AFAL-TR-68-290
PART II



# LIGHTNING AND STATIC ELECTRICITY CONFERENCE, 3-5 DECEMBER 1968

## PART II. CONFERENCE PAPERS

TECHNICAL REPORT AFAL-TR-68-290, PART II

MAY 1969



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AIR FORCE AVIONICS LABORATORY
AIR FORCE SYSTEMS COMMAND
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# LIGHTNING AND STATIC ELECTRICITY CONFERENCE, 3-5 DECEMBER 1968

## PART II. CONFERENCE PAPERS

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# FOREWORD

The Air Force Avionics Laboratory, in cooperation with the Society of Automotive Engineers, sponsored the "Lightning and Static Electricity Conference," on 3-5 December 1968 in Miami Beach, Florida. This report presents in two parts the proceedings of this conference, as follows: Part I, Abstracts; Part II, Conference Papers. For this conference, C. R. Austin, Air Force Avionics Laboratory, served as conference organizer and compiled the information for the report. This is the final report of the conference.

Research described herein represents the efforts of many persons in many organizations. The papers as presented here differ somewhat from those originally presented because they have been edited for compliance with requirements for an AFAL Technical Report. This report was submitted in January 1969.

The Air Force Avionics Laboratory wishes to express its gratitude and appreciation to the following for their contributions in the conduct of this conference and preparation of this report:

Members of the steering committee, including Messrs R. Peterson, E. Rivera, T. Horeff, J. Moe, J. Robb, W. D. McKerchar, R. Banning, H. Schwartz, K. Moore, R. Bostak; Dr. J. Nanevicz and Dr. D. R. Fitzgerald; and R. L. Johnson, Major, USAF.

Mr. Philip J. Klass, Senior Editor, Aviation Week and Space Technology, for a stimulating presentation on "UFO's - An Atmospheric Mystery" and Professor M. M. Newman and associates of Lightning and Transients Research Institute for a spectacular demonstration of simulated lightning on the research vessel THUNDERBOLT.

Mr. C. E. Seth, ASNAC-30, and Mrs. Linda L. Dill, AVWE, who spent many hours in assisting in organizing the conference and preparing material.

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In the editing and preparation of the report manuscript, unusual unrewarded efforts were put forth by the editor, Mrs. Ruby Connerton, and by Mr. H. M. Bartman, Project Engineer for EMC research.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

Joseph A. Dombrowski

JOSEPH A. DOMBROWSKI
Lt Colonel, USAF
Chief, Electronic Warfare Division

## ABSTRACT

This report presents the text of the unclassified papers presented at the Lightning and Static Electricity Conference, 3-5 December 1968. These papers discuss problems of lightning and static electricity as they pertain to aerospace vehicles. Research to solve these problems conducted by numerous US and foreign agencies, both governmental and industrial, is discussed. The sessions covered include fluids and fuels, grounding and bonding techniques, survivability of nonconductive materials in a lightning environment, and control of static electricity effects on nonconductive materials.

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Organizer - J. D. Robb

Lightning & Transients Research Institute

Chairman, R. G. Stimmel

Air Force Avionics Laboratory

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# A REVIEW OF LIGHTNING PROBLEMS TO FLIGHT VEHICLES WITH SUGGESTED SOLUTIONS AND RELATED INTERFERENCE REJECTION TECHNIQUES

H. M. Bartman, Air Force Avionics Laboratory

## I. INTRODUCTION

Lightning hazards and related interference to avionics systems have been of concern to both military and commercial flight operations since the early days of aviation. The problems the Air Safety Board as early as July 1939 believed to be of importance included the following:

a. Will a serious fire occur in an aircraft if a lightning strike burns out a radio antenna?

b. Can a lightning strike freeze the bearings of a control surface?

c. Will the increase in aircraft size and speed allow the aircraft to accumulate still greater charge?

These questions were stated by Col A. F. Binney, U.S. Marine Corps, in his paper presented at the November 1948 symposium on Lightning Protection for Aircraft (Reference 1).

What has been accomplished since the last symposium and is being accomplished now will, I am sure, be covered thoroughly here. I would like to review what has been accomplished under Air Force sponsored programs and what must still be done.

Aircraft lightning protection has progressed from a simple spark gap on radio antenna bushings to extensive techniques for the protection of radomes, plastic sections, wiring, fuel systems, VHF antennas, etc. A few areas in which serious problems still exist include thunderstorm crossfields and induced surge voltages penetrating avionic subsystems. These problems have become more serious with increasing use of plastics and the resultant reduced shielding of internal wiring. Semiconductor devices are surge-vulnerable, even to short pulses. Since lightning current can puncture fuel tank skins, this must be considered in using new materials and techniques for aerospace vehicles.


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## II. HISTORICAL REVIEW

In the early '30's when communication and navigation equipments were first installed in commercial and military aircraft, it was found that severe radio frequency interference occurred in receiving systems whenever the aircraft was operated in precipitation containing ice crystals and in or near thunderstorms. Since the interference was often sufficiently severe to disable communications for hours at a time, it presented a serious flight safety hazard. The hazard was particularly serious on overseas flights, where fuel reserves often prevented avoiding thunderstorm activities. Although a great deal of information concerning atmospheric interference and its control was being accumulated, this noise source was still considered a serious flight hazard in 1943; therefore, a joint Army-Navy precipitation static project was initiated. This extensive program involved flight test to devise ways to obtain more effective interference control techniques. Several new methods of radio interference control were proposed, the most important of which were the use of wick dischargers and dielectric coated antenna wire. Both of these became standard equipment on aircraft. Upon the completion of the precipitation static project in 1946, the Avionics Laboratory, then known as the Communication and Navigation Laboratory, separated precipitation static and lightning electrical hazards into two task areas. Since that time, research has been pursued under Project 4357 as specialized research efforts involving the reduction of aircraft lightning hazards and related radio interference. New problems are being encountered due to the fast-changing techniques in aircraft design and the rapid development of missile systems, space vehicles, materials, and interface technology.

Considerable progress has been made during the past years. Practically all U. S. military and civilian aircraft now incorporate lightning protective equipment developed under Air Force sponsored contracts. Evidence indicates that the protection techniques for antenna systems, radomes, canopies, fuel systems, etc., has made the difference between destruction and survival of jet aircraft struck by lightning. Figure 1 shows milestones of interest.

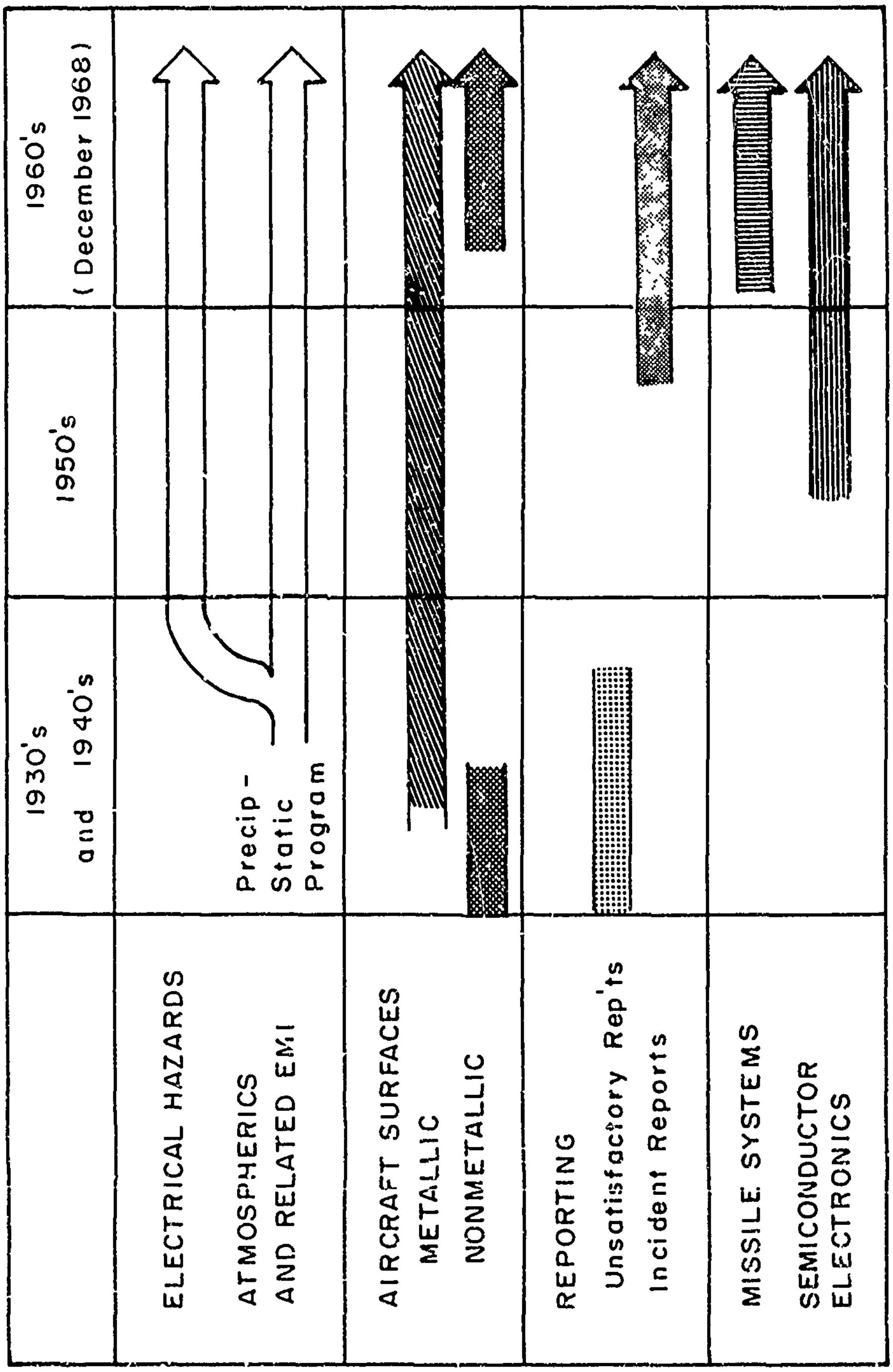


Figure 1. Lightning Hazards and Related Interference Problems

1. LIGHTNING SYMPOSIUM

Through the years, the Air Force and Navy have continuously supported specialized research programs for the reduction of aircraft lightning hazards and related radio frequency interference. The Navy's prime area of interest has been lightning protection to airships and early warning radar pickets aircraft. This symposium should be a major milestone in the exchange of information important to the implementation of the latest technology to present and future airborne systems. Certainly, it should result in a comprehensive updating of the lightning hazard control handbook.

