluid should not come in contact with electrical equipment or wiring as a result of gravity, airflow, or battle damage. The electrical system components should be located, and suitably mounted, in areas where anticipated impacts will be minimal and where maximum anticipated structural deformation will not result in structural impingement on either components or wiring. Wires should have 6-in.-diameter loops near their component connections to accommodate any wire tensioning resulting from structural deformation. All wire connections should be made on a component's least vulnerable side. Batteries, inverters, and generators should be mounted in compartments lined with tough, nonconductive shields. The shields will prevent sparking between terminals or severed wires and the aircraft structure. The components should be mounted to the aircraft with structural attachments capable of withstanding 30 G loads in any direction.

Electrical wires should be routed along the strongest structural members and should not, in general, traverse areas of anticipated severe structural deformation, e.g., in leading edges of wings or in the lower regions of the fuselage. Wires that must pass through areas of anticipated structural deformation should be approximately 20 to 30 percent longer than necessary. The extra length should be accumulated in the form of loops or S-shaped patterns and located at the areas of anticipated structural deformation. When wires pass through structural openings or bulkhead holes, the openings should be 8 to 12 times larger than the wire diameter and appropriate grommets should be provided. The wires should be attached to the aircraft structure with clamps or ties that will fail before breaking the wire. Nonconductive shields should surround all areas where wire abrading or cutting may occur. Wires should not be routed near flammable fluid sources.

The mounts for antennas and lights should be attached to the aircraft with frangible structures. The wires should incorporate a shielded covering and/or a breakaway capability. A suggested installation technique for a rotating beacon is illustrated in Figure 38. Structural impingement upon the component will be difficult because the frangible mounting structure will allow the beacon to displace. The extra wire contained in the loop can allow for considerable beacon movement without failing; if massive displacement is anticipated, shielded failure points can be used. These same techniques apply for all similar types of components.

4.5.2 Engine

The two principal engine ignition sources are (1) intake, combustor, and exhaust flames, and (2) hot metal surfaces. The differences between these two relate to the time that these sources persist after a crash, the manner in which ignition occurs, and the mode of propagation of the resulting fire out of the engine.

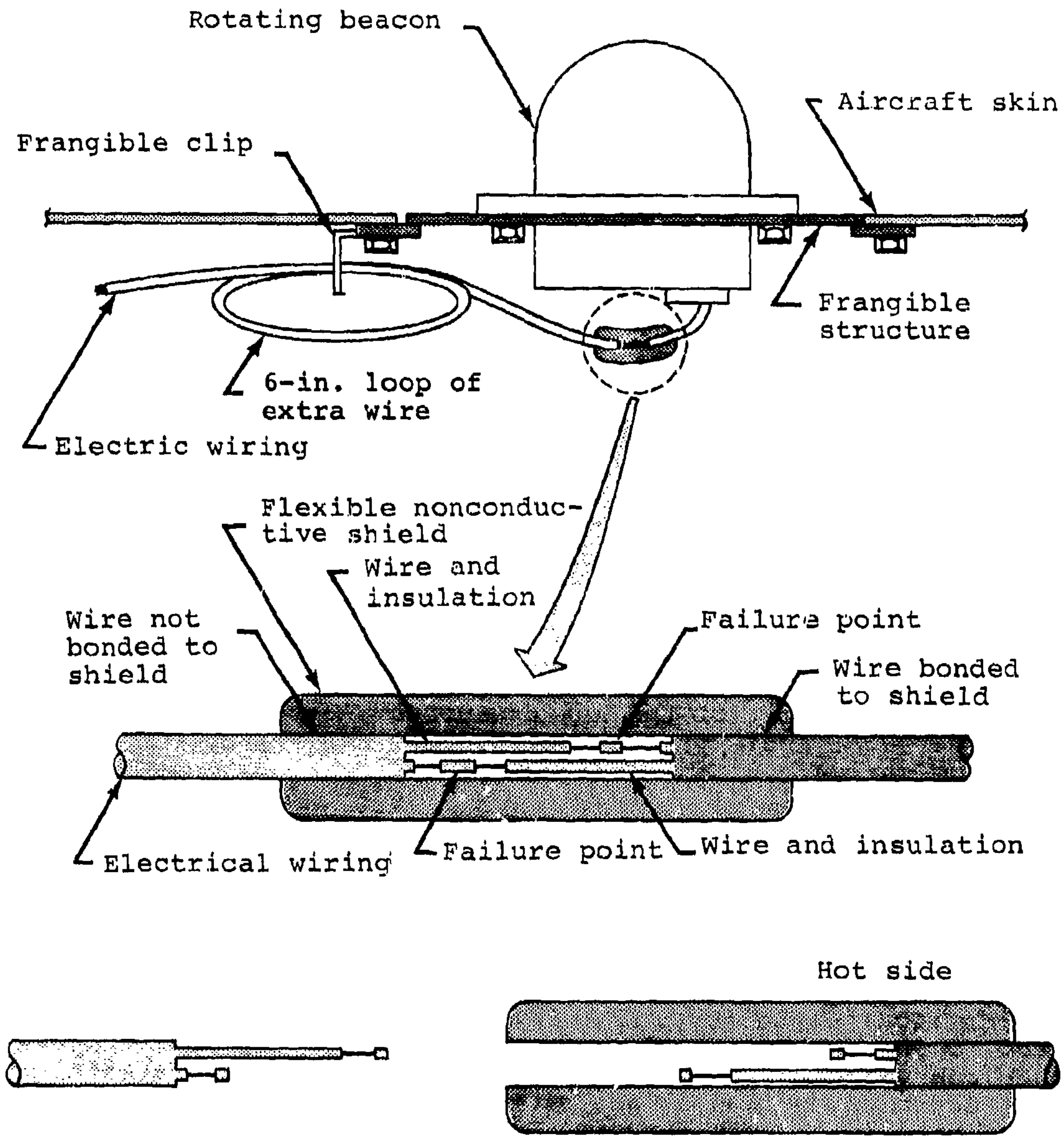

Figure 38. Rotating beacon installation.

4.5.2.1 <u>Flames</u>: Engine inlet and exhaust flames are responsible for the ignition of many crash fires. During the crash sequence, flames often appear at these locations due to engine breakup or rapid changes in engine loading, as can occur when a drive shaft is severed or a propeller is sheared.

Flames also appear at these locations when engines ingest spilled fuel. Turbine engines are highly susceptible to fuel ingestion because of the relatively long period of time required for the turbine to coast to a stop. The ingested fuel-air mixture enters the downstream end of the combustor, where flames may persist for up to 18 sec after fuel cutoff. The ingested mixture then burns in the tailpipe downstream of the turbine (Reference 26). Also, under the proper conditions, the combustor flame may propagate upstream through the ingested mixture and exit at the engine inlet.

The occurrence of engine inlet and exhaust flames and the resulting ignition hazard can be reduced by stopping the fuel flow, by inerting the flame source, and by providing shielding to prevent fuel spillage from entering anticipated flame areas. The engine fuel valves should be closed, but the ignition system should be left on to permit normal burning of ingested fuel in order to prevent undesirable exhaust or inlet flames.

4.5.2.2 Hot Surfaces: The probability of flammable fluid ignition due to contact with a heated surface during a crash is high and can remain so for several minutes after a crash. The circumstances leading to ignition are somewhat involved; generally, they are dependent upon the type of flammable fluid involved, temperature of the fluid, composition of the heated surface, temperature of the heated surface, geometry of the heated surface, ratio of the fuel to air, and the degree of fuel atomization.

Ignition temperatures vary widely. As a general rule, hydraulic and lubricating oils ignite at lower flat-plate temperatures than aviation gasoline. JP-4 also has a lower flat-plate ignition temperature than aviation gasoline. The lower grades of gasoline have lower ignition temperatures than the higher grades. Kerosene has a lower ignition temperature than JP-4.

The ignition temperature of a flammable fluid is directly related to the initial fluid temperature. While the fuel temperature can vary considerably, depending on temperature at altitude, on the ramp, and at the storage facility, the temperature of the oils is of more concern. As mentioned, oils can ignite at a temperature lower than most fuels, and since they are carried in the heated state, low hot-surface temperatures and exposure times will provide ignition. Oil fires can, in turn, act as ignition sources for the fuel.

26. Black, D. O., CRASH-FIRE PROTECTION SYSTEM FOR A J57 TURBOJET ENGINE USING WATER AS A COOLING AND INERTING AGENT, NASA Technical Note D-274, National Aeronautics and Space Administration, Washington, D. C., February 1969.

The time between fuel contact with the heated surface and fuel ignition is directly related to the temperature of the heated surface. As shown in Figure 39, the hotter the surface, the faster ignition can occur. Also, it can be seen that ignition can occur at a much lower surface temperature if the exposure, or residence time, is longer.

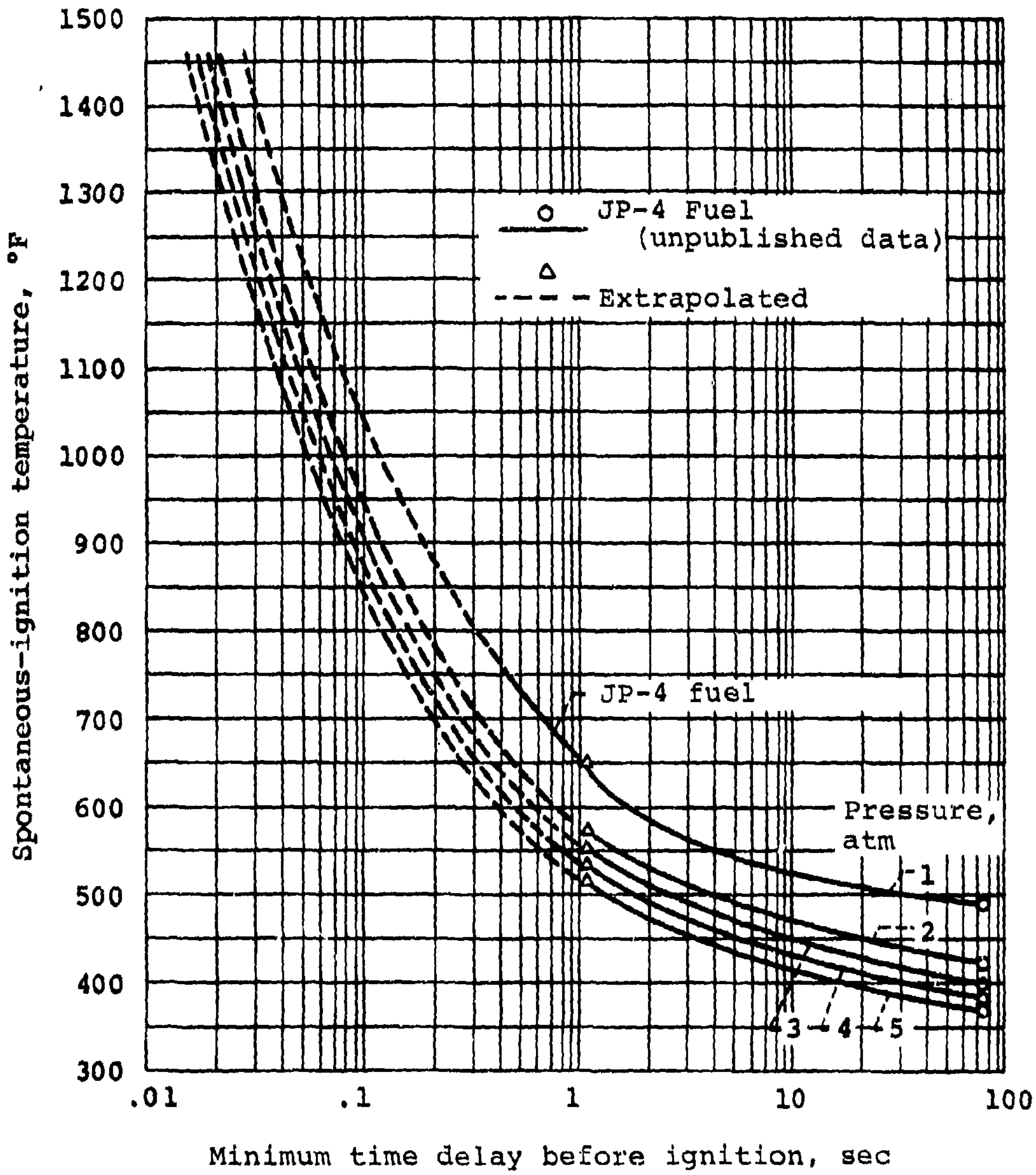

Figure 39. JP-4 ignition delay versus surface temperature. (Taken from Reference 25)

The ratio of fuel to air also governs the probability of ignition. The designer must assume that the proper ratio does exist somewhere within the spillage area.

The potential hot surface ignition sources on aircraft with reciprocating engines are the intake systems, the exhaust gas disposal systems, heaters, and the higher temperature regions of the cylinders. Ignition sources on turbojet engines include the internal areas downstream of the compressor, and external areas from the compressor aft, including the tailcone and tailpipe, and, in some designs, the bleed air system. The gas flow through a turbojet engine may be too rapid to permit the ignition of ingested combustibles on hot metal in contact with the main gas stream. However, a portion of the engine airflow is diverted to hot surfaces not in the main gas stream where ignition may occur.

The hot surface ignition hazard can be reduced by methods analogous to those used with the inlet and exhaust flame hazards. An inerting system can be used to reduce the temperatures of hot surfaces to predetermined acceptable levels and to surround the hot surfaces with an inert atmosphere to prevent ignition from occurring should flammable fluids be spilled on these surfaces. In previous studies, hot surfaces have been cooled to temperatures ranging from 400°F to 760°F with satisfactory results (References 24 and 27). The temperatures to which hot surfaces must be cooled to prevent ignition must be determined as a function of the fuel and the engine configuration to be used; however, 400°F should be used as the upper limit for safe hot surface temperatures.

Shielding also can be used to prevent spilled flammable fluids from reaching the hot surfaces.

4.5.2.3 Inerting Systems: The function of inerting systems is to render ignition sources harmless and, therefore, to prevent fire or explosion. Inerting systems can be designed to surround hot surfaces with an inert atmosphere. With an inerting system, there is not sufficient oxygen to support combustion when flammable fluids contact the hot surfaces. These systems also can be designed to perform the additional function of cooling the hot surfaces to temperatures below the ignition temperatures of flammable fluids. Knowledge of temperature gradients and cooling rate characteristics for each particular engine design is required in order to design an adequate cooling and inerting system.

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27. Pinkel, I. I., et al., MECHANISM OF START AND DEVELOPMENT OF AIRCRAFT CRASH FIRES, NACA Technical Note 2996, Lewis Flight Propulsion Laboratory, National Advisory Committee for Aeronautics, Cleveland, Ohio, 1953.

The system must be capable of providing a high-discharge-rate liquid spray for rapid cooling, a lower-discharge-rate liquid spray over the more massive engine surfaces for a longer period of cooling and inerting, and a follow-up inerting spray. Water is a preferred coolant because of its high latent heat of vaporization, its availability, and its low cost. Additives to the water can be used to protect against freezing and corrosion of the piping system. Nitrogen and carbon dioxide are among other inerting agents that have been used successfully as follow-up sprays.

Several successful hot surface and/or flame inerting systems have been designed, tested, and incorporated into current military aircraft. A schematic diagram of a hot-surface inerting system used on a reciprocating engine is shown in Figure 40 (Reference 24). Systems such as this have been installed on aircraft and crash tested. They successfully inerted the engine and exhaust systems, thus preventing crash fires.

Testing of a pyrotechnic gas-generator-type extinguisher system indicates that it offers performance improvements over the pressurized nitrogen-type system (Reference 28). The pyrotechnic system was more effective at low temperatures and with less volatile extinguishing agents, and it eliminated problems associated with the mixing of the nitrogen and the liquid agent.

Flame-source inerting systems are designed to extinguish combustor flames, which linger long after the fuel has been cut off. The flames are a result of the ignition of fuel that remains in the fuel manifold and continues to drip into the combustor. A schematic diagram of a flame-source inerting system used on a reciprocating engine is shown in Figure 41. Upon actuation by either a manual or a crash-actuated switch, high-pressure CO_2, or a comparable inert gas, is released into the engine air intake. Concurrently, the high-pressure gas is used to activate linkages that close the fuel inlet valve, the oil inlet valve, and the air intake opening. The large volume of inert gas released into the engine interior quickly dilutes the incoming air to the point of nonflammability, thereby eliminating the flames. As the engine continues to coast to a stop, the inert mixture is pumped through the engine and expelled at the exhaust outlet.

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28. Klueg, E. P., et al., AN INVESTIGATION OF IN-FLIGHT FIRE PROTECTION WITH A TURBOFAN POWERPLANT INSTALLATION, Report No. NA-69-26, Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, April 1969, AD 686045.

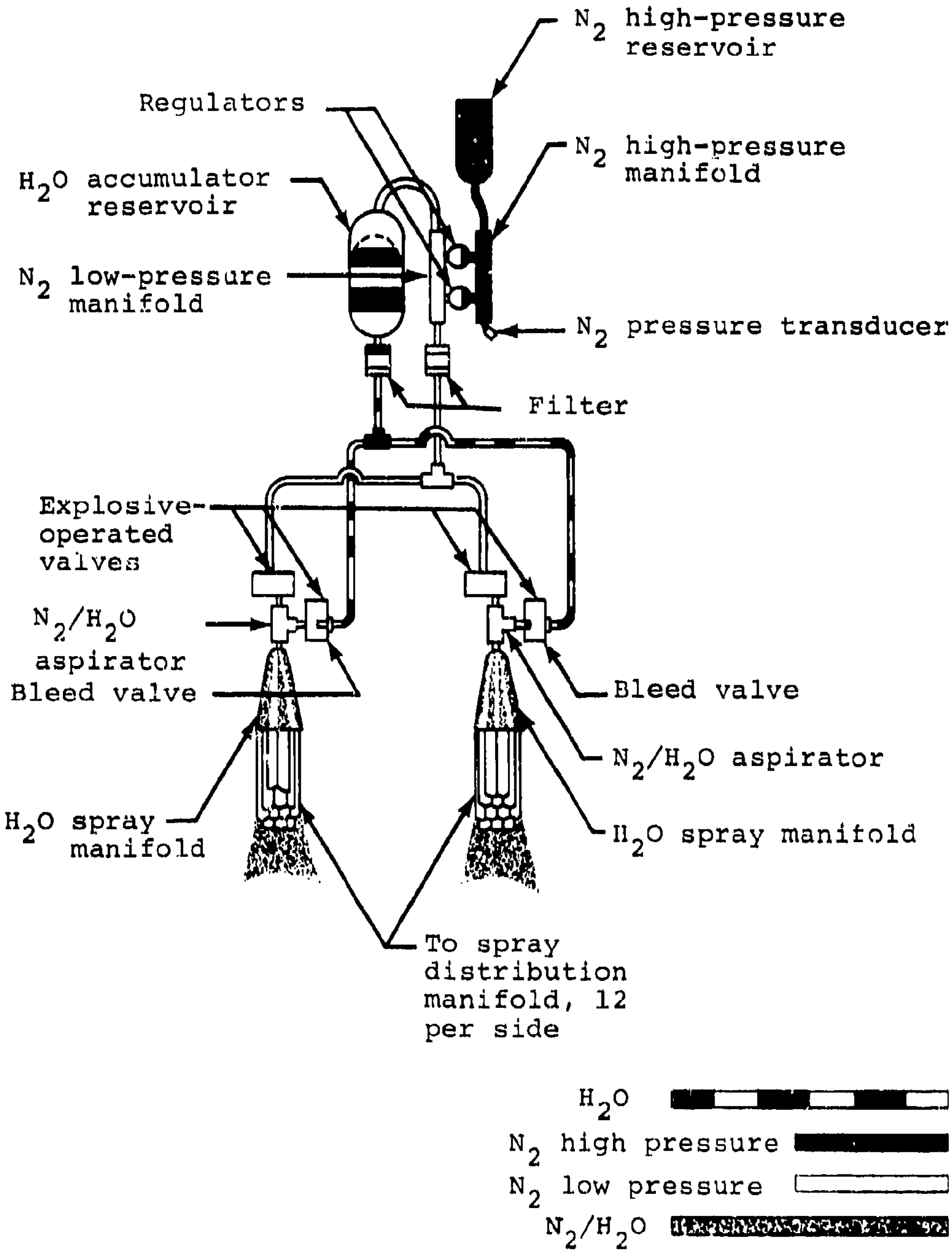

Figure 40. Hot surface inerting system.

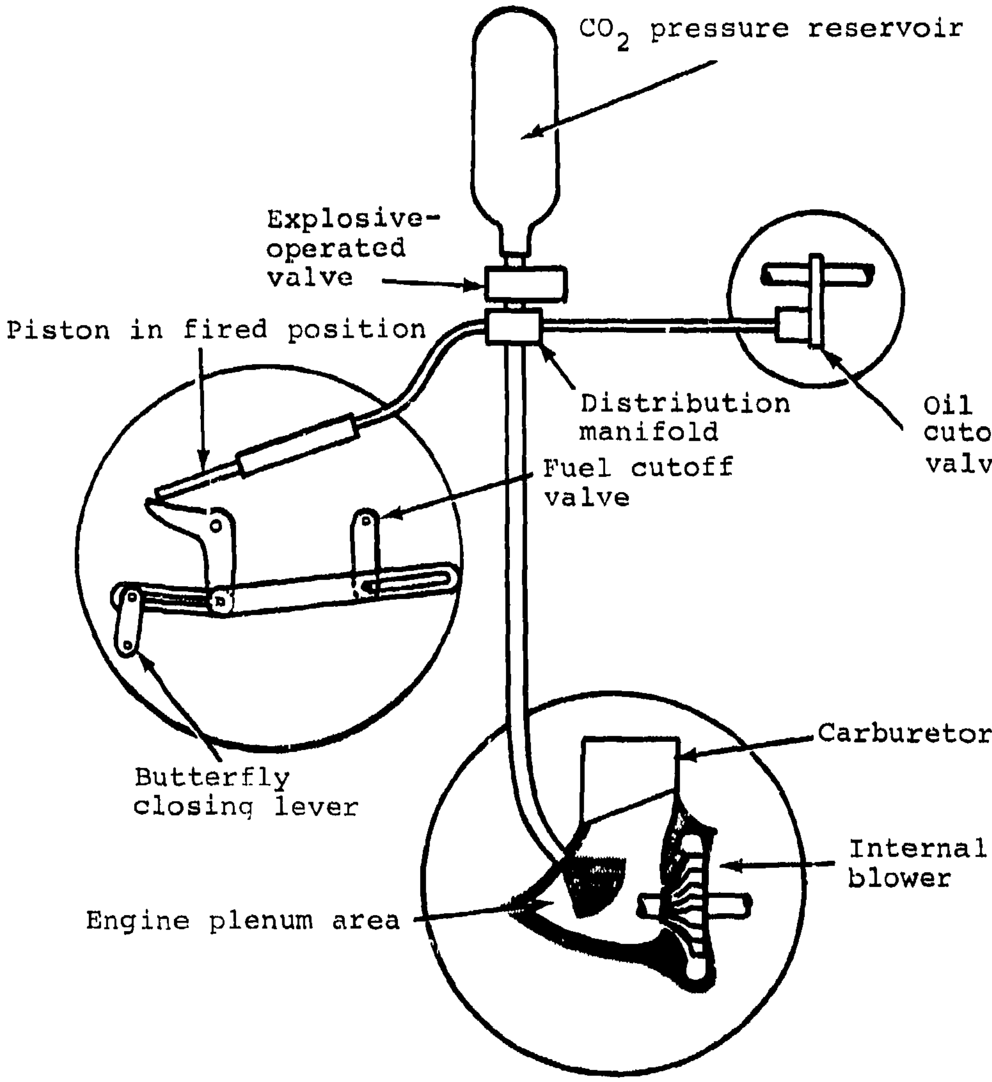

Figure 41. Flame-source inerting system.

In a turbojet application, the combustor flames were eliminated by providing for rapid fuel shutoff and draining of the fuel manifold. The system is shown in Figure 42 (Reference 26). The fuel was shut off by a pneumatically operated valve installed in the fuel line between the engine fuel-control unit and a modified pressurizing and dump valve. Simultaneously, the manifold drain valves and the modified pressurizing and dump valve opened and vented the fuel manifold overboard. The

Modifications

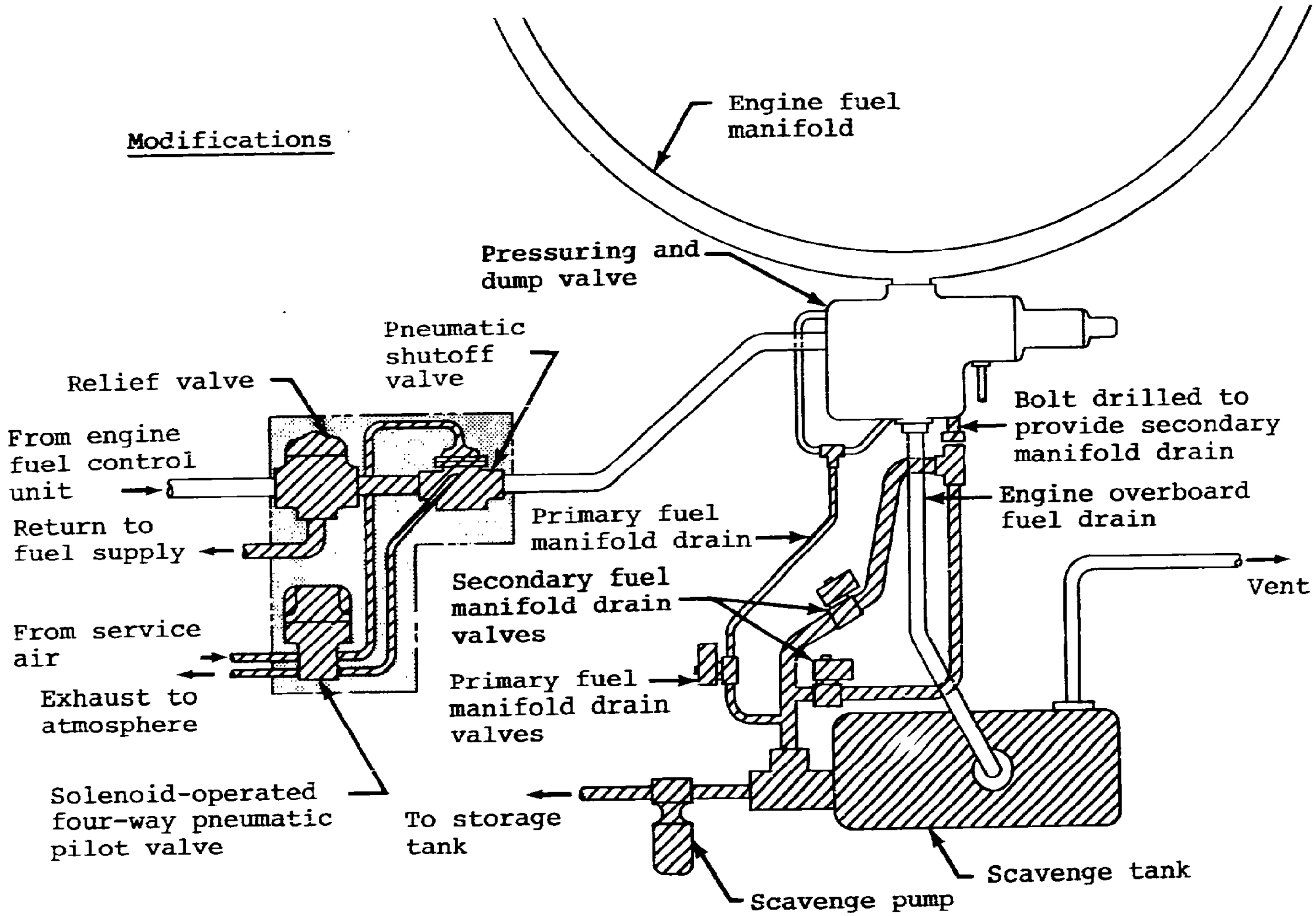

Figure 42. Schematic diagram showing fuel shutoff and drain system in J57 engine crash-fire protection system.

combustor air pressure, available at the instant the fuel shut-off valve closed, then reversed the fuel flow in the nozzles and manifold through the overboard fuel drains. The combustor flame was extinguished in 0.23 sec by this method.

As is the case with electrical system de-energizing devices, the inerting systems should be operable within 0.20 sec of the sensing of a crash.

4.5.2.4 <u>Shielding</u>: Shielding is an effective method of preventing flammable fluids from reaching potential ignition sources. Shielding can take many forms; however, there are three general methods in use, with a fourth now in the development stage.

The first method of shielding uses baffles. Metal or other rigid paneling will not satisfy the shielding requirement because of its inability to maintain an effective seal in areas of large structural displacement. Sealed curtains or baffles made of fire-resistant cloth or similar material can perform satisfactorily. The only requirement is that they must seal all openings through which flammable fluids could travel to an ignition source. To accommodate the anticipated structural displacement, it is suggested that all curtains and shields be at least 30 to 40 percent larger than the minimum size required to protect a given area. Figure 43 illustrates how the flexible curtain concept was used to keep fuel from entering the occupiable areas on an experimental test helicopter.

The second method of shielding uses spillage flow diverters or drip fences. Once liquid has settled onto a sloping surface, it flows to the lowest point. It can flow on top of a surface, or it can cling to the underside. In either case, it can travel a considerable distance to an ignition source. Chordwise drip fences should be located on the wing on each side of wing-mounted engines. Drainage holes should be strategically located within the aircraft structure to drain internal spillage. All areas containing electrical components should be surrounded with a spillage gutter, or drainage trough, to prevent flowing spillage from entering those areas.

Each engine and exhaust system mount should incorporate a drip fence. Figures 44, 45, and 46 illustrate several types of drip fence fuel-flow diverters.

The third method of shielding uses nonconductive flexible paneling. This type of paneling should be used as a liner for electrical compartments and other regions where electrical components are installed. It should surround areas of electrical wire groupings such as terminal strips and power control areas. Nonconductive flexible shielding also can be used for shrouding

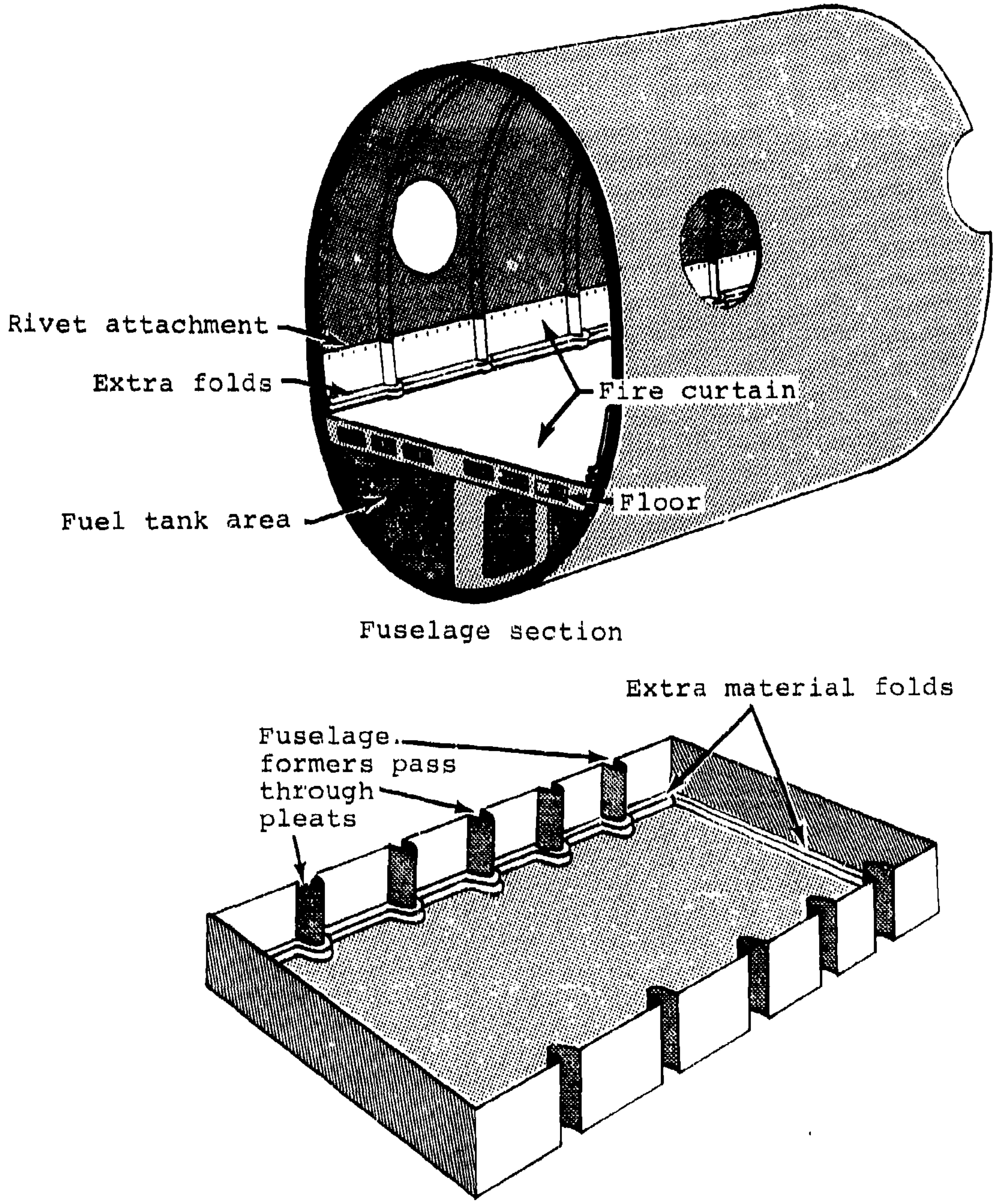

Figure 43. Fuel baffle and fire curtain concept.