2. LIGHTNING ELECTRICAL HAZARDS REDUCTION DEVELOPMENT

Lightning protection technology resulting from Air Force and Navy cosponsored programs is extensive. As shown by the time line in Figure 1, starting with the initial efforts under the precipitation static program to the present, such devices as the wick discharger for precipitation static interference control, lightning arrestors, radome and canopy lightning protection, and lightning divertors have been developed (Reference 2).

3. ATMOSPHERICS AND RELATED RADIO INTERFERENCE

Since the advent of sensitive broadband receivers, communications and navigation equipment have been plagued by loss of information during natural disturbances such as precipitation static and atmospheric static interference. The time line in Figure 1 shows that, parallel to lightning electrical hazards studies, consideration was given to precipitation static and atmospherics as radio noise sources. A need was established to develop interference control techniques, such as the interference blanking circuitry. Developmental efforts were conducted in the late 50's, which have been reactivated in recent years because improved solid state devices have become available (Reference 3).

4. METALLIC AND NONMETALLIC AIRCRAFT

The time line in Figure 1 indicates interest in regard to nonmetallic aircraft, or canvas-covered aircraft, in the early thirties and the attendant electrical hazards. Recent interest has resulted from the increased use of plastics and other nonmetallic composites in the structure and skin of advanced aircraft designs. A current study in this area sponsored jointly by the Air Force Materials Laboratory and the Air Force Avionics Laboratory should provide a better understanding of the hazards associated with the use of these new materials (Reference 4).

### 5. UNSATISFACTORY REPORTS AND INCIDENT REPORTS

Prior to the late forties, the Unsatisfactory Report was the channel through which higher headquarters was informed of equipment problem areas, including incidents of lightning damage. These reports were phased out after World War II. A new approach initiated by flight safety activities, known as the Incident Report, was started in the late fifties to report lightning damage incidents. These reports have proved useful in determining future research requirements and quick reaction fixes to flight hazards.

### 6. MISSILE SYSTEMS AND SEMICONDUCTOR DEVICES ELECTRONICS

In the middle fifties, these problems involved such matters as electrical transients from a nearby high current surge or electrical power line transient causing an airborne computer to malfunction. The rapid development of missile systems, space vehicles, solid state materials, and interface technology has resulted in new problems in the sixties.

## III. ELECTRICAL HAZARDS

Naturally, a major surge in research activities has occurred after each major accident in which lightning was possibly the cause. These temporary spurts of activity have undoubtedly increased the pace of lightning protection research and the development of techniques, but the moderately funded, continuously supported efforts, such as the Air Force - Navy programs, have provided steady progress in solving serious electrical hazards problems. The following examples of protection techniques developed during the past years are indicative of lightning electrical hazards experienced in the past with the corrective actions taken (References 5 and 6).

1. FUEL VENTS

One of the most hazardous aircraft lightning problems, both past and present, is probably the high surge current penetrating the surface and causing electrostatic sparking inside of aircraft fuel systems. Solutions to fuel hazards problems are difficult because control techniques, such as the use of a flame arrestor at the fuel vent, can introduce other hazards, such as icing of the flame arrestor, which may be more serious than the lightning.

2. DIVERTERS

Since little can be done to prevent lightning from striking an aircraft, some means was sought to protect the most vulnerable areas. A graded resistance lightning diverter rod was developed to direct strikes to points on the aircraft designed to carry high current strokes without damage to the surface or structure and where excessive radio interference would not be produced.

3. ANTENNA AND AVIONIC EQUIPMENTS

All possible points where the stroke might enter the aircraft, such as the antenna lead-in conductors, must be protected. Airborne antenna lightning arrestors for protecting communications equipment have been developed to lead a lightning discharge on the aircraft antenna lead-in safely to the airframe; this arrestor consists of series blocking capacitors and by-pass shunt gaps. This approach is not a cure all, however. Care must be exercised in the application of this technique to prevent parallel components to the discharge gap having a lower impulse breakdown voltage from being damaged by the lightning stroke.

4. RADOME LIGHTNING PROTECTION

As the use of radomes has become more common, lightning protection has been provided in many cases by arranging solid metallic conductors in such a way as to produce minimum interference with radar operation and maximum protection against lightning discharge to the antenna elements within the radome. Aluminum foil strips were also considered for this purpose. Conductive coatings were generally used over the entire radome surface to reduce precipitation charge streamering and the possibility of static puncture, as well as the electrical gradient and the tendency for corona discharge from the protection strips.

## IV. APPLICATION OF PRESENT TECHNIQUES

Present lightning hazards reduction techniques would be difficult to apply on a handbook basis, but they could greatly reduce possible hazards if they were judiciously applied to new designs or design techniques. Generally, the most difficult problem for the aircraft designers is to recognize hazardous design configurations that require protective techniques. A guiding philosophy may be derived from considering the unusually good record of the DC-6 aircraft. The outer skin of this aircraft was very close to being a perfect faraday shield, with wing-to-wing resistance of the order of 15 microhms. With a fuel which is overrich throughout most of the flight regime, particularly in the temperature range at which most lightning discharges are recorded (+32° ±5°), any deviation from this protection for this type of aircraft could be dangerous.

A unique, experience was encountered in applying state-of-the-art lightning protection techniques to the C-141 aircraft. The techniques were applied early in the design of the aircraft and were exposed to full-scale component testing. Decisions as to fuel vent locations were made early in the program, with full consideration given to lightning protection aspects. The C-5A weapons systems is also being developed along these general guidelines.

To provide good lightning protection, new aircraft designs should be reviewed for such potential hazards as plastic sections through which discharges could penetrate and interfaces that could channel a discharge into the interior of the vehicle or produce internal sparking.

## V. CURRENT DEVELOPMENTAL EFFORTS

Current technical efforts include the investigation of nonmetallic materials and components as related to lightning strike effects, methods to define critical lightning hazards problem areas, practical solutions for upgrading present specifications, and the development of impulse interference rejection techniques (References 7 through 10).

### 1. NONMETALLIC MATERIALS

Advanced composites, such as boron and graphite fiber reinforced plastics, which have recently been developed, are being considered for use in major structural components of aircraft such as the wings, fuselage, and empennage. Lightning protection systems developed for radome structures should be adaptable in principle to these reinforced plastics. Fortunately the Air Force Materials Laboratory and the Air Force Avionics Laboratory have been able to join forces in determining the effects of lightning strikes to these advance composites; this should prevent costly redesign later.

2. RELATED INTERFERENCE REJECTION TECHNIQUES

Since their inception, sensitive broadband receiver communications and navigation equipment have been plagued by the loss of information during such natural disturbances as precipitation static and atmospheric static interference. It is not always possible to remove the source of this interference. Crossfield and thunderstorm conditions, which produce corona static discharges on aircraft extremities, and RF interference produced by induced current surges from a lightning discharge and electromagnetic pulse (EMP) cannot be reduced at the source. The most practical method of effectively reducing such interference is to use electronic interference rejection techniques (circuitry). An HF receiver recently developed with such circuitry has an impulse blanking gate, RF limiter, and squelch circuit; this device exhibits a nominal signal improvement of 50 db. The block diagram in Figure 2 illustrates this novel approach.

a. Interference Blanker System

The problems plaguing previous blankers have fallen generally into two categories; (1) placing the blanking action too far back in the receiver, necessitating a long blanking time and thereby losing information, or (2) a blanker that could not discriminate between transmitted RF signals and atmospheric noise. The blanker shown in the block diagram of Figure 3 overcomes these shortcomings. An interference pulse entering the antenna is split four ways by a delay line to the receiver and three tuned circuits. The center frequency of the amplifiers is in the tunable range of the receiver. The output of these amplifiers is a pulse that is detected and compared to a reference level threshold. If the pulse exceeds the threshold level, the output of the NAND-gate blanks the receiver for a specific length of time; this time is adjustable.

If a strong RF signal not in the pass band of one of the amplifiers is present, it will not blank the receiver because the NAND-gate needs three positive

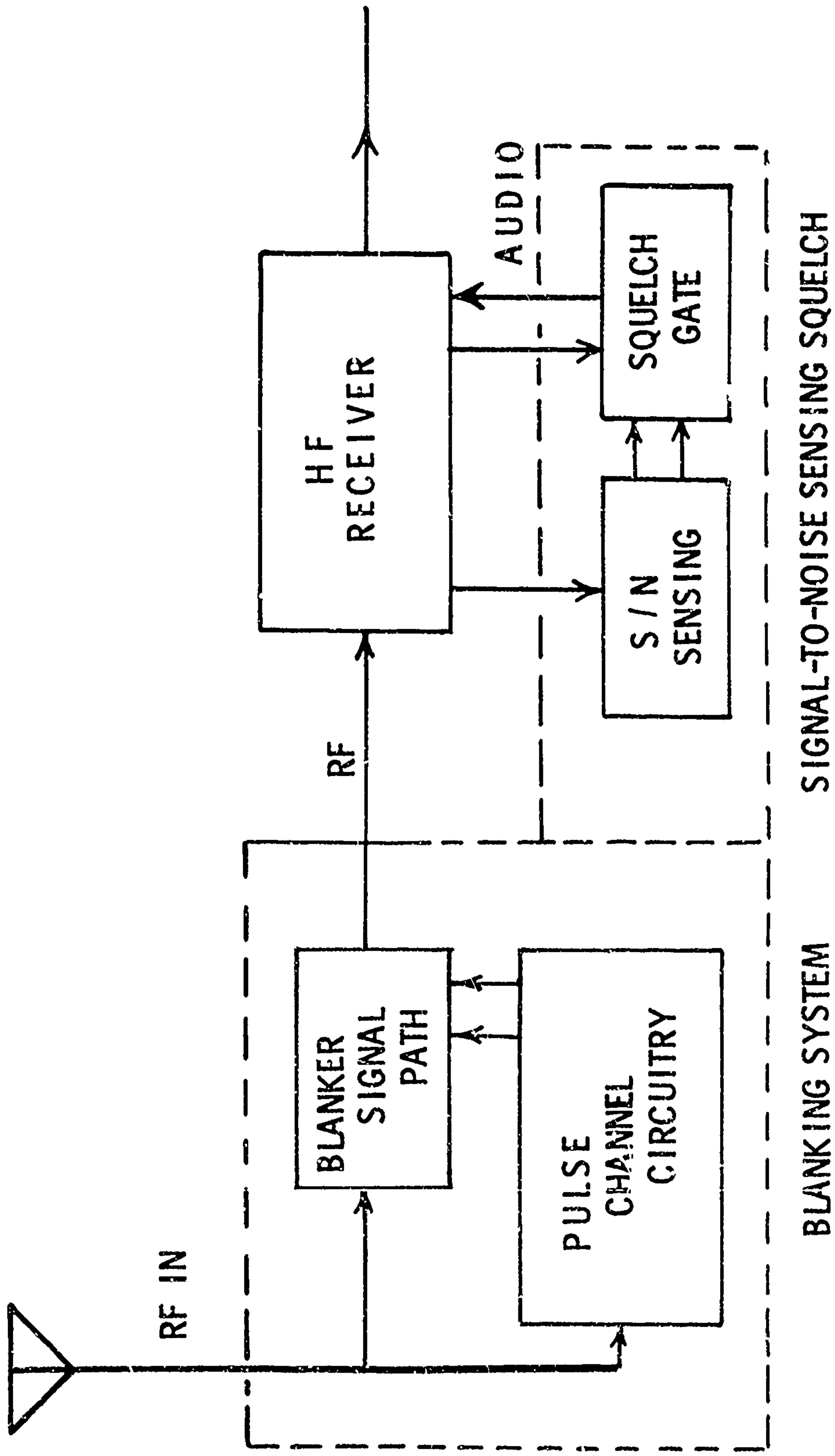


Figure 2. Interference Reduction System

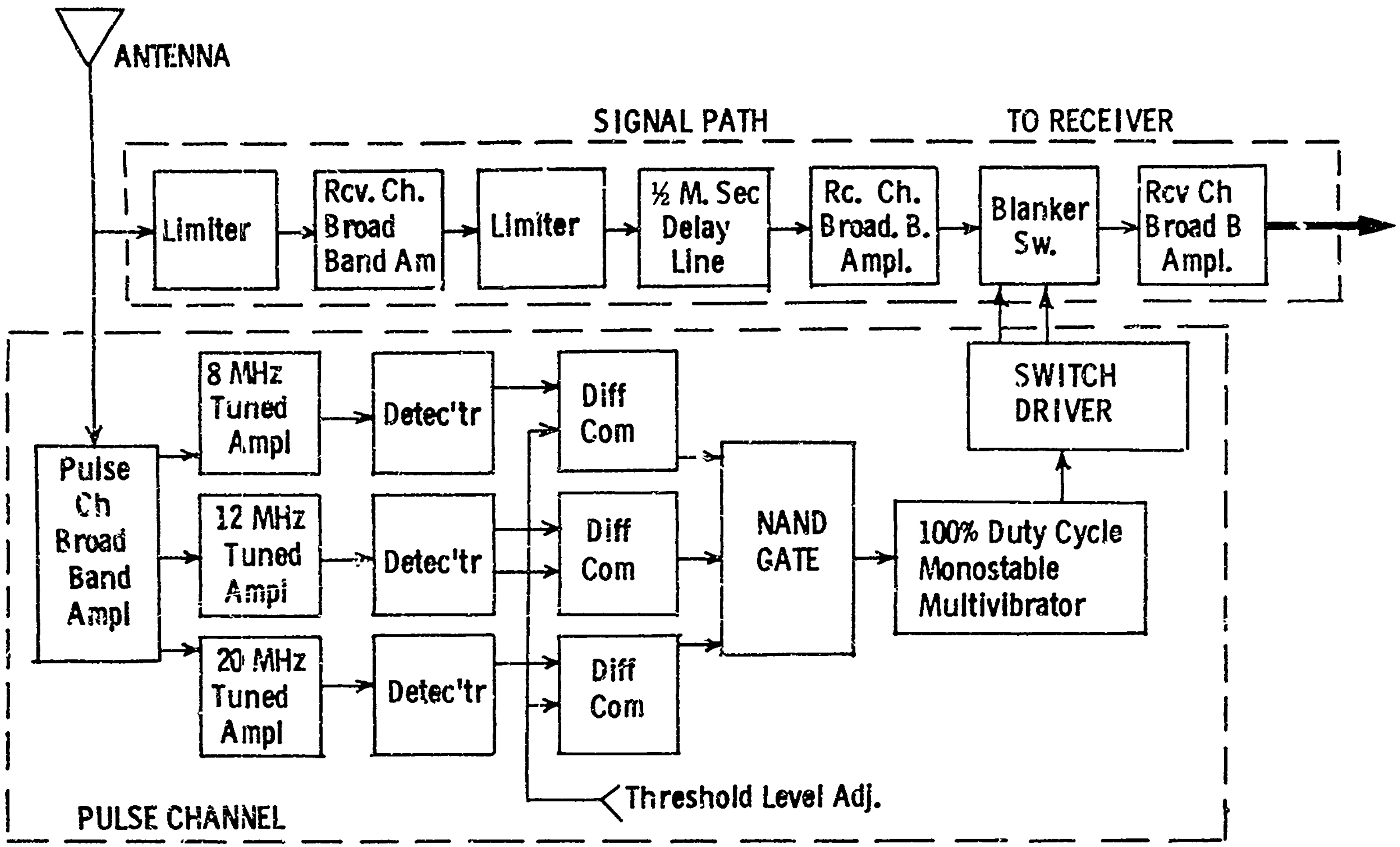


Figure 3. Blanking System Block Diagram

coincidental inputs from an impulse. Even if all three impulse amplifiers are receiving signals (an unlikely condition), the blanker will only blank once because the detectors are AC coupled. This system makes use of the spectral content of desired signals versus impulse noise. The blanking system under development, which is all solid state, is used with an HF receiver and can be used with an external receiver in that frequency range.

b. RF Limiter

A solid state RF limiter has also been developed which limits the interference pulses to amplitudes in the order of +6 dbm with input levels as high as +20 dbm. The limiter, which also acts as a low pass filter, has been incorporated in the input of the blanking system as shown in Figure 3. It can also be used as a separate component between an antenna and a receiver input. Maximum insertion loss over the frequency range of 0.1 to 30 MHz is 0.8 db.

c. Squelch Circuit

A squelch circuit to be used in reducing atmospheric noise effects was also developed and appears in block diagram form in Figure 4. The principal of this circuit is based on carrier-to-noise ratio rather than AGC voltage. This gives the advantage of keeping the squelch enabled. No audio output occurs when atmospheric noise alone is present; that is to say, in a normal receiver the atmospheric noise could be interpreted as a signal disabling the squelch to deliver audio atmospheric noise. When atmospheric noise comes into the 455 KHz IF, it is amplified, limited, and detected to produce audio noise. A 455 KHz notch filter removes all 455 KHz present in the noise at the output of the detector. This is followed by a power detector which operates the squelch. When a desired signal is received, the limiter reduces the noise power, since the limiter keeps a constant total power output, and the circuit will pass the audio signal. The threshold can be varied over a range of approximately 40 db. The gate driver and squelch gate operation is such that no audible switching clicks are heard when the gate is activated.

## 3. UPDATING OF SPECIFICATIONS AND HANDBOOKS

Available techniques developed under AF sponsored contracts are, with few exceptions, adequate for existing flight vehicles. However, wide gaps exist in

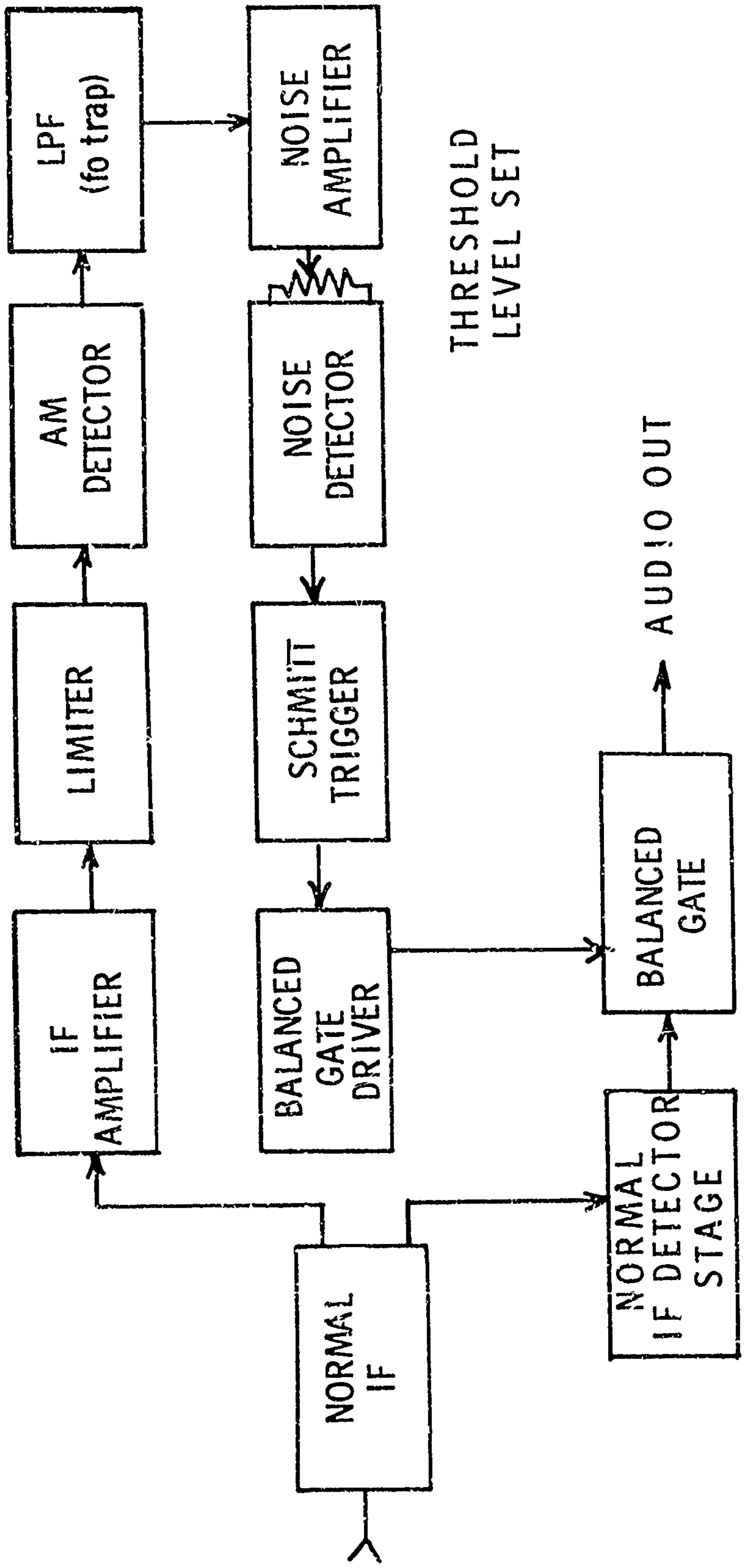


Figure 4. Squelch Circuit

knowledge concerning the power levels that may be induced in new generation avionics subsystems and control devices. Because of the complexity of this problem and because the quantity of current and voltage that lightning strikes can induce into these highly sensitive avionic components is an unknown quantity, we must be accurate in defining all parameters applicable to the adverse lightning environment. High standards of lightning protection should be carefully considered in the early design stages of aerospace vehicle systems to reduce the systems lightning vulnerability at minimum cost. Unfortunately, lightning protection design techniques are not amenable to handbook specification.