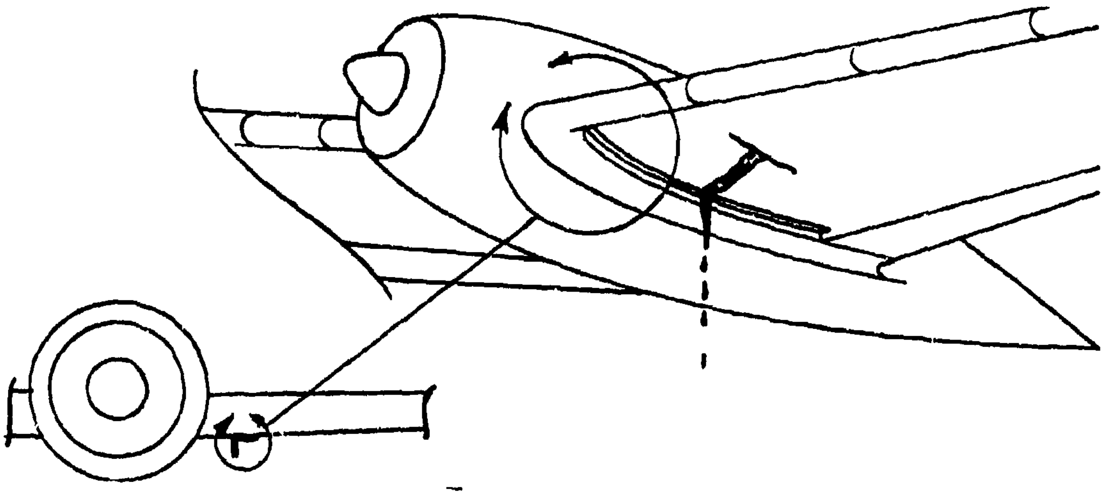

Figure 44. External drip fence.

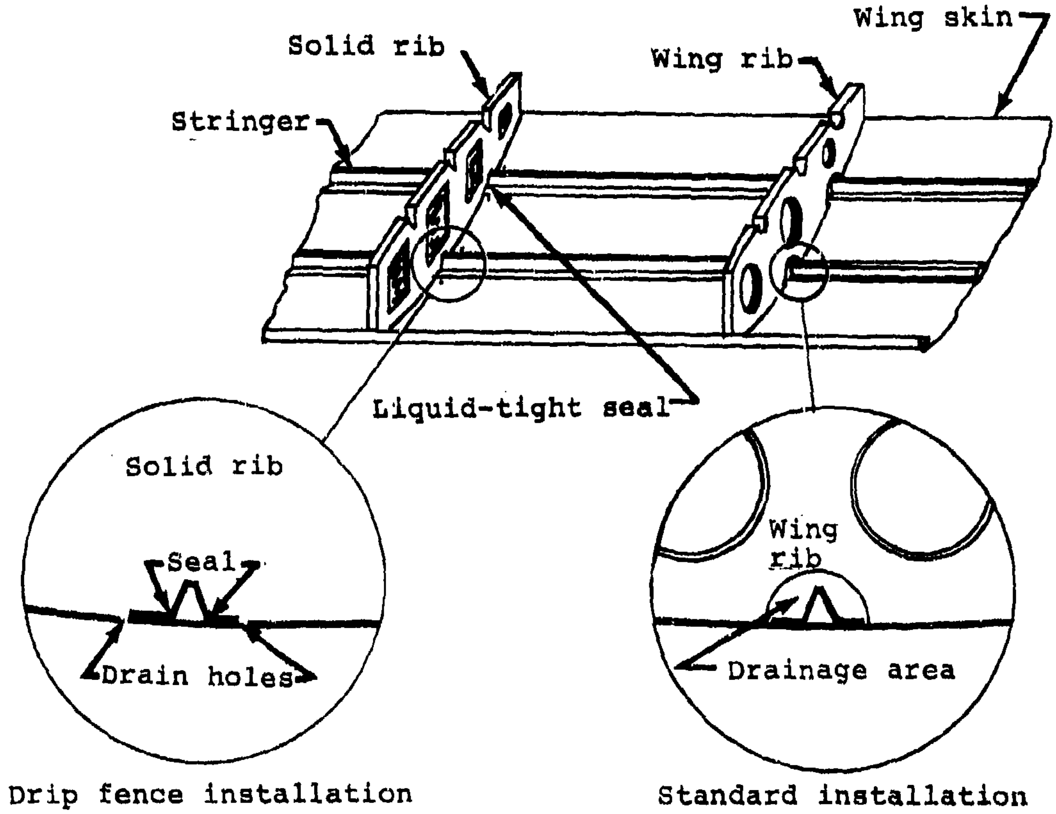

Figure 45. Internal drip fence.

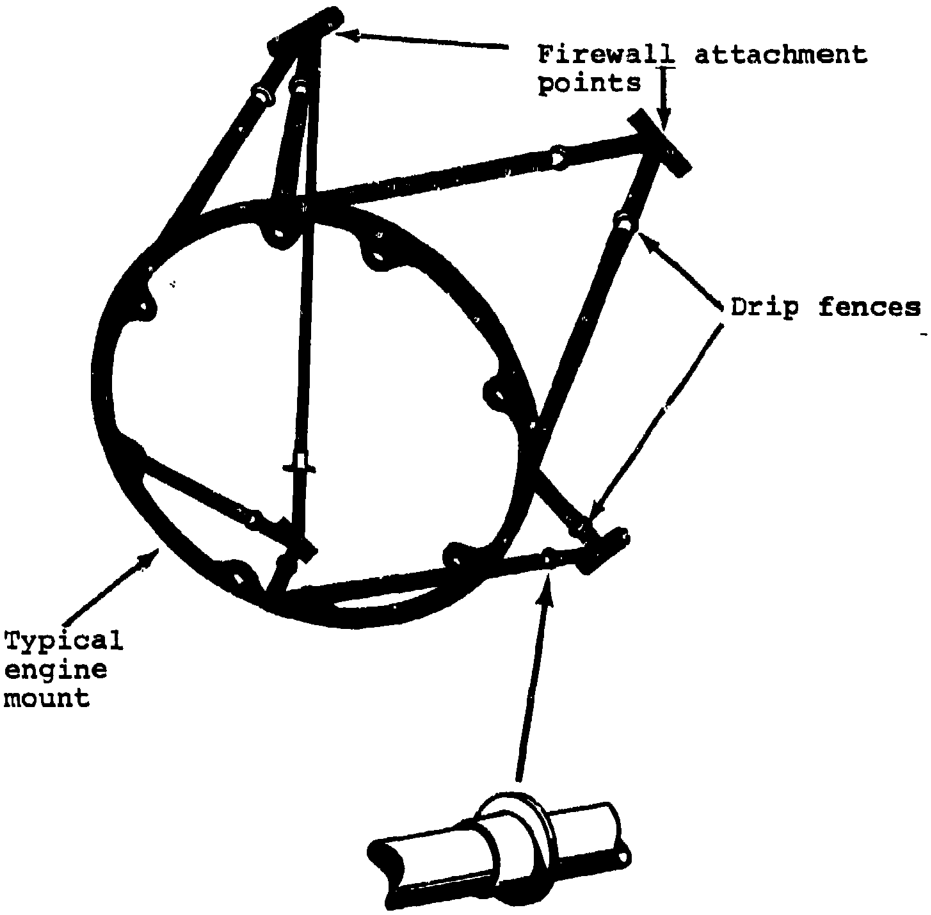

Figure 46. Fuel flow diverters.

or enveloping electrical wiring. The shielding should be used in all places where structural shift or collapse could cause an impingement on electrical wiring or related components.

The fourth method of shielding uses protective coatings or surfaces. Recent studies have produced materials that, when heated or exposed to other environments, expand to insulate and protect the surface to which they have been applied. Intumescent paints are an example of this form of shielding. This field of protective surfacing is fairly new and is an area recommended for future research.

4.5.3 Heaters

Heating units often are provided in aircraft cockpits and passenger compartments. These units, which also supply deicing air, may be either combustion or engine bleed-air types.

Bleed-air heaters normally use air from the compressor section of the engine. Hot-surface ignition sources on turbojet engines are downstream of the compressor section, and if the temperature is below 400°F, the piping system that carries the bleed air to a mixing chamber should not be an ignition source. If a temperature survey indicates that the system produces temperatures above 400°F, suitable inerting and/or shielding should be provided.

Combustion heaters will produce hot metal surfaces that should be treated as potential ignition sources. The surfaces of the heater that become hot enough during normal operation to cause ignition of crash-released flammable fluids must be determined, and a cooling and inerting system must be designed. The coolant must not be an irritant to aircraft occupants. A water-detergent solution was used in the cooling and inerting system described in Reference 29.

4.5.4 Sparks

Two types of sparks should be considered potential ignition sources: friction sparks and electrostatic sparks. The friction spark is a particle abraded from a parent material through contact with a moving surface. Initially, the particle is heated by friction. If the friction is great enough, the particle can burn, thus increasing its temperature. Electrostatic sparks result from the discharge of an electrostatic charge accumulated on parts during normal operation. The discharge is triggered during the crash when the parts are separated due to crash forces.

4.5.4.1 Friction Sparks: Friction sparks become possible ignition sources when portions of aircraft structure are scraped along the ground. While all common metals can be abraded, not all spark sufficiently to ignite spilled fluids. Ignition occurrence depends on the thermal energy of the spark. The thermal energy is a function of the bearing pressure with which the metal is abraded, the slide speed of the metal structure,

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29. Jones, R. B., et al., AN ENGINEERING STUDY OF AIRCRAFT CRASH FIRE PREVENTION, Walter Kidde and Company, Inc.; Technical Report 57370, Wright Aeronautical Development Center, Ohio, June 1958, AD 155846.

the hardness of the metal, and the temperatures at which the metal particles will burn.

NACA has conducted research on the friction spark ignition hazard relative to crashed aircraft (References 27 and 30). Some results of this research are listed in Table 6. These studies indicated that aluminum was the safest of the metals tested, since it produced no visible sparks and did not ignite combustible mists at the highest bearing pressure and greatest slide speeds tested. Of all the metals tested, titanium ignited the combustible mist most readily; however, stainless steel, chrome-molybdenum steel, and magnesium all ignited the mist at slide speeds and bearing pressures less than those expected during a crash.

TABLE 6. MINIMUM CONDITIONS UNDER WHICH CERTAIN ABRADED METAL PARTICLES WILL IGNITE

Metal	Minimum bearing pressure ($lb/in.^2$)	Drag speed (mph)
Titanium	21-23	Less than 5
Chrome-molybdenum steel	30	10
Magnesium	37	10-20
Stainless steel	50	20
Aluminum	1455*	40

*Ignition was not obtained with aluminum.

No data are currently available on the spark hazards of the new composite materials. Tests such as those mentioned above should be conducted with these materials to determine their friction spark ignition potentials.

30. Campbell, J. A., APPRAISAL OF THE HAZARDS OF FRICTION SPARK IGNITION OF AIRCRAFT CRASH FIRES, NACA Technical Note 4024, Lewis Flight Propulsion Laboratory, National Advisory Committee for Aeronautics, Cleveland, Ohio, May 1957.

There are two practical methods of reducing the friction spark hazard. One is to use shielding to prevent the fuel from reaching the spark-producing area, and the other is to build the probable contacting surface out of materials having little or no spark-producing tendencies.

As stated above, aluminum was the least likely metal to ignite spilled flammable fluids. Building all aircraft structures likely to come in sliding contact with the ground out of aluminum can reduce the spark hazard; however, it must be pointed out that aluminum also is easily abraded. It can tear when sliding, thereby exposing other metals that might spark and ignite the spilled fuel. Therefore, the areas most apt to come in sliding contact with the ground should be reinforced so that longer contact times are possible without skin failure due to abrasion. Particular attention should be given to attachment points for hoists, landing gears, and other components located in anticipated impact areas. Also, particular attention should be given to the location of steel bolts, nuts, and washers. All too often an otherwise spark-free area is contaminated by locating a spark-producing bolt or nut within it.

4.5.4.2 Electrostatic Sparks: During the course of the NACA research, it was noted that electrostatic discharge from a wheel strut caused ignition of a fuel mist and, ultimately, the destruction of the test aircraft (Reference 27). This ignition source was produced by a combination of environmental conditions that would occur infrequently. It may be possible to reduce electrostatic charge buildup by applying coatings to those parts of the aircraft likely to be separated in a crash. Additional research is required to develop methods of eliminating this hazard.

4.5.5 Initiating Systems

A crash-fire prevention system should include an initiating system that senses the existence of crash-fire conditions and causes action to be taken to suppress the ignition sources. The initiating system must meet the following design requirements:

- The system must not be capable of accidental operation as the result of malfunctioning sensors or short circuits.
- The system must be designed to operate automatically upon receipt of coincident signals from redundant sensors.

- The pilot must be capable of operating the system manually and of overriding the automatic signals.

- The system must be designed for positive airborne and ground check-out, with reset capability provided.

Sensors and the discriminating circuitry used to derive the automatic signals must be carefully selected and developed. The skill and knowledge of the designers also are important in determining the location and installation of the sensors. A discussion of sensors and criteria for aircraft application are contained in Chapter 8 of this volume (Crash Locator Beacons).

The activating circuitry must be designed to avoid inadvertent operation of the crash-fire protection system. Crash signal redundancy is the key element in any such fail-safe system. This design philosophy is illustrated by the activating circuitry shown schematically in Figure 47. This circuitry was developed for reciprocating, multiengined aircraft by NACA (Reference 31). A signal from any one of three switches will result in the inerting of one of the engines. A fuel tank penetration switch indicates when the wing has been penetrated and will result in de-energizing the electrical circuits within the wing. Either the inerting of an engine or the de-energizing of a wing's electrical circuits will cause a signal to be sent to an arming control box. This signal must be combined with signals from two ground contact switches to actuate the entire inerting system. This requirement for simultaneous signals from different types of initiating switches reduces the possibility of the entire inerting system operating while the aircraft is still in the air.

A schematic of activating circuitry that could be applied to rotary- and fixed-wing single-engine aircraft is shown in Figure 48 (Reference 32). The average reading of four proximity switches is compared with the aircraft's normal landing height.

31. Moser, J. D., and Black, D. O., PROPOSED INITIATING SYSTEM FOR CRASH-FIRE PREVENTION SYSTEMS, NACA Technical Note 3774, National Advisory Committee for Aeronautics, Cleveland, Ohio, December 1956.

32. Drummond, J. K., STUDY TO DETERMINE THE APPLICATION OF AIRCRAFT IGNITION-SOURCE CONTROL SYSTEMS TO FUTURE ARMY AIRCRAFT, Dynamic Science; USAAMRDL Technical Report 71-35, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, June 1971, AD 729870.

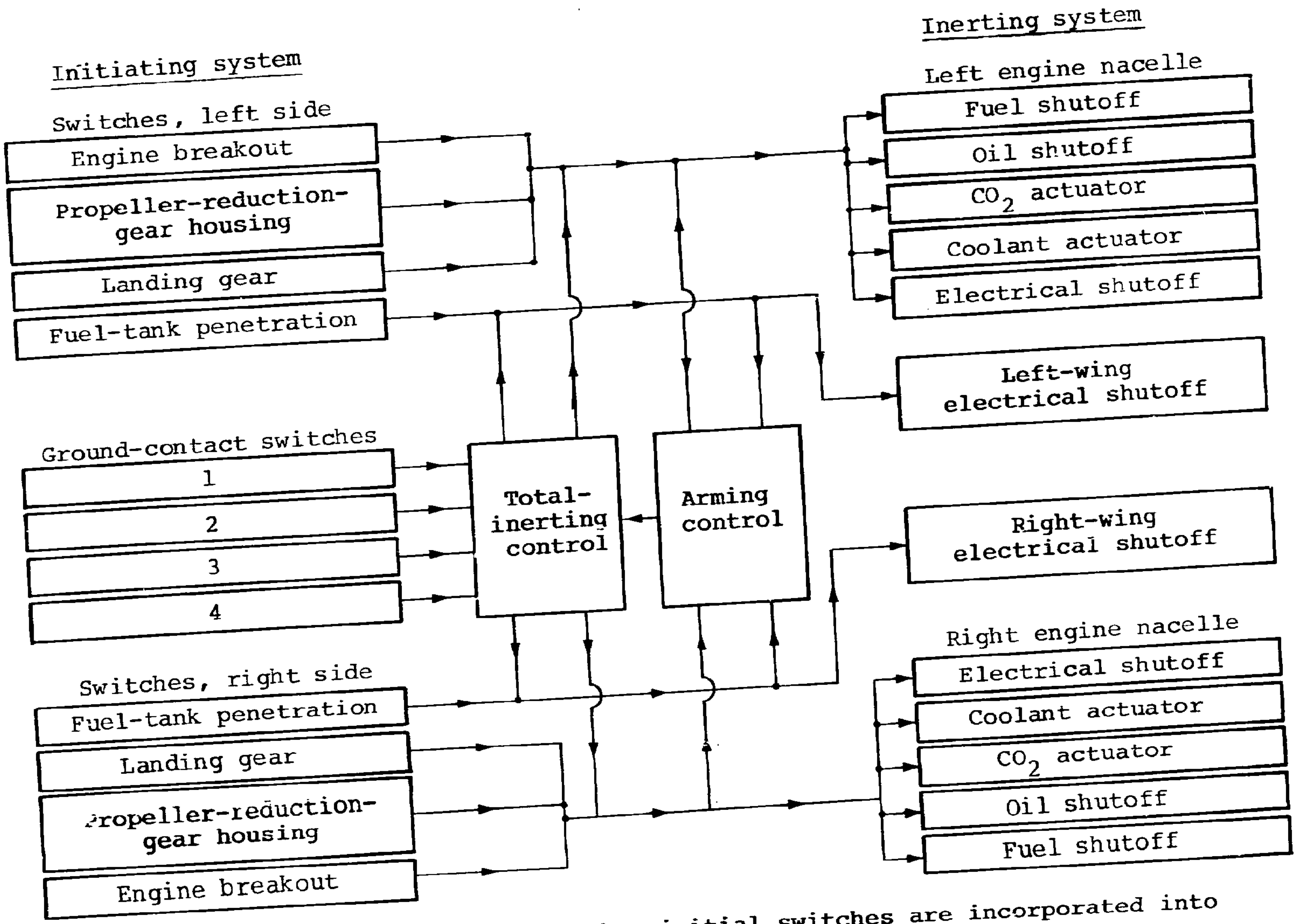

Figure 47. Block diagram showing how initial switches are incorporated into a crash-fire-prevention system for twin-engined airplane.

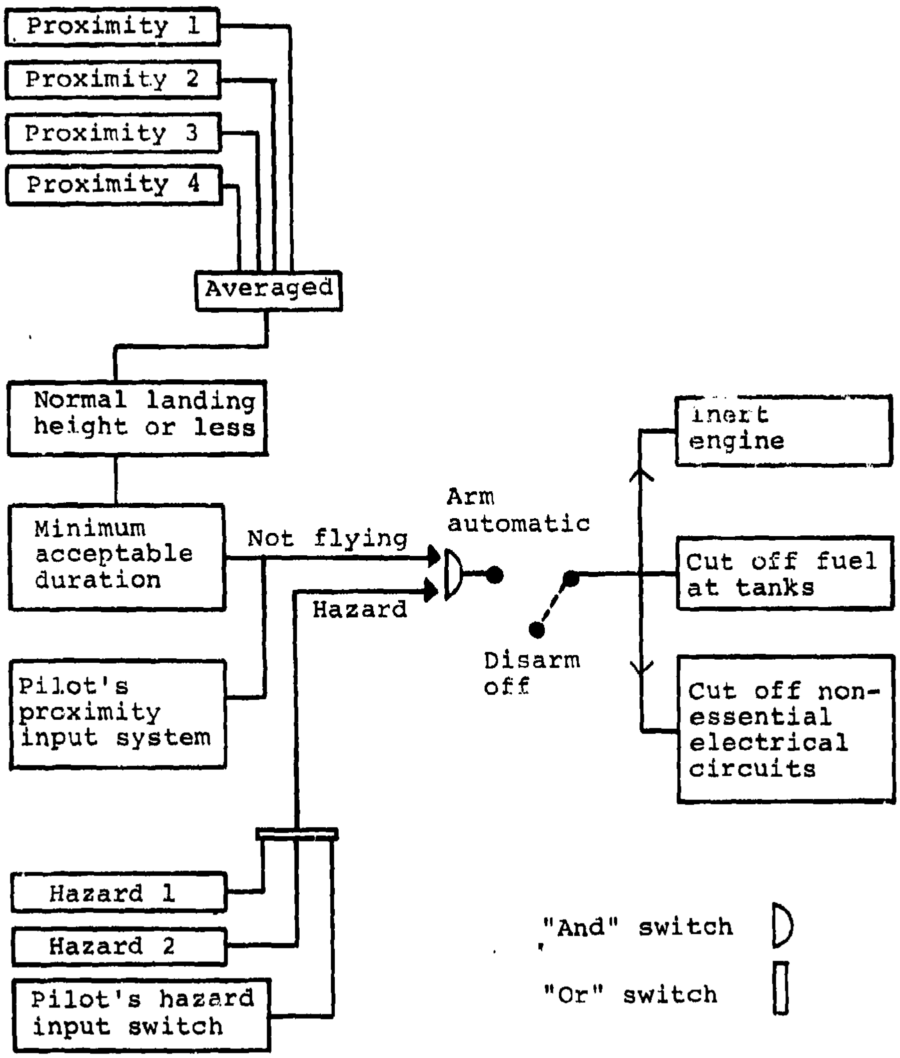

Figure 48. Potential activating circuitry for single-engine aircraft.

If the average is less than the normal landing height for a period of time that exceeds a preset minimum duration, an arming signal is initiated. A second, independent arming signal provided by a hazard switch is required before automatic operation of the ignition-source suppression system. This hazard switch may be any of the sensors previously discussed. Provisions also are included for pilot input to the arming signal and for pilot override of the entire system.

5. INTERIOR MATERIALS

5.1 INTRODUCTION

While every effort should be made to prevent a major postcrash fire by containing the fuel, as much protection as possible should be provided for the occupants in case a fire does start at any time. Careful selection of interior materials can slow the spread of smaller fires and give occupants time to evacuate the aircraft safely or to be rescued by other personnel. The protection afforded against in-flight fires is as important as postcrash fire protection. Often, fire-hardening of the aircraft interior can result in a controllable in-flight fire incident rather than a catastrophic fire accident.

It would be desirable to present the designer with a concise list of materials that should be used in aircraft interiors. Unfortunately, this is not possible at the present time for two major reasons. First, the selection of interior materials is dependent on several varied and sometimes conflicting, design criteria. For instance, seat cushion materials must possess compressive modulus and rebound characteristics necessary for crashworthiness, restraint webbing must meet definite elongation criteria, and seat upholstery must possess a minimum wear resistance. At the same time, these materials should provide maximum fire resistance. Many materials currently available cannot meet all of the criteria simultaneously; thus, priorities must be established and trade-offs must be made. One factor compounding this problem is that, at the time this volume is being written, there is no one place where all material properties, including flammability data, are available. This situation should be rectified shortly when the Transportation Systems Center data bank is fully operational (see Section 5.5).

The second major reason that precludes a listing of recommended materials is that a great deal of activity has been directed toward the development of materials and testing methods in the last few years. This field is still very active and new materials and tests are being developed constantly. The designer should be aware of this and select the best possible materials for the aircraft interior. Many improved materials only now are becoming available or will become available before the Design Guide is revised again.

Because the aircraft designer must choose the materials for the aircraft interior, considering all the necessary criteria that these materials should meet, it is essential that the designer have the knowledge upon which to base intelligent selections and trade-offs. Therefore, the following sections present in some detail the various aspects of material flammability hazards, current testing methods, and flammability properties of some currently used and newly developed materials. Guidelines for making trade-offs between conflicting criteria also are presented. This background information should assist the designer in evaluating and selecting interior materials that will provide maximum fire protection while still meeting necessary design requirements.

5.2 FIRE BEHAVIOR OF MATERIALS

Interior materials can contribute to the overall fire hazard not only by their flammability, but also by their release of smoke and toxic gases during combustion. Although the three factors of flammability, smoke, and toxic gases are discussed separately in the following sections, all three must be considered together when evaluating any material for its fire safety.

5.2.1 Flammability

The principal factors to be considered in evaluating the flammability of a material are:

- Ease of ignition.
- Flame spread rate.
- Heat release rate.
- Flash fire potential.

5.2.1.1 Ease of Ignition: Ease of ignition can be defined as the ease with which a material can be ignited under given conditions of temperature, pressure, and oxygen concentration. Almost any material can be made to ignite with enough heat, oxygen, and time. Ease of ignition can, therefore, be measured by the amount of heat required under fixed conditions of oxygen and time, by the amount of oxygen required under fixed conditions of heat and time, or by the amount of time required under fixed conditions of heat and oxygen.

Ease of ignition can be inferred from minimum radiation intensities required to ignite the material, from the auto-ignition temperature of the material, or from the minimum amount of oxygen that permits steady burning of the material. These parameters are highly dependent on the conditions under which they are determined. Test parameters such as sample configuration and size, ventilation, type of ignition source, superimposed heat input (heat flux), and heat losses can profoundly affect the test results. Thus, the relative flammability ranking of materials may vary with the combustion test used, since a material may perform well in one test and poorly in another.

Generally, the ignition temperature of a material is lower when the material and the ambient atmosphere are uniformly heated, as compared to situations in which only the material is heated. This is illustrated in Table 7, which lists the minimum autoignition temperatures (AIT) obtained in a closed vessel and the hot plate ignition temperatures in which only the samples were heated (Reference 33).

TABLE 7. MINIMUM AUTOIGNITION TEMPERATURES (AIT) AND HOT PLATE IGNITION TEMPERATURES OF SHEET-TYPE COMBUSTIBLES IN AIR (FROM REFERENCE 33)

	Ignition temperature, °F	
Material	AIT	Hot plate
Cotton sheeting	725	870
Conductive rubber sheeting	735	895
Paper drapes	750	880
Plexiglas sheeting	840	1105
Nomex fabric	960	>1110
Blanket wool	1005	>1110
Cellulose acetate sheeting	1020	>1110
Polyvinyl chloride sheeting	1040	>1110

The time required to ignite a material with a pilot flame is dependent on the intensity of any superimposed radiant heat flux. For instance, under identical test conditions, the time from flame exposure to burning for particle board varies from approximately 1.7 min at a radiant heat flux of 0.9 Btu/sec,ft^2

33. Kuchta, J. M., FIRE AND EXPLOSION MANUAL FOR AIRCRAFT ACCIDENT INVESTIGATORS, Pittsburg Mining and Safety Research Center, Bureau of Mines, AFAPL Technical Report 73-74, Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio, August 1973, AD 771191.

to 0.5 min at a heat flux of 2.5 Btu/sec,ft² (Reference 34). The minimum oxygen concentration required for combustion also is dependent on the heat flux seen by the test sample, as shown in Figure 49 (Reference 35).

The relative flammability hazards of different materials can be determined for any specific set of test conditions. For instance, the radiation intensity required for ignition during tests using a heat flux of 48.7 Btu/sec²,ft² was about 50 Btu/ft² for cotton sheeting and between 90 and 120 for wood and paper sheeting (Reference 33). In comparison, neoprene, nylon, and polyvinyl chloride sheeting appeared to be nonignitable in air with the same radiation source.

Whenever comparisons are made between materials, however, one must remember that those relative rankings are valid only for the set of conditions imposed by the test, and may or may not be valid for other test conditions. The data presented in Figure 49 clearly show the changes in relative rankings that can occur under varying test conditions.

5.2.1.2 Flame Spread Rate: Surface flame spread can be defined as the rate a flame front travels across a material under given conditions of burning. This characteristic provides a measure of fire hazard in that surface flame spread can transmit fire to more flammable materials in the vicinity, thus enlarging the overall fire, although the transmitting material itself may contribute little fuel to the fire.

Flame spread rates are markedly influenced by such factors as the presence of a superimposed radiant heat flux, oxygen concentration of the atmosphere, density of the material, and orientation of the material. Generally, flame spread rates increase with increasing radiant heat exposure, as illustrated in Figure 50 (Reference 35). The magnitude of the change in flame spread rates can be startling and, at times, misleading if more than one test condition is not considered. For instance, Smith found that a rigid polyurethane foam that was self-extinguishing up to a heat flux of 0.5 Btu/sec,ft² changed to a combustible material with a high flame travel rate at a heat flux between 0.5 and 1.0 Btu/sec,ft² (Reference 34).

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34. Smith, E. E., PRODUCT FIRE HAZARD EVALUATION, Journal of Fire & Flammability/Consumer Product Flammability, Vol. 2, March 1975, pp. 58-69.

35. Brauman, S. K., EFFECT OF SAMPLE TEMPERATURE ON COMBUSTION PERFORMANCE OF POLYMERS, Journal of Fire & Flammability, Vol. 8, April 1977, pp. 210-224.

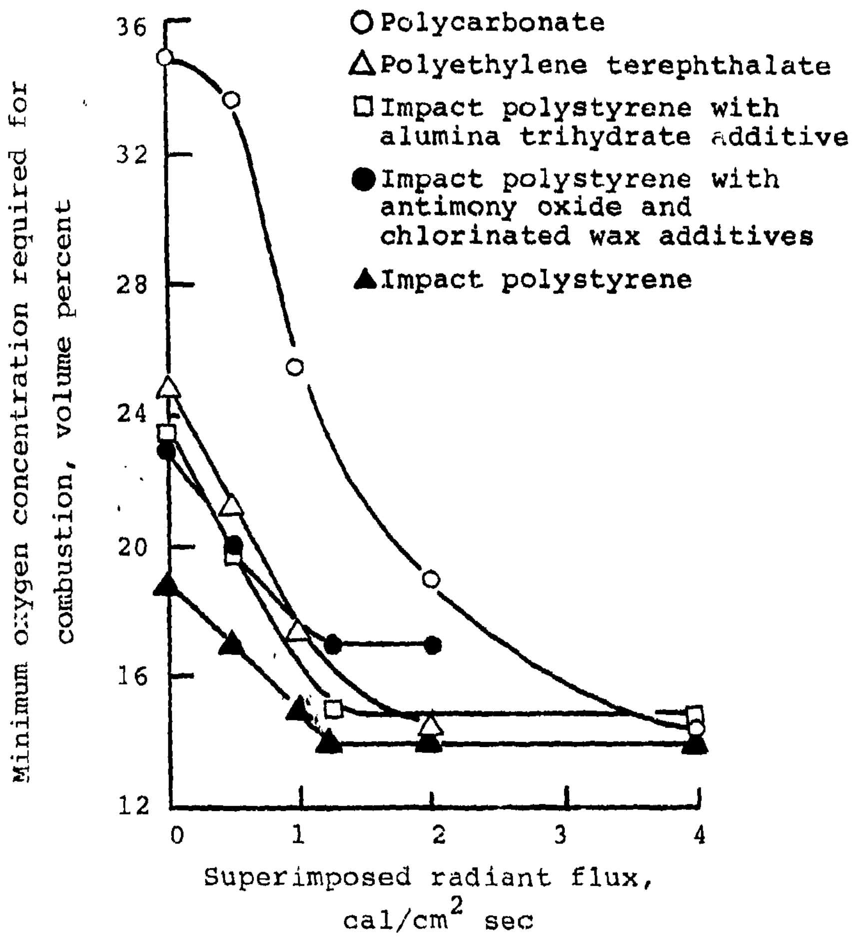

Figure 49. Influence of superimposed heating on minimum oxygen concentration for various polymer systems burning in the driven-rod configuration. (From Reference 35)

The orientation of the test sample also can markedly influence flame spread rates. Upward burning of a vertical sample will generate a higher flame spread rate than will the burning of a horizontal sample under the same conditions. Also, it has been observed that the flame spread rate of cotton sheeting is about 40 times greater with upward burning than with downward burning for specimens in a vertical position (Reference 33).