Theoretical models of electrical hazards phenomena, such as the generation, propagation, and effects of electrical surges on aerospace vehicles, have been quite difficult to obtain. A number of the techniques now available were empirically derived and have limited theoretical backing. There is interest in the development of theoretical models for obvious reasons, although the primary objective up to the present time has been to develop practical solutions to electrical hazards problems.

4. METHODS OF DEFINING CRITICAL LIGHTNING HAZARD PROBLEMS

Analysis and measurement techniques for the designer to use in either predicting or assessing the magnitude of a lightning electrical hazards problem should be considered. The capability for analyzing unique or generally unpredictible problem areas, where flight vehicle vulnerability could result in loss of the vehicle or personal injury, would be a great step forward in preventing future aircraft from incidents or accidents resulting from lightning or related natural interference phenomena.

## VI. FUTURE PROBLEM AREAS

Unique lightning hazards and related RF interference problems are continuously being encountered in new aircraft. New problems arise because of the fast-changing techniques in aircraft design and the rapid development of missile systems, space vehicles, materials and interface technology and require careful consideration in the early design stages of the systems. Reduction of lightning vulnerability of these systems can be accomplished early in the concept and design definition phases at greatly reduced cost. Some of the problem areas requiring serious consideration are the use of nonmetallic materials, integrated antenna and interface techniques as related to electrical transients, and development of modeling techniques to aid the designer in the prevention of lightning electrical hazards and related interference to aerospace flight vehicles and avionics systems.

## VII. SUMMARY

This paper has presented a brief review of flight vehicle lightning hazards problems and related RF interference, with consideration of what has been accomplished up to the present and what must still be done to reduce system vulnerability to lightning and static electricity. Through the years from 1946 to the present, the Air Force and Navy have had jointly sponsored efforts to reduce aircraft lightning hazards and related RFI. Lightning protection technology resulting from these programs is extensive and has been applied to many military aircraft. A few areas still exist in which the state of the art is deficient in the area of electrical hazards reduction and lightning protection for new aircraft. New problems are continuously arising due to the fast-changing techniques in aircraft design and rapid development of missile systems, spacecraft, materials, and interface technology.

The loss of communication and navigation information during natural disturbances, such as precipitation static or atmospheric static, has necessitated the development of electronic interference rejection circuitry. A new blanking approach using solid state components is now available for the control of RF interference from lightning discharges, electromagnetic pulses, precipitation static, and atmospheric static.

## ACKNOWLEDGEMENTS

The author has drawn extensively from Air Force project files for much of the information presented herein. The following are a few who have contributed research efforts that have been utilized herein without reference: Messrs P. W. Couch, H. C. Storch, R. G. Stimmel, H. S. Schwartz, and C. R. Austin of the Air Force; M. M. Newman and J. D. Robb of the Lightning and Transient Research Institute; and G. J. Palladino, and R. Sugarman of the American Electronics Laboratories.

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# U. S. NAVY RESEARCH AND DEVELOPMENT PROGRAMS ON LIGHTNING PROTECTION FOR AIRCRAFT

Emilio R. Rivera

Naval Air Systems Command

Formal lightning protection programs by the U.S. Navy date back to the first high-frequency antenna lightning arrestors for aircraft developed under a contract with Lightning and Transients in 1948 as a part of a general lightning research program. The aim of the contract was to develop and originate information on lightning protection of HF antennas. Flight research programs had not produced a great deal of information regarding severe natural lightning discharges which strike aircraft. Special arrestors were designed, as illustrated in Figure 1. These arrestors not only provided protection for the aircraft HF radio system, but also provided data as to the current crests and charge transfers which strike the HF antenna.

As illustrated in Figure 2, the system utilized a series capacitor between the antenna and the radio equipment, which blocked the lightning stroke energies. A shunt spark gap to the aircraft fuselage bypassed the lightning stroke energies safely into the aircraft skin. The spark gaps had a replaceable cartridge made of a brass spherical electrode for indicating the charge transfer and a small magnetic link cartridge to indicate the current peaks. A shunt resistance was used across the spark gap in order to bypass precipitation-static charge to the aircraft skin without producing radio interference. In terms of modern oscillographic recording of lightning currents, these represent a fairly crude device. In terms of the technique and equipment available at the time, and particularly in view of the expense involved in installing this equipment on an entire fleet, however, the units represented a good compromise. Spark gaps also provided data on the severe natural lightning strikes that occasionally occur to commercial aircraft. Photographs of the spark gap and cartridges, opened up to illustrate the severe gap pitting, are shown in Figure 3; also shown for comparison is the pitting produced by typical severe laboratory lightning discharges.

These lightning arrestors provided the first quantitative data on natural lightning discharges to aircraft. In addition, they provided protection for the high frequency radio equipment. The Air Force extended this development to the high-frequency, high-voltage lightning arrestor for the flush type antennas in current use. Private equipment manufacturers have extended these designs for use on most of the modern commercial and military jet transport aircraft.

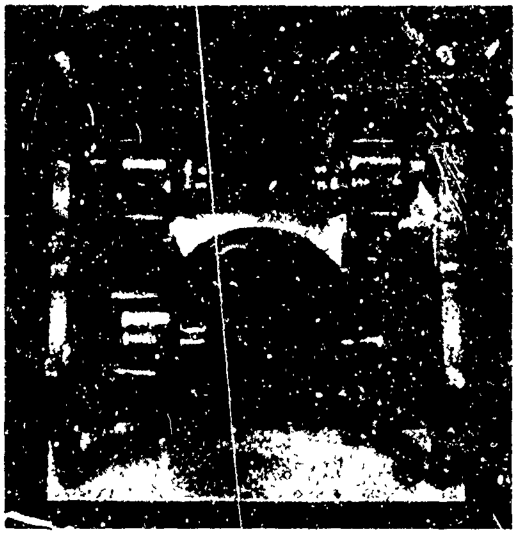

Figure 1. Photograph of Early Lightning Arrestor Design

Other earlier Navy lightning protection development included protection programs for large radomes such as those on early warning aircraft. Early warning aircraft are subjected to severe storms, as they are required to stay at a given geographic location regardless of weather. Their incidence of lightning strikes is extremely high during the thunderstorm seasons. Protection techniques were extended to include radomes. Because of the broad frequency ranges used, special protection techniques such as the use of aluminum-particle paint to enhance surface flash-over on the outside of the radome were required to reduce the liklihood of puncture. Other programs included the development of a protection system for the nose observer dome on patrol aircraft. Injury to personnel in these domes led to developing a protection strip across the forward section of the dome. Of course, lightning can produce shock, injury, and even fatality to personnel through acrylic enclosures if proper protection is not provided. The thick high-dielectric-strength acrylic, such as is used on fighter aircraft and observation domes generally, reduces the likelihood of puncture, but fatalities have occured from puncture of fighter aircraft canopies. Shocks to personnel, including numbed hands or feet, are not uncommon.

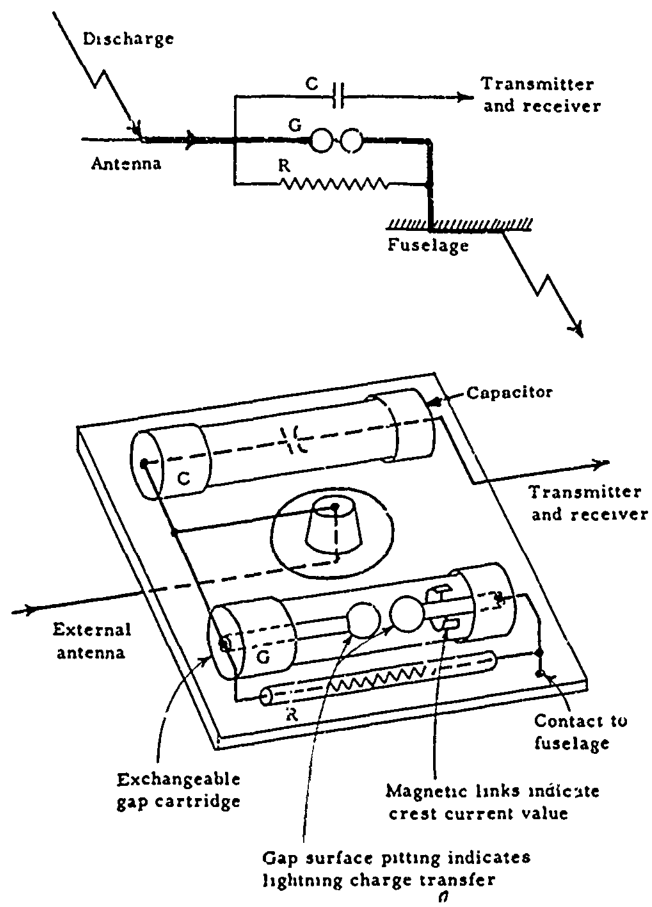


Figure 2. Schematic of early lightning arrestor showing series capacitor, shunt spark gap and static drain resistor.

Navy research programs have been directed toward preventing lightning damage to aircraft structures, such as MAD booms, radomes, and electronic and electrical systems. More recently, under a cooperative program with the Air Force, efforts have been directed toward the problem of protecting boron epoxy and graphite composite structures. These materials are suggested for use in new aircraft for their excellent mechanical properties. (The mechanical properties will be discussed in separate papers.) However, these materials are particularly

Figure 3. Photographs of spark gap cartridge and pitting marks on sphere gap which has passed moderately severe natural lightning discharge.

vulnerable to lightning damage if unprotected. The most promising protective approach appears to be to coat the outside of the aircraft with spray aluminum or thin foil aluminum. This coating provides weather protection as well as one-shot lightning protection. The principal objection to this approach is the maintenance and repair costs.

Aircraft with metal-aluminum skins have provided safety to personnel and equipment during thunderstorms and lightning strikes and present lightning protection devices have saved lives and prevented equipment damage. However, new proposed plastic aircraft skins will require other types of lightning protection and static reduction devices. Such changes in construction are nearly as great from the lightning protection standpoint as were the changes from wood to metal.