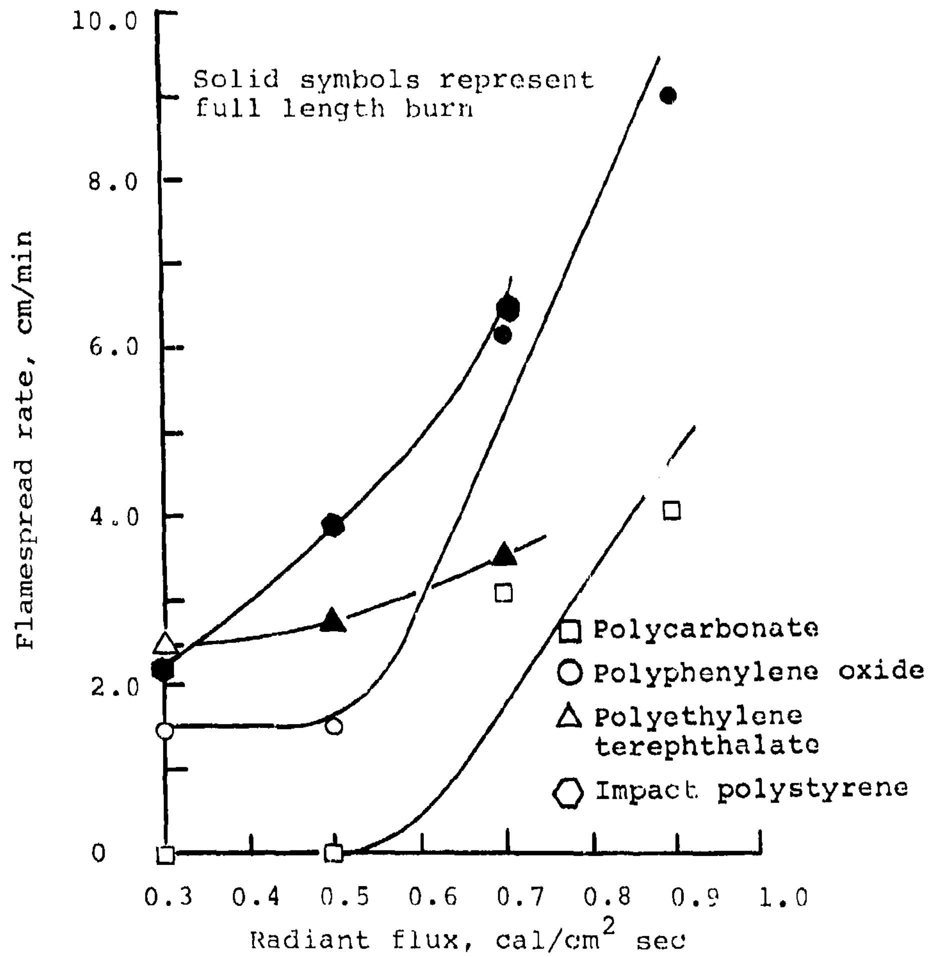

Figure 50. Influence of superimposed heating on flame spread rates of various polymers. (From Reference 35)

5.2.1.3 Heat Release Rate: Heat release can be defined as the heat produced by the burning of a given weight or volume of material. This characteristic provides a measure of fire hazard; i.e., a material that burns with the evolution of little heat per unit quantity burned will contribute less to a fire than a material that generates large amounts of heat.

Of more importance in relation to the spread of a fire, and thus, the available escape time, is the rate of heat release. Heat release rates can give a comparative measure of the contributions of various materials to a developing fire. However, heat release rate values by themselves cannot adequately describe the contribution of a material in a real fire. In order to accurately assess the fire hazard of a material, the heat release rate must be determined as a function of time.

As with the other parameters used to assess a material's flammability, heat release rates are a function of exposure. In order to predict performance in a fire, heat release data must be obtained over a range of heat flux levels. Most cellulosic materials exhibit a uniform change in ignitability, flame travel rate, and maximum rate of heat release with change in exposure (Reference 34). However, different materials do not necessarily respond to changes in exposure in the same manner. For instance, certain "self-extinguishing" fire-retarded polymers that do not support combustion at low exposure levels will change to highly combustible materials when exposed to a higher heat flux. This type of behavior can result in two materials, examined and rated at one set of conditions, having their ratings reversed at another set of conditions. Wool and nylon carpet, as well as some of the polymers, go through such rating reversals. At low heat flux levels, nylon is less combustible, while wool is less combustible at higher heat fluxes (Reference 34).

5.2.1.4 Flash Fire Potential: A flash fire is a flame front that propagates through a fuel-air mixture as a result of the energy released from the combustion of the fuel vapor. These fires occur when combustible vapors evolve from burning materials and accumulate elsewhere as substantial volumes of flammable fuel-air mixtures, which then come in contact with an ignition source.

Screening tests for the flash-fire propensity of materials have been proposed based on the concentration of the flammable gases evolved when the materials are pyrolyzed (Reference 36). The gases analyzed during the tests were the hydrocarbons methane, ethylene, ethane, and carbon monoxide. Test results showed that those materials with the highest propensity for flash fires, such as polyethylene and polyurethane, had significantly higher hydrocarbon concentrations in their pyrolysis products than did wood, which appeared to have the least propensity for flash fires. Also, materials that melted, such as polyethylene and polyurethane foam, had larger concentrations of the more flammable hydrocarbons (ethylene and ethane) than materials that intumesced, such as bisphenol A polycarbonate, or charred, such as wood.

36. Hilado, C. J., and Cumming, H. J., HYDROCARBON CONCENTRATIONS IN FIRE TOXICITY TESTS AS AN INDICATION OF FLASH FIRE PROPENSITY, Journal of Fire & Flammability, Vol. 8, April 1977, pp. 235-240.

5.2.2 Smoke

Combustion of organic materials yields gaseous products in which small solid particles of carbon and ash, as well as liquid droplets, are frequently dispersed. This mixture of gases, solids, and liquids can be defined as smoke. In general practice, however, smoke is often defined as the combination of solid and liquid particles that lead to vision obscuration, while the gaseous products are treated separately.

The primary hazard of smoke (excluding toxic gases) is the reduction of visibility. The degree of light or sight obscuration due to smoke is generally expressed in terms of optical density, defined as $D = \log 100/T$ (where T = percent light transmission).

The amount of smoke generated by a burning material depends on the surface area involved, and the degree of obscuration depends on the available volume and light path length for any given amount of smoke. A quantitative measure, the specific optical density, has been defined to allow comparisons of smoke generation between different materials (Reference 37). The specific optical density is defined as:

$$D_s = \frac{V}{AL} \log \frac{100}{T} \qquad (2)$$

where

$$\left.\begin{array}{l} D_s = \text{specific optical density} \\ V = \text{chamber volume} \\ A = \text{area of sample exposed to burning} \\ L = \text{path length of light} \end{array}\right\} \text{all in consistent units}$$

T = percent light transmission

Ideally, the change in D_s with time and the maximum D_s (sometimes designated D_M) would depend only on the thickness of the material specimen, its chemical and physical properties, and the test exposure conditions. The visibility in any size compartment could then be calculated from the D_s obtained during the laboratory testing.

37. Gross, D., et al., SMOKE AND GASES PRODUCED BY BURNING AIRCRAFT INTERIOR MATERIALS, Building Science Series 18, U. S. Department of Commerce, National Bureau of Standards, Washington, D. C., February 1969.

It is difficult to precisely extrapolate specific optical densities to human visibility in burning aircraft compartments. This is because a number of major assumptions must be made in the extrapolation: the smoke generated is uniformly distributed and is independent of the amount of excess air available; for any given smoke, the optical density is linearly related to concentration; and human and photometric vision through smoke, expressed in terms of optical density, are similar. However, the specific optical density does offer a valid means of comparing smoke generated by various materials and can be used to screen out those materials generating the greatest amount of smoke.

Smoke levels generated by burning materials are dependent on both physical and chemical parameters of the material involved and on the burning conditions. In an extensive series of tests on aircraft interior materials, Gross found that the maximum smoke level depended on the thickness and density of the specimen and could be expected to increase with thickness, but not always in direct proportion (Reference 37). He also found that, although most materials produced more smoke during the flaming exposure test, some materials produced significantly more smoke in the absence of open flaming (smoldering).

Although the addition of flame retardants has significantly reduced the flammability of many polymeric materials, these chemical additives often have resulted in increased smoke emissions during fire exposure. Figures 51 and 52 illustrate the effect of concentrations of reactive and nonreactive flame retardants on the light obscuration times in rigid urethane foams (Reference 38). It should be noted that the reactive fire retardant, which imparts the greatest degree of protection, produces more rapid light obscuration. The addition of flame retardant to a flexible urethane foam tested by Gross not only resulted in an increase in overall smoke levels, but also led to a reversal of the relative smoke concentrations from smoldering versus open flaming.

38. Einhorn, I. N., PHYSIO-CHEMICAL STUDY OF SMOKE EMISSION BY AIRCRAFT INTERIOR MATERIALS, PART I: PHYSIOLOGICAL AND TOXICOLOGICAL ASPECTS OF SMOKE DURING FIRE EXPOSURE, University of Utah; Report No. FAA-RD-73-50-1, Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, July 1973, AD 763602.

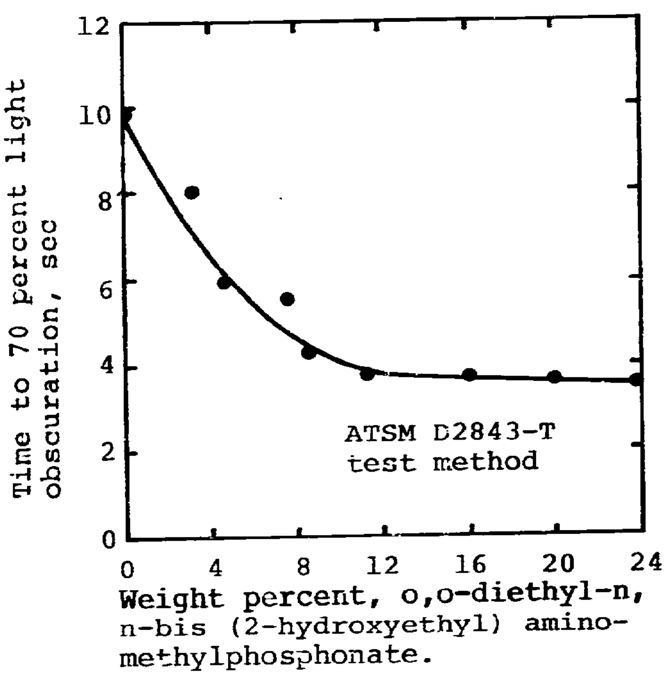

Figure 51. Effect of concentration of the reactive fire retardant o,o-diethyl-n,n,-bis (2-hydroxyethyl) aminomethylphosphonate on light obscuration in rigid-urethane foams. (From Reference 38)

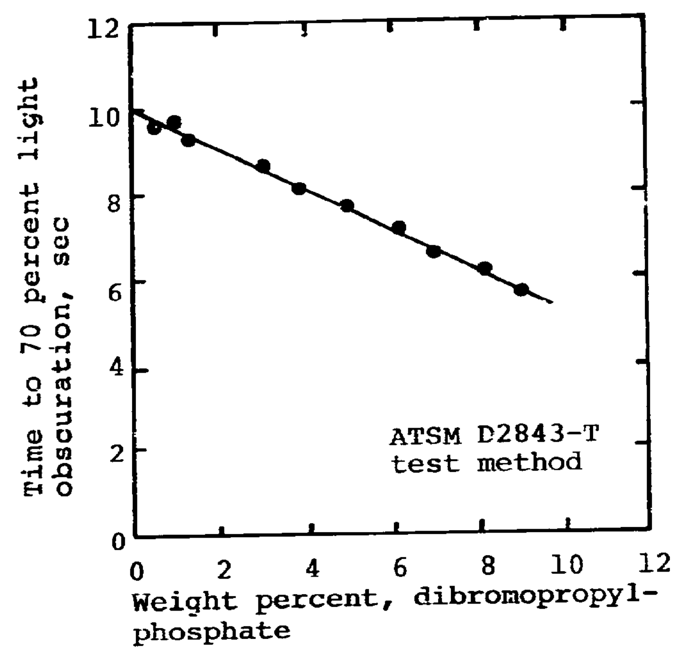

Figure 52. Effect of the nonreactive fire retardant tris, 2,3-dibromopropylphosphate on light obscuration in rigid-urethane foams. (From Reference 38)

5.2.3 Toxic Gases

The most common gases generated during the combustion of any organic material are carbon monoxide and carbon dioxide. Several other toxic gases also may be produced, depending on the chemical composition of the involved material. The results of the extensive series of burn tests on aircraft interior materials conducted by Gross showed that carbon monoxide (CO) was produced by almost all the samples in varying amounts depending on the type of material (Reference 37). Most materials produced significant amounts of other toxic gases in addition to CO. Table 8 summarizes those results. The addition of flame retardants can contribute to the generation of taxic gases, as noted in Table 8, when comparing urethanes identical in all respects except for the presence of a flame retardant.

TABLE 8. TOXIC GASES PRODUCED BY BURNING AIRCRAFT INTERIOR MATERIALS

Material	CO	HCl	HCN	Other
Nylon	X	-	-	-
Wool	X	-	X	-
Polyvinyl chloride (PVC)	X	X	-	-
Modacrylic	X	X	X	-
Polyamid (aromatic)	X	-	X	NO_2
Polyvinyl fluoride (PVF)	X	-	-	HF
Urethane	X	-	X	-
Urethane (flame retarded)	X	X	X	-
Acrylonitrile/butadiene/ styrene (ABS)	X	-	X	-
Polysulfone	X	-	-	SO_2
Rubber	X	-	-	SO_2
Propylene	X	-	-	-
Polycarbonate	X	-	-	-

Of course, the amount of toxic gas generated will depend on the amount of material burned. However, Gross found that the amount of a given gas produced and its rate of generation are strongly temperature dependent. This was confirmed during additional testing by Spurgeon, et al., who also found that varying oxygen concentrations will affect the yield of combustion gases (Reference 39). No generalizations could be made, however, since the observed effects seemed to depend on the composition of the test material. The same is true when comparing the yields of gases during flaming or nonflaming conditions. Most of the materials tested by Gross yielded higher concentrations of gases under flaming conditions. There was little difference for some materials, however, while others generated more gases during nonflaming conditions.

Although approximate human toxicological data are available for many of the individual gases given off by burning materials, little is known of the synergistic effects of two or more gases inhaled at the same time. Since the majority of materials give off more than one gas, and since many interior components are actually combinations of materials, relative toxicities of different components must be determined in a manner that assesses the total effect of the toxic gases given off. This can be accomplished by using small animal toxicity tests. In an attempt to correlate analytical test methods with small animal toxicity tests, the FAA tested a number of aircraft materials using both methods (Reference 40). Although most nitrogen-containing materials indicated a correlation between HCN concentration and time to animal incapacitation, one material (76 percent wool, 24 percent PVC) showed a much higher than expected toxicity. This toxicity could not be explained on the basis of HCN concentrations or a simple synergistic response due to the combination of PVC and wool. One possible explanation for the observed toxicity is the zirconium fluoride flame-retardant treatment that the material had received. Whatever the cause, the unexpected toxicity illustrates the value of animal tests in assigning relative toxicities to interior materials.

39. Spurgeon, J. C., Speitel, L. C., and Feher, R. E., OXIDATIVE PYROLYSIS OF AIRCRAFT INTERIOR MATERIALS, Journal of Fire & Flammability, Vol. 8, July 1977, pp. 349-363.

40. Spurgeon, J. C., A PRELIMINARY COMPARISON OF LABORATORY METHODS FOR ASSIGNING A RELATIVE TOXICITY RANKING TO AIRCRAFT INTERIOR MATERIALS, Report No. FAA-RD-75-37, Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, October 1975, AD A018148.

5.3 MATERIAL TESTING

The number of material tests has increased in direct proportion to the increasing importance placed on fire safety over the last few years. Unfortunately, the proliferation of tests has not generated any degree of consensus in selecting the "best" test(s) for material screening and selection. There is a great deal of controversy among those working in this field as to the validity of the various tests in predicting a material's performance in a real fire. Thus, there are no generally accepted test methods or criteria for material performance at the present time. A brief review of the different types of tests is presented in the following sections so that the designer can understand the various performance ratings assigned to materials by the use of different tests. Those tests recommended for materials in U. S. Army aircraft are discussed in Section 5.5.

5.3.1 Laboratory Testing

5.3.1.1 Flammability Tests: There are several different types of tests for material flammability, with each type testing a specific aspect of flammability, such as ease of ignition, flame spread, heat release, and fire endurance. In addition, there are several different test methods for each type of test. A review of these numerous tests has been compiled by Hilado (Reference 41).

The simplest tests for ease of ignition provide fixed conditions of heat, oxygen, and time, and the sample either ignites or does not ignite under those conditions. A somewhat more sophisticated test is the ASTM D 1929 (Setchkin) ignition test in which a specimen is exposed to heated air at successively higher temperatures until ignition occurs. The lowest temperature of air that evolves combustible gases in a sufficient amount to be ignited by a small pilot flame is defined as the flash-ignition temperature of the material. The self-ignition temperature is the lowest air temperature at which the material ignites by itself, in the absence of any external ignition source.

A different type of ignition test, one being used increasingly for aircraft interior materials, is the ASTM D 2863 oxygen index test. In this test, a vertical specimen is ignited at its upper end by a flame that is then withdrawn; then the atmosphere (mixture of oxygen and nitrogen) that just permits steady burning is determined. The limiting oxygen index (LOI) is the

41. Hilado, C. J., FLAMMABILITY TESTS, 1975: A REVIEW, Fire Technology, Vol. 11, No. 4, November 1975, pp. 282-293.

minimum concentration of oxygen in the oxygen-nitrogen mixture that will just permit the sample to burn. The higher the LOI, the less flammable the material is.

In addition to the above two widely used tests, Hilado lists eight other tests for ease of ignition. He points out that many tests that measure flame spread are actually tests for ease of ignition because failure to ignite or to sustain ignition is the most desirable response. Hilado lists 10 tests in this category, including the FAR 25.853 vertical test (Reference 42).

The latter test is currently required by the FAA for compartment interior materials in transport category airplanes. In this test, the lower edge of a vertically mounted sample is exposed to a burner flame for either 60 or 12 sec, depending on the type and application of the material. The flame is then removed and flame time, burn length, and flaming time of drippings are recorded. All materials must be self-extinguishing; i.e., average flame time after removal of the flame source must not exceed 15 sec. In addition, the average burn length must not exceed 6 or 8 in., again depending on the type and application of materials.

There are even more tests for surface flame spread rates; Hilado lists 35 tests in this category, including the FAR 25.853 test. Two of the most widely used tests for flame spread are the ASTM E 84 tunnel test and the ASTM E 162 radiant panel test.

The tunnel test involves igniting the sample at one end and measuring the rate at which the flame travels across the surface. Flame spread classification is determined on a scale where asbestos cement board is zero and select-grade red oak flooring is 100. No additional heat flux is supplied beyond that generated from the burners and the burning sample itself. The radiant panel test, however, does provide additional heat beyond that generated by the sample. The specimen is mounted in a semivertical position in front of a radiant heat source maintained at 670°C (1238°F). The top of the specimen is then ignited and the flame spread index determined as in the tunnel test.

Many authorities feel that the radiant panel test is superior because it more nearly duplicates the exposure the material would experience in an actual fire. (The importance of the increased heat exposure in assessing a material's "true" flammability has been discussed in Section 5.2.1.) The Urban Mass

42. Federal Aviation Regulations, AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY AIRPLANES, Part 25, Section 25.853.

Transportation Administration (UMTA) has recommended the radiant panel test in their guidelines for flammability specifications applicable to combustible materials used in transit systems (Reference 43). The National Materials Advisory Board of the National Research Council also has recommended the use of ASTM E 162 for aircraft interior materials testing (Reference 44).

There are several tests for heat release, but currently they are not as widely used as the more established tests for ease of ignition and flame spread. The Factory Mutual calorimeter test and the Ohio State University heat release rate tests are the most commonly used at the present time.

Fire endurance tests measure the resistance offered by a material to the passage of fire normal to the exposed surface. The fire resistance can be measured by the burn-through time or by the relative difference in temperature between the flame side and the back face of the specimen. There are several tests for fire endurance. The NASA Ames T-3 test (Reference 45) seems to be the most widely used endurance test for aircraft materials and is well suited for testing components containing several different materials, such as interior fuselage wall panels.

5.3.1.2 Smoke Evolution Tests: One of the earliest procedures for measuring smoke density was the ASTM D 2843 test. This test measures the light obscuration over a 1-ft optical path inside an enclosed chamber containing the burning sample (Reference 38). Smoke evolution also can be measured during flammability testing in the ASTM E 84 tunnel test, the ASTM E 162 radiant panel test, and the heat release test developed at Ohio State University. The most widely used test for aircraft materials, however, is the National Bureau of Standards (NBS) smoke density test first developed by Gross, et al. (Reference 37).

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43. Transportation Systems Center, PROPOSED GUIDELINES FOR FLAMMABILITY AND SMOKE EMISSIONS SPECIFICATIONS, (Unofficial) U. S. Department of Transportation, Cambridge, Massachusetts.

44. FIRE SAFETY ASPECTS OF POLYMERIC MATERIALS, VOLUME 6 - AIRCRAFT: CIVIL AND MILITARY, National Materials Advisory Board, National Academy of Sciences, Washington, D. C., 1977, p. 11.

45. Fish, R. H., AMES T-3 FIRE TEST FACILITY - AIRCRAFT CRASH FIRE SIMULATION, Journal of Fire & Flammability, Vol. 7, October 1976, pp. 470-481.

The NBS test, conducted in a completely closed cabinet, exposes a vertical sample to 2.5 W/cm^2 (2.2 Btu/sec,ft^2) thermal radiation from an electric heater. Light absorption is measured by a photometer over a vertical light path 3 ft long. Tests are performed under both flaming and nonflaming (smoldering) conditions. (A small pilot flame applied to the bottom of the specimen induces open flaming.) Smoke measurements are expressed in terms of specific optical density, D_s (refer to Section 5.2.2). Both the FAA and UMTA have proposed the NBS smoke density test for interior materials (References 43 and 46). The proposed FAA requirement specifies that the D_s of interior materials must not exceed 100 in 4 min for textiles, air ducting, thermal insulation, and insulating covering. The D_s of all other interior materials shall not exceed 100 within 90 seconds, nor 200 within 4 min.

5.3.1.3 Toxic Gas Tests: Concentrations of potentially toxic gases can be determined by chemical analysis of the combustion products. However, the results of this analysis depends on the accuracy of the analytical method employed, the effectiveness of the sampling technique used, and the number of different gases analyzed. Even though the previous factors might be optimized, the problem of relating the chemical results to physiological hazards still remains. Synergistic toxicological effects of various gas combinations are not amenable to analysis. The possibility also exists that some toxic components will not be anticipated and, therefore, will not be considered in the analysis. This latter situation arose during FAA tests comparing chemical analysis versus animal toxicity tests in assigning relative toxicity hazards to aircraft materials (Reference 40).

Both the FAA and the University of San Francisco, under NASA sponsorship, have done extensive animal toxicity testing of

46. Federal Aviation Administration, Proposed Regulations, TRANSPORT CATEGORY AIRPLANES: SMOKE EMISSION FROM COMPARTMENT INTERIOR MATERIALS, Federal Register, 40 F.R.6506, 12 February 1975.

aircraft materials (Reference 47 and 48). Both laboratories expose rodents (the FAA uses rats, USF uses mice) to the pyrolysis products of materials that are thermally degraded in a tube furnace. The exposure conditions vary, however, since the FAA tests maintain a nearly normal concentration of oxygen in the exposure chamber, while the USF tests do not. The FAA conducts its tests with sufficient ventilation in the pyrolysis furnace to assure near normal oxygen concentrations, while USF runs its tests with or without air flow through the furnace. Both laboratories report times to incapacitation and times to death.

Although the above tests cannot duplicate the gas concentrations found in a real fire, the tests do reflect the relative toxicities of different materials to rodents under the specific test conditions. Thus, the animal toxicity tests are useful in screening out the most hazardous materials. The FAA also found that dose-response relationships for the systemic toxins (CO, HCN) are very similar for rodents and humans (Reference 47).

5.3.2 Large-Scale Testing

Because the validity of laboratory tests as a means of predicting material behavior in a real fire has been of increasing concern, large-scale testing is being used more and more as a final test for system performance in a fire. Part of the problem with the predictability of the laboratory tests lies in the fact that a system of materials, not just one material, is involved in actual fires. The types of materials, amounts of each material, and the locations of the different materials all affect the development of a fire in an aircraft fuselage. The behavior of one material will affect that of another, possibly altering its behavior markedly from that demonstrated in laboratory tests.

Large-scale tests of bus and rail car interior assembly mockups illustrate the total effect of all the materials comprising a

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47. Crane, C. R., Ph.D., et al., INHALATION TOXICOLOGY: I. DESIGN OF A SMALL-ANIMAL TEST SYSTEM: II. DETERMINATION OF THE RELATIVE TOXIC HAZARDS OF 75 AIRCRAFT CABIN MATERIALS, FAA Civil Aeromedical Institute; Report No. FAA-AM-77-9, Department of Transportation, Federal Aviation Administration, Office of Aviation Medicine, Washington, D. C., March 1977, AD A043646.

48. Hilado, C. J., FIRE RESPONSE CHARACTERISTICS OF NONMETALLIC MATERIALS: A REVIEW OF RECENT PAPERS AND REPORTS, Journal of Fire & Flammability, Vol. 7, October 1976, pp. 539-558.

system (Reference 49). The assemblies consisted of one or two seat assemblies, wall paneling, and glazing, as would be found in the actual vehicle. Effects of various seat cushions, seat backs, and glazing materials were studied during fire tests started by igniting newspapers on the seat. The tests showed that with urethane seat cushions the system failed at approximately 6 min (flashover occurred) irrespective of the glazing or wall-covering material. Replacing the urethane with less flammable neoprene cushions resulted in increasing the importance of other materials in the system performance. For example, acrylic glazing panels led to system failure (total involvement near flashover conditions) at 7 min, even with the presence of the neoprene seats. However, the fire was confined to the seat of origin in tests using neoprene seat cushions and polycarbonate glazing.

McDonnell Douglas Corporation has developed a Cabin Fire Simulator (CFS) for use in testing commercial aircraft interiors (Reference 50). The CFS is a double-walled steel cylinder 12 ft in diameter and 40 ft long, equipped with a ventilation system, exhaust scrubber, and nitrogen-extinguishing system. To date, over 200 individual fire tests have been conducted in the CFS. These tests have been valuable not only in assessing material interactions but also in evaluating design changes. For instance, in a full cabin lavatory fire test, the lavatory module failed to contain the fire and the fire erupted into the cabin area. Analysis showed that the failure to contain the fire was primarily due to a utility panel falling from the ceiling, resulting in the escape of combustible gases into the cabin. Covering the panel considerably improved system performance.

5.4 SELECTED MATERIAL PROPERTIES

Until recently, most efforts to reduce the flammability of materials were centered on flame-retardant treatments of polymers with halogens or phosphorous. This led to materials such as Fluorel, Refset, and Durette, as well as flame-retarded polyurethanes and cottons, wools, and other fabrics. Many recent efforts, however, have been centered on developing thermally stable, char-forming polymers such as polyimide, polyphosphazene, and polybenzimidazole (PBI). Interest in the thermally

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49. Nelson, G. L., et al., MATERIAL PERFORMANCE IN TRANSPORTATION VEHICLE INTERIORS, Journal of Fire & Flammability, Vol. 8, July 1977, pp. 262-278.

50. Dusken, F. E., FIRE TESTING OF AIRCRAFT CABINS, Journal of Fire & Flammability, Vol. 8, April 1977, pp 193-201.

stable polymers has increased with the finding that most flame-retarded materials produce significantly greater amounts of smoke and toxic gases than do the thermally stable polymers.

Although the listing of flammability properties for the numerous materials already being used or being developed for use in transportation vehicle interiors is beyond the scope of this volume, a brief overview of some current and newly developed materials for aircraft interiors follows.

5.4.1 Seat Cushion Foams

Polyurethane foam is the most common seat cushioning material currently used for aircraft seats. Studies conducted in the late 1960's showed that fire-retarded polyurethane foam was considerably less flammable than nontreated foam (Reference 51). Since that time, considerable efforts have been expended in trying to improve the flame resistance of polyurethane. Einhorn concluded, after his studies, that major improvements could be accomplished in the flammability characteristics of rigid polyurethane foams by modifying the chemical structure and formulation (Reference 52). However, the flexible foam system did not possess the necessary chemical structure to permit the formulation of truly flame-resistant systems.

Einhorn's conclusions seem to be validated by the results of NASA tests that exposed fire-retardant-treated, Fluorel-coated, and untreated, uncoated polyurethane foam seats to a large flaming ignition source located 12 in. below the seat cushion (Reference 53). These test results indicated that the improved

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51. Marcy, J. F., and Johnson, R., FLAMING AND SELF-EXTINGUISHING CHARACTERISTICS OF AIRCRAFT CABIN INTERIOR MATERIALS, Report No. NA-68-30 [DS-68-13], Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, July 1968, AD 673084.

52. Einhorn, I. M., Kanakia, M. D., and Seader, J. D., PHYSIOCHEMICAL STUDY OF SMOKE EMISSION BY AIRCRAFT INTERIOR MATERIALS, PART II: RIGID- AND FLEXIBLE-URETHANE FOAMS, University of Utah; Report No. FAA-RD-73-50,II, Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey, July 1973, AD 763935.

53. Fewell, L. L., et al., eds., CONFERENCE ON THE DEVELOPMENT OF FIRE-RESISTANT AIRCRAFT PASSENGER SEATS, NASA Technical Memorandum X-73144, Ames Research Center, National Aeronautics and Space Administration, Washington, D. C., August 1976, pp. 115-125.

state-of-the-art polyurethane foams without the added fire retardant and coating treatments were not significantly better than untreated, older, less fire-resistant foams. However, by treating and coating the state-of-the-art foams, production of toxic gases was delayed, and destruction of the foam was limited. It should be noted that relatively high levels of hydrogen cyanide were detected in each test, indicating that polyurethane foam may be the major contributor to similar high levels found in large-scale tests. Figures 53 and 54 show the temperatures of the top portions of the seat backs and the hydrogen cyanide concentrations during the tests.

Neoprene foam, used extensively in mattresses, has been advocated for seat cushions and is currently being used for that purpose in some mass transit vehicles. Its flammability characteristics are superior even to flame-retarded polyurethane foam. One of the earliest large-scale tests comparing neoprene foam with other seat cushioning materials was conducted by the FAA using a simulated airplane cabin. The test data comparing urethane and neoprene foam seat cushions are summarized in Table 9 (Reference 54). These tests were some of the earliest studies to document the now well-known flashover phenomenon encountered with unretarded urethane foam. Although the flame-retarded urethane foam was effective in reducing fire temperatures, neoprene was more effective. The smoke levels in these tests showed that neoprene provided significantly longer times to 50 percent smoke obscuration, a result repeated in large-scale bus and rail car tests (Reference 49).

The smoke results in the large-scale tests of neoprene foam contrast markedly with NBS smoke chamber data that indicate neoprene releases more smoke than urethane. This anomaly illustrates many of the problems in trying to extrapolate laboratory data to real fire situations. Under actual fire conditions, neoprene decomposes at a slower rate than it does in the NBS laboratory test, thus producing smoke at a lower rate. University of San Francisco tests also have shown that the neoprene foam produces less toxic smoke than does the polyurethane foam (Reference 55). Efforts now are under way to develop improved neoprene foam formulations that have superior smoke evolution properties. Table 10 summarizes comparative data on the standard and improved neoprene foams.

54. Morford, R. H., THE FLAMMABILITY OF NEOPRENE CUSHIONING FOAM, Journal of Fire & Flammability, Vol. 8, July 1977, pp. 279-299.

55. Hilado, C. J., THE CORRELATION OF LABORATORY TEST RESULTS WITH BEHAVIOR IN REAL FIRES, Journal of Fire & Flammability, Vol. 8, April 1977, pp. 202-209.

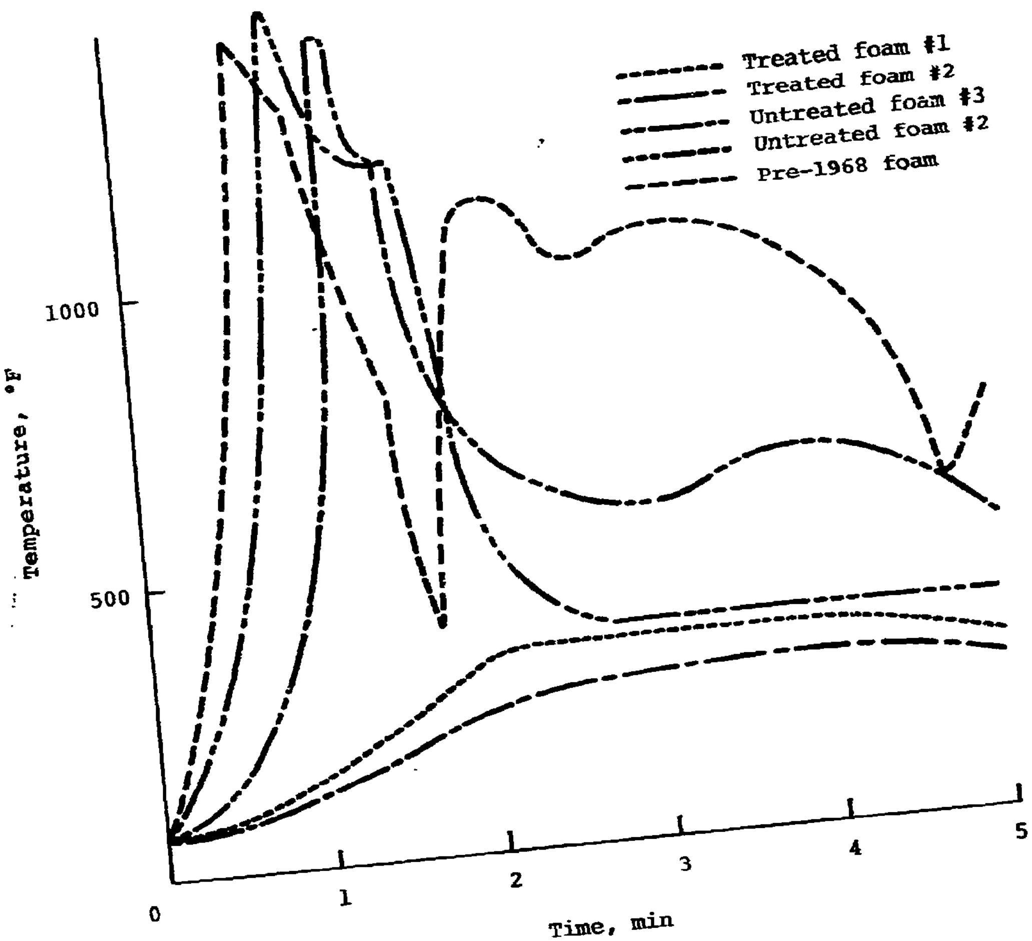

Figure 53. Seat back temperatures during burn tests of coated and uncoated polyurethane foam seats.