Another area of concern is lightning protection for helicopters. These aircraft are inherently much more vulnerable to lightning damage because of the exposure of moving structural parts such as tail rotor and main rotor blades. Problems with helicopters are expected to increase because of more extensive operation in bad weather. This automatically requires greater attention to proper design of fuel sys' 'ms, structures, plastic surfaces of all kinds, navigation lights, and antennas. Additional effort is required to develop protection devices that will enhance their safety of flight during all-weather operation.

In conclusion, Navy lightning protection investigations have been carried on for the past twenty years. Under a mutual Navy-Air Force program, joint investigations and cooperation in this important aspect of aviation safety have been conducted. It is expected that investigations will continue to solve the more serious problems anticipated in the new generation of aircraft.

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PART II

# FAA LIGHTNING STRIKE EXPERIENCE ON AIRCRAFT AND DESIGN REQUIREMENTS FOR FUEL SYSTEMS

Robert J. Auburn

Federal Aviation Administration

The general problem of protecting against the possible hazards of lightning strike discharges has been recognized for many years and is reflected in the requirements and practices which the aircraft industry follows in designing and building aircraft. Nevertheless, it became apparent during the investigation of the 1963 accident at Elkton, Maryland, that perhaps the effects of lightning strikes on aircraft were not sufficiently understood to enable the design of adequate protection. Witnesses to the accident reported the occurrence of extremely severe lightning discharges in the area at the time of the inflight explosion and fire. Lightning burn marks and pitting were discovered on wing tip surfaces found in the wreckage. Examination of the wreckage showed that explosive forces had occurred in the fuel tanks.

After an exhaustive study of all the evidence recovered in that accident, a report (Reference 1) was published by the Civil Aeronautics Board (now National Transportation Safety Board). This report constitutes a comprehensive review of the results of the investigation, analysis of the evidence, and the results of specific investigations carried out under the FAA sponsored programs. It states that wreckage distribution, supported by the fire damage pattern, leads to the conclusion that the initial explosion was in the left reserve tank and that fire damage did not occur prior to the explosion. Since there was a direct correlation in time and location between the lightning discharge observed in the area and the accident, and the finding of lightning damage of recent origin on the wing skin, it was reasoned that the ignition of the flammable vapors was associated with the lightning strike. The Board determined that the probable cause was lightning-induced ignition of fuel air mixtures in the No. 1 reserve tank with resulting explosive disintegration of the left outer wing, followed by loss of control.

EARLY PRECAUTIONARY MEASURES

As bits and pieces of information were received and the accident investigation progressed, a number of precautionary actions were instituted by the FAA so that the maximum amount of protection could be realized during the period of time prior to a firm determination of probable cause, and availability of results from a number of research and development activities.

Notices were issued seeking cooperation between pilots and traffic controllers in exchanging information and avoiding active lightning areas. The FAA recommended to all operators of turbine-powered aircraft that the full complement of static discharger units be installed and maintained on all aircraft at approved locations. A survey was instituted to re-examine other aircraft in service and evaluate the effectiveness of their lightning protective features. This included examinations of the fuel tank locations in relation to extremities, thickness of fuel tank skin, location of vent outlets, and bonding.

The Agency issued two airworthiness directives requiring improvements to the type aircraft involved in the accident. The first was to improve the electrical bonding and electrical continuity of the fuel tank access doors on the bottom tank surface so as to assure that they would not be a source of internal sparks. The second was to require the application of a thin metal overlay in the area of the fuel surge tank to improve the capability of the skin to resist penetration by a lightning strike. In addition, the Agency issued an advisory circular on "Flammability of Jet Fuels" giving characteristics and properties of fuels being used in jet transport aircraft and describing relative hazard factors.

INTER-AGENCY COMMITTEE

Shortly after the Elkton accident, the need to carry out an organized effort on this subject was recognized and an effort was made to bring together all available expertise. The FAA formed an interagency technical committee composed of representatives from the Civil Aeronautics Board, Federal Aviation Administration, United States Weather Bureau, United States Navy, United States Air Force, and National Aeronautics and Space Administration. This committee was charged with the review of all technical matters related to the investigation of the accident and the FAA program of studies, investigations, experiments, and evaluation of ways and means for the protection of aircraft fuel systems. The committee provided guidance, coordination, and advice concerning the programs. Committee members individually served as liaison channels to their respective agencies. This proved to be an excellent arrangement because it made possible frequent re-examination of the test work and on-the-spot recommendations concerning interpretations of results.

During the two-year period that the committee was in existence, it became clear that a continuing research and development program should be pursued; therefore, listings of areas of work were accumulated. One of the concluding products was a proposed research and development program on lightning. Priorities were assigned to each work area, based upon what was known in each area and how important each item was judged to be to safety goals. The proposed program is outline in Appendix I.

SHORT RANGE R&D

Many hypothetically possible explanations of how lightning may have ignited fuel vapors in the Elkton case were considered. The most feasible of the explanations could be placed in one of two categories.

Category I — Fuel vapors ignited by sparking inside the tank, induced by the lightning strike currents:

(a) A spark may have occurred at the fuel quantity gauge capacitance probe inside the tank.

(b) A spark may have occurred at the joints of the tank access covers.

(c) Sparking may have occurred at the over-wing filler cap.

(d) Internal sparking may have occurred at an undetermined point.

Category II — Fuel vapors ignited in the venting system, with subsequent propagation into the fuel tank, causing an explosion:

(a) By lightning strike at or near the vent outlet.

(b) By streamering formations at the outlet.

With these possible explanations in mind, the FAA initiated two separate high-priority research and development efforts, involving contracts for s[illegible]es, analysis, and test work. Both were designed to reveal information quickly by making use of existing test facilities and test specimens.

A contract with the Atlantic Research Corporation of Alexandria, Virginia covered the study and investigation of the propagation of flame through venting passages to the fuel tanks, the evaluation of flame arresters and explosion suppressors, as well as an evaluation of other ways to accomplish the same purpose. Among other things, this program produced information on feasible methods of testing fuel tank venting systems for flame propagation characteristics and for evaluating flame arrester designs for effectiveness. Flame propagation tests using simulated lightning discharges directly to the vent outlet showed that flame speeds higher than the usual turbulent speeds are achieved due to the large mass flows caused by an organ pipe action of the vent system. Flame arresters near the vent outlet were compromised while those further inboard were effective (Reference 2).

A contract with Lightning and Transients Research Institute covered determination of the effects of lightning strikes on a typical jet transport aircraft wing tip and fuel tank sections. A wing tip section was instrumented to detect internal arcing and sparking when lightning discharges were applied to the tip and at other points on the wing panel. The program included evaluation of ways to protect against the occurrence of internal arcing, the effectiveness of diverters, and a proof test of the most effective flame arrester developed by the Atlantic Research Corporation in their portion of the program. This work (Reference 3) produced information that eliminated certain mechanisms of ignition advanced as possible explanations for the Elkton accident from consideration. For example, measurements of the induced voltages in the fuel quantity gage wiring were made to learn whether or not the ignition source was at the fuel gage tank probe units; results indicated this to be very unlikely. A large number of tests were conducted in which the complete wing panel fuel tank was subjected to simulated lightning strikes; no signs of internal sparking were found which would indicate this as a possible explanation of the Elkton accident.

The test work demonstrated how filler caps and access cover plates can be designed to withstand direct lightning strokes without causing internal sparking.

ASSISTANCE BY COORDINATING RESEARCH COUNCIL

Because many questions have been raised concerning the relative safety of the different types of fuels used in turbine-engine aircraft, FAA requested a study of this subject be made by the Coordinating Research Council of New York City. The CRC appointed an ad hoc group to assemble and review all available technical information and experience having a bearing on fuel safety, to define the "state-of-the-art" with respect to the operational hazards of fuels, to identify the areas in which more knowledge is needed, and to recommend ways and means by which necessary additional research might be undertaken. Upon completion of this study, a report (Reference 4) was issued which concluded that the adoption of a single type of fuel by the entire industry would not significantly improve the overall excellent safety record of commerical aviation. The ad hoc group cited the need for additional research into the nature and effect of lightning strikes and electrostatic discharges, and pointed out the lack of information available regarding vapor-air mixtures in a fuel tank under actual flight conditions. Recommendations made by the group were taken into account in the planning of follow-on work by FAA.

AIRLINE REPORTS OF LIGHTNING STRIKE DAMAGE

One of the early actions taken by FAA after the Elkton accident was to request information from every available source concerning lightning strikes on aircraft. A large amount of information was collected. However, making direct use of the collected information was difficult since it was given in a variety of forms. As a consequence, it has not been possible to construct a summary of this information, although a review of the individual reports indicated a correlation with current basic knowledge of aircraft lightning strike experience from the standpoint of areas struck, severity of the strike, and the flight condition under which the strikes occur.

Data concerning strike experience was reviewed by the Interagency Technical Committee on lightning protection for transport aircraft fuel systems and was taken into account in their recommendations concerning the FAA lightning program, and in recommending the direction of follow-on research and development efforts. They agreed, however, that there was continuing need for accumulating data on experience, especially information showing the magnitude of energies involved and the degree of damage resulting. Also, statistics and trends indicating the types of turbine-powered aircraft currently predominating in airline operation were lacking.

The Air Transport Association was contacted and a large number of member operators agreed to report lightning strike damage on a standardized form. The FAA then tabulated the results and prepared periodic consolidated reports. This reporting arrangement continued for a two-year period, from January 1965 through December 1966. The data collected produced a reasonably accurate account of lightning strike damage patterns during that period. The results of the survey were summarized in bar chart form and published as Appendix II of Reference 5.

Reports in the survey showed lightning strike was experienced on 17 different aircraft models, including 8 turbojet, 3 turboprop, and 6 piston-engined types. A total of 945 strikes producing damage were recorded. The distribution of the damage over the surfaces of the aircraft was as follows:

| DAMAGE POINT | PERCENT OF TOTAL |
|---|---|
| Wing | 17% |
| Fuselage | 20% |
| Vertical Stabilizer | 11% |
| Aileron | 5% |
| Rudder | 9% |
| Elevator | 9% |
| Radome | 14% |

In general, the results of the survey served to confirm that the expected strike patterns were actually being experienced, and produced reasonably accurate figures on the frequency of strike damage occurrence in airline service.

FOLLOW-ON RESEARCH & DEVELOPMENT

After the "short range" research and development program was completed, the FAA contracted for a number of additional studies, which have contributed to the overall knowledge of lightning stroke characteristics, provided additional information on fuel vent design, and helped to define areas for future efforts.