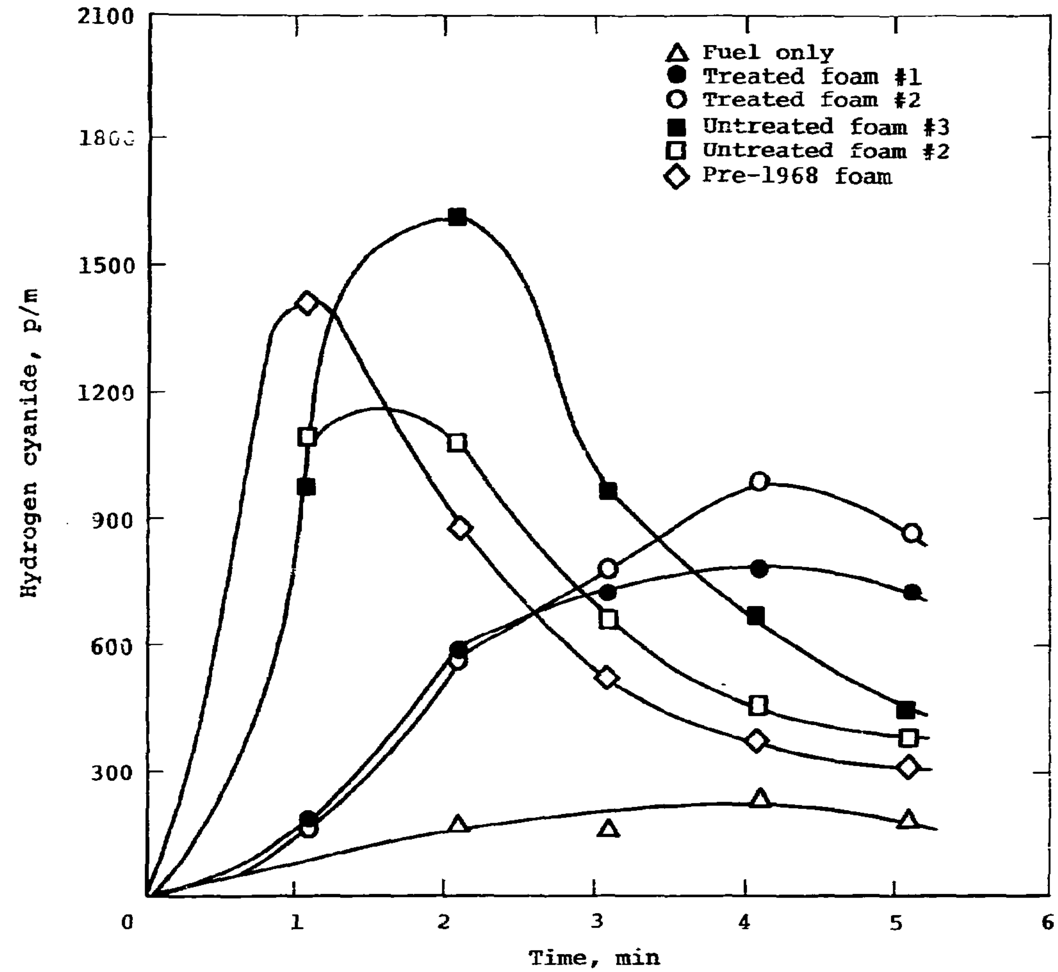

Figure 54. Hydrogen cyanide concentrations from burn tests of coated and uncoated polyurethane foams.

TABLE 9. RESULTS OF LARGE-SCALE AIR TRANSPORT CABIN FIRE TESTS USING URETHANE OR NEOPRENE SEAT CUSHIONS (FROM REFERENCE 54)

Property	Test number			
	4	11	22A	22B
Seat cushion material	Urethane foam	F.R. urethane foam	Neoprene foam	Neoprene foam
Maximum ceiling temperature, °F	1420	560	230	100
Time to 50 percent smoke obscuration, min	1.9	1.7	3.4	12.4
Minimum oxygen concentration, percent	2.5	20.0	18.3	20.5
Maximum CO concentration, percent	1.5+	0.6	0.21	0.05
Time to flashover, min	2.6	∞ (4 min)	∞	∞

TABLE 10. TYPICAL PROPERTIES OF STANDARD AND IMPROVED NEOPRENE FOAM (FROM REFERENCE 54)

Property	Standard foam	Improved foam
Density, lb/ft^3	4	7
Tensile strength, $lb/in.^2$	9	7
Elongation, percent	120	100
Compression set (50 percent deflection, 22 hr @ 212°F)	7.5	7.5
ASTM E-162-75 flame spread rating	6	4
NBS smoke chamber results		
D_s, 90 sec	280	93
D_s, 4 min	380	195
D_s, maximum	380	250
Time to maximum D_s, min	4	7

Advanced state-of-the-art materials being developed for seat cushions include polyphosphazene and polyimide foams (References 56 and 57). Both of these foams have superior flammability properties compared to typical fire-retarded urethane foams, as can be seen in Table 11. Future optimization and production of these types of foams may make them viable alternatives for improved aircraft seat cushions.

TABLE 11. COMPARISON OF POLYPHOSPHAZENE, POLYIMIDE, AND FIRE-RETARDED POLYURETHANE FOAM PROPERTIES

Property	Typical polyphosphazene foam	Typical polyimide foam	Typical F.R. urethane foam
Density, lb/ft^3	4.0 - 9.0	1.3 - 1.4	4.5 - 8.5
Tensile strength, psi	20 - 80	10 - 13	40
Elongation, percent	80 - 125	20 - 23	100
Flame spread index	14	-	30
Limiting oxygen index	43 - 45	44 - 54	20
Maximum smoke density, D_s			
Flaming	40 - 150	0 - 0.5	250
Nonflaming	-	0 - 1	-

56. Parker, J. A., Ph.D., POLYPHOSPHAZENE SEAT CUSHION APPLICATIONS, American Research Center; in Conference on the Development of Fire-Resistant Aircraft Passenger Seats, Fewell, L. L., et al., eds., NASA Technical Memorandum X-73144, National Aeronautics and Space Administration, Washington, D. C., August 1976.

57. Gagliani, J., FIRE RESISTANT RESILIENT FOAMS - FINAL REPORT, Solar Division, International Harvester; Report No. N76-18278, National Aeronautics and Space Administration, Lyndon B. Johnson Space Center, Houston, Texas, February 1976.

Another recent approach to improving the fire resistance of aircraft seats is the use of an interliner or thermal barrier between the upholstery fabric and the seat cushion foam (Reference 58). However, experimental evaluation of specific combinations of materials on systems is essential if this approach is used.

5.4.2 Upholstery and Other Fabrics

Current seat upholstery and other fabrics can be divided into two classes: uncoated and coated. Typical uncoated fabrics include wool, cotton, nylon, rayon, polyester, modacrylic, or combinations of these fibers. Marcy found that the majority of these fabrics, even when treated with flame retardants, had unacceptable burn rates during both the FAA vertical flammability test and the ASTM radiant panel test (Reference 51). Modacrylic fabrics were the only self-extinguishing fabrics. However, animal toxicity tests revealed that modacrylic fabrics were the most toxic of 75 different interior materials that were tested (Reference 47).

The coated fabrics, on the other hand, were all self-extinguishing during the vertical burn tests. These materials consisted in large part of vinyl- and acrylic-coated glass fabrics. Toxicity tests of cotton, nylon, and polyester fabrics coated with polyvinyl chloride (PVC) showed that these coated fabrics were much less toxic than their uncoated counterparts.

Advanced state-of-the-art fabrics include Nomex, Kynol, polybenzimidazole (PBI), and polyimide (e.g., Kapton) fabrics. These fabrics exhibit superior flammability characteristics, as shown in Table 12 (from Reference 59). Kynol and PBI fabrics emit very little smoke and essentially no toxic gases during burning. Nomex is readily available, while Kynol and PBI

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58. Tesoro, G. C., Ph.D., FABRICS FOR FIRE RESISTANT PASSENGER SEATS IN AIRCRAFT, Massachusetts Institute of Technology; in Conference on Fire Resistant Materials (Firemen), Kourtides, D. A., ed., NASA Technical Memorandum 78523, National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California, October 1978.

59. Tesoro, G. C., Ph.D., STATE OF THE ART MATERIALS USED IN AIRCRAFT PASSENGER SEATS, Massachusetts Institute of Technology; in Conference on the Development of Fire-Resistant Aircraft Passenger Seats, Fewell, L. L., et al., eds., NASA Technical Memorandum X-73144, National Aeronautics and Space Administration, Washington, D. C., August 1976.

TABLE 12. FLAMMABILITY PROPERTIES OF VARIOUS FABRIC MATERIALS (From Reference 59)

Property	Cotton	Polyester	Nomex	Kynol	PBI
Ignition in air					
Calrod temperature, °C	<550	-	871	788	927
Time, sec	Inst.	-	1	-	6
Flame impingement heat flux protection	Nil	(Melt)	Good	Good	Good
Char yield characteristics	Low	(Melt)	High, friable	High, strong	High, strong
Smoke	Moderate	Low	Moderate	Low	Low
Off gases (toxicity)	-	-	Toxic	CO_2/H_2O Predom.	CO_2/H_2O Predom.
Thermal stability temperature degradation, °C	-	-	437	-	590-680
Percent approximate weight loss at 900°C	-	-	60	40	30
Limiting oxygen index (percent O_2)	16 - 18	20 - 21	27 - 29	29 - 30	38 - 43

currently are subject to limited availability. However, availability of the latter two fabrics should increase as demand increases.

5.4.3 Structural Components

All major transport aircraft manufacturers and NASA are engaged in efforts to increase the fire safety of interior structural components, such as sidewall, floor, and ceiling panels. These efforts encompass the selection of single candidate materials and the fabrication of multimaterial assemblies.

Candidates for improved thermoplastic materials are listed in Tables 13 and 14, along with their physical and chemical properties (Reference 60). All of these materials exhibit greater fire resistance and lower smoke and toxicity than the majority of aircraft interior materials currently in use. Other promising materials include the char-forming polyisocyanurate and PBI foams (Reference 61). Fire-hardening of honeycomb panels has been accomplished by filling the honeycomb core with PBI or isocyanurate foam, as well as with phenolic-impregnated fiberglass batting (References 50 and 62). Further improvements can be made by replacing flammable adhesives (e.g., acrylate adhesive) with more fire-resistant compounds such as fire-retarded epoxy adhesives or polyamide adhesives (Reference 61).

5.4.4 Aircraft Insulation

Suitable insulation installed in the interior walls of crew and passenger compartments can help protect occupants from heat

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60. Silverman, B., AIRCRAFT INTERIOR THERMOPLASTIC MATERIALS, Lockheed Missiles and Space Company; in Conference on the Development of Fire-Resistant Aircraft Passenger Seats, eds. Fewell, L. L., et al., NASA Technical Memorandum X-73144, National Aeronautics and Space Administration, Washington, D. C., August 1976.

61. Parker, J. A., Ph.D., et al., FIRE DYNAMICS OF MODERN AIRCRAFT FROM A MATERIALS POINT OF VIEW, in Aircraft Fire Safety, AGARD Conference Proceedings No. 166, North Atlantic Treaty Organization, Advisory Group for Aerospace Research and Development, Neuilly-sur-Seine, France, October 1975, pp. 10.1-10.11, AD A018180.

62. Anderson, R. A., et al., EVALUATION OF MATERIALS AND CONCEPTS FOR AIRCRAFT FIRE PROTECTION, Boeing Commercial Airplane Company; Report No. NASA CR-137838, National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California, April 1976.

TABLE 13. PRELIMINARY PROPERTIES OF CANDIDATE COMPRESSION MOLDING MATERIALS

Property	Polyether sulfone	Polyphenylene sulfide	Polysulfone	Mod-polycarbonate
Tensile strength, psi	11,000	9,500	10,000	8,500
Elongation, percent	-	1.5	40	50
Flexural strength, psi	16,000	13,000	15,000	12,000
Heat deflection temperature, °F, @ 264 psi	390	275	330	270
Specific gravity	1.37	1.3	1.25	1.20 to 1.26
Impact strength (notched izod) ft-lb/in. of notch	1.6	1.5	1.3	9.0
Mod of elasticity, psi	350,000	500,000	340,000	300,000
Compressive strength, psi	12,000	15,000	13,500	12,000
Smoke density flaming, D_s (6 min.)	20	100	80	130
Limiting oxygen index (LOI)	37	44	30	23

TABLE 14. PRELIMINARY PROPERTIES OF CANDIDATE THERMOFORMED MATERIALS

Property	Mod-polycarbonate	Mod-polysulfone	Chlorinated-PVC	Mineral-filled polyethylene
Tensile strength, psi	8,500	8,000	5,400	2,300
Elongation, percent	70	-	40	200 Takes permanent set
Flexural strength, psi	12,000	12,500	10,000	3,800
Heat deflection temperature, °F, @ 264 psi	220	-	200	160
Specific gravity	1.26	1.26	1.57	1.7
Impact strength (notched izod) ft-lb/in. of notch	10.0	9.0	6.6	12.0
Mod of elasticity, psi	300,000	320,000	300,000	450,000
Smoke density flaming, D_s (6 min)	130	105	140	20
Limiting oxygen index (LOI)	23	30	42	36

generated from an exterior postcrash fire as long as the fuselage remains intact. However, not all types of insulation are suitable, and some insulations, such as glass wool or fiberglass, may worsen, rather than lessen, the postcrash fire hazard.

Recent developments in foam insulation may offer promising results for the aircraft industry. Tests conducted under the sponsorship of the NASA/Ames Research Center provided evidence of the effectiveness of a modified polyisocyanurate semirigid foam. A C-47 aircraft fuselage was separated into two zones: one a reference zone with no modification, and the other, a zone provided with the foam insulation, as shown in Figure 55. Fuel-fed ground fires were lighted next to the aircraft fuselage and allowed to burn for approximately 10 min. Temperatures recorded inside the aircraft fuselage are shown in Figure 56. Little smoke or gas was evolved. Occupant survival time inside the reference section would have been no more than 1-1/2 min, while an occupant in the insulated section could have survived for approximately 9 to 10 min. (See Section 3.3.1 for a detailed discussion of human tolerance to heat.)

The application of similar technology to rotary-wing aircraft has not been as successful. Fire tests of a CH-47 and a UH-1D helicopter showed that only limited protection could be attained for occupants for the first few minutes following the onset of a postcrash fire (Reference 63). Two main factors accounted for the limited protection attained. These were (1) the unreliability of the protective wall materials because of problems in suitably applying the materials to the helicopter wall structures, and (2) the poor fire resistance of currently used Plexiglas helicopter windows.

The walls of the two helicopters posed different insulation problems because of their different structures. Fire penetrations in the CH-47 walls occurred where the isocyanurate foam could not be applied because of the presence of wiring, air ducts, and hydraulic oil tubes. The UH-1D walls did not lend themselves to foaming because of the absence of ribs and formers. The sodium silicate hydrate panels used to protect the interior walls partially collapsed because of the absence of structural support.

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63. Atallah, S., and Buccigross, H., INVESTIGATION AND EVALUATION OF NONFLAMMABLE, FIRE-RETARDANT MATERIALS, Arthur D. Little, Inc.; USAAMRDL Technical Report 72-52, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, November 1972, AD 906699L.

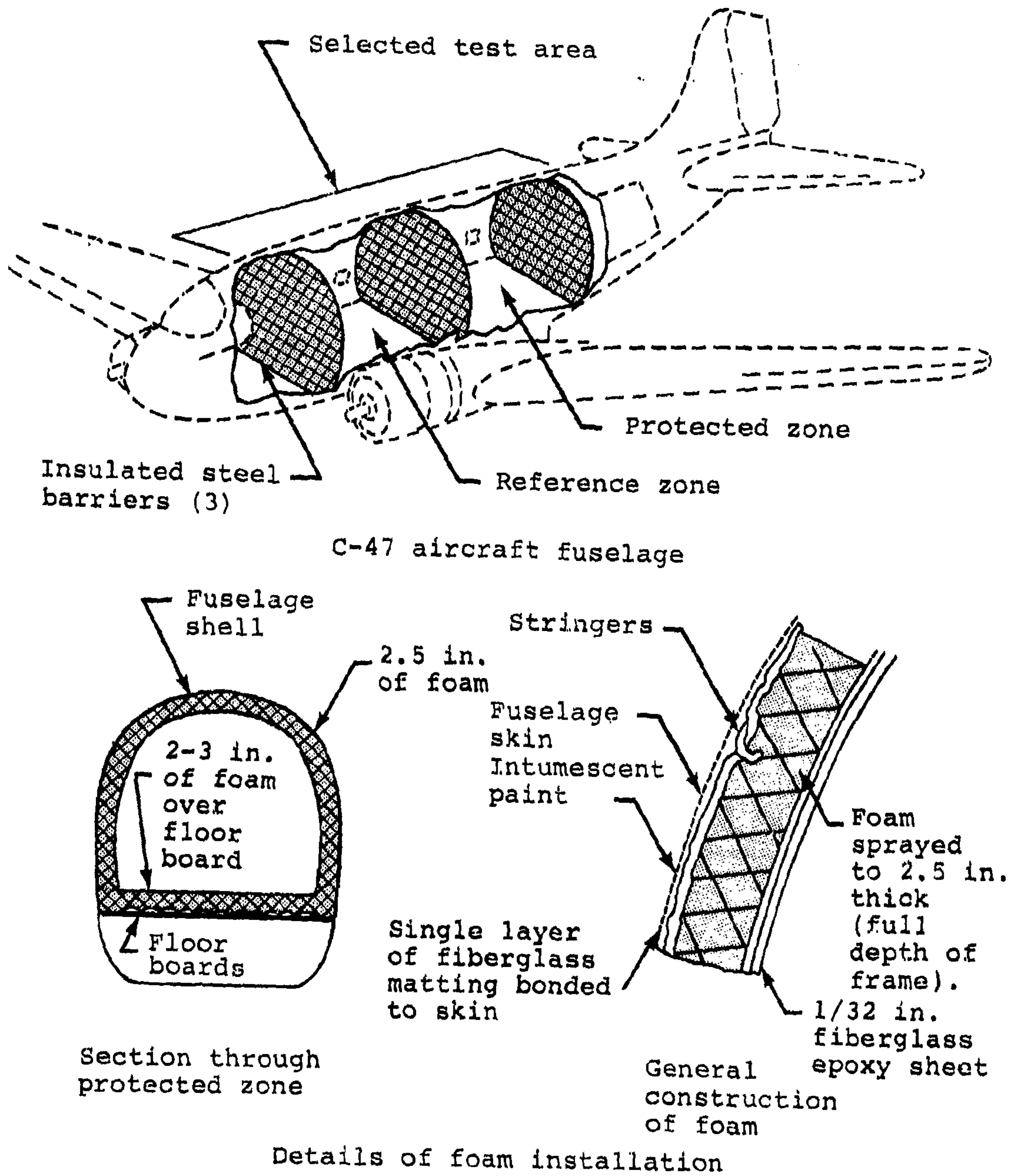

Figure 55. Semirigid foam insulation test configuration.

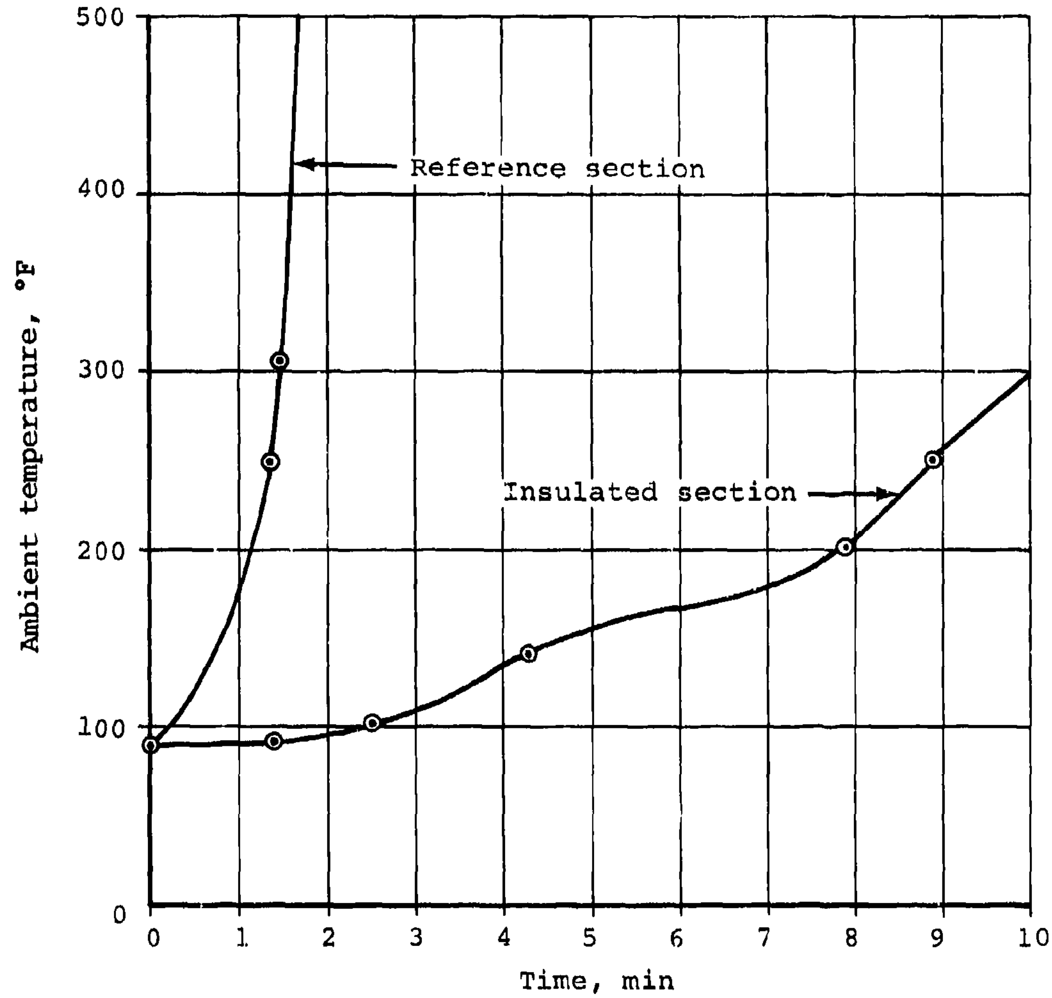

Figure 56. Compartment ambient temperature versus time from fuel ignition during foam insulation tests.

Although the postcrash fire protection was limited, in-flight simulation tests conducted during the same program indicated that it should be possible to protect the habitable compartment against a fire occurring in an adjacent compartment. Sodium silicate hydrate panels lining the fire compartment successfully contained the fire and kept temperatures in the adjacent cabin far below human tolerance levels.

Intumescent paints and coatings provide another method of thermal insulation. These fire-retardant materials react to the heat of a fire by swelling and forming a thick, low-density, polymeric coating or char layer, thus protecting the coated surface from the full effects of the fire. An intumescent paint was sprayed on the interior fuselage wall in the CH-47 foam test described above to provide a void fill between the frame-foam interface to prevent burn-through at that point.

More research must be done in this area, however, before these materials can provide postcrash fire protection by themselves. Furnace test results for eight commercially available intumescent paints and coatings indicated that none of them could provide the desired fire protection (Reference 63). Although these materials intumesced readily, the fire gases eroded the char very quickly when the coatings were exposed directly to the fire. The coatings were not effective on the nonfire side of the aluminum panel because the char could not support itself once the panel burned away. Furthermore, most of them produced noxious fumes, and therefore should not be applied in habitable compartments. These materials did show promise, however, as linings for potential fire compartments to protect occupants against in-flight fires resulting from fuel or hydraulic line leakage. Flight-critical components next to a fire zone can also receive partial protection from these paints.

5.5 MATERIAL SELECTION CRITERIA

5.5.1 General Considerations

As stated in the introduction to this chapter, the selection of interior materials is governed by several varied, and sometimes conflicting, requirements. All interior materials must effectively meet their original intended uses. For instance, seat cushions must provide a certain degree of comfort, must possess specific crashworthy characteristics of compressive modulus and rebound, and must meet minimum durability criteria. The additional requirements of low flammability and low smoke and toxic gas emissions can create a problem in finding materials that meet all of the criteria, and trade-offs must sometimes be made. The following guidelines should be considered in establishing priorities for trade-offs.

The importance of material flammability characteristics increases as the fuel load of any material(s) in the aircraft increases. Those materials most prevalent in the aircraft, such as noise insulation and interior structural components, should possess as low a flammability rating as possible. Low flammability characteristics for restraint systems, on the other hand, are not as critical because of the small amount of material used. Seat cushions and upholstery, depending on the quantity used, might fall somewhere in between in regard to the importance of their flammability characteristics. This should not be construed as deleting flammability requirements for those materials used in lesser quantities. Every effort should be made to select the least flammable material available for each end use.

Those items involved directly in crashworthiness, such as seat cushions and restraint systems, must satisfy all crashworthy requirements as a first priority. If there is no material available that can satisfy both the crash and postcrash (flammability) requirements, some reduction in optimum flammability characteristics might have to be tolerated. However, the designer should first consider protecting the more flammable material with a less flammable one. The effectiveness of the latter approach must be confirmed by flammability tests of the candidate system to ensure that the system performs as anticipated.

If materials that cannot fulfill all of the flammability requirements contained in the following section must be selected, materials that present the least amount of fire hazard should be chosen. The material should not ignite easily and should have as low a flame spread rate as possible. Care must be taken to avoid selecting a material with high flashover potential, such as a nonfire-retarded polyurethane foam.

Smoke and toxic gas emissions also should be held to minimum possible levels. Those materials known to emit significant amounts of toxic gases, such as modacrylics, should not be used.

5.5.2 Flammability Test Criteria

The large number of laboratory tests available, plus the controversy among experts as to which test or tests are preferred, has precluded the adoption of one, or even one set of tests for all materials. There are, in fact, several tests recommended for vehicle interior materials.

At the present time, the FAA flammability requirements specified in FAR 25.853 (Reference 42) are the only specific, mandatory requirements for aircraft interior materials. The FAA proposed standard for smoke emission has not yet been adopted. In 1977, the FAA held extensive hearings on the feasibility of developing a comprehensive set of integrated flammability, smoke emission, and toxic gas emission standards for interior materials in transport category aircraft. Currently, the matter is under study, and no recommendations have been issued.

Recommended specifications for transit system combustible materials have been proposed by UMTA (Reference 43). These specifications are guidelines only and do not have any official status. However, many manufacturers in the transit industry are voluntarily following these guidelines. The guidelines establish specifications for both flammability and smoke emission, but none, as yet, for toxic gas emission.

Although many of the other tests can and should be used for screening materials during the selection process, until some agreement is reached among experts and regulators as to their suitability for standards, U. S. Army criteria should be based on the FAA requirements or the UMTA guidelines. As a minimum, interior materials in all U. S. Army aircraft must meet the requirements for FAR 25.853. Those aircraft that can carry passengers (i.e., utility and cargo aircraft as opposed to attack or observation aircraft) should meet the more stringent guidelines for both flammability and smoke emission issued by UMTA.

5.5.2.1 FAR 25.853 Flammability Requirements: Materials used in each compartment occupied by the crew or passengers must meet the following requirements:

- Ceiling panels, wall panels, partitions, structural flooring, etc. Must be self-extinguishing when tested vertically by applying a 1550°F flame to the lower edge of the specimen for 60 sec. Average burn length not to exceed 6 in.; average flame time after removal of test flame not to exceed 15 sec. Drippings may not continue to flame more than an average of 3 sec.

- Floor coverings, textiles (including upholstery), seat cushions, paddings, insulations (except electrical insulation), etc. Must be self-extinguishing when tested vertically by applying a 1550°F flame to the lower edge of the specimen for 12 sec. Average burn length not to exceed 8 in., average flame time after removal of test flame not to exceed 15 sec. Drippings may not continue to flame more than an average of 5 sec.

- Acrylic windows, signs, restraint systems, etc. May not have an average burn rate greater than 2.5 in./min when tested horizontally by applying a 1550°F flame to the specimen edge for 15 sec.

The reader is referred to Reference 44 for the complete text of the regulations and test requirements.

5.5.2.2 UMTA Flammability and Smoke Emission Guidelines: Combustible materials used in transit systems are required to possess the following flammability characteristics:

- Seat cushions and insulations (except electrical insulation). Must pass ASTM E 162-67 Radiant Panel Test with a flame propagation index (I_s) not exceeding 25, with the added provision that there shall be no flaming, running, or dripping.

- Wall and ceiling panels, seat frames, partitions, etc. Must pass ASTM E 162-67 Radiant Panel Test with a flame propagation index (I_s) not exceeding 35, with the added provision that there shall be no flaming dripping.

- Upholstery materials. Burn length must not exceed 6 in. when tested by FAR 25.853 vertical test. Average flame time after removal of flame source may not exceed 10 sec. Flaming dripping not allowed.

- Carpeting (tested with its padding). Must pass NBS flooring Radiant Panel Test, NBSIR-74-495, with a minimum critical radiant flux of 0.6 W/cm².

- Plastic windows. Must pass ASTM E 162-67 Radiant Panel Test with a flame propagation index (I_s) not exceeding 100.

- Flooring. Must withstand requirements of ASTM E 119 when underside is exposed to a flame up to 1400°F for 15 min.

- Elastomers. Must pass the requirements of ASTM C542-71A, with the added requirement of no flaming dripping.

When tested in accordance with the National Fire Protection Association Standard No. 258 (1974) in both flaming and nonflaming modes, combustible materials should meet the following smoke emission requirements:

- Upholstery, air ducting, insulation (except electrical insulation). Optical density (D_s) must not exceed 100 within 4 min after start of test.

- All other materials, (except foam seat cushioning, electrical insulation, and carpeting). Optical density (D_s) must not exceed 100 within 90 sec after start of test, nor exceed 200 within 4 min after start of test.

If fire-retardant coatings are used for fabric and trim materials, the effects, if any, of routine maintenance and cleaning procedures must be assessed. If the coatings can be removed by routine cleaning procedures, the flammability and smoke/toxic fume tests should be repeated after a representative number of cleaning cycles.

The above guidelines are being voluntarily used by several transit authorities and manufacturers although the guidelines are not Government standards and have no official status. The reader is referred to Reference 43 for the complete text of the regulations and test requirements.

5.5.3 Evaluation of Materials

The flammability properties of polymeric materials currently used in commercial aircraft have been qualitatively assessed by the Committee on Fire Safety Aspects of Polymeric Materials of the National Materials Advisory Board (Reference 44). This information is presented in Table 15 to assist the designer in evaluating materials. Table 15 should not be construed as a list of acceptable materials, since the assessment is based on flammability properties only. Specific application to U. S. Army aircraft, which necessarily have different functional requirements than do commercial aircraft, must be considered in final material selection.

Also, a great deal of effort is being expended in the development of newer, less flammable, less toxic interior materials. The flammability properties of these materials are scattered throughout the literature. Fortunately for the designer, the Transportation Systems Center (TSC) of the U. S. Department of Transportation has established a computerized materials information bank that will be continually updated as new materials and testing methods are evaluated. The data bank contains flammability, smoke, and toxicity properties obtained by a variety of standard testing procedures for candidate materials. These data are supplemented with available physical and mechanical properties, as well as durability and maintainability data. The data in the bank can be accessed only from TSC, located in Cambridge, Massachusetts. The data will be made available to designers, transit authorities, and other Government agencies.

TABLE 15. FLAMMABILITY APPRAISAL OF MATERIALS USED IN COMMERCIAL AIRCRAFT (FROM REFERENCE 44)

Use	Presently Used and Available: Generally Acceptable	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Presently Used and Available: Needs Improvement	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Possible Future Development	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish
1. Acoustical thermal insulation	a. Fiberglass insulation with paper backing	+	+	+	*	+	+	Flexible polyurethane foam	–		–	–	–	–	a. Crosslinked phenolic foam	+	+	+	*	+	+
	b. Glass wool	0	+	+	*	+	+								b. Ames polyurethane foam	+	+	+	*	*	+
															c. Polybenzimidazole	+	+	+	*	*	+
															d. Polyimide	+	+	+	*	*	+
															e. Fiberfrax	+	+	+	*	*	+
2. Adhesives	a. Polyimides	0	0	0	+	0	0	a. Epoxides	0	0	0	–	*	0	a. Polybenzimidazole	+	+	+	*	*	+
								b. Neoprenes	0	0	0	–	–	0	b. Polyphenylquinoxaline	+	+	+	*	*	+
								c. Buna N phenolics	0	0	0	*	*	0							
								d. Acrylics	–	–	–	*	*	*							
3. Air duct								Urethane foam	–	–	–	–	–	–	a. Polyimide foam	+	+	+	*	*	+
															b. Polybenzimidazole	+	+	+	*	*	+
4. Baggage	a. Phenolic/glass laminate	0	0	0	0	*	0	a. PVC-coated veneer honeycomb	*	*	*	*	–	*	a. Furan/glass laminate FR paint	+	+	+	+	+	+
								b. PVF on phenolic/glass honeycomb	*	*	*	*	–	*							
5. Blankets	a. Aramid	0	0	0	0	*	*	a. Wool	–	–	0	–	–	0	a. Polybenzimidazole	+	+	+	+	*	+
	b. Aramid/phenolic	0	+	0	0	*	*	b. FR wool/rayon 85/15	0	0	0	–	–	0	b. FR modified aramid	+	+	+	*	*	+
								c. FR wool	0	0	0	–	–	0	c. Polyimide	*	+	+	+	*	*
								d. Wool/PVC	+	+	0	–	–	+							
								e. Wool/modacrylic	–	0	0	–	–	0							
								f. FR cotton	0	0	0	–	*	+							
								g. PVC/PVOH	0	0	0	–	–	0							
								h. FR modacrylic/Kynel	0	0	0	*	*	0							
								i. FR rayon	0	0	0	–	–	+							
6. Cabinet								a. FR polyethylene skins on epoxy/glass, aramid/phenolic core	0	0	0	0	–	*	a. Furan with FR epoxy covering	+	+	+	*	*	+
								b. PVF covering on epoxy/glass, aramid/phenolic core	*	0	0	*	–	*	b. Phenolic with FR epoxy covering	+	+	+	*	*	+

KEY: (–) = unsatisfactory, (+) = good, (*) = further study needed, and (0) = satisfactory.

TABLE 15. FLAMMABILITY APPRAISAL OF MATERIALS USED IN COMMERCIAL AIRCRAFT (FROM REFERENCE 44) (CONT'D)

Use	Generally Acceptable (Presently Used and Available)	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Needs Improvement (Presently Used and Available)	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Possible Future Development	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish
7. Cargo Liners	Polyimide resin/glass	+	+	+	*	*	+	a. Rigid polyurethane sheet on aramid honeycomb	0	-	0	-	-	-	a. Bismaleimide resin/glass	+	+	+	*	*	+
								b. Polyester resin laminates	*	*	0	-	*	*	b. Polybenzimidazole/glass cloth	+	+	+	+	*	+
								c. Polyurethane resin laminates	*	*	0	-	*	*							
8. Carpets (FR latex FR backing)	a. Aramid	*	*	*	-	*	*	a. Modacrylic	*	*	-	-	-	0	a. Polybenzimidazole	*	*	*	*	*	*
								b. FR wool	*	*	-	-	-	0	b. FR aramid	*	*	*	*	*	*
								c. Wool/nylon	0	-	-	-	-	*	c. Novoloid	*	*	*	*	*	*
								d. Wool	0	-	-	-	-	*							
9. Ceiling panels and covering								a. Vinyl-coated epoxy/glass laminate honeycomb	0	0	0	-	-	+							
								b. PVC-coated cotton	*	*	*	-	-	-							
								c. PVF-epoxy/phenolic over glass-filled aramid honeycomb	*	*	*	*	*	+							
10. Curtains and draperies	a. Aramid	0	0	0	0	*	0	a. Wool	-	-	-	-	-	0	a. Polybenzimidazole	+	+	+	+	*	+
	b. Fiberglass	*	*	+	+	+	*	b. Modacrylic	0	0	0	-	-	0	b. FR aramid	+	+	+	*	*	+
	c. Novoloid	0	0	0	0	*	0	c. FR wool	0	0	0	-	-	0	c. FR aramid/novoloid	+	+	+	*	*	+
	d. Aramid/novoloid	0	0	0	-	*	0	d. FR cotton	0	0	0	-	*	-							
								e. Cotton/rayon	-	-	-	-	*	-							
								f. PVOE/PVC	0	0	0	-	*	0							
								g. Modacrylic/phenolic	0	0	0	-	*	0							
11. Emergency slides								Neoprene-coated nylon	-	-	*	-	-	-	See No. 17 (life vests)						
12. Floor coverings								a. PVC laminates	0	0	0	0	-	0	FR epoxy coating	0	0	0	*	*	+
								b. Vinyl/ABS	0	-	-	-	-	-							

KEY: (-) = unsatisfactory, (+) = good, (*) = further study needed, and (0) = satisfactory.

TABLE 15. FLAMMABILITY APPRAISAL OF MATERIALS USED IN COMMERCIAL AIRCRAFT (FROM REFERENCE 44) (CONTD)

Use	Presently Used and Available: Generally Acceptable	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Presently Used and Available: Needs Improvement	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Possible Future Development	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish
13. Floors	a. Metal-faced balsa	+	+	0	*	+	*	a. Epoxy/fiberglass aramid honeycomb	0	+	0	–	*	0	a. Polyimide/fiberglass-faced aramid honeycomb	+	+	0	*	*	+
	b. Metal-faced aramid honeycomb	+	+	0	*	*	*	b. Phenolic-faced Styrofoam	0	*	*	–	*	*	b. Phenolic/fiberglass-faced aramid honeycomb	+	+	0	*	*	+
															c. Furan/fiberglass-faced honeycomb	+	+	0	*	*	+
14. Hoses	Silicone	0	0	*	0	0	*	a. Neoprene	0	0	*	–	–	–	a. Phosphazenes	+	+	+	*	*	+
								b. Buna N	0	0	*	–	–	*							
								c. Viton	0	0	*	0	–	0							
15. Lavatory paper								Paper and tissues	–	–	–	0	0	–	FR paper: improved dispensing and discard	+	+	+	+	*	+
16. Life rafts								a. Urethane-coated nylon fabric	–	–	–	–	–	–	See No. 17 (life vests)						
								b. Neoprene-coated nylon fabric	–	–	*	–	–	–							
17. Life vests								a. Neoprene-coated nylon fabric	–	–	*	–	–	–	a. FR neoprene-coated nylon fabric	+	+	+	–	–	+
								b. Urethane-coated nylon fabric	–	–	–	–	–	–	b. FR neoprene-coated aramid fabric	+	+	+	–	–	+
															c. Fluoroelastomer on aramid	+	+	+	*	–	+
															d. Phosphazene-coated aramid	+	+	+	*	*	+
18. Movie film								a. Polyester	–	–	–	–	*	–							

KEY: (–) = unsatisfactory, (+) = good, (*) = further study needed, and (0) = satisfactory.

TABLE 15. FLAMMABILITY APPRAISAL OF MATERIALS USED IN COMMERCIAL AIRCRAFT (FROM REFERENCE 44) (CONTD)

Use	Presently Used and Available: Generally Acceptable	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Presently Used and Available: Needs Improvement	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Possible Future Development	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish
19. Paints (must be considered with substrate)								a. Nitrocellulose, tung oil lacquers	-	-	-	-	-	-							
								b. Alkyds and urethane	-	*	0	-	*	*							
								c. Epoxy and polyester primes	-	,	0	-	*	*							
								d. Fire-retardant paints	0	0	0	-	*	0							
								e. Intumescent paints	0	0	C	-	*	C							
20. Partitions, clothes closet, and side walls								a. PVC or PVF on phenolic/fiberglass honeycomb	*	0	0	*	-	*	a. FR epoxy coating, furan/glass face, phenolic honeycomb	0	0	0	*	*	*
								b. PVC laminate over veneer-faced honeycomb	*	*	*	-	-	*	b. FR epoxy coating phenolic/glass face phenolic/glass honeycomb	0	0	0	*	*	*
								c. PVC laminate over epoxy/glass-faced honeycomb	*	*	*	-	-	*	c. Same as "a" with aramid honeycomb	0	0	0	*	*	*
								d. FR epoxy, aramid/phenolic core, PVF film	0	0	*	*	*	*	d. Same as "b" with aramid honeycomb	0	0	0	*	*	*
21. Pillows	Cover//fill							Cover//fill							Cover//fill						
	a. Aramid//aramid	0	0	0	-	*	0	a. Cotton//PET	-	-	-	-	-	-	a. Polybenzimidazole//novoloid	+	+	+	*	*	+
	b. FR cotton//aramid	0	0	0	-	*	0	b. Cotton//polyurethane	-	-	-	-	-	-	b. FR aramid//novoloid	+	+	+	*	*	+
	c. Aramid//FR cotton	0	0	0	-	*	0	c. FR paper//PET	-	-	-	-	-	-							
	d. Novoloid/aramid//aramid/novoloid	0	0	0	-	*	0	d. FR cotton//FR cotton	0	0	*	-	-	0							
								e. FR paper//PVC/PVOH	0	0	*	-	-	0							
22. Sealants								a. Polysulfide	-	-	*	*	0	-	Polyphosphazenes	+	+	+	*	*	+
								b. Fluorosilicone	0	0	*	C	-	0							
								c. Viton	0	0	*	*	-	0							
23. Seat belts	a. Nylon	0	0	0	0	*	0	Nylon/rayon	-	-	*	*	*	-	Polybenzimidazole	+	+	+	+	*	+
	b. Aramid	0	0	0	-	*	0														
24. Seat cushions (foam)								FR polyurethane foam	-	-	-	-	-	-	a. Steel spring/FR batting	+	+	+	*	*	+
															b. Phosphazene foams	+	+	+	*	*	+

KEY: (-) = unsatisfactory, (+) = good, (*) = further study needed, and (0) = satisfactory.

TABLE 15. FLAMMABILITY APPRAISAL OF MATERIALS USED IN COMMERCIAL AIRCRAFT (FROM REFERENCE 44) (CONCLUDED)

	Presently Used and Available																				
Use	Generally Acceptable	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Needs Improvement	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish	Possible Future Development	1. Ease of Ignition	2. Flame Spread	3. Heat Release	4. Smoke	5. Toxicity	6. Ease of Extinguish
25. Seat upholstery	a. Aramid	0	0	0	-	*	0	a. Wool	0	0	-	-	-	*	a. Polybenzimidazole	+	+	+	*	*	+
	b. Aramid/novoloid	0	0	0	-	*	0	b. FR wool	0	0	0	-	-	0	b. FR aramid	+	+	+	*	*	+
								c. Rayon/nylon	-	-	-	-	*	-							
								d. Nylon	0	-	-	-	*	*							
								e. FR cotton/rayon	0	-	-	-	*	*							
								f. Nylon/neoprene	0	-	-	-	-	-							
								g. FR rayon/novoloid	0	0	0	*	*	0							
								h. Modacrylic/aramid	0	0	0	-	-	0							
								i. PVC/PVCH	0	0	0	-	-	0							
26. Seat trim	a. Polyphenylene oxide	0	0	*	*	*	*	a. ABS/vinyl	-	-	-	-	-	-							
	b. Polysulfone	0	0	*	*	*	*	b. ABS/polyester	-	-	-	-	-	-							
								c. ABS	-	-	-	-	-	-							
								d. PVC-acrylic alloy	0	*	*	-	-	*							
27. Thermoplastic parts	a. FR polycarbonate	+	*	0	*	*	*	a. Bisphenol A polycarbonate	0	*	*	-	*	*							
	b. FR PPO blends	0	0	0	0	*	-	b. PPO blends	0	0	0	-	*	-							
								c. FR-ABS	0	-	-	-	-	-							
28. Trash can liner								Polyethylene	-	-	-	0	0	-							
29. Wall covering								PVF over PVC film	0	0	0	-	-	0							
30. Windows (transparent aircraft enclosures)								a. Acrylates	-	-	-	-	*	*	a. Epoxy-TMB	*	*	*	*	*	+
								b. Bisphenol A polycarbonates	0	*	*	-	*	*	b. Phenolphthalein polycarbonate	+	+	+	*	*	*
															c. Polyarylsulfone	+	+	+	*	*	+
31. Window shade assembly	a. FR bisphenol A polycarbonate	0	0	*	*	*	0	Bisphenol A polycarbonate	0	*	*	-	*	*	Furan/glass laminate, FR coating	+	+	+	*	*	+
	b. Polysulfone	0	0	*	*	*	0														
32. Wire insulation	a. Polyimide with Fluorocarbon outer jacket	*	*	+	+	*	+	a. PVC with fiberglass and nylon jacket	*	*	*	-	-	*	Fluoroelastomer with fiberglass sleeving	+	+	+	+	*	+
	b. Silicone-covered fiberglass sleeve	+	+	+	+	+	+	b. Polytetrafluoroethylene	+	+	+	+	*	+							
								c. Polychlorotrifluoroethylene (as conduit)	+	+	+	+	*	+							

KEY: (-) = unsatisfactory, (+) = good, (*) = further study needed, and (0) = satisfactory.

6. DITCHING PROVISIONS

6.1 INTRODUCTION

Since U. S. Army aircraft are frequently flown over water, unplanned water landings are not uncommmon. The crash parameters, survival envelope criteria, and human tolerance limits presented in Volume 2 of this design guide are equally applicable to water and land impacts. However, the water environment during the postcrash phase presents additional unique problems that do not occur on land. This chapter addresses those problems and offers general design concepts and guidelines to increase occupant survival in ditching situations.

6.2 THE DITCHING ENVIRONMENT

An aircraft ditching is a forced landing of an aircraft in the water. It is not to be confused with an uncontrollable crash into a water environment. Ditching is a premeditated maneuver deliberately executed by the pilot with the specific intention of abandoning the aircraft. In general, it is an act that offers reasonable hope of escape and survival. In fact, premeditated ditchings should have an equal or greater number of survivors than forced landings on land if adequate postcrash survival provisions are present. Analysis of the ditching environment shows survival can be enhanced by adequate (large, numerous) egress openings, highly visible lighting around escape openings, and, especially for passenger-carrying helicopters, truly effective aircraft flotation devices.

6.2.1 Aircraft Configuration and Survivability Characteristics

The majority of fatalities in light fixed-wing and rotary-wing aircraft ditchings are due to drowning. However, the behavior of the aircraft and consequent egress difficulties vary somewhat between the two different aircraft configurations.

6.2.1.1 Fixed-Wing Aircraft: Fixed-wing aircraft generally will remain afloat for a sufficient length of time to permit occupant evacuation. In a study of 306 light aircraft ditchings, Snyder and Gibbons found that, although actual flotation times were not clear in many cases, the known data indicated 90 to 95 percent of the aircraft stayed afloat long enough for safe egress (Reference 64). This finding is reflected in the

64. Saczalski, K., et al., eds., AIRCRAFT CRASHWORTHINESS, Charlottesville, Virginia, University Press of Virginia, 1975, pp. 121-139.

relatively high survival rates determined from the study: 88.5 percent survival for both pilots and passengers. The authors also concluded that at least 50 percent of the resulting fatalities were caused by drowning after a successful egress. Thus, fatalities were related more often to lack of emergency personnel flotation devices than to impact trauma or egressing difficulties.

Aircraft configuration seems to be a factor in ditching incident survival. This same study determined that fixed-gear aircraft, whether high- or low-wing, are less successfully ditched than retractable-gear configurations. Occupants of high-wing, multiengine aircraft seem to have significantly less chance of surviving a ditching than do occupants of other types of fixed-wing aircraft.

6.2.1.2 Rotary-Wing Aircraft: Unplanned landings on water are difficult for rotary-wing aircraft because their high center-of-gravity configurations are inevitably unstable in this environment. The rotors cannot be relied upon to help keep the aircraft upright since waves may induce an early rolling tendency, causing the rotor blades to strike the water. Also, compressor blades very often become salt-incrusted and stall shortly after touchdown. In addition, a significant number of helicopter ditchings involve autorotation onto the water. Flaig, in a study of U. S. Navy helicopter ditchings from 1960 through 1974, found that 24 percent of controlled, unplanned water landings involved autorotation (Reference 65).

As with fixed-wing aircraft, most fatalities in helicopter ditchings are due to drowning. During a study of 78 Navy helicopters involved in water accidents resulting in the loss of 63 lives over a 4-year period, it was found that only 10 deaths were due to injuries (Reference 66). Twenty-five deaths were attributed to drownings and the remaining 28 were lost at sea. Twenty-one, or 40 percent, of those recovered drowned or lost at sea were last seen still in the aircraft. The overall survival rate seemed to correlate with the helicopter flotation time, as shown in Table 16.

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65. Flaig, J. W., HELICOPTER FLOTATION AND PERSONNEL SAFETY IN UNPLANNED WATER LANDING FROM FY 1960 THROUGH FY 1974, Letter Report 75-61, Naval Air Systems Command, Washingtion, D. C., June 1975.

66. Rice, E. V., and Greear, J. F. III, UNDERWATER ESCAPE FROM HELICOPTERS, Naval Safety Center, Norfolk, Virginia; paper presented at the 11th Annual Symposium of the Survival and Flight Equipment Association, Phoenix, Arizona, October 7-11, 1973.

TABLE 16. SURVIVAL RATE VERSUS HELICOPTER FLOTATION TIME (FROM REFERENCE 65)

Flotation time	Survivors	Fatalities	Percent Fatalities
<1 min	165	26	13.6
1-5 min	42	4	8.7
>5 min	83	5	5.7

In correlating fatality rates with specific helicopter models, however, Flaig found that the helicopter flotation capability did not correlate with the fatality rate of occupants who survived the impact but perished because the helicopter did not stay afloat long enough (Reference 65). This seeming inconsistency results from the finding that, in larger helicopters (more than four crew members), safety decreases faster with the number of people than it increases with relatively good flotation. Another finding, which bears on the issue of helicopter size versus flotation, is that passengers were much more likely to be fatalities than regular crew members. Of particular significance is the fact that 76 percent of the crew fatalities and 92 percent of the passenger fatalities were due to drowning.

6.2.2 Underwater Escape

Since the majority of ditched helicopters roll inverted and sink in only a few minutes, inrushing water might be expected to hinder emergency egress. Interviews of helicopter ditching survivors have confirmed this supposition, with inrushing water reported as a deterrent to escape far more frequently than any other problem (Reference 66). Inrushing water was the only egress problem encountered by 43 survivors. However, in addition, it was reported in conjunction with several other egress problems, as shown below:

Egress Problem	Number of Survivors
Inrushing water only	43
Inrushing water plus:	
Reaching hatch	34
Confusion/disorientation	26
Releasing hatch	16
Darkness	12
Fire/smoke/fuel	11
Releasing restraints	9

To determine the effectiveness of escape hatch illumination on ease of egress from submerged, inverted helicopters, simulation tests were conducted by the Naval Submarine Medical Research Laboratory utilizing trained divers (Reference 67). Tests were conducted using three different window escape hatches under day and night, light and no-light conditions. The only two variables that showed statistically significant effects were the window used for egress and the presence or absence of window lighting. (The lighting consisted of high intensity electroluminescent lights at the tops and sides of the windows.)

The one window showing significantly longer egress times required the occupant to move the seat back support from the window in order to egress, and to exit from the window without striking a sponson support.

More rapid egress occurred when the windows were illuminated than when they were not. There was no significant difference between the speed of night and day egress under either the light or no-light conditions. Even with the use of trained divers and controlled conditions, there were 16 recorded instances when subjects became disoriented, lost, and/or entangled within the helicopter. Fifteen of these instances occurred in the absence of illumination, and one occurred with the lights on.

6.3 EMERGENCY EGRESS OPENINGS

6.3.1 General Provisions

Emergency escape provisions are discussed in detail in Chapter 7. Although the provisions in Chapter 7 apply to all aircraft, the unique problems encountered in escaping a ditched aircraft, especially rotary-wing aircraft, dictate special consideration for egress openings. Maximum egress time prior to helicopter rollover into an inverted position and submergence can vary from a few seconds to a few minutes. Therefore, occupant survival is highly dependent on egressing from the aircraft in a timely manner.

Since the ditching survival rate is dependent on the number of occupants in rotary-wing aircraft, more and larger emergency exits should be provided in passenger-carrying helicopters than might normally be provided. The configuration of each aircraft

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67. Ryack, B. L., et al., THE EFFECTIVENESS OF ESCAPE HATCH ILLUMINATION AS AN AID TO EGRESS FROM A SUBMERGED HELICOPTER, Report No. 856, Naval Submarine Medical Research Laboratory, Groton, Connecticut, November 1977.

model dictates the potential available escape routes. Consideration should be given to providing additional escape hatches, which can be opened if necessary, in the overhead, deck, and tail sections to facilitate escape, especially if the aircraft sinks on its side.

6.3.2 Explosively Cut Exits

Explosive cutting charges can be used to provide quick-opening emergency exits in downed aircraft. These systems definitely should be considered for use in passenger-carrying helicopters operating over water environments. Their rapid initiation time (less than 0.1 second) and immunity to the crash environment would provide the rapidity of opening and accessibility required of emergency exits in unplanned water landings.

Linear shaped charges should be placed around and extend beyond existing windows and hatches to preclude the problem of jammed or stuck exits. Strategically placed shaped charges in the overhead, deck, and empty bulkhead spaces could provide the additional emergency exits required in the ditching environment. Each exit should be capable of being actuated manually and independently from the rest so that only desired exits are opened, since opening of submerged exits may result in more rapid sinking. However, automatic actuation by water pressure could be used after all exits are submerged.

A detailed discussion of explosively created exit systems may be found in Section 7.2.7.

6.4 UNDERWATER EMERGENCY LIGHTING

Adequate emergency exit lighting is essential for rapid evacuation of any aircraft under conditions of reduced visibility. It is critical in the ditching environment because of the disorientation of aircraft occupants and the limits of underwater visibility, even during daylight conditions. The following sections discuss the particular problems of underwater visibility and the criteria necessary for adequate emergency exit lighting under water. Emergency lighting in general is discussed in Section 7.3.

6.4.1 Underwater Visibility

The ability of an observer to detect an object depends not only on the intensity of illumination but also on the visual threshold of the observer's eye. Smith, et al., found that luminance* thresholds in water are higher than those in ambient air by about 1.5 log units (Reference 68). A principal reason for this is the loss of the unprotected eye's focusing power in the water. This loss produces severe hyperopia; that is, the focal length of the eye is increased and the viewed target cannot be brought into focus within the plane of the retina. In water one does not see a sharply defined target light, but rather a diffuse blur whose apparent size is much larger than it would be if viewed in the air. The increase in size has the effect of spreading the light intensity over too large an area to be compensated for by spatial summation in the retina, thus resulting in an increase in the luminance required for detection.

The initial adaptation level of the eye also influences the luminance threshold. When an observer looks from a brighter field to a dimmer field, his eyes must adapt to the change in light intensity. Thus, increased target brightness or longer viewing times are required to compensate for the temporarily lower threshold sensitivity experienced when changing from higher to lower adaptation levels. The rate of adaptation in water parallels that in air, as illustrated in Figure 57 (Reference 68). Thus, the difference in visibility thresholds between air and water mediums is approximately 1.5 log units at all adaptation levels.

6.4.2 Emergency Lighting Requirements

Since the curves for threshold luminance in air can be used to predict the sensitivity of the eye in water if the curves are shifted downward by 1.5 log units, Smith, et al., have proposed the following method for determining light levels necessary for helicopter escape (Reference 68).

Bouguer's exponential law of absorption may be used to obtain the luminance (L) required of a light source to be just visible at a distance (d) by an individual whose threshold sensitivity is (S) in water with attenuation coefficient (a):

*Luminance is the photometric brightness or luminous intensity of a surface in a given direction per unit of projected area. It is measured quantitatively in foot-lamberts (fL) or candelas per square meter (cd/m^2). One foot-lambert = 3.426 cd/m^2.

68. Smith, P. F., Lurial, S. M., Ryack, B. L., LUMINANCE-THRESHOLDS OF THE WATER-IMMERSED EYE, Report No. 857, Naval Submarine Medical Research Laboratory, Groton, Connecticut, March 1978.

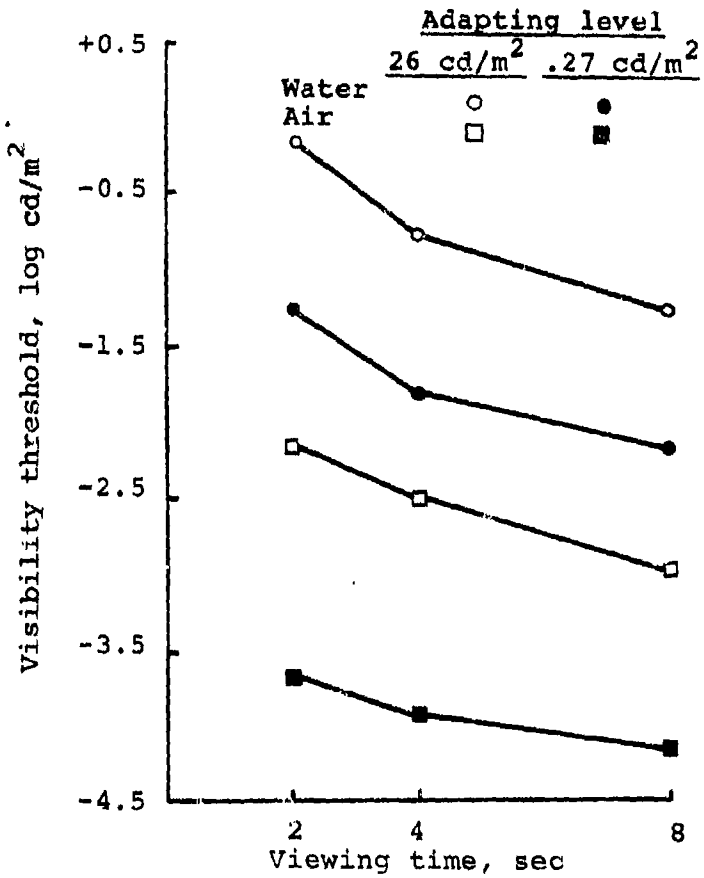

Figure 57. Comparison of luminance thresholds at two adapting levels in water and in air.

$$L = SV \qquad (3)$$

where $V = e^{ad}$

The attenuation coefficient varies with climatic and water conditions. Representative values of the coefficient (a) are shown in Table 17. In open water, the coefficient (a) generally varies from 0.08 to 0.125. Values in harbors, bays, and gulfs may vary from 0.167 to 0.7, while estuaries and coastal waters tend to be much more turbid. Conditions within ditched helicopters may be such that the coefficient depends more on debris or oil rather than the water in which it is ditched, but this factor has not been evaluated.

TABLE 17. REPRESENTATIVE ATTENUATION COEFFICIENTS (a) FOR VARIOUS WATER SOURCES

Water source	a	Year of determination
Pacific Countercurrent	0.083	1951
Pacific North Equatorial Current	0.083	1951
Gulf of Mexico (Panama City)	0.100	1967
Pacific South Equatorial Current	0.111	1951
Caribbean Sea	0.125	1951
Caribbean Sea (Roosevelt Roads)	0.300	1969
Long Island Sound	0.700	1967
Thames River (Connecticut)	3.500	1969

Viewing distance (d) will vary with seating arrangements and escape hatch placement.

Sensitivity (S) will vary among the aircraft crew and flight conditions. Occupants not looking outside the aircraft may be exposed to adapting fields of 15 to 50 cd/m^2. Pilots and crew members who must look outside the aircraft will be exposed to much higher levels. For example, a pilot flying in a hazy sky can experience 25,000 to 35,000 cd/m^2. During ditching, pilots will be looking at the water, which is generally less bright than the sky, and adaptation levels will be reduced to approximately 350 cd/m^2.

The minimum output levels (L) for escape hatch lights may be determined by substituting appropriate values of V and S in Equation (2). The value of V can be found from Figure 58, which gives values of V for various attenuation coefficients and the distance from the light to the observer. Figure 59 provides threshold sensitivity levels for elapsed times following extinction of an adapting field with luminances from 0.35 to 350 cd/m^2.

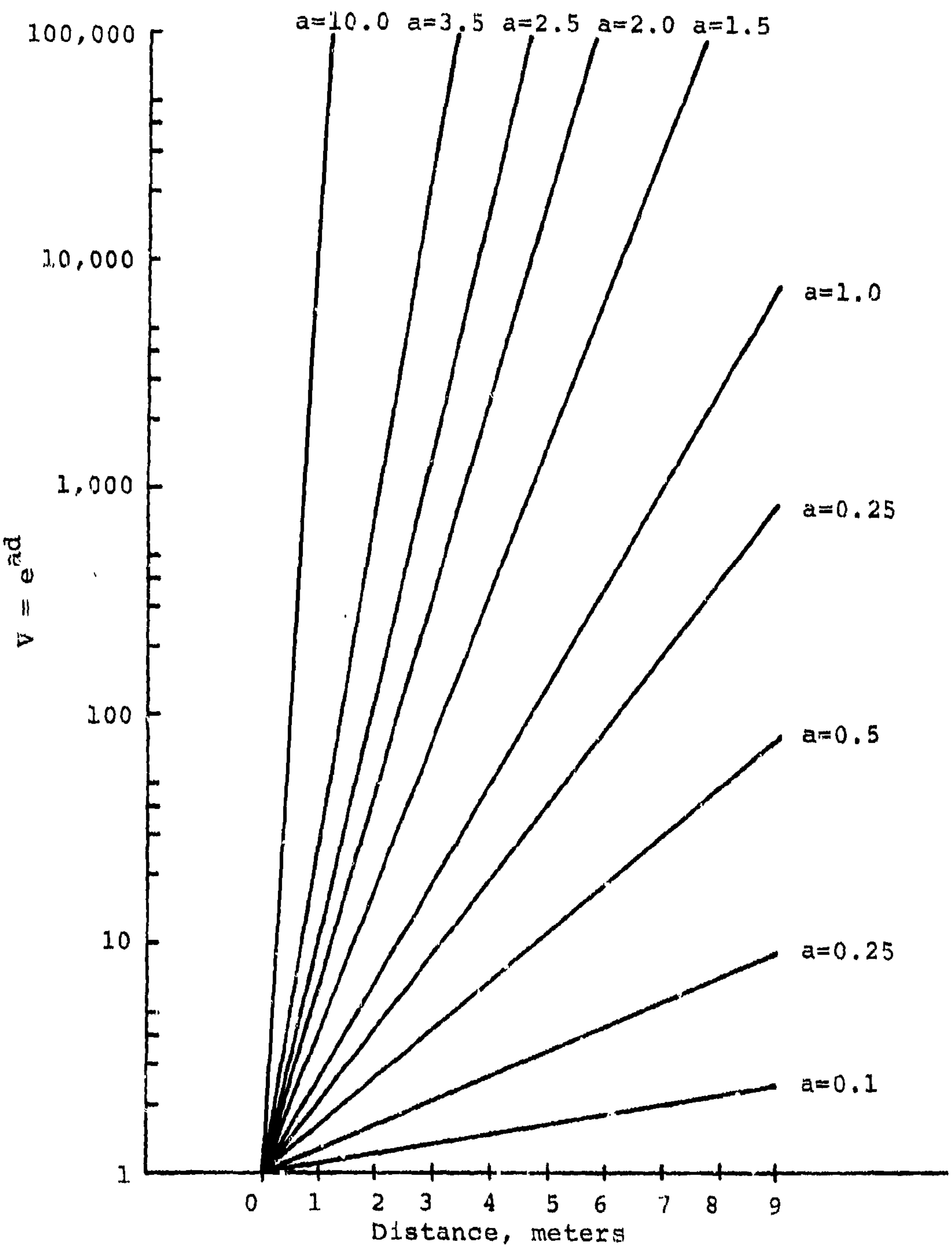

Figure 58. Values of V in Equation (3) for distances from light source to observer with various water attenuation coefficients (a). (From Reference 68)

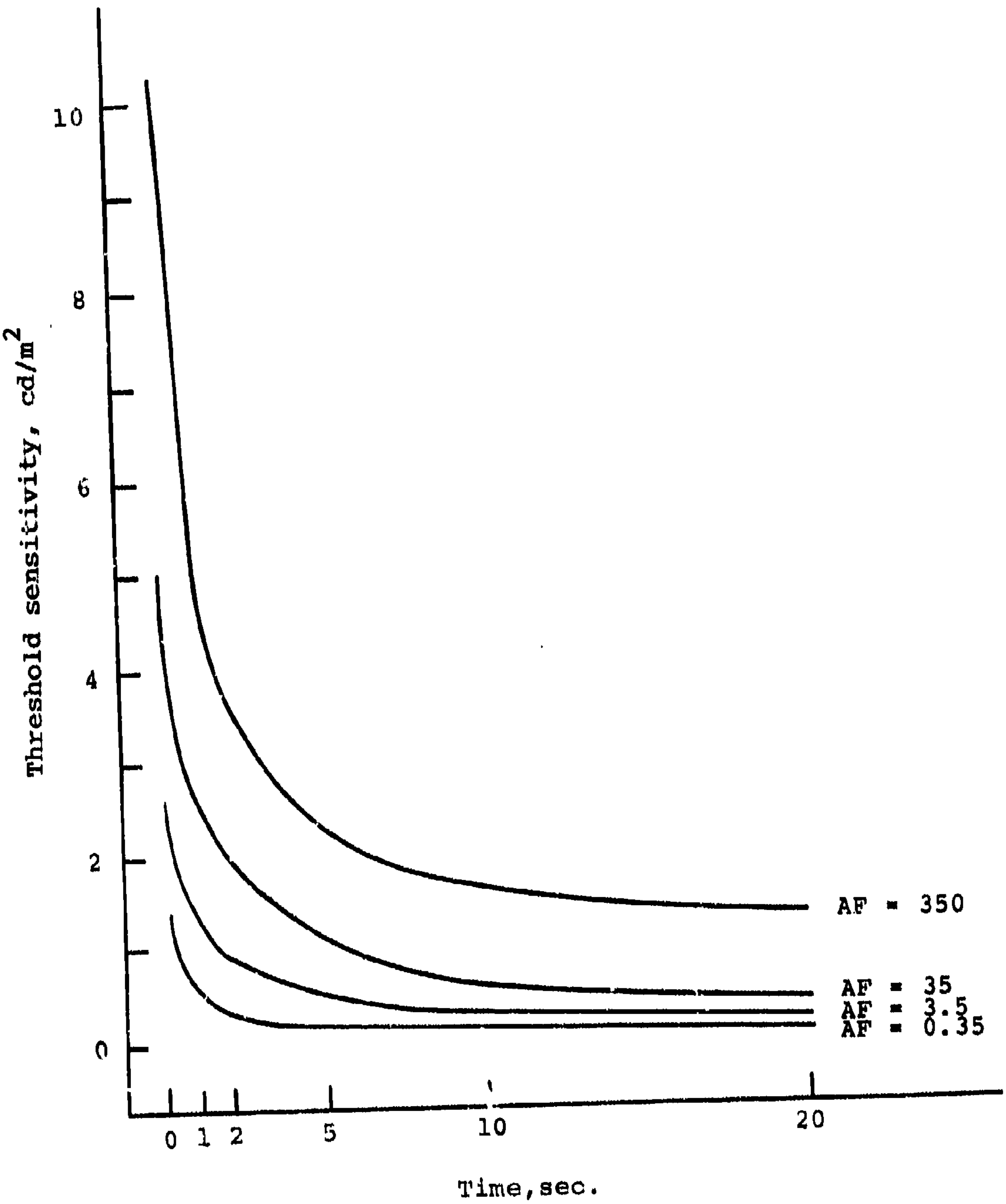

Figure 59. Threshold sensitivity (S) levels for elapsed time after ditching for adaption fields (AF) from 0.35 to 350 cd/m². (From Reference 68)

A degree of latitude is available in selecting the sensitivity level. Several seconds are required during a ditching for conditions to stabilize enough to permit escape. During this time, ambient light levels are dropping, thus lowering the adaptation levels. Requiring the lights to be visible 2 sec after ditching, rather than immediately, significantly lowers the threshold sensitivity (S), as may be seen in Figure 59. Consequently, the lower value of S will reduce the required brightness of the exit lights.