A cooperative program by FAA, USAF, Sandia Laboratory, and Lightning and Transients Research Institute included in-flight measurements of lightning strikes to aircraft and experiments with the triggering of natural lightning discharges to the LTRI research vessel. The results of these studies and experiments are published in References 6, 7, and 8.

Studies of the effects of airflow over fuel vent outlets were carried out by LTRI at it's Miami facility, which produced information on the probability of fuel vapors igniting at the vent outlet under flight conditions. The final report (Reference 9) was published in July 1967.

An investigation of the fuel vapor conditions that exist within aircraft fuel tanks was accomplished by the Naval Air Propulsion Center at Philadelphia. The final report of this investigation (Reference 10) describes, among other things, the flammability characteristics of the fuel tank vapor space under environmental conditions representative of in-flight operation.

DEVELOPMENT OF IMPROVED STANDARDS

Assisted by the Interagency Technical Committee, the FAA proceded to convert the available body of knowledge into safety standards and criteria. The initial step was to prepare guidance material which outlined the key facts relative to lightning protection, present a classification scheme which could be used to determine type and degrees of protection required, present methods of evaluating design measures, and outline testing methods. The guidelines were published in Reference 11.

The guidelines in the Advisory Circular (Reference 11) provided the basis for the development of rule changes by amendment to Federal Aviation Regulations, Part 25, adopted September 10, 1967. The amendment (Reference 12) states the specific conditions under which the fuel system must be protected against ignition of fuel vapors. It requires that all aircraft certificated under FAR, Part 25, must have a fuel system designed and arranged to prevent the ignition of fuel vapors within the system by (a) direct lightning strikes to areas having high probabilities of stroke attachments, (b) swept lightning strokes to areas where swept strokes are highly probable, and (c) corona and streamering at fuel vent outlets.

The adoption of amendments to FAR, Part 25, coincided with FAA's issuing an airworthiness directive (Reference 13) which required that Boeing 707/720 aircraft fuel venting systems be given improved protection against lightning hazards. This directive implemented the requirements of the amendment to FAR, Part 25, and the gudielines of the Advisory Circular.

The approach used in both the guidelines and the standards is to protect the aircraft fuel system against the kinds of lightning strikes experience showed to be probable. The most probable types are those that occur at the extremities of the aircraft and those that sweep aft from projections. In addition, streamering can produce ignition with certain types of vent outlet configurations. The protection is then designed based upon zone boundaries with respect to the fuel system: Zone 1, the area where direct contact with a stroke is expected; Zone 2 where a stroke may be swept rearward; Zone 3 all other areas.

For areas identified as Zone 1 or 2, certain protective features have been found to be acceptable. For example, if a fuel tank extends in to Zone 1, it should be protected against skin penetration and internal sparking at semi-insulated parts.

Semi-insulated parts, such as those designed for ready removal for servicing or accessibility, can produce internal sparking when subjected to a direct lightning strike. Two basic solutions to prevent internal sparking have been used: (1) provide a continuous electrical contact around the entire part so that no sparks develop, or (2) design the part so that any sparking would take place on the outside of the fuel tank rather than the inside. Experience has shown the difficulties in making a reliable evaluation of such parts without resorting to specimen testing. The advisory circular describes an acceptable method of testing, involving the application of electrical discharge while the part is mounted on a light-tight box and viewed photographically.

FUTURE ACTIVITIES

A close watch is being maintained on lightning strike experience. Service records so far have clearly indicated that the protective features now used are effective. However, the FAA is striving for an improved level of protection because:

(a) The strike points on an aircraft, while generally following a pattern, sometimes include random strike points.

(b) The natural occurrence of lightning can be expected to undergo extreme variability.

(c) With air traffic increasing, the frequency of aircraft exposure to lightning is likely to increase.

(d) The design and testing of protective features involves an element of uncertainty.

These are reasons for continued concern about lightning strike protection of aircraft fuel systems. An appropriate safety goal is the attainment of complete immunity from fire hazards in the fuel storage vessel.

The FAA has recently formed an advisory committee to assist in seeking improvements in fuel system fire safety. It will act as a coordinating medium with all interested segments of industry, collect information on fuel system fire hazards and methods of protection, determine the relative merits of systems, and provide recommendations to the FAA regarding airworthiness requirements. There is every expectation that new systems on the horizon for commercial application may offer complete protection regardless of the type and nature of the ignition source. It is the Agency's hope that the efforts of the advisory committee will facilitate the ultimate adoption of standards and requirements which achieve this goal.

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# APPENDIX I

## PROPOSED RESEARCH AND DEVELOPMENT PROGRAM ON LIGHTNING

Prepared by the Inter-Agency Lightning Technical Committee on Lightning Protection for Aircraft Fuel Systems.

September 21, 1964

PRIORITY I.

1. Flight Investigation of Characteristics and Hazards of Natural Lightning.

A. Characteristics of natural lightning.

(1) Continuation or reactivation of "Rough Rider" or similarly oriented program.

(a) Measurement of characteristics of natural lightning strokes.

(b) Determination of characteristics of swept strokes.

(c) Measurement of shock-wave pressures near lightning strokes.

B. Development of a magnetic-link or other instrument for recording maximum exposures of in-service aircraft to atmospheric electricity.

C. Detection of lightning discharges. Investigate instrumentation to warn of possibility of imminent lightning strike.

D. Problems associated with fuel tanks.

(1) Development of techniques for determining conditions within fuel tanks.

(2) Determination of transient physical conditions within fuel tanks and composition of fuel and vapor during all flight conditions.

2. Vent System Studies.

A. The effect of vent location and configuration upon its susceptibility to hazards of ignition by lightning.

B. The effect of streamering upon various configurations and locations of vent systems.

PRIORITY II.

1. Aircraft electrical continuity.

A. Development standards for evaluation of electrical continuity of joints, bonds and other areas involving continuity.

B. Develop equipment for measuring parameters associated with electrical continuity so that potentially hazardous conditions may be detected.

2. Electrostatics inside aircraft, including fuel tank.

A. Charge generation and transients in flight.

(1) Development of an experimental technique for measuring the charge and charge distribution in the aircraft and fuel tanks under conditions expected in flight.

(2) Determination of spark parameters that lead to fuel ignition, including the effects of tank wall surface coatings and surface conditions.

(3) Determination of the effects of lightning upon the state of charge and spark parameters.

PRIORITY III.

1. Electrical transients throughout aircraft when struck.

A. Determination of effect of aircraft geometry on induced transients and electrical system.

(1) Evaluation of inductive effects.

(2) Evaluation of service deterioration and disruptive damage.

(3) Correlation of transients with measured electrical characteristics of airframe.

2. Determination of optimum vent configurations based upon studies of flame propagation characteristics and various configurations.

A. Effects of configuration upon efflux composition and distribution.

3. Investigation of other possible methods of vent protection.

4. Penetration and heating of tank surfaces by lightning.

PRIORITY IV.

1. Investigation of various inerting systems.

2. Investigations of other possible means of overall fuel system protection.

PRIORITY V.

1. Development of design criteria to minimize hazards on account of electrostatic charges inside the aircraft.

2. Investigation of the capability of static discharges and diverters.

3. Development of design criteria for the application of diverters to aircraft.

DEFERRED (no priority assigned)

1. Studies of plasma generation.

2. Investigation of natural lightning using triggered strikes.

3. Development of countermeasures to eliminate or neutralize the hazards of lightning.

4. Development of advance techniques for the simulation of lightning or its effects.

5. Extensions of the Faraday cage to plastic sections.

6. Development of specific flame arrester criteria, including investigation of the effects of geometry, configuration, location, and materials; identification of factors bearing on the margin of safety; and studies of flame-holding properties of flame arresters.

7. Investigation of the mechanism of spark ignition to fuel surfaces and evaluation of preventive measures.

AFAL-TR-68-290
PART II

# GENERAL LIGHTNING PROTECTION REVIEW SUMMARY

M. M. Newman

Lightning Research Oceanic Laboratory

in cooperation with

Lightning & Transients Research Institute

Minneapolis, Minnesota

## 1. BACKGROUND

Since its creation in 1946, the Lightning and Transients Research Institute (LTRI) has specialized in researches involving reduction of lightning hazards to aircraft and related radio interference. New problems have continued to be encountered due to changing techniques in aircraft design and the rapid development of space vehicle technology.

Considerable progress has been accomplished. Practically all U.S. jet transports have incorporated antenna and radome system lightning protection equipment developed at LTRI, and there has been direct evidence that the protective measures developed for antenna systems and radomes of fast jet aircraft, for example, have made the difference between destruction and survival of aircraft struck by lightning.

The most hazardous aircraft lightning problem is probably electromagnetic penetration and internal sparking in aircraft fuel systems. Solution of the fuel hazard problem is admittedly difficult because such obvious control techniques as the use of flame arresters at fuel vents can introduce other potential hazards, such as icing of flame arresters, which may be more serious than the lightning. A major spurt in research has occurred after every accident in which lightning appeared to have been a possible cause of the accident. Although the spurts of activity undoubtedly increased the pace of lightning protection research and development temporarily, there remains a need for steady progress: the LTRI-Industry cooperative program has provided continuous progress in solving the serious problems in the fuel hazards area.

Good design information and uniform standards of lightning protection should be carefully considered in the early design stages of aerospace vehicle systems, when decisions can be made to greatly reduce lightning vulnerability. Unfortunately, lightning protection design techniques have not proved amenable to handbook specification; a thorough understanding of the principles involved is necessary, along with full-scale testing at some point in the program.

The application of presently known techniques can greatly reduce possible hazards arising from new designs or design techniques. Generally, the most difficult problem for aircraft design groups is to recognize possible hazards in new design configurations. New designs should be examined, for example, for plastic sections into which discharges could penetrate or areas that could channel discharges into the interior or produce internal sparking. Demonstrations of possible sparking under different conditions have illustrated the value of laboratory tests with a comprehensive lightning environment test facility.

With high-voltage — high-current facilities combined with wind tunnels such as have been available on the LTRI Research Vessel Thunderbolt, a natural lightning environment can be reproduced under controlled laboratory conditions. Some typical flight data from aircraft struck by natural lightning, the lightning environment, and its laboratory reproduction are summarized briefly in the following section.

## 2. LIGHTNING PROTECTION "NOTEBOOK"

Fundamentally, lightning protection principles are simple; there are so many variations in the environmental characteristics of the lightning discharge, as well as in aircraft structures and associated electronic systems, however, that the design of lightning protective systems becomes relatively complex.