Table 18 lists the values of L obtained from Equation (3) when the adapting level is 350 cd/m^2 and the light must be visible immediately upon ditching. The effect of increasing turbidity (higher attenuation coefficient) on light level requirements is apparent. The calculated luminance (L) is the minimum value at which the light is just barely visible. The higher the value that L is above the threshold level, the greater the probability of detection. Therefore, the highest brightness level of light permitted by other design conditions should be employed for the escape lighting system. Based on the data in Table 18 and the sizes of passenger-carrying U. S. Army helicopters, the minimum brightness of the lights should be at least 120 fL (411 cd/m^2).

TABLE 18. EXAMPLES OF VALUES FOR MINIMUM LUMINANCE (L) IN cd/m^2 FOR ESCAPE HATCH LIGHTS AT DISTANCES TO LEVEL OF 350 cd/m^2 (From Reference 68)

Distance (meters)	Attenuation coefficients		
	2.5	1.0	0.5
0.5	34	17	14
1.0	127	27	17
1.5	439	45	21
2.0	1,538	75	27
2.5	5,331	127	34
3.0	18,610	206	45
3.5	64,950	339	58
4.0	*	562	75
5.0	-	1,528	127
6.0	-	4,152	206
7.0	-	11,289	339
8.0	-	30,680	562
9.0	-	83,399	925

*Values below here become very large and are perhaps prohibitive.

6.5 AIRCRAFT FLOTATION SYSTEMS

Several methods currently being used in attempts to provide ditched helicopters with flotation capabilities include inflatable bags, large sponsons, sealed hulls, and combinations thereof. Some of these methods have not been particularly successful in preventing postcrash fatalities, since they were unable to provide adequate flotation times for the escape of all occupants from larger helicopters. For instance, although one type of Navy helicopter has floated upright for more than 2 min in 70 percent of its ditchings, it has a high fatality rate (Reference 65).

If large numbers of people are to be carried, the flotation provisions must be very effective to lower the fatality rate. As might be expected, the number of inadequate flotation incidents will decrease as more flotation provisions are incorporated in any given helicopter. Thus, consideration should be given to using a combination of flotation methods, such as sponsons in conjunction with flotation bags, sealed hulls, etc.

6.5.1 Sponsons

Although sponsons are not intended to permit extended periods of operation on water, they can help stabilize the aircraft in relatively calm seas. However, to be of any value in providing flotation, the sponsons must be quite large to counteract the inherent instability due to a helicopter's high center of gravity.

The sponson buoyancy required to stabilize an aircraft for small angles of rotation may be estimated by using the following equation (Reference 69):

$$\tan\theta = \frac{F_s e}{dW} \qquad (4)$$

where θ = heeling angle, deg

F_s = maximum single sponson buoyance, lb

e = horizontal distance from aircraft centerline to the center of buoyancy of the sponson, ft

d = vertical distance of the aircraft center of buoyancy to the aircraft center of gravity, ft

W = normal gross weight of the aircraft, lb

The heeling angle calculated for Equation (4) should be verified by data from tests performed on the aircraft or on a scale model thereof.

6.5.2 Flotation Bags

Inflatable gas bag flotation systems have been developed and are currently being used on several aircraft. Their success to date, however, has been limited. Reliability problems have yet to be solved satisfactorily (Reference 65). In addition, buoyancy requirements of truly effective flotation bags pose design problems relative to the size and location of the deployed bags.

The flotation bag buoyancy required to stabilize a helicopter to any desired heeling angle may be estimated from the following equation (Reference 69):

$$\tan \theta = \frac{F_s e_s + F_b e_b}{dW} \qquad (5)$$

where θ = heeling angle, deg

F_s = maximum single sponson buoyancy, lb

e_s = horizontal distance from aircraft centerline to center of buoyancy of the sponson, ft

F_b = buoyancy of inflated bag, lb

e_b = horizontal distance from aircraft centerline to center of bag, ft

d = vertical distance of aircraft center of buoyancy to aircraft center of gravity, ft

W = normal gross weight of aircraft, lb

As may be seen, the maximum heeling angle determined from Equation (5) is dependent on the buoyant force of the bag (bag size) and the distance of the bag from the aircraft.

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69. Saczalski, K., et al., eds., AIRCRAFT CRASHWORTHINESS, Charlottesville, Virginia, University Press of Virginia, 1975, pp. 645-667.

To achieve maximum effectiveness, the bags should be inflated simultaneously, prior to or at water contact at a low airspeed. Reliability considerations of the flotation system are of prime importance. The failure of both bags to inflate, or the separation of both bags from the aircraft upon water contact, will destroy any effectiveness the system might have. Moreover, the loss of buoyancy on one side could cause the aircraft to list and possibly sink faster than it would without the system.

6.6 DITCHING EQUIPMENT

Suitable tiedown or stowage facilities should be provided for life rafts, life preservers, survival kits, and miscellaneous ditching equipment. Restraint devices and supporting structures for equipment should be designed to restrain the equipment to static loads of 50 G downward, 10 G upward, 35 G forward, 15 G aftward, and 25 G sideward. All survival equipment should be readily available and easily released from their restraining devices by the occupants after ditching. More details on the design requirements for containing emergency equipment may be found in Volume 3 under Aircraft Ancillary Equipment.

Provisions for carrying life rafts should be included in all aircraft whose mission requires frequent flight over water, especially if the aircraft mission also includes troop transport. Research has shown that individuals are not able to tolerate exposure in 32°F (0°C) water for more than 90 min, or 50°F (10°C) water temperatures for more than 18 hr (Reference 70). Figure 60 shows that a life raft between the sea and the individual provides a significant buffer that extends the tolerance time for a period of days. A raft with an effective spray canopy can make the difference in survival of aircraft occupants in the sea.

The design and location of life raft mountings or restraining devices should be such that rafts can be removed from their mounts or enclosures and deployed outside the aircraft within 30 sec from the time that release or removal action is initiated by the operator.

When exterior installations for life rafts or other survival equipment are provided, the mountings, retention devices, or enclosures should be designed to preclude inadvertent actuation or damage in flight or when ditching. Such equipment should be

70. Veghte, J. D., COLD SEA SURVIVAL, Aerospace Medicine, Vol. 43, No. 5, May 1972.

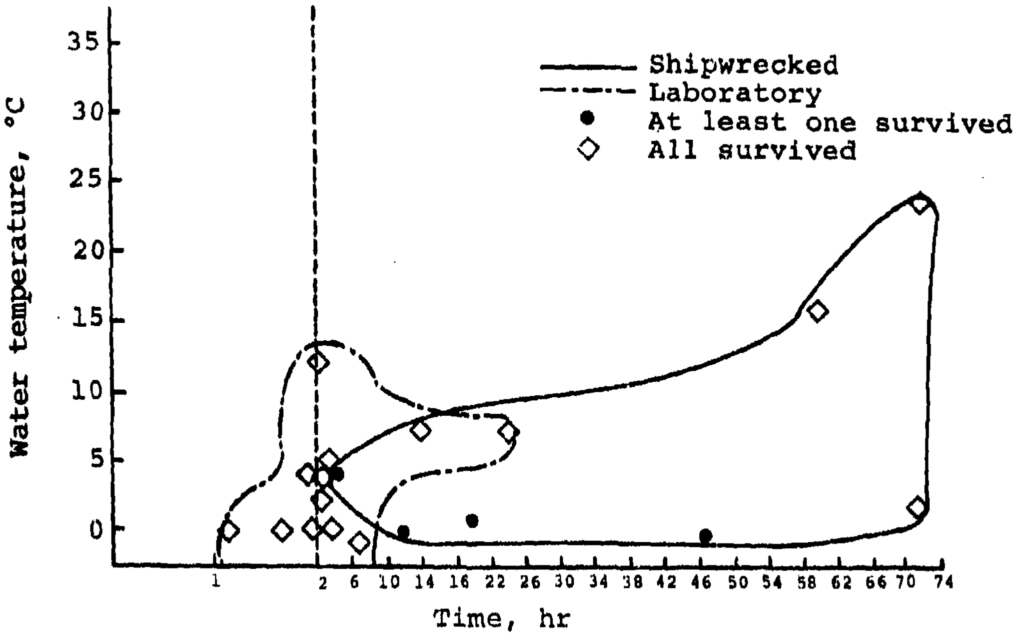

Figure 60. Tolerance time for different water temperatures while in life rafts.

recoverable by occupants from an exit intended for use in ditching. Release mechanisms should minimize the possibility of jamming due to structural deformation that might be incurred upon ditching.

7. EMERGENCY ESCAPE PROVISIONS

7.1 INTRODUCTION

Even though an occupant has survived a crash, the problem of surviving the postcrash environment still remains. Nonsurvivable postcrash conditions occur in a relatively small percentage of accidents, but they account for a disproportionately large number of injuries and fatalities. The key to postcrash survival is the time between the initial crash sequence and the onset of nontolerable conditions. The primary postcrash hazards are fire and water. The occurrence of either can reduce the available escape time to seconds. Therefore, effective emergency escape provisions are essential as integral portions of the aircraft design.

7.2 EMERGENCY EXITS

7.2.1 Emergency Exit Requirements

Two factors that largely determine emergency exit requirements are (1) the amount of available time before the postcrash environment exceeds human tolerance limits, and (2) the attitude and condition of the aircraft structure after it comes to rest.

Research has shown that the available escape time from helicopters involved in postcrash fires is only 7 to 16 sec (see Chapter 3). Thus, all occupants must be able to evacuate the aircraft within 10 sec if they are to survive. However, the allowable evacuation time can be extended to 30 sec if a crash-resistant fuel system is installed in the aircraft. The emergency exit criteria presented in this chapter are predicated on the installation of such a fuel system and should allow all occupants of an aircraft to evacuate within 30 sec.

Providing sufficient exits for 30-sec evacuation of the maximum number of personnel to be carried would seem to meet the emergency requirements. However, it is not unusual for several exits to be blocked following a crash. For instance, if a rotary-wing aircraft comes to rest on its side, all exits on that side will be unusable. Also, exits can be blocked by outside objects, such as trees, or by deformation of the aircraft structure. Therefore, emergency escape provisions should allow the maximum number of aircraft personnel to evacuate in 30 sec with only one-half of the aircraft exits available for egress.

Evacuation times should be demonstrated by actual tests using personnel approximating 95th-percentile troops with full combat equipment for passengers and 95th-percentile aviators with arctic flight gear and body armor for crew members. (Anthropometric data for U. S. Army aviators can be found in Volume 2.) The following sections present emergency exit design criteria to assist the designer in meeting the above requirements.

7.2.2 Types of Exits

Aircraft exits are provided to facilitate either normal or emergency exit and entry. Generally, these exits are classified as follows:

- Class A Exit: A door, hatch, canopy, or other exit intended primarily for normal entry and exit.
- Class B Exit: A door, hatch, or other exit intended primarily for service or logistic purposes (e.g., cargo hatches and rear loading ramps or clamshell doors).
- Class C Exit: A window, door, hatch, or other exit intended primarily for emergency evacuation. Exit closures for Class C exits must be capable of being removed from the exit opening within 5 sec regardless of the aircraft's attitude.

A Class C exit constitutes the minimum requirement for an emergency exit. A Class A exit with emergency jettison provisions is normally considered superior to a Class C exit because of its large size, and, in most cases, it can be used in lieu of a Class C exit. Despite its superiority, however, each Class A exit with emergency jettison provisions can replace only one Class C exit. Class B exits also may be used in lieu of Class C exits if adequate emergency release provisions are installed; however, the functional design of Class B exits usually makes their use less desirable for emergency exit. In order for Class A and B exits to qualify for use in lieu of Class C exits, the exit closures must be capable of being removed from the exit opening within 5 sec regardless of the aircraft's attitude.

7.2.3 Size of Exits

All exits, including Class C exits, must be large enough to accommodate 95th-percentile troops and aviators as specified in Section 7.2.1. Furthermore, the exits must be large enough to allow these personnel to evacuate the aircraft rapidly.

Class C exits should be at least 22 in. square with 6-in.-radius corners, or 22 in. in diameter. This exit size is an Air Force requirement and is considered to be minimum for the evacuation of troops at the rate of 1.5 sec per person (Reference 71). This size must be considered an absolute minimum since the anthropometric data given in Volume 2 lists the shoulder breadth of a 95th-percentile U. S. Army aviator as 20.3 in. Therefore, it is strongly recommended that all Class C exits be larger than the minimum 22 in. Other shapes may be used also, providing the minimum dimensions are met or exceeded. In any case, all exits must be sufficient in size and shape to allow 95th-percentile troops and aviators, equipped as specified in Section 7.2.1, to pass through the exit at a rate of 1.5 sec per person or less.

7.2.4 Number of Exits

7.2.4.1 Crew Compartment (Cockpit): Each flight crew member must have access to at least one usable emergency exit regardless of aircraft attitude after impact. Thus, if a single cockpit enclosure is used for a single crew position, two Class C exits on opposite sides of the cockpit should be provided. This arrangement assures an alternate means of escape if the aircraft rolls on its side, blocking one exit. One Class A exit with an emergency release provision may be substituted for each Class C exit if desired. The Class A exit may be the normal entry/exit door with an emergency jettison capability.

The minimum emergency escape exit requirement for cockpit enclosures with side-by-side crew positions is also two Class C exits. One exit should be installed on each side of the fuselage. Although two Class C exits are required, any combination of Class C and Class A exits may be substituted, provided the Class A exits have an emergency jettison feature.

Cockpit enclosures with tandem crew positions should be provided with two Class C exits and one Class A exit with an emergency release provision. This requirement assumes that the two crew positions are mutually accessible. Mutual accessibility means that a 95th-percentile crew member dressed in arctic flight gear and body armor could, without undue difficulty, climb from one crew position to the other in order to exit the aircraft in an emergency. If the exits in such a cockpit are not mutually accessible to the crew members because of intervening structure or installed equipment, each crew member should be provided with a Class C exit and a Class A exit with

71. Air Force Systems Command, DESIGN NOTE 3Q6; in AFSC: DHL-6, System Safety, 4th Edition, Wright-Patterson Air Force Base, Ohio, July 1974.

an emergency release provision. When sliding or clamshell canopies are used, Class C exits or other suitable means should be provided for crew escape when the aircraft is inverted or otherwise malpositioned on the ground and the canopy cannot be jettisoned. Accident records for aircraft with canopy-type cockpit enclosures indicate crew members are often trapped in the cockpit when the aircraft flips over on its back and the canopy cannot be jettisoned. Experience with such accidents indicates that knives, axes, or other tools carried in the cockpit for chopping through the Plexiglas canopy are not adequate solutions for emergency exit when postcrash fire or occupant injury is present. When the primary means of escape is blocked, an alternative means is clearly necessary.

7.2.4.2 <u>Passenger or Troop Compartments</u>: The minimum emergency escape exit requirement for troop/passenger sections, exclusive of exits provided in cockpit sections, is two Class C exits. One exit should be installed on each side of the fuselage. If one of the two exits becomes blocked for any reason, the other exit will serve as the primary means of escape. Normally, a Class A exit is required for passenger/troop compartments. If normal passenger or troop entry and exit in a particular aircraft is through the troop/passenger compartments, a Class A exit with emergency release provisions and a Class C exit will be more realistic and satisfy the emergency exit requirements.

In addition to the minimum number of exits, additional exits may be required depending on the maximum number of personnel carried in the passenger/troop cabin. Class C exits at a ratio of at least one exit for every 10 persons expected to occupy the section should be provided. An additional exit in excess of the 1-to-10 ratio should be provided when the specified capacity of the section is not evenly divisible by 10 (e.g., if the capacity is 21, three exits are required). The requirement for the passenger/exit ratio of 1-to-10 is based upon the possibility that at least 50 percent of the exits may be blocked if the aircraft comes to rest on its side. This would then leave a 20-to-1 ratio, assuming that both sides of the aircraft have an equal number of exits. A 20-to-1 ratio is, from a theoretical point of view, considered adequate to evacuate all occupants within 30 sec at the exit rate of 1.5 sec per person (assuming no troop debilitation and all exits open). However, at least two exits must be provided even if the number of occupants is less than 10.

The exit requirements cited above also are applicable to cargo compartments if the compartments have a dual capability for troop transport.

7.2.5 Location of Exits

7.2.5.1 Side Exits: Exits intended for emergency use should be equally divided on each side of the aircraft and, if feasible, should not be directly opposite each other. The primary reason for dividing emergency exits equally on both sides of the fuselage is that an alternate means of escape is provided if, for any reason, the exits on one side become blocked. Exits should not be located directly opposite each other because of the probability of crowding in one particular area when both sides of the aircraft may be used for evacuation. By staggering the exits, the tendency to crowd up is diminished.

Since any aircraft may be operated over water, at least one emergency exit on each side of the fuselage should be well above the anticipated waterline under the most adverse conditions expected after a water landing.

7.2.5.2 Overhead Exits: In aircraft where the width of the crew and troop compartments is too great to permit easy access to fuselage up-side exits if the aircraft comes to rest on the opposite side following an accident, Class C overhead exits should be provided at a ratio of one exit for every 20 occupants. Where the capacity of the compartment is less than 20, at least one Class C exit should be present. These overhead exits are in addition to the normal requirements for Class C exits.

When an aircraft comes to rest on its side, blocking the exits on that side, the exits on the other side of the aircraft could be the only means of evacuation. These exits, now on the topside of the rolled aircraft, may be useless if the width of the fuselage is such that they cannot be reached easily. In an aircraft resting on its side, overhead exits would be more accessible than the normal up-side exits. A fuselage width of 5 ft or more between side exits is considered too great to permit easy access to up-side exits by troops with minor debilitating injuries following a crash.

In helicopters with engines, transmissions, major controls, etc., located over personnel compartments, bottom or fore and/or aft exits may be substituted for the overhead exits. Alternatively, side exits may be located where interior aircraft components, such as seats and consoles, can be used as steps to gain access to the up-side exits. If this type of arrangement is used, the designer must ensure that these components will maintain their structural integrity and attachment to the aircraft during a survivable crash. Such component-steps must be able to support a 300-lb occupant to accommodate fully equipped 95th-percentile crew members and troops.

If the aircraft has a high-wing arrangement, overhead exits should be provided to facilitate escape following ditching. These overhead exits will be in addition to the normal requirements for emergency exits. Overhead exits constitute the only practical means of escape in a rapidly sinking aircraft of this type because the occupiable portion of the fuselage in high-wing aircraft sinks below the surface of the water rapidly following a ditching. The opening of side exits causes flooding of the interior at a high rate, decreasing escape time.

7.2.5.3 Exit Location Relative to Fuselage Distortion: To provide maximum accessibility to aircraft occupants following a crash, emergency exits should be located in areas least vulnerable to distortion. Insofar as it is feasible, exits should not be located in close proximity to the main landing gear because of the possibility of the gear being driven upward and/or inward against the aircraft, causing a blocked or jammed exit. Exits should not be located under heavy components mounted on the top of the fuselage, such as engines and transmissions, because of the possibility of fuselage distortion in crashes where high vertical forces are present. In high-wing aircraft, a crash landing is likely to cause structural deformation below the wing; therefore, exits located under the wing should be avoided as much as possible.

7.2.5.4 Exit Location Relative to Obstructions: Class C exits should be located where it will not be necessary to move equipment, cargo, or furnishings to gain access to them. Insofar as it is feasible, all exits that might be used in emergencies should be located where external components such as propellers, turbine engine inlets, turrets, armament, and tail surfaces will not interfere with occupant escape.

7.2.5.5 Exit Locations Relative to Ignition Sources: Exits should be located as far as possible from fuel spillage areas and from major ignition sources (e.g., exhaust stacks, hot engine parts). Where the occupiable portion of the aircraft is mainly aft of the power units and fuel tanks, it is desirable to locate at least one Class A or B exit with an emergency jettison feature as far aft as possible. In the case of rear-mounted engines, an A- or B-type exit should be as far forward as possible. Such an arrangement may increase escape time in the event of a postcrash fire.

7.2.6 Exit Operation

7.2.6.1 Exit Operational Design: The method of emergency exit operation should be simple, obvious, and natural to all personnel expected to be aboard the aircraft. Exit operation also should be as rapid as possible. Therefore, exits intended for emergency use should be designed so that no secondary operation such as moving or unlocking locks, catches, stops, bolts, or bars is necessary. (Such a requirement does not preclude the use of easily removable protective covers intended to prevent inadvertent actuation of exit release handles.)

An emergency exit should be capable of being completely opened within 5 sec. The time requirement of 5 sec to remove the exit closure (window, door, hatch, etc.) from its opening is based upon the need for all possible haste in evacuating burning aircraft and a realistic estimate of the time-motion requirements for actuating a simple, continuous-motion release mechanism without secondary operations. The measurement of time should begin when the operator places his hand on the release handle and end when the exit closure is free and clear of the exit opening.

Only the single operation of pulling or pushing the exit closure into the clear should be necessary, once the release handle has been actuated. Unless the aircraft is pressurized, all emergency exit closures should be arranged to fall free, or: to be easily pushed outward for side exits, to be pushed upward for overhead exits, to be pushed downward for bottom exits, when the emergency release mechanism is actuated. To remove the exit closure inward would add to the congestion and impede escape. In a pressurized aircraft, exit closures must be removed inwardly, but, if at all possible, the closure should then be canted at an angle and pushed out the exit opening in order to avoid congestion inside the aircraft.

Emergency exits should be designed to permit removal of the exit closure when seal vulcanization occurs, when the fuselage is covered with ice accumulated in flight, and when minor fuselage deformation occurs. A peripheral clearance of at least 0.20 in. provided between the exit closure and its frame will help accomplish this goal.

The 0.20 in. specified should be considered the minimum clearance between the exit closure and its frame. It is probable that some aircraft with relatively light fuselage construction could use more than 0.20-in. clearance in this area, since greater fuselage distortion in such aircraft is likely when a crash occurs. With a 0.20-in. peripheral clearance, the exit frame could theoretically deform inward for 0.40 in. on any one of its four sides before binding occurs.

Consideration also should be given to designs that cause the exit closure to eject itself from its frame when large structural deformation due to impact occurs. This type of design is particularly appropriate for the simple Class C type of exit that contains no release mechanism but needs only to be pushed out of its mountings to open.

The use of Class A and B exits that slide to open is probably unavoidable on certain types of aircraft. Such a design should not be incorporated on exits intended for emergency evacuation because fuselage distortion, which is common in aircraft crashes, can cause the exit to bind on the tracks attached to the fuselage.

7.2.6.2 Release Mechanism Design: The exit release mechanism is the primary handle, lever, or latch used to open the emergency exit closure. Handles may be of the T- or L-shaped design that turns, the D-ring type that pulls, or the lever type that slides fore and aft. However, the number of different types of handles in the aircraft should be held to a minimum. It is recognized that some types of emergency exits will not use exit release handles. One common type of exit uses a release method whereby a panel held by a flexible mounting is simply pushed out.

The method of operation of the exit release mechanism should be simple, obvious, and natural to the operator. In order to facilitate rapid emergency egress, exit release mechanisms should be designed to permit release handle actuaton and exit opening by one person using one hand. The Air Force specifies an actuation/operating force of 10 to 30 lb to meet this requirement (Reference 71). Release and opening mechanisms also should allow all exits to be removed successfully in an emergency when the aircraft is in other than an upright position.

The shape and direction of operation of exit release handles should conform to the "form follows function" rule, where the releasing action is most natural to the position of the operator initiating the action. According to McFadden and Swearingen, "In general, the best position for applying force to a handle is one in which a subject can use his legs and lift. The next best is in pushing down and using body weight. The least effective method is the employment of an over or under motion. The under motion is slightly superior." (Reference 72).

Specific considerations for different types of handles are as follows:

72. McFadden and Swearingen, FORCES THAT MAY BE EXERTED BY MAN IN THE OPERATION OF AIRCRAFT DOOR HANDLES, Journal of the Human Factors Society of America, Vol. 1, No. 1, September 1958.

- T- or L-Shaped Emergency Release Handles: Internal emergency release handles with a T- or an L-shaped design should be capable of actuating the release mechanism in both clockwise and counterclockwise directions. The arc of rotation in this case should not exceed 90 degrees. If only one direction of handle rotation is permitted, rotation should be counterclockwise and the arc of rotation should not exceed 180 degrees. Stops that prevent rotation in the wrong direction should be provided.

- D-Ring Type Emergency Release Handles: If the release handle is a D-ring type that requires pulling for release action, the grip of the D-shaped handle should be parallel to the aircraft's vertical axis for side exits and parallel to the aircraft's longitudinal axis for overhead exits. The direction of pull should be toward the operator in the same straight line as the natural position of the extended forearm holding the handle prior to release action.

- Lever-Type Emergency Release Handles: Internal emergency release handles incorporating a lever or bar that slides fore and aft along the x axis of the aircraft should be capable of opening the exit in both directions.

Exit release mechanisms should be designed so that the entire operation of the release handle is a continuous motion from start to finish without sharp changes in direction except for external installations where the release handle must be pulled from countersunk recesses before actuation. In any type of release handle, the final motion of the handle should contribute to the opening of the exit.

Release handle shapes and dimensions should be designed for normal hand grip limitations and incorporate handle-to-hand contact areas that ensure adequate load applications to the handle. Release handles on external installations should provide clearance to allow gripping of the handle with gloved hands, since rescue crews normally wear heavy gloves to protect hands from jagged and hot metal surfaces. Standard fire fighter's asbestos gloves should be used for testing. The release handle should be mounted on the exit closure itself or immediately adjacent to the exit opening so that it is readily accessible to any occupant attempting to use that exit. Remote exit release mechanisms should be avoided. The release handles on the exit closure or on the adjacent airframe should not be located in a position that would allow the handle to snag clothing or impede escape through the exit opening even if the exit is, for some reason, limited to partial opening. Similarly, the exit actuating mechanism should be designed so that the final position of the release handle upon opening will not obstruct the removal of the exit closure.

Emergency exit release handles in cockpits and troop compartments should be located where it is not necessary for crew members to unlock their shoulder harnesses in order to turn the handles. This is very important in cockpits and at crew chief or special crew stations, primarily because it is sometimes desirable to release emergency exits just prior to crash impacts. This is especially true for ditching. If a shoulder harness has to be unlocked to release the exit, there may be insufficient time available to relock it before impact. This requirement also is applicable to those emergency exits that are adjacent to certain seats in the passenger/troop compartment. This, however, should not be construed as a recommendation to remove exits prior to crash impact in every case. The openings of such exits can sometimes critically reduce the time otherwise available for occupants to escape, since fire can develop on the outside, causing flash fires within the compartment. As a general rule, the chances of surviving a crash involving fire are less if doors and exits are open prior to impact.

Accidental release of exits in flight can be extremely dangerous in rotary-wing aircraft. Exits that have been released in flight have been known to fly into the main or tail rotor system, causing disintegration of the system and subsequent loss of the aircraft and crew. An unguarded or unshielded exit release handle can make a convenient hand-hold for inexperienced troops. Therefore, release mechanisms should be designed so that improper or incomplete closing of the exit will be obvious. On both external and internal installations, a locked-position indicator, such as a detent to indicate positive locking, should be provided.

In the event that crash victims become trapped in the aircraft or become otherwise unable to escape without help, it is essential that all emergency exits be capable of being opened by rescue personnel from the outside of the aircraft. The actuation of an internal release handle must not preclude the simultaneous actuation of an external release handle. If "push-out" type Class C exits are provided, they should be as easy to open from the outside as from the inside. Means to prevent icing of the outside release and handle mountings should be provided to ensure positive operation under adverse weather conditions.

If the cockpit enclosure consists of a canopy that slides back and forth or opens on hinges in a clamshell fashion, an emergency jettison feature can provide rapid egress for the crew. The jettison mechanism should allow complete removal of the enclosure from its mounting within 5 sec from the time that mechanical action is initiated. In addition to the internal jettison release, external canopy jettison controls should be provided on both sides of the fuselage. The canopy jettison feature does not eliminate the necessity for additional emergency exits since the postcrash aircraft attitude might preclude successful jettisoning of the canopy.

7.2.7 Explosively Created Exits

Explosive systems have been developed and successfully used to provide quick-opening emergency exits in military aircraft. These systems can cut emergency exits through existing doors and windows and through fuselage structures. The systems provide the advantages of extremely rapid release times, simplicity of operation, and immunity to jamming by structural deformation, ice, or seal vulcanization. The following sections discuss factors that must be considered during the design of an effective and operationally safe explosive exit system.

7.2.7.1 Overall System Design: An explosively operated exit system contains four basic components or subsystems: (1) an arming/firing system, (2) primer and/or detonating cord, (3) a linear shaped cutting charge, and (4) an actuation mechanism. The relationship of these components to each other can best be illustrated by considering the design of an actual system--in this case, the Emergency Lifesaving Instant Exit System (ELSIE) developed for the U. S. Air Force (Reference 73).

The ELSIE system is composed of an electromechanical safe/arm mechanism, dual shielded mild detonating cord lines, a flexible linear shaped cutting charge, and interior and exterior initiation handles attached to firing lanyards. The relationship of the components is shown schematically in Figure 61. The safe/arm mechanism requires only momentary application of power to arm or disarm. The system remains armed or disarmed, even if power is lost, since the mechanism is mechanically locked in position. Once the system has been armed, it can be actuated

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73. Nicholson, D. E., and Burkdoll, F. B., AN EMERGENCY LIFE-SAVING INSTANT EXIT SYSTEM FOR CARGO, CARGO-TRANSPORT AND PASSENGER AIRCRAFT, Explosive Technology; ASD Technical Report 71-41, Aeronautical Systems Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, June 1971, AD 736056.

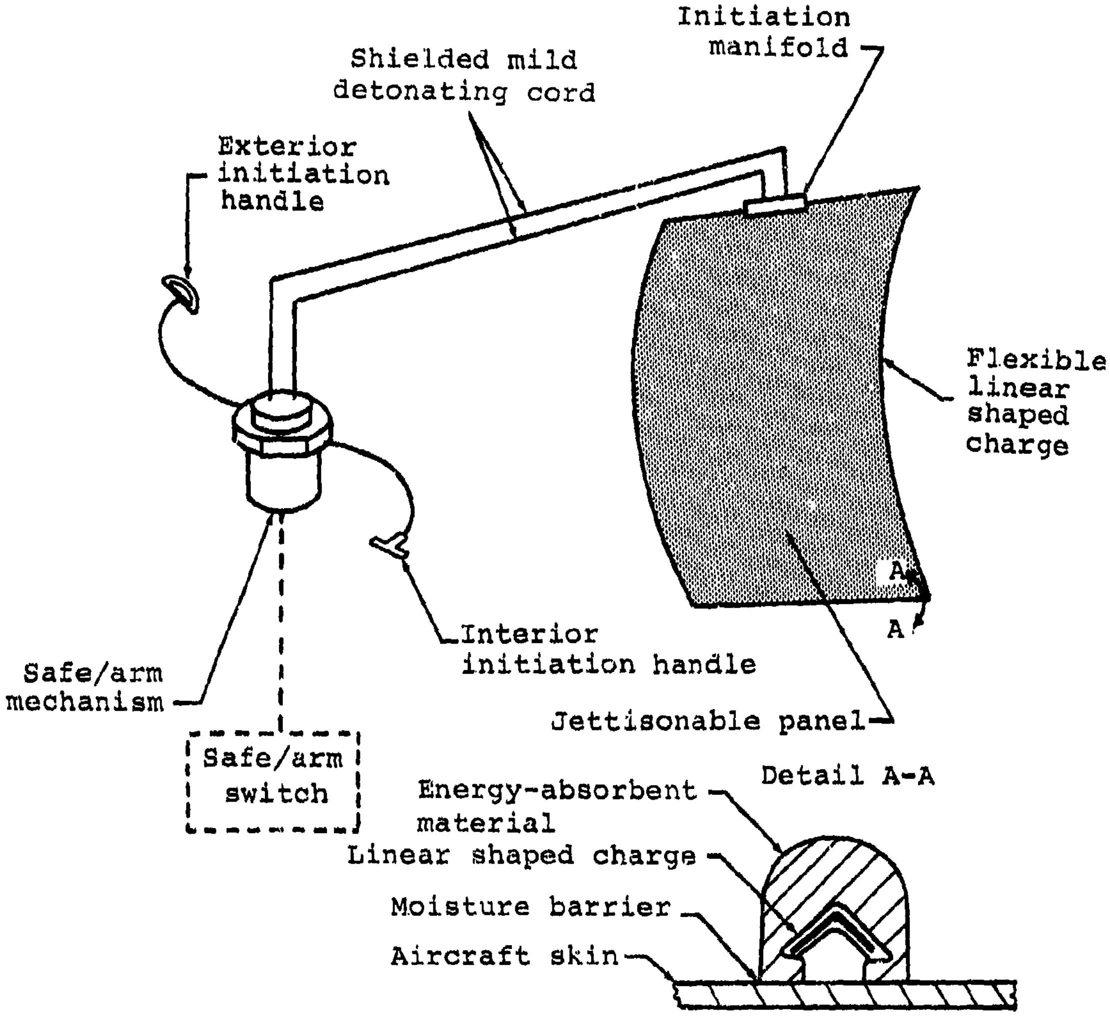

Figure 61. Schematic of ELSIE system.
(From Reference 73)

either from inside or outside the aircraft by pulling the handle in either location. The handles operate a mechanical striker that fires the dual detonating cord lines. These redundant lines, in turn, initiate the shaped charge that cuts the egress opening in the aircraft and ejects the cut panel outward. Tests on the ELSIE system show that the elapsed time from pulling the initiation handle until the egress opening is available for use is less than 0.027 sec.

7.2.7.2 Arming/Firing System: The arming/firing system must be designed for simple and rapid actuation of the explosive system and yet provide maximum safety against inadvertent actuation. Operational safety must be assured by preventing inadvertent actuation due to environmental conditions, system component failures, or human error.

To provide maximum operational safety, arming and firing must be accomplished in two separate and deliberate actions. The arming function always should be under the control of the flight crew. Thus, the arming mechanism should be located only in the cockpit and at the crew chief's station. If cockpit enclosures with tandem crew positions are used, each crew member must be provided with an arming mechanism unless the two positions are mutually accessible. System status indicators should be provided at all pertinent flight crew stations.

Once armed, the system should be capable of being fired by any of the aircraft occupants. Each exit should be capable of being actuated independently from the rest since it is not always desirable to open all available emergency exits, especially in case of a postcrash fire or a ditching. A firing mechanism should, therefore, be located immediately adjacent to each exit for actuation of that particular exit only. This means the arming and firing mechanisms will, of necessity, be physically separated from each other. An exception to this practice might be acceptable when the exits are located quite near each other, as in tandem cockpit configurations. Then the adjacent exits could be fired simultaneously from one firing mechanism, although a firing mechanism must still be available to each crew member.

Once the system is armed, it must stay armed until it is disarmed by a crew member or rescuer. The reverse also is true; once the arming mechanism is in a disarm, or safe, position, it must remain that way until a deliberate arming action is initiated. Any type of system or component failure must not change the position of the safe/arm mechanism. For instance, if arming is accomplished by electrical power, loss of power must not allow the mechanism to switch from arm to safe or vice versa. The mechanism also must be immune to any environmental or crash load input. Disarming capability must be provided to permit safing the system even though normal safing modes are inoperable following a crash.

In order to provide the highest degree of both operational and crash safety, the firing mechanism should be independent of any external energy source, such as the aircraft electrical system. This requirement dictates that the firing mechanism be manually operated. The design considerations for emergency exit release mechanisms discussed in Section 7.2.6.2 also apply to the firing handles used in explosive exit systems. In addition to those considerations, the external release handle must be designed to allow rescue personnel sufficient separation from the aircraft before actuation to prevent their being struck by debris when the exit is opened. It is also strongly recommended that all arming mechanisms and firing handles be completely separated from each other, even in those cases where it might seem feasible to combine them (e.g., pilot's crew station). If the arming and firing mechanisms are combined into one package, it is essential that the operations of arming and firing be distinctly separate from each other, such as turning the handle to arm and pulling the handle to fire.

7.2.7.3 Explosive System: All explosives used in the exit system should possess as high a thermal limit as possible, not only to ensure that the system is safe in high-temperature operating environments but also to provide as much safety as possible in case of a postcrash fire. The system must be able to function when exposed to temperatures up to the limits of human tolerance to heat (approximately 400°F, based on ambient air temperature), yet not function inadvertently during brief exposure (30-60 sec) to postcrash fires. The latter requirement is necessary to prevent flames coming through an unintentionally opened exit of an occupied aircraft. The thermal limits of the explosives used in the ELSIE system, which meet the above requirements, are below.

Explosive	Thermal limit (°F)
HNS (22', 44', 66' hexanitrostilbene)	618
Lead azide	635
M-426 primer	425

The linear shaped charge must be held securely in position against the aircraft structure it is to cut. The size of the exit opening should conform to Class C requirements given in Section 7.2.3. The jettisonable section must be ejected outward to preclude its obstructing the exit opening. Energy-absorbing backup material must be placed behind the shaped charge to control the backblast of the explosive and prevent fragments from entering the cockpit or cabin (refer to Figure 61).

The explosive system must be designed to minimize the possibility of system actuation igniting any fuel that might be spilled during a crash. The amount and duration of any exposed flame must be minimal. The ELSIE system successfully functioned during a series of fuel spray tests without igniting the fuel because the explosive charge was designed to penetrate only 90 percent of the aircraft skin thickness. The remaining 10 percent was severed by the pressure created by the detonation of the shaped charge and the momentum already imparted to the jettisonable section. This design allowed the combustion products around the periphery of the cut to cool significantly before the metal skin was completely severed. Because of this, the only flames exterior to the aircraft skin were at the initiation points of the shaped charge and lasted less than 10 msec for most of the tests.

7.2.8 Access to Exits

7.2.8.1 Exit Obstructions: Access from aisles to all exits must be provided so that the exits will not be obstructed by troop seat components, seat back webbing and webbing support bars, litter installations, or other protrusions to an extent that would reduce the effectiveness of the exit.

A common problem with troop-carrying aircraft is that, in order to carry the maximum number of troops, some emergency exits are blocked by the installation of troop seat back webbing and webbing support bars. These components are normally designed to be pulled away from the emergency exit in order to provide access. It is desirable, of course, to avoid obstructing exits, but if it is necessary to do so, seat backs or other potential obstructions should be readily collapsible or movable to provide access to exits during an emergency evacuation.

7.2.8.2 Aisle Widths: The width of aisles at any point between seat rows should be sufficient to allow unobstructed movement of 95th-percentile troops with full combat equipment. Current criteria suggest a minimum width of 17 in. In aircraft where it is necessary to pass through seat rows to gain access to exits during an emergency, longitudinal spacing between seat rows should be sufficient to permit these troops to move at a rate consistent with the capacity of the exit (1.5 sec per person or less).

7.2.8.3 Compartment Doors: Doors on hatches separating various interior compartments should be located and have a direction of opening so as not to impede or block passage to other exits or interfere with the use of emergency equipment such as axes and fire extinguishers.

The doors or hatches should be large enough to permit crew and troop movement from one compartment to another during emergency evacuation. The openings should have no protrusions that would impede movement through them. Provisions should be made for securing compartment doors in the open position during takeoff and landing. Such doors should be capable of remaining open and latched when exposed to crash forces of survivable magnitude.

Compartment doors should have release handles designed so that the method of operation is a single, obvious, and natural motion in a single plane. Round or spherical handles or knobs should unlatch when gripped and turned in either direction. The handles should not snag on clothing or equipment.

7.3 EMERGENCY LIGHTING

Emergency lighting provides the illumination required for emergency evacuation and rescue when the normal aircraft lighting is not available. There are three basic types of emergency lighting: (1) interior lighting for personnel orientation following aircraft accidents at night, (2) lighting for the purpose of reading exit operating instructions and releasing the exits, and (3) exterior lighting to illuminate exits and paths of escape.

7.3.1 Interior Emergency Lighting

When an aircraft crashes at night or is filled with dust or smoke, disorientation of embarked crew and troops is likely to occur. Since escape time is critical, interior emergency lighting units should be installed in sufficient number and possess adequate brightness to permit personnel orientation in all compartments during emergency evacuation situations. The emergency lighting should provide sufficient illumination throughout the cockpit and cabin areas to permit occupants to locate emergency exits and survival equipment, perceive escape paths, and avoid obstacles while moving toward the exits. This criteria may not be necessary for some aircraft using overhead canopies.

Interior lighting fixtures may be mounted as aisle, ceiling, or cornice lights. Regardless of where the lights are mounted, they must furnish adequate illumination near floor level to allow occupants to see exit paths and avoid any obstructions. The emergency lighting requirement for both civil and military aircraft is a minimum average illumination in clear air of 0.05 foot-candle (fc) measured 20 in. above the floor (or at armrest height) along passageways leading to each exit and in front of each exit (References 74 and 75). (1 foot-candle is the intensity of illumination 1 ft from a source of 1 candela.) Full moonlight is approximately 0.04 fc, while recommended reading intensities are 20 to 50 fc (Reference 76).

7.3.2 Emergency Exit Lights

Supplementary emergency lighting units should be provided at or near each emergency exit with adequate brightness to permit untrained personnel to identify exits, to read exit operating instructions, and to actuate the exit mechanism without difficulty during periods of reduced visibility. The identity and location of each emergency exit should be recognizable under limited visibility (darkness, smoke, etc.) from a distance equal to the width of the cabin.

Exit light requirements must take into account the fact that the illumination at any distance from a light source is inversely proportional to the square of the distance from the source. Thus, at a distance of 5 ft, the brightness of a light will theoretically diminish to only 4 percent of the brightness measured 1 ft from the source. The same rapid decrease has been measured in the brightness of internally illuminated aircraft exit signs during an FAA program to evaluate current exit signs and markers (Reference 77). The decrease in average exit sign brightness with increasing distance from the signs is shown graphically in Figure 62.

Exit light effectiveness also is reduced by the presence of smoke. Measured light output for all units tested by the FAA diminished proportionately in a 90 percent smoke environment, as shown in Figure 62.

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74. Federal Aviation Regulations, AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY ROTORCRAFT, Part 29, Section 29.811.

75. Military Specification, MIL-L-6503, LIGHTING EQUIPMENT, AIRCRAFT, GENERAL SPECIFICATION FOR INSTALLATION OF, Department of Defense, Washington, D. C., 2 April 1975.

76. Mendenhall, C. E., et al., COLLEGE PHYSICS, 3rd Edition, Boston, Massachusetts, D. C. Heath and Company, 1950, p. 476.

77. Garner, J. D., and Lowrey, D. L., EXIT SIGN COMPARISONS IN CLEAR AIR AND SMOKE, unpublished paper presented at an A-20 meeting, Society of Automotive Engineers, New York, December 1976.

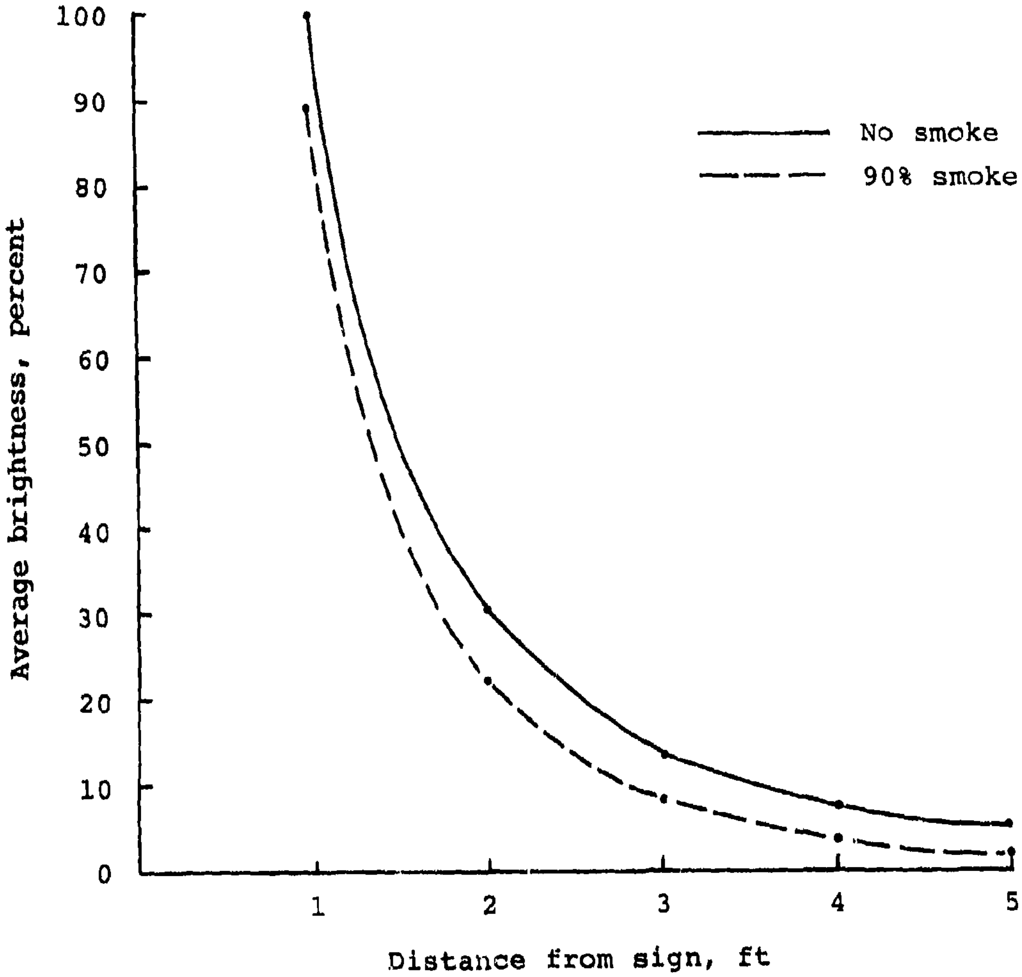

Figure 62. Relative brightness of internally illuminated aircraft exit signs at various distances from signs (average of 10 different signs).

Current FAA requirements for large transport-category airplane emergency lighting include internally or self-illuminated signs at each exit with a minimum luminance (brightness) of at least 25 fL (Reference 78). Small (9 seats or less) transport airplanes and transport-category rotorcraft need only have exit signs with a brightness of 160 microlamberts (0.15 fL) (References 74 and 78). Although the above requirements might be

78. Federal Aviation Regulations, AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY AIRPLANES, Part 25, Section 25.811.

sufficient in clear air, the rapid drop in brightness due to the presence of smoke makes the sufficiency of even the brighter (25 fL) requirement questionable.

Most current transport airplane exit lights exceed the 25 fL requirement, but lights far brighter than those currently used are available. Figure 63 presents the results for two of the 10 exit lights tested by the FAA (Reference 77). This figure shows that, under all conditions, the experimental light was approximately 10 times brighter than the typical currently used exit light. This is most important during smoke conditions and at some distance away from the exit sign. For instance, at a 6-ft distance under 90 percent smoke, the current sign transmitted only 0.017 fc of light while the experimental sign transmitted 0.13 fc. It is noteworthy that the experimental aircraft sign is currently used in building installations and uses less battery power than some current aircraft signs. Other newly developed lights, which are much brighter than current lights, also are available.

Based on the results of the FAA tests, all passenger- or troop-carrying aircraft should contain internally illuminated exit signs with a minimum average brightness of at least 25 fL. However, it is strongly recommended that the exit signs be even brighter.

The diminishing of exit light effectiveness when the aircraft is submerged has already been discussed in Chapter 6. Any aircraft whose mission requirements include troop transport over water should contain exit sign lighting meeting the requirements specified in Chapter 6.

7.3.3 Exterior Emergency Lighting

For noncombat missions, exterior emergency lighting should be considered at each exit to illuminate the ground near the exit and areas where escape and survival equipment will be deployed. MIL-L-6503 specifies that the light intensity on the ground below normal and emergency exits should be 0.02 fc minimum (Reference 75).

7.3.4 Structural Considerations

All emergency lighting units must be self-contained, explosion-proof, operable under water, and accessible for periodic maintenance. All units must be capable of operating independently of the main aircraft lighting system.

The emergency lighting system should be designed, installed, and located so as to minimize damage to or loss of any portion

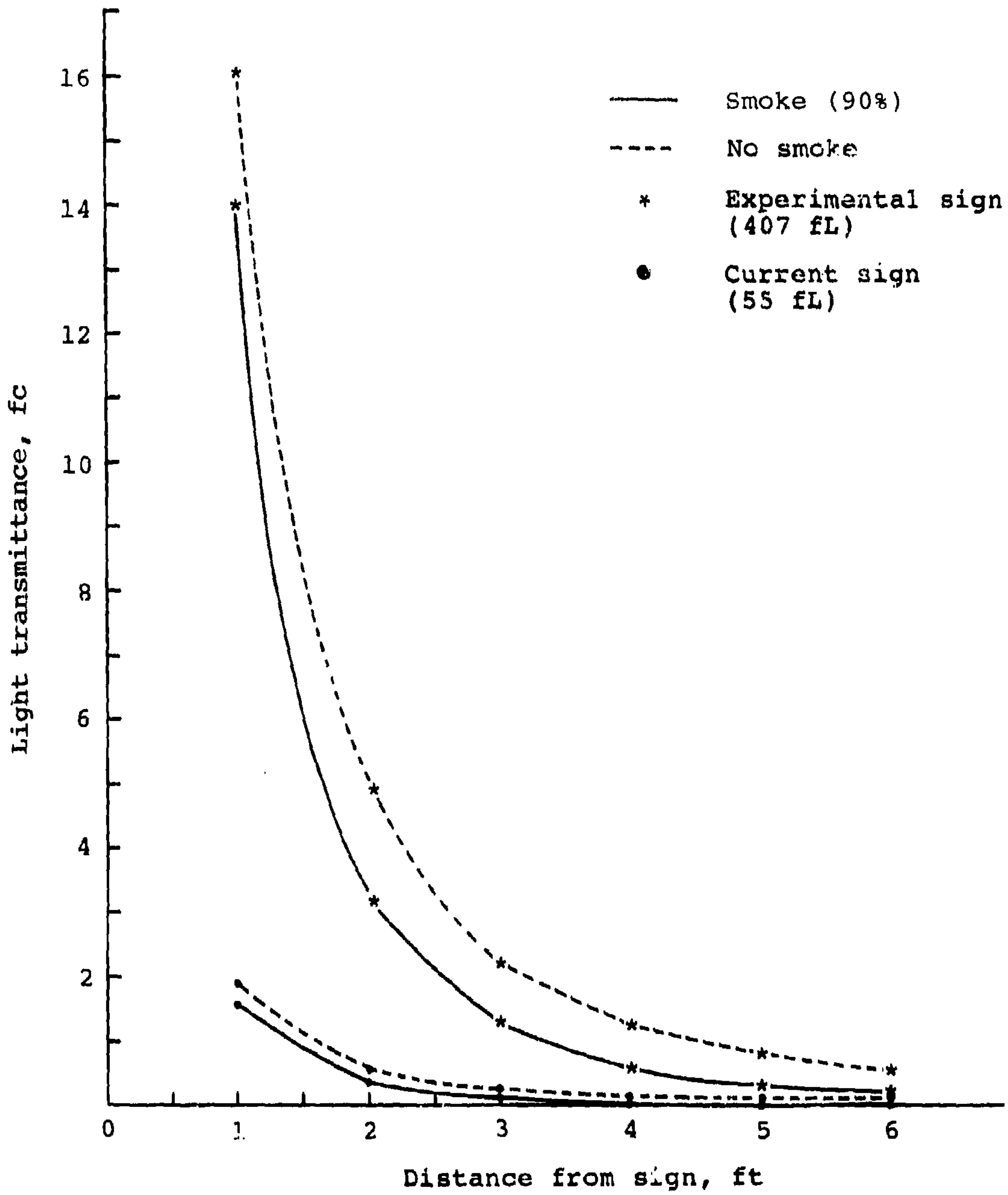

Figure 63. Light transmittance of exit signs in clear air and in smoke at various distances from sign.

of the emergency illumination as a result of ditching or emergency landing. To ensure structural integrity and continued operation after a crash, the lighting system, including all components necessary to provide the required illumination, should be capable of withstanding the following crash loads: 50 G downward, 10 G upward, 35 G forward, 15 G aftward, 25 G lateral. Breakup of the fuselage should not render any portion of the emergency illumination inoperative except for those lights directly destroyed by the break.

7.3.5 Emergency Lighting Power Source

Emergency lighting power sources should be independent of the main electrical power source for the aircraft and should contain power sufficient to ensure effective illumination for a minimum of 15 min.

It is believed that a power source strong enough to provide at least 15 min of effective illumination following a crash at night is adequate. If a postcrash fire does not occur within 15 min, it is likely that one will not occur at all. Personnel who are stunned or otherwise unable to evacuate the aircraft during the 15 min of emergency lighting could, in all probability, evacuate in the darkness if they were physically able.

7.3.6 Actuation of Emergency Lighting Units

Emergency lighting units should be designed to actuate both automatically and manually. If inadvertent actuation occurs, the unit should be capable of being reset manually.

7.3.6.1 Manual Actuation: There are circumstances where it would be desirable to manually turn on the emergency lighting. One such instance would be when a crash was imminent, but some time was available prior to the crash. By turning on the emergency lights manually, the aircraft occupants would have time for their eyes to adjust from normal lighting or darkness to emergency lighting. This also would permit all normal aircraft lighting to be turned off in order to reduce potential postcrash fire ignition sources. Therefore, a manual actuating switch should be placed in the cockpit, and another should be placed in the passenger/troop compartment close to the crew chief's station.

7.3.6.2 Automatic Actuation: The emergency lighting units should be automatically actuated in as many survivable accidents as possible. This can be accomplished by using inertia sensors responsive to the crash pulse parameters typical of lower-severity accidents. The sensor criteria should be identical to those specified for crash locator beacons (see Chapter 8).

The crash sensors may be contained in each lighting unit, or the units may be actuated from one or more common sensors located remotely from the lights. The circuits for the lights should be such that they will be energized if the circuits between the lights and the sensors are broken.

There may be circumstances, such as forced or crash landings in enemy territory, where it would not be desirable to automatically actuate the emergency lighting. A circuit breaker or other device to nullify the automatic feature therefore is desirable. Such a device could be utilized by the crew upon entering enemy territory.

7.4 MARKING OF EMERGENCY EXITS

Emergency exits must be clearly marked both inside and outside the aircraft so that occupants and rescuers can find them rapidly. The markings must be distinctive to set them apart from the numerous other markings found on the aircraft. In addition to identifying the exits, instructions for releasing the exit closures must be clearly marked beside the exit release mechanism. The time required to determine how to release the exit closure could well mean the difference between survival or nonsurvival.

All U. S. Army aircraft must be painted and marked according to the requirements in TB 746-93-2 (Reference 79). The requirements contained therein for marking of emergency exits are summarized in the following sections. The reader is referred to TB 746-93-2 for complete details.

7.4.1 Internal Identification of Exits

An orange-yellow band must mark the complete periphery of the escape exit on olive drab backgrounds. A gloss black band is used on light backgrounds. The band must be between 1 and 2 in. wide and divided equally, if possible and practicable, between the mounting of the exit and the exit itself.

If soundproofing (or lining) covers the identification band on the inside of the aircraft, it also must be appropriately marked.

The words EMERGENCY EXIT, in orange-yellow, must be marked or stenciled on the escape hatch, door, or exit in the most readily visible location. Preferably, letters should be 2 in. high, but they cannot be less than 1 in. high.

79. Technical Bulletin 746-93-2, PAINTING AND MARKING OF ARMY AIRCRAFT, Department of the Army, Washington, D. C., January 1971.

7.4.2 External Identification of Exits

Markings identifying escape hatches, doors, and exits on the outside of aircraft must be marked gloss yellow on dark surfaces and gloss black on light surfaces. On camouflaged aircraft, emergency exit markings are painted with lusterless yellow lacquer.

7.4.3 Identification of Secondary Openings

Secondary openings, such as auxiliary exits and windows, are usually smaller than primary openings, making entrance or exit more difficult. If the structure immediately surrounding a secondary opening is free from heavy structural members; oxygen, fuel, and oil lines; and battery leads it should be marked with a broken orange-yellow band on olive drab or a broken black band on light surfaces. The band must be placed on the extreme boundary of the clear area around the secondary opening, inside and outside of the fuselage. Segments of the broken band must be 1/2 in. wide, 1 in. long, and approximately 12 in. apart. CUT HERE FOR EMERGENCY RESCUE must be marked inside of, parallel with, and adjacent to the broken band identifying the area on the outside of the aircraft where forced entry can be made for rescue purposes. CUT HERE FOR EMERGENCY EXIT must be marked in a similar location inside the aircraft. Letters must be 1 in. high.

On camouflaged aircraft, the corners of emergency exits and rescue exit areas are outlined with right-angle corner bands 1 in. wide and 3 in. long at each leg. The corner markings are painted with lusterless yellow lacquer.

7.4.4 Marking Instructions for Exit Operations

7.4.4.1 Internal Markings: Small handles or levers used to actuate doors or hatches must be identified by alternate 1/8-inch-wide orange-yellow and black stripes, painted on the background of the exit. Background striping must be applied at a 15-degree angle from the vertical, rotated clockwise. The striping must not interfere with other types of markings or codings. Large levers or exit controls must be marked with alternate orange-yellow and black stripes, 1/8 to 1/4 inch wide, painted directly on the lever or control.

7.4.4.2 External Markings: All external releases for operation of emergency exits must be marked EXIT RELEASE on the outside of the aircraft to facilitate quick identification. Letters preferably should be 2 in. high and must not be less than 1 in. high.

7.4.4.3 Operating Instructions Markings: Operating instructions to identify and explain the emergency release operation must be marked on the exit door, or hatch, or aircraft structure, whichever is nearer the release. Minimum lettering heights specified are 1/2 in. internally and 1 in. externally. Preferably, the descriptive wording should be 1 in. high on the inside of the aircraft and 2 in. high on the outside. The 1/2-in. minimum specified in TB 746-93-2 for internal wording is not sufficient for easy reading under reduced visibility conditions, such as darkness or the presence of smoke.

The instructions should be as simple and concise as possible consistent with clarity of meaning. Standard English terminology, such as PULL, PUSH, TURN, or SLIDE, must be used.

The painting and marking schemes for in-service aircraft contained in TB 746-93-2 show liberal use of nonverbal symbols in exit operating instructions. Symbols are particularly useful in delineating directions of motion for handles, levers, etc. The use of symbols in conjunction with words will often lead to quicker understanding of the operation to be performed. Symbols are invaluable when the wording cannot be deciphered, as might be the case under reduced visibility conditions, or when viewed by non-English-speaking personnel. Thus, although not stated as a specific requirement in TB 746-932, symbols should be used in exit operating instructions whenever possible. Some symbols in current use are shown in Figure 64.

7.5 CREW CHIEF STATIONS

There is a great need for experienced crew chief personnel to provide the necessary leadership and guidance for embarked troops during emergency evacuation. Accident records indicate that on many occasions crew chiefs have been responsible for successful emergency evacuations of large numbers of troops from aircraft under severe conditions.

At least one crew chief station should be located in each troop compartment. The location of the crew chief's station should provide as complete surveillance of the troop compartment as is practicable. The station should be located as near the main or emergency exits as possible. For aircraft requiring two crew chiefs, their respective stations should be as far apart as practicable; e.g., one in the forward end of the compartment and one in the aft end.

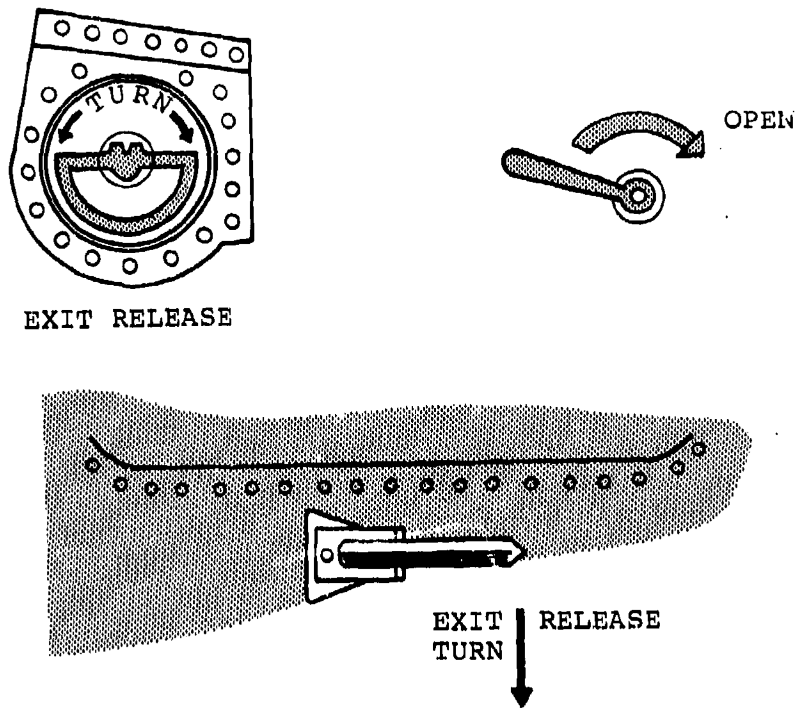

Figure 64. Typical exit release instructions incorporating symbols.

8. CRASH LOCATOR BEACONS

8.1 INTRODUCTION

After a survivable crash has occurred, rescue time becomes paramount in determining the ultimate survival chances of the occupants. Air Force records indicate that the life expectancy of injured survivors decreases as much as 80 percent during the first 24 hr following an accident, and the chances of survival of uninjured occupants rapidly diminishes after the first three days (Reference 80). Therefore, the installation of a crash locator beacon in the aircraft can greatly enhance the occupant's survival chances by reducing the amount of time between crash and rescue. The following sections present criteria that should be followed to ensure the satisfactory operation of a crash locator beacon installed in an aircraft.

8.2 TYPES OF LOCATOR BEACONS

Crash locator beacons can be either automatically or manually activated, and they can be permanently installed (fixed), portable, or automatically deployed.

8.2.1 Automatic and Manual Activation

Automatic activation of the transmitter is triggered when the crash sensor detects a preset impact condition. Such activation requires no previous action on the part of the crew and, for that reason, is the preferred method. However, in military aircraft there are obviously times when it is not desirable to activate a locator beacon on a downed aircraft. For this reason, an arming switch should be included to provide the option of automatic activation depending on the aircraft mission. A manual activation switch also should be provided so that the crew can activate the beacon after the crash if the arming switch is not on or if, for any other reason, the beacon is not automatically activated.

8.2.2 Fixed, Portable, and Deployable Equipment

Fixed equipment is permanently mounted in the aircraft. The transmitter, antenna, and power supply need not be contained in one package, although their close proximity to each other lessens the changes for connecting circuitry to be damaged in the impact. Portable and deployable beacons, on the other hand, must include the transmitter, antenna, and power supply in one package.

80. EMERGENCY LOCATOR TRANSMITTERS: AN OVERVIEW, Report No. NTSB-AAS-78-1, National Transportation Safety Board, Washington, D. C., 26 January 1978.

Portable equipment is designed to be easily removed from its installation in the aircraft by crew members for use in a location remote from the downed aircraft if so desired. However, its installation in the aircraft must be secure enough to protect it from impact damage. In addition, an externally mounted antenna should be provided so that the beacon can function in the desired manner if it is not removed from the aircraft.