Naturally, insofar as possible, protective requirements should be defined in specifications, but it is difficult to foresee and define all possibilities in advance. We have found, in working on lightning researches with industry, that protective design can be facilitated by presenting illustrative examples in "notebook" form so that state-of-the-art advancements can be added continually. Eventually, it is hoped, the evolving "notebook" will be consolidated and form part of an invited text book on "Lightning Protection for Aircraft." Thus, for the past several years, we have had annual LTRI-Industry cooperative program seminars, following the "notebook" topic outline below as an indication of the scope of the lightning protection problems:

1. Thunderstorm electromagnetic environment.

2. Lightning discharge measurements and laboratory reproduction.

3. Experience with lightning stroke effects on aircraft, and laboratory comparisons.

4. Aircraft antenna system protection techniques.

5. Plastic discontinuities in the aircraft metallic shell and protective techniques.

6. Electromagnetic structural stressing, and possible protective diverter techniques for controlling current discharge paths.

7. Possible discharge streamer fuel ignition sources at fuel vents.

8. Possible effects of lightning pressure waves on flame propagation in ducts and plasma penetration at fuel vents.

9. Structural electrical discontinuities at fuel tank filler caps, inspection doors, and fasteners; possible internal sparking hazards and protective techniques.

10. Discharge burning through fuel tank metal skins, and "protective armor" techniques of aluminum foil-fiber glass sandwiches.

11. Continuing studies on natural lightning channel discharge characteristics, with shipborne rocket-triggered lightning discharges to ground, measurements of pressure wave, possible plasma oscillations, and possible metastable "ball lightning" type energy states that might penetrate fuel vents.

12. Fuel tank internal electrification hazards, from dielectric inner surfaces and possible interrelation with lightning-induced transient discharges.

13. Questionnaire analyses including additional accumulation of data on jet aircraft skin pitting.

14. Experimental field checks on full-size aircraft of applied protective techniques.

15. Application of protective techniques to helicopters, light aircraft, and airships.

16. Special problems with new type Mach II and III aircraft, including electromagnetic surge penetration into more exposed and more vulnerable complex electronic systems, and vulnerability of new materials such as composite structures (boron and graphite epoxies) and titanium.

17. Special problems of aircraft vulnerability on the ground, and related problems concerning rocket launching systems.

18. Concluding summary of general protective techniques and further researches.

The 1968 seminar, to review recent advances in the above topics, will follow after the main conference on Friday, December 6. Any of the Conference attendees who wish to may participate in the LTRI-Industry Cooperative research program.

## 3. PROTECTIVE TECHNIQUES AND STATE-OF-THE-ART

Lightning protective techniques are difficult to express in terms of simple specifications because of complex interaction of many factors, including the many distinct effects of various types of strokes, components of different aircraft materials and construction, complicated by the wide statistical variation in the stroke magnitudes and types. Studies of the various effects have been the subject of about 300 LTRI reports, and most of the protective development is being summarized in "notebook" format as a design guide that can be kept in step with state-of-the-art advances. A few representative publications on LTRI's lightning protection studies are referenced at the end of this paper.

It has been generally found that practically every aerospace vehicle prototype needs individual evaluation. Full-scale checking under a laboratory-reproduced lightning environment is also highly desirable, as is brought out in the following state-of-the-art summary.

a. Protection of the aircraft HF antenna system from directly intercepted discharges is essentially solved with the lightning arrester designs already evolved. Various arresters are now available commercially, but complete system checks are essential for individual aircraft designs. Most important are tests of the arrester connections and any shunt elements, such as VHF isolation transformers which are in parallel with the arrester. Shunt fed antennas which are essentially grounded eliminate the requirement for a large lightning arrester but do not prevent small inductive transients from entering the communications equipment. Thus, small arresters or surge suppressors may still be required to protect the radio equipment.

b. Extension of protective techniques to special UHF and VHF mast antennas, particularly through the use of grounded antennas, is now feasible. Now required are guide lines and standardized specifications.

c. Radome protection has been evolved with diverter strip systems of various kinds, and production designs which had been laboratory checked have worked out satisfactorily under natural lightning conditions. Care must be taken with foil strip systems so that the strips are not confined to produce internal pressures to damage the radome. Foil strips can be cracked by differential expansion with thick paint layers to produce severe precipitation static radio interference. While foil is cheap and simple to apply in comparison with the permanent but heavier aluminum bars, it requires frequent maintenance checks for continuity and cracks. Heavier aluminum strip designs have withstood occasional 200,000 ampere discharges.

d. Further development work is needed on new special radome designs where dimensions, materials, and internal gradients from new antennas, etc., may introduce new problems.

e. Individual laboratory lightning tests are essential to check new design prototypes. There have been many examples where apparently unimportant production modifications have proved unsuitable. With so many interrelated factors, an artificial lightning check of the final production design as an integrated part of the overall system is considered important and is certainly safer and more economical than waiting for flight damage experience.

f. The fuel system hazard from lightning discharge currents is an area in which the state-of-the-art is still uncertain. Further researches are in progress under government and industry support to determine the degree of the hazards. The industry cooperative program is developing interim protective approaches to reduce potential hazards, even though their extent is statistically uncertain.

As detailed in referenced reports, facilities for reproduction some of the more severe lightning discharge effects have been developed by LTRI under this program. Consequently, it was possible to effectively carry out some related research under FAA sponsorship on fuel system hazards, using a full-size jet aircraft wing tank for the experiments. Information in deficient state-of-the-art areas is summarized as follows:

(1) Sparking of fuel tank components was demonstrated, in the case of direct lightning currents to (a) filler caps of all types, (b) access doors of some types, and (c) semi-insulated bolt fasteners. Earlier bonding techniques at filler caps and access doors have been shown to be ineffectual in reducing sparking, but adequate protective techniques, specifically complete peripheral metal-to-metal bonding, have been shown to be feasible and effective.

(2) Streamering directly at shielded fuel vents has not been obtained at voltages up to 10 million, whereas much lower voltages had been sufficient to produce streamering on other types of fuel vents. It is probable that recessed vent edges in a flight environment would also be free of any cross-field streamering. However, since the potential of an aircraft contacted in flight by a lightning discharge could be raised to 30 to 100 million volts instead of the 10 million volts used in the test, it would appear desirable to extend the tests to reproduce

the higher transient potential and study more fully the effects of approaching lighining discharges to the fuel vent area, including both effluxing fuel vapor mixtures and typical air flow.

(3) Induced effects on fuel gage wiring did not prove to be excessive on the wing in earlier tests (a factor of about 10 below hazard level) but definitely could be hazardous with other wiring and shielding configurations.

(4) Further researches are necessary beyond the earlier FAA-sponsored short-range study on induced effects and fuel vent streamering, but it can be concluded that the direct stroke effects are the most important. If the fuel tank system were made invulnerable to all possible swept or direct strokes, protection would also automatically be attained for induced effects as well.

(5) Promising techniques for preventing internal arcing from direct strokes have been suggested for each of the above hazard areas and have been checked and demonstrated as feasible, though considerable development and further tests would be necessary in some cases to be sure other hazards are not introduced. Improved filler cap designs were mocked up and demonstrated to eliminate internal sparking. Similarly, improved access plates and fastenings have been demonstrated, but experience to date indicates that considerable development is required in any new design.

(6) Other hazard areas were also considered. Semi-insulated rivets and gasketed bolts can pose a hazard from direct strikes; however, this also was shown to be greatly reduced or eliminated when they are covered by suitable sealants. Proper thickness control of the sealant greatly reduced hazards in this connection, though further researches toward better fastenings should be continued. Sparking was possible in areas adjacent to fuel tanks that may contain trapped fuel vapors.

(7) Potential vent hazards from direct strokes swept into the vent opening were conditionally confirmed. Separating the effects of contributing factors including corona streamering, plasma heating, contact spark showering, possible pressure wave detonation, and the related high velocity flame propagation effects require further study in relation to ignition with and without flame arresters.

(8) Flame arrester variations checked at LTRI in conjunction with Atlantic Research Corporation (presented in their reports) have indicated considerable promise — providing, of course, that they do not introduce other hazards, such as icing. Some yet unknown factors of possible high pressure waves and intense ionization effects from nearby direct strokes require further basic study of the nearby zone of natural lightning discharges.

(9) Until flame arrester or flame quenching developments are successful, it may be worthwhile to apply LTRI-developed diverter techniques at the wing tip to keep lightning discharges as far from potentially vulnerable wing areas as possible. Experiments to check quantitatively possible gains indicated that a single trailing diverter would probably provide a factor of 3 improvement, and if it could be flown at 45° angle outward, perhaps a factor of 10 may be achieved. (This is a possibility that would need further experimental developments and verification, particularly as problems of mounting and optimum locations would need to be carefully worked out to be sure other hazards were not introduced.)

(10) In experiments with explosive mixtures, fuel vapors were ignited by internal spark showers off the fuel filler caps and access doors before "fixes" were installed, but no explosions occurred after installation of the "fixes." The effectiveness of the "fixes" was demonstrated by firing to a fully fueled tank section. This test also checked the possibility of unknown ignition sources inside the fuel tank.

In summary, LTRI has demonstrated a number of very definite hazards and possible improvements under the recent short-range FAA program. We consider the effort particularly worthwhile in having effectively demonstrated that real hazards exist in relation to lightning and fuel systems, even though no direct accident evidence has been established; in some cases of low energy ignition sources, such as cross-field discharge streamers, direct pitting evidence probably never can be established.

Certainly, much more work is necessary; perhaps a two-year program at the pace of the initial FAA-sponsored four-month effort, including basic phases also, should go far toward solving the problems. Areas we consider important to extend, and which we are, in our own specialized Institute, proceeding with (at a rate of effort and progress naturally reflecting the degree of cooperative program support) are as follows:

(1) We have applied suitable air flow environment equipment to researches on fuel vent streamering, diverter operation with swept discharges, and

(2) We are continuing to determine the lateral extent of the hazard of a lightning discharge channel including spark showers, plasma extent, shock waves at the discharge channel center, and ionization effects, in answer to the question: How near a fuel vent may a lightning discharge safely strike? The Research Vessel "Thunderbolt," which has been specially equipped for triggering and measuring natural ligtning discharges, provides a unique facility for the close study of the lightning channel and should provide answers to many of these questions through continuing direct measurement of natural lightning near zone pressures, temperature distributions, and ionization extent.

Other important problem areas already under study but requiring further work include:

(3) Transient current flow through the aircraft structure.

(4) Further studies of aircraft continuity through joints, rivets, and access doors.