Automatic deployable equipment is automatically ejected from the aircraft when the crash sensor experiences crash forces equal to or greater than the present values. The equipment must be designed to minimize or withstand ground impact forces after ejection. Also, it should be buoyant, self-righting, and stable when floating in water, and not adversely affected by immersion in fresh or salt water for the life of the power supply. Automatically activated and deployable equipment should be considered for any aircraft that spends a considerable portion of its operating time over water.

8.3 COMPONENTS OF LOCATOR BEACONS

All aircraft-installed crash locator beacons contain the same basic components: a crash sensor, transmitter, antenna(s), power supply, activating switch, and associated electrical circuitry.

8.3.1 Crash Sensors

8.3.1.1 Types of Crash Sensors: Several general types of sensors can be used to detect aircraft crashes. These include:

- Ground contact switches that sense abnormal landings in which the fuselage or a wing tip is in contact with the ground.
- Proximity switches that sense the altitude of the aircraft and determine if the aircraft is below the normal level at which it would be supported on its landing gear.
- Deformation or damage indication switches that sense deformations of landing gear or wings and/or displacement of engines.
- Inertial switches that sense the decelerative loads applied to the aircraft.
- Attitude switches that sense sustained odd attitudes typical of aircraft wreckage after a crash.

- Aircraft system sensors that detect excessive pressure drops in fluid systems.

- Tachometer-operated switches that detect excessive changes in the speed of rotor blades.

If inertia switches are used, they must be capable of integrating acceleration over time so they can differentiate between a crash situation and normal aircraft operational accelerations. Two such sensors are currently available for aircraft. One is based on a mass held in place by a magnet, and the other is based on a mass resisted by a spring. These sensors are described in References 32 and 81.

Currently, the majority of crash locator beacons use inertia sensors to activate the transmitter. The FAA has specified crash force criteria for transmitter activation, although alternate means may be used if they can be shown to be substantially equivalent in their response to the specified crash forces (Reference 82). Regardless of the type of sensor used, the sensor must be responsive to the majority of survivable aircraft accidents, including those accidents in which the crash forces and damages are minimal. In this latter regard, an inertia sensor is probably more universally applicable than a sensor that responds to only local conditions in various parts of the aircraft.

8.3.1.2 <u>Criteria for Inertia Crash Sensors</u>: Inertia sensors must be designed to respond to the aircraft crash loads but to ignore normal operational loads. These two environments must be defined before the sensor criteria can be determined.

8.3.1.2.1 Crash Environment: A detailed analysis of over 500 survivable Army aircraft accidents has resulted in the definition of crash pulses for both light fixed-wing and rotary-wing aircraft. (This effort is discussed in Volume II.) The analysis provided frequency of occurrence data for average fuselage floor accelerations and aircraft velocity changes experienced during the crashes.

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81. Johnson, N., and Sanderson, S., SPILLED FUEL IGNITION SOURCES AND COUNTERMEASURES, Ultrasystems, Inc., Dynamic Science; Report No. DOT-HS-801 722, U. S. Department of Transportation, National Highway Traffic Safety Administration, Washington, D. C., September 1975.

82. Federal Aviation Technical Order, EMERGENCY LOCATOR TRANSMITTERS, (TSO) C91.

The median average longitudinal acceleration was only 2.7 G, with 50 percent of the accidents experiencing lower accelerations. Since the crash locator beacon should activate in as many survivable accidents as possible, any G-type sensor must respond to acceleration levels as low as 2 G to capture 75 to 80 percent of the accidents. However, combinations of vibrational loads and flight loads can sometimes exceed this acceleration value. Therefore, some other parameter, such as velocity change, also must be specified to prevent inadvertant sensor actuation.

The median longitudinal velocity change determined for the aircraft crashes was 28 ft/sec, with 95 percent of the accidents experiencing a velocity change of 13 ft/sec or more (see Figure 65). Thus a G-sensor could be designed to filter out the vibration and flight environment, even with a low 2 G threshold level, if it also must detect a velocity change typical of crash rather than operational conditions.

Fixed-wing aircraft rarely have sizable vertical crash forces without also experiencing large longitudinal forces. Helicopters, on the other hand, can experience large vertical crash loads with minimal longitudinal loading. Average vertical accelerations were determined during the aircraft accident analysis mentioned previously. The median vertical acceleration was 3.2 G, slightly higher than that of the longitudinal accelerations. However, the average vertical acceleration for 80 percent of the accidents was 2 G or more, the same as for the longitudinal accelerations.

The median vertical velocity change was 24 ft/sec, with 95 percent of the accidents experiencing a velocity change of 10 ft/sec or more (see Figure 66).

The aircraft accident survey produced insufficient data to yield an accurate distribution of lateral crash accelerations or velocity changes. However, significant lateral accelerations were present in the crashes studied, particularly in some types of helicopter accidents. The data did suggest that the maximum lateral accelerations and velocity changes were somewhat less than those along the longitudinal and vertical axes.

8.3.1.2.2 Operational Environment: Operating conditions for fixed-wing aircraft can be estimated by using the vibration test loads specified by the Radio Technical Commission for Aeronautics (RTCA) in their performance standards for emergency locator transmitters (Reference 83). These loads are:

83. MINIMUM PERFORMANCE STANDARDS - EMERGENCY LOCATOR TRANSMITTERS, Document No. DO-147, Radio Technical Commission for Aeronautics, Washington, D. C., 5 November 1970.

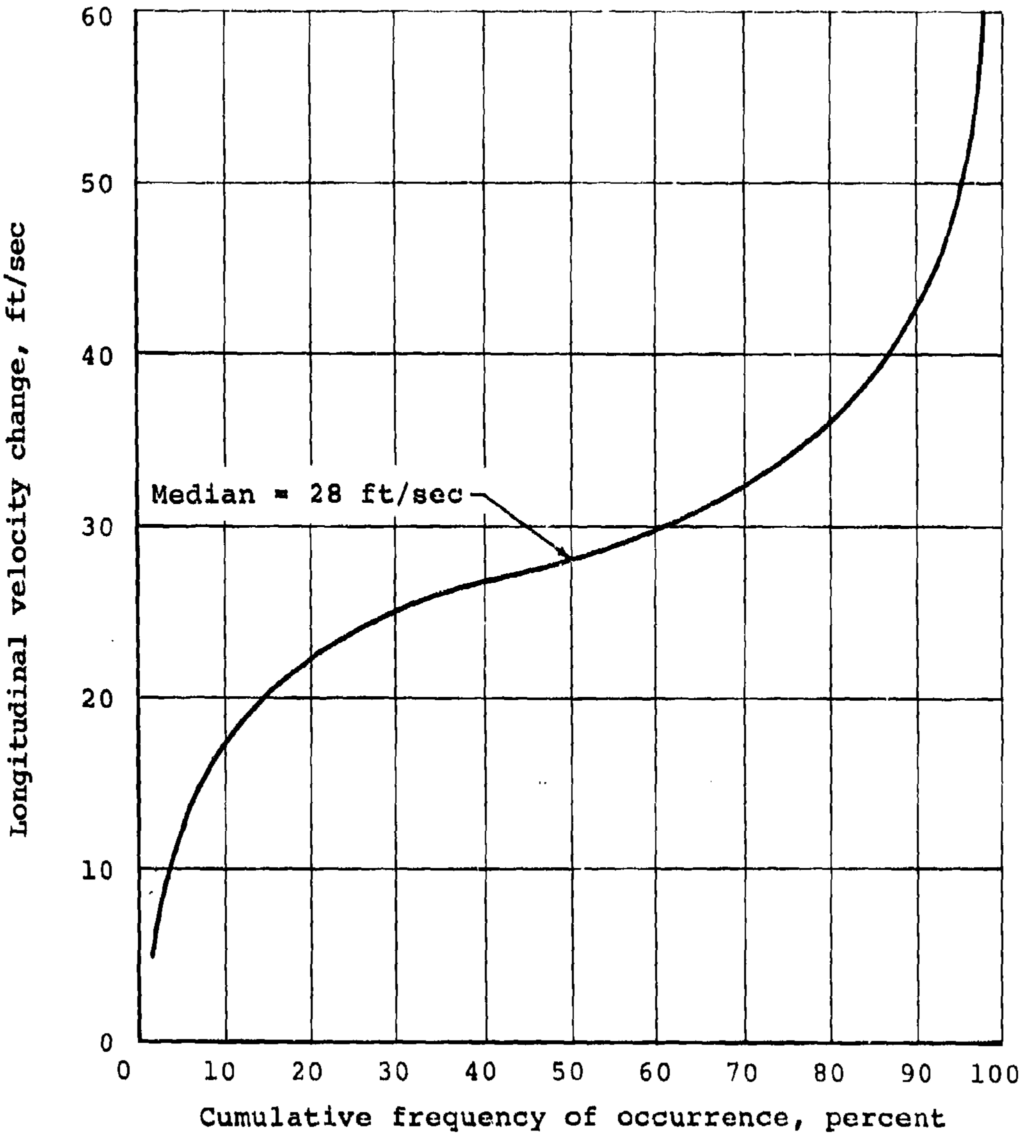

Figure 65. Longitudinal velocity changes for survivable rotary- and light fixed-wing aircraft accidents.

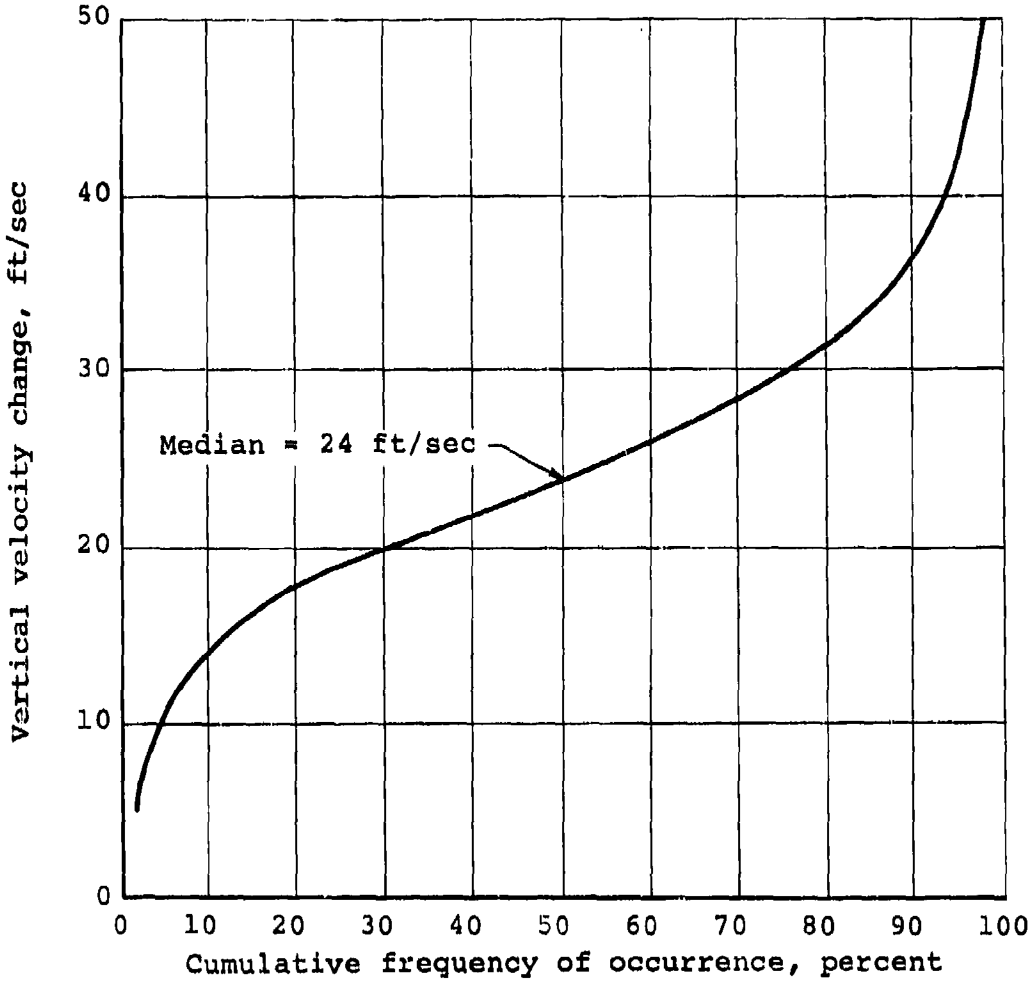

Figure 66. Vertical velocity changes for survivable rotary- and light fixed-wing aircraft accidents.

- Constant total excursion of 0.100 in. from 5 Hz to that frequency where a peak-to-peak acceleration of 10 G is reached.

- From the frequency determined above (31.28 Hz) to 2,000 Hz, a constant peak-to-peak acceleration of 10 G.

The total velocity changes under the half-sine waves corresponding to the above vibration spectrum are listed in Table 19. Although peak accelerations are greater than the desired 2 G

TABLE 19. VIBRATION SPECTRUM CHARACTERISTICS BASED ON RADIO TECHNICAL COMMISSION FOR AERONAUTICS, DO-147 REQUIREMENTS

Frequency (Hz)	Acceleration (±G)	Double amplitude (in.)	Velocity change (ft/sec)
5	0.13	0.10	0.262
10	0.51	0.10	0.524
20	2.07	0.10	1.047
30	4.62	0.10	1.571
31.28	5.00	0.10	1.633
40	5.00	0.061	1.279
50	5.00	0.039	1.038
100	5.00	0.001	0.513

threshold, the velocity changes due to the vibration spectrum are considerably less than those experienced during 95 percent of the aircraft crashes.

However, as pointed out by the Crash Research Institute, the above treatment is too simplistic because the vibration environment is actually superimposed on the flight environment (Reference 84). In developing crash sensor criteria for fixed-wing aircraft, the Institute assumed a possible 1.5 G longitudinal flight force (which could occur under the unlikely conditions of a flight load limit of 6 G and pitch attitudes of ±15 degrees of the flight path) and superimposed it on top of the worst vibration load listed in Table 19. The results are depicted graphically in Figure 67. The velocity change under these conditions (the shaded area under the curve) was found to be 2.7 ft/sec, still considerably less than accident velocity changes.

84. DEVELOPMENT OF ELT CRASH SENSOR PERFORMANCE SPECIFICATION AND TEST PROCEDURES - FINAL REPORT, Crash Research Institute, Robertson Research, Inc.; Contract No. DOT-FA76WA-3842, Department of Transportation, Federal Aviation Administration, Washington, D. C., June 1977.

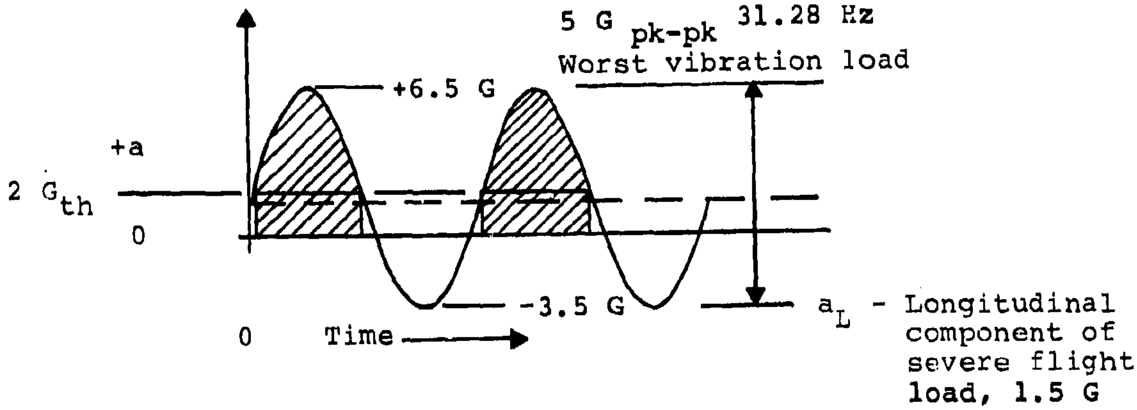

Figure 67. Worst vibration loads superimposed on severe longitudinal components of flight loads.

This same type of procedure can be used to estimate operating conditions for helicopters. The vibration spectrum used is that specified in MIL-STD-810B for equipment installed in aircraft and helicopters (Reference 85). The vibration test curve used (curve Z of Figure 415-1, Reference 85) is more severe than any of the other curves with the exception of those for equipment installed in the engine compartment, or pylon, or on the engine. The vibration spectrum characteristics for this curve are listed in Table 20. Although there are two inflection points in the MIL-STD-810B vibration curve as compared to only one in the RTCS curve, the velocity changes for each of the two curves are in the same general range.

Another parameter to be considered in aircraft performance is the vertical acceleration experienced by the aircraft during operational conditions. Vertical flight loads for helicopters can be significant and must be taken into account in determining the operational environment of rotary-wing aircraft. Assuming a 2.5 G vertical flight load (specified for the Army's Advanced Attack Helicopter) on top of the worst vibration loads of Table 20, the maximum vertical velocity change is 3.0 ft/sec. The maximum velocity change using the vibration curve of Table 19 is very similar -- 2.9 ft/sec.

8.3.1.2.3 Criteria: Aircraft accident data have shown that an inertia sensor must have an actuation threshold as low as 2 G acceleration in order to detect 80 percent of all aircraft crashes. However, the analysis of the previous section

85. Military Standard, MIL-STD-810B, ENVIRONMENTAL TEST METHODS, Department of Defense, Washington, D. C., 11 June 1967.

TABLE 20. VIBRATION SPECTRUM CHARACTERISTICS BASED ON MIL-STD-810B REQUIREMENTS

Frequency (Hz)	Acceleration (±G)	Double amplitude (in.)	Velocity change (ft/sec)
5	0.13	0.10	0.266
10	0.51	0.10	0.522
20	2.00	0.10	1.024
30	2.00	0.043	0.582
33	2.00	0.036	0.621
40	2.94	0.036	0.742
50	4.60	0.036	0.942
74	10.00	0.036	1.383
80	10.00	0.031	1.280
100	10.00	0.020	1.024
500	10.00	0.001	0.205

has shown that vibration and flight loads during normal operations can exceed this limit. The specification of a minimum velocity change that must be detected by the sensor before it will actuate, in addition to a threshold G level, can prevent actuation during normal conditions while assuring actuation during crash conditions.

The Crash Research Institute has recommended a threshold acceleration limit of 2 G and a minimum velocity change of 3 ft/sec for crash sensors in light fixed-wing aircraft (Reference 83). These specification limits are shown in Figure 68.

The velocity change of 3.0 ft/sec is above the maximum 2.7 ft/sec longitudinal velocity change considered typical of the light fixed-wing operational environment. This value also should be satifactory for helicopters, which experience even lower maximum longitudinal flight loads. The 3.0 ft/sec velocity change is only marginally satisfactory in the vertical direction, though, since expected vertical velocity changes

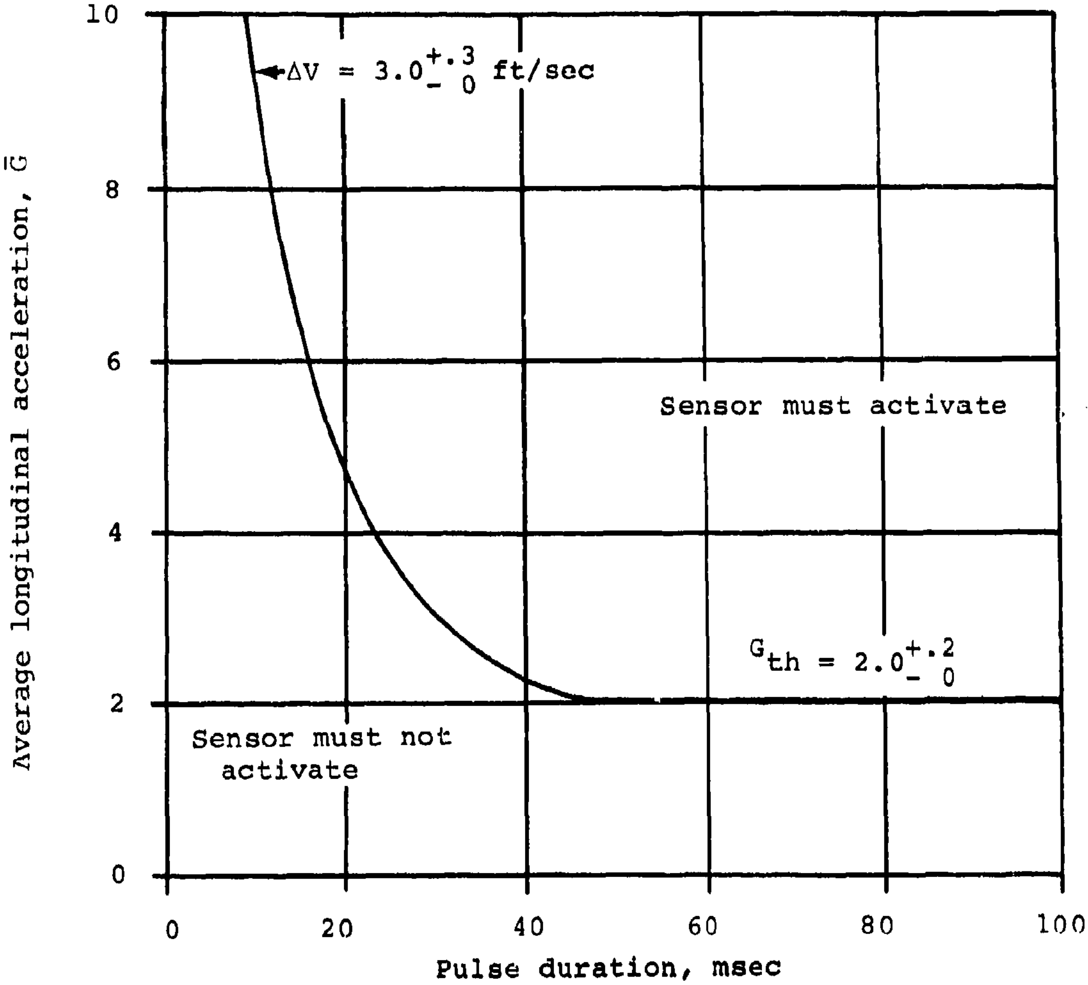

Figure 68. Proposed specification for crash force sensors in light fixed-wing aircraft.

for helicopter flight operations can be expected to equal or slightly exceed this limit. The problem is compounded by the inclusion of energy-absorbing landing gear that will allow a normal landing at sink speeds up to 10 ft/sec.

Since 95 percent of the aircraft accidents experienced vertical velocity changes of 10 ft/sec or more, the sensor criteria can reasonably be modified in the vertical direction to a minimum 10 ft/sec velocity change with a 2 G threshold level. The specification limits for this case are shown in Figure 69. In

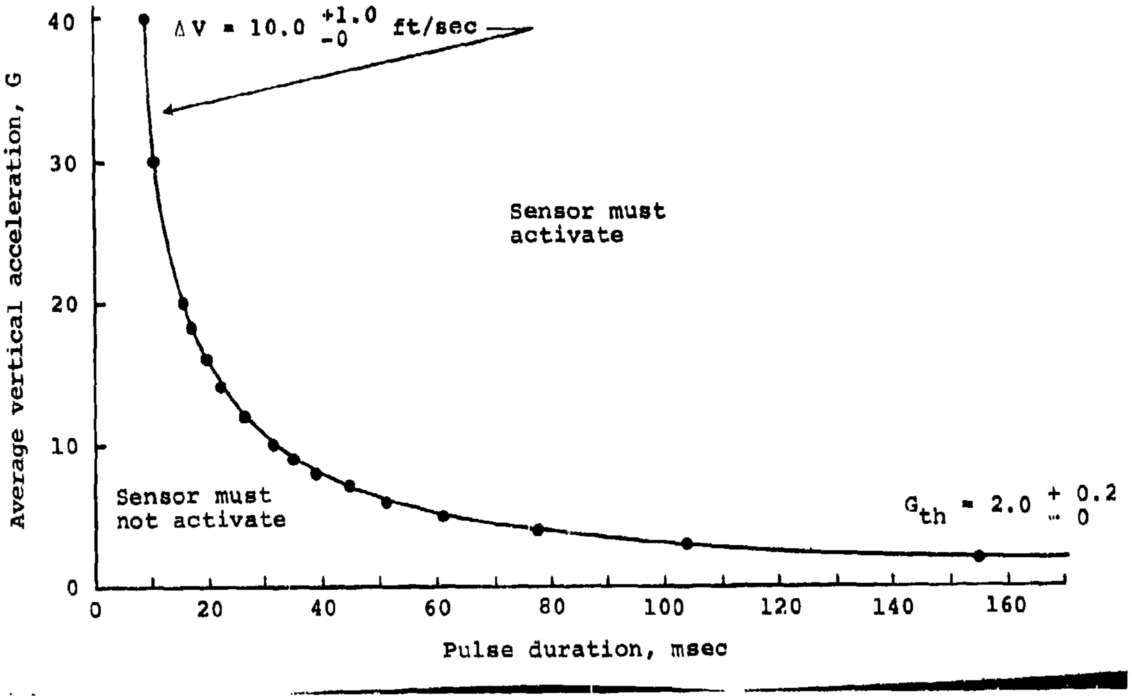

Figure 69. Proposed specification for vertical crash force sensors in rotary-wing aircraft.

fact, any desired velocity change can be combined with the 2 G threshold level to generate a family of curves similar to those in Figures 68 and 69. However, to prevent inadvertent actuation during flight maneuvers, the specified velocity change should not be below 3 ft/sec. Any velocity change above 10 ft/sec will reduce the number of accidents that will be sensed. (A velocity change of 15 ft/sec will sense approximately 85 percent of the accidents.)

Another parameter that must be considered is whether the crash sensor should be unidirectional, bidirectional, or omnidirectional. Inasmuch as most fixed-wing aircraft accidents have a major longitudinal component of velocity and force, a unidirectional sensor mounted with the active axis forward in the direction of the longitudinal axis of the aircraft should be sufficient. The situation is different for helicopters, which may have large vertical crash forces with minimal longitudinal forces. Thus, a vertically oriented crash sensor, as well as a longitudinal crash sensor, should be used in helicopters. As more data become available, lateral crash sensors might be indicated also.

The sensor must be able to withstand the impact forces associated with severe survivable crashes and still function. Thus, the sensor must withstand shock pulses equal to or greater than those listed in Table 21.

TABLE 21. SUMMARY OF DESIGN CONDITIONS FOR CRASH LOCATOR BEACONS

Impact direction	Velocity change (ft/sec)	Peak acceleration (G)
Forward	50	35
Downward	42	50
Lateral	30	25
Upward	-	10
Aftward	-	15

8.3.1.3 Criteria for Other Sensors: There are no generally applicable criteria for other types of crash sensors at the present time. Accident data to determine optimum sensor locations or deformations in the majority of survivable crashes are not available. However, because of the vibratory and installation problems that can occur with inertia sensors, the Crash Research Institute is currently working with NASA to develop a crash locator beacon that may use an "odd attitude-long exposure" type of crash sensor. This type of switch senses abnormal attitudes that persist for relatively long periods of time (e.g., 1 min). As yet, however, no criteria are available for this type of sensor.

8.3.1.4 Sensor Mounting: The inertia sensor criteria recommended in the preceding section are based on crash forces present in survivable crashes. These are the forces seen at the aircraft floor and thus are typical of the forces transmitted to the occupant compartment. Therefore, the crash sensor must be located in an area that will experience crash forces representative of those that will be seen in the occupant compartment. Of course, the sensor must be protected from any impact damage that could render it useless before it is able to activate the transmitter.

The sensor must be mounted to rigid structure, not flexible sheet metal bulkheads that may amplify or attenuate flight or crash loads. Amplification could lead to inadvertent actuation, while attenuation could result in a less severe crash being missed. For the same reasons, the crash sensor must be rigidly mounted to the structure. Soft mounting techniques, such as flexible straps or Velcro fasteners, must not be used. If portable beacons are installed, and if the sensor is integral with the rest of the equipment, the beacon must be rigidly mounted to the structure. Alternatively, the sensor can be rigidly mounted remotely from the rest of the equipment and a quick-disconnect provided for the connecting circuitry. This would allow easy-access installation of the portable equipment.

8.3.2 Transmitter

8.3.2.1 Mounting: The transmitter must be protected from any impact damage that could render it useless. Therefore, the transmitter must be located in an area that is not subject to impact damage. Generally, this would be above the aircraft floor and near the center or rear of the fuselage.

The transmitter must be able to withstand the impact forces of a severe survivable accident and still function as designed. This means, in addition to retaining its component and structural integrity, the transmitter's mounting to the aircraft must meet the same static attachment strengths as those specified for all ancillary equipment. These are:

- Downward: 50 G
- Upward: 10 G
- Forward: 35 G
- Aftward: 15 G
- Sideward: 25 G

8.3.2.2 Activation: The transmitter should be capable of being either manually or automatically activated. An arming switch that will allow the automatic activation capability to be selected or not, as desired, should be provided. Manual activation must always be available in case the sensor malfunctions or unusually low-level accelerations fail to trigger the sensor. The cockpit should be provided with a warning light that could alert the crew to inadvertent transmitter activation. A manual override switch should be provided so that the transmitter can be turned off whenever desired.

8.3.2.3 Operating Characteristics: Operating frequencies and transmitter ranges (power) must be determined by the procuring activity according to its own special needs. The FAA has specified a 75 mW peak effective radiated power since this would provide a transmission range of 50 nautical miles under unfavorable crash site conditions (Reference 82).

8.3.3 Antenna

Antennas, except for those used in portable and automatic deployable equipment, are usually mounted outside the aircraft to ensure the proper radiated signal strength and shape. Since survival of the antenna is critical to the successful operation of the crash beacon, care must be taken in deciding its location. It must be kept out of primary impact zones, such as the front or bottom of the aircraft, and it also should be kept out of secondary impact zones. These zones include wing or tail surfaces likely to impact trees, etc., and those portions of helicopters apt to experience rotor blade strikes during impact. The strength of the antenna attachment also must be sufficient to withstand decelerative impact forces.

8.3.4 Power Supply

The crash locator beacon must have its own independent power supply so that it is not dependent on aircraft power for its operation. The power supply should be capable of providing necessary power for optimum transmitter operation over a specified time period and under specified environmental conditions. These conditions should be specified by the procuring activity dependent on particular mission requirements. The FAA, in following the RTCA requirements, has specified that the power supply must be able to provide continuous operation for at least 48 hr under the condition of maximum power consumption (Reference 83). Maximum low and high operating temperatures are specified as -20°C (-4°F) and 55°C (131°F), respectively. However, the National Transportation Safety Board (NTSB) has pointed out that the low temperature standard is not sufficient to ensure operation of the crash locator beacon during winter in many areas of the United States, and recommends a low temperature standard of -40°C (Reference 80).

The power supply, if not integral with the transmitter, must be mounted to the aircraft so that it will not be torn loose or damaged during impact. It should be mounted in a location away from anticipated impact areas and should have an attachment strength equal to that specified for the transmitter.

8.3.5 Electrical Wiring

All electrical wiring between components of the system must be protected from impact damage unless the components are packaged together. Protection can be accomplished by routing the wires along the strongest structural members of the aircraft and away from anticipated areas of structural deformation. The wires should be attached to the aircraft structure with clamps or ties that will fail before the wires break. Twenty to 30 percent extra length in the wires, in the form of loops or S-shaped patterns, will allow the wires to move with deforming structure rather than be pulled apart. Nonconductive shields should surround the wires in all areas where structural crushing could occur.

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APPENDIX A

RELATION OF PAIN THRESHOLD TIME TO HEAT SOURCE TEMPERATURE

The pain threshold curves in Figure 7 (Section 3.3.1.1), which apply to visible or exposed areas of the skin, were generated from data in References A1, A2, A3, A4, A5, A6, and A7. The curves that account for variable radiating surfaces (F) were determined in the following manner.

The most significant variables that determine the rate of heat absorption by heat radiation are: (1) temperature of radiative source, (2) fraction of visible hemisphere at elevated temperature (F), and (3) emissivity of radiative source. Taking these factors into consideration, the rate of radiative heat absorption is

$$q_R = a \cdot e \cdot \sigma \cdot F \cdot T^4 \qquad \text{(A-1)}$$

where a = absorptivity of skin surface

e = emissivity of radiative source

σ = Stephan-Boltzmann constant

$= 4.88 \times 10^8$ kcal m^{-2} hr^{-1} $(^oK)^{-4}$

F = fraction of visible hemisphere occupied by radiating surface

T = temperature of radiative surface, °K

Assuming that skin absorptivity and source emissivity are both equal to 0.85, Equation (A-1) becomes

$$q_R = 3.50\ F\ \left(\frac{T}{100}\right)^4 \quad [\text{kcal } m^{-2}\ hr^{-1}] \qquad \text{(A-2)}$$

or

$$\left(\frac{T}{100}\right) = 0.73\ \left(\frac{1}{F}\right)^{1/4}\ (q_R)^{1/4} \quad [^\circ K] \qquad \text{(A-3)}$$

Equation (A-3) relates the temperature of the emitting source to the rate of heat absorption per unit area by the skin. (The emitter occupies fraction F of the visible hemisphere.)

Equation (A-3) was used to plot the radiative burn curves for four cases (F = 1.0, 0.5, 0.25, and 0.10) in Figure 7.

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READER'S SERVICE LETTER

(For use in submitting comments, recommendations, corrections, and revisions for *Aircraft Crash Survival Design Guide*)

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REMARKS:

SIGNATURE: ______________________________ **DATE:** ____________________