(5) Lightning metastable ionization as a possible ignition source.

(6) Further experimental work on vent locations and designs.

(7) Lightning swept stroke penetration of fuel tank walls.

(8) New aircraft designs, including STOL, VTOL, and SST types, with new materials and techniques such as boron composite structures, also require further study in relation to lightning hazards.

4. CONTINUING RESEARCHES

Model studies of aircraft and other aerospace vehicles are generally carried out at LTRI as a first step in determining the probability distribution of stroke location. The measurements must be carefully conducted so as not to introduce unwanted stroke direction bias, and the interpretation of the results involves evaluation of the scale factor influence relative to natural lightning conditions. Usually a large number of simulated strokes are used to bring out low probability factors, and some checks with added air flow are sometimes desirable to duplicate "swept stroke" effects.

For field penetration and lightning damage studies, however, it is usually preferable to work with full-scale sections or mockups. Models must be large enough for internal instrumentation; if an actual section is used, the questions of scale factor effects and accuracy of model representation do not arise. Thus, we conducted such measurements with an actual wing section in one case, and a mockup containing the obvious electrical discontinuities that might prove hazardous. Field penetration of a full-scale section is being studied, with high currents and high-rates-of-rise current available from the LTRI lightning simulators.

a. <u>Electromagnetic penetration into electronic subsystems and protective techniques</u>

Some present electromagnetic field penetration problems are closely related to our work started under FAA programs and early radio interference, shielding, and lightning studies for the Air Force and Navy. These problems could be further studied using improved techniques with the new LTRI facility providing increased currents and rates of rise of current. One of these problems is the accidental activation or dudding of an electronic component due to voltages induced in the unit or its associated circuitry. The technique of using an internal battery-powered oscillograph with an observer, where possible, proved effective.

One suggested method of studying magnetic-field penetration of enclosures consists of coating the inside of a conducting shell with a magnetic material and, after passing a unipolarity simulated stroke current through the enclosure, checking field magnitude on the inside surface and plotting current paths with a magnetometer. To determine the electromagnetic energy required to cause malfunction in a hypothetical electronic system, such a system can be set up and the required energy determined by using simulated circuits or, if the destructive energies together with circuit impedances are known, the radiated external energy required to induce sufficient energy into the internal circuits can be measured.

In applying the results of the studies to actual systems, it is important to conduct final full-scale tests to bring out variations in materials and techniques that may appear minor but can actually destroy the shielding effectiveness, and to find unexpected hazards. For example, in one very important application, shielding effectiveness was destroyed when a cable sheath arced to a nearby ground; the arc burned away the shield and made the system vulnerable to the next surge or stroke.

Measures proposed to protect sensitive elctronic circuitry due to the high currents and fields of a lightning discharge are usually those that improve shielding. These steps include:

(1) Check full-scale sections or mockups of representative enclosures with simulated lightning, i.e., high voltage for stroke probability tests, high current for stroke damage and field penetration, and high charge transfer for burning effect.

(2) Apply lightning protective techniques to the mockup where required. Improve conductivity about joints and other discontinuities and work out and check other measures on the section or mockup.

(3) Finally, evaluate a full-scale production sample of the system by inspecting and measuring with simulated lightning to reveal shielding deficiencies. Correct deficiencies and verify adequacy of the final system shielding.

b. Full scale field study of aircraft systems

Checks, preferably of an actual aircraft, can hardly be overemphasized. We frequently find that slight changes between test mockup and the production aircraft result in complete failure of the protection system. Many examples can be given of the problems encountered because of failure to make a final system check, but the best illustration is this: a change in the angle at which a radome protective bonding conductor was brought into an attachment point, in one case, resulted in a failure and serious destructive results, this would have shown up immediately in laboratory tests.

The need for full-scale tests has been recognized by the Royal Canadian Air Force; they have sponsored full-scale tests at LTRI of the Argus aircraft (reported at this conference by H. R. Shaver). The results of the full-scale aircraft tests again showed the practicality of system tests of a full-size aircraft, and we strongly recommend this type of investigation for all aerospace vehicles, including civil aircraft, that are susceptible to lightning.

c. Study of molecular plasma metastable states as possible ignition sources at fuel vents

All possible fuel ignition mechanisms consistent with modern physical theory should be considered objectively in relation to aircraft safety. Dr. E. L. Hill, Professor of Physics at the University of Minnesota and working part time at LTRI, conducted initial studies which indicated metastable molecular plasma as a source of ignition for aircraft fuel vapors. Specifically, the metastable ions formed in the lightning discharge channel retain nearly all of their ionization energy and can be triggered to the ground state by shock waves or turbulence. Thus, the metastable ions formed by a strike near a vent could be ingested into a dry vent where flow turbulence or subsequent shock waves could trigger them to produce visible ionization and fuel vapor ignition. Laboratory experiments should be made to verify or deny the existence of this mechanism. A motion picture sequence (Fastax recording) of LTRI-triggered natural lightning discharges illustrates some of the plasma characteristics of the discharge.

d. Study of swept natural lightning discharges over aircraft fuel tank surfaces

At present, the most serious unsolved problem is the very low but finite probability of a "swept-stroke" long duration, or restrike, discharge component burning a hole in a fuel tank or an access door or being "swept" into a fuel vent.

Recent studies at LTRI have shown that the aircraft skin surface condition and treatment is probably the key factor in determining a stroke "hang-on" and the amount of charge transfer acting at any single location except the trailing edge. Aircraft lightning damage reports confirm the conclusion that severe pitting hardly ever occurs midchord on bare aluminum surfaces; it only occurs on insulated surfaces covered with anodize or paint. These are preliminary conclusions, however; evidence is hardly sufficient on which to suggest changing a specification involving safety of flight. Thus, an accelerated supplementary program is suggested to examine the various mechanisms involved in stroke sweeping, and evaluate possible solutions to the problem, such as an exterior surface to enhance stroke movement and reduce charge transfer concentration and consequent excessive heating at a single point.

Continuation research is expected to include the evaluation of surface coatings, both those in use and those which might be used to enhance stroke movement, based on the results of the initial studies; some of these are being coordinated by the Surfaces and Finish activities to ensure their practicability.

Also to be studied are the effect of configuration or component geometry, such as around fuel filler caps and access doors, and the ability of various types of fasteners in holding the stroke at a single location.

## 5. CONCLUDING DISCUSSION

This summary was kept relatively brief so that a number of related invited papers can help expand the discussion with different points of view. In this concluding discussion, we will present a few of the 100 illustrations from the referenced LTRI Cooperative Program "Notebook" to illustrate some specific points.

### a. Swept Stroke Current Characteristics.

The problem of possible holes burning through the aircraft fuel tank surface is perhaps the most crucial in the design of new aircraft, and may even be a factor in selecting paints for fuel tanks of present aircraft. Figure 1b shows that most discharges intercepted by aircraft (flight path A) are inside or above clouds and are relatively low-current low-charge transfers of portions of a main lightning discharge. The probability of intercepting a cloud-to-ground discharge (flight path B) is much lower but may involve high currents to 200,000 amperes (as illustrated in Figure 2), high charge transfers of about 500 coulombs (Figure 3) and high rates of current rise of possibly 100,000 amperes/second (Figure 4).

Most ground lightning discharges are probably of multiple stroke type as illustrated in our triggered natural lightning record as documented in Figures 5 through 9, which was a relatively low current (40,000 ampere max) discharge about fifteen component strokes totaling about 100 coulombs. This probably corresponds to natural lightning, such as caused holes in the wing tip tank shown in Figure 10.

We early considered that "swept stroke" charge transfer as the aircraft flies through the channel (Figure 11) would be much lower at hang-on points along the flat surface than at an extremity, such as the approximately 500 coulomb charge transfer to the tail section that was bypassed by the lightning arrester gap of Figure 3. From the discharges of Figures 6 through 9, we might expect an initial component of about 30 coulombs over a 3 millisecond period as more typical of smaller discharges that would hang on at localized points in the path of the aircraft sweep through the lightning channel at a rate of .6 ft/sec at a 400 mile per hour speed. However, there is a wide range of lightning discharge characteristics; that particular triggered stroke was definitely a small one; obviously larger charge transfers are easily possible. Thus, a reasonable factor-of-four increase would bring us up to the equivalent of the 300 coulombs estimated for the burned hole (Figure 12). Figure 10 shows swept strokes along tip tanks with over one foot of distance between arc hang-on points, which is in accord with recent experimental duplication of swept strokes illustrated in Figure 13. In these tests, the discharge hung on for a relative aircraft travel of several feet. We have shown that is probably due to anodized and painted skin surfaces. Thus, there is little question that swept strokes can burn through anodized .065" thick aluminum.

Reports have been received of tip tanks burning, being exploded, and being blown off fighter aircraft. It has been suggested that a tip tank is an "extremity,"

Figure 1 (a). Laboratory illustration of charge accumulation on a highly insulated epoxy paint layer, over a propeller spinner system, discharging in multiple paths to rod B as the lightning discharge to ground path shown in Figure 1(b).

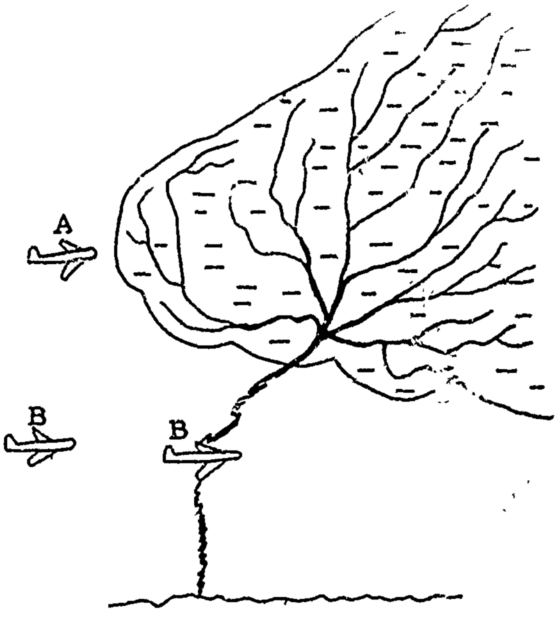



Figure 1 (b). Illustration of the relative aircraft stroke interception areas with high probability of low energy cloud charge tapping branches in the cloud going thru the aircraft, and much lower probability of a ground stroke B diverter through aircraft passing nearby.

Figure 2. Artificial lightning discharge of 200,000 amperes crest required to duplicate electromagnetic pinch effect in laboratory (upper left) produced by natural lightning discharge to aircraft rudder bow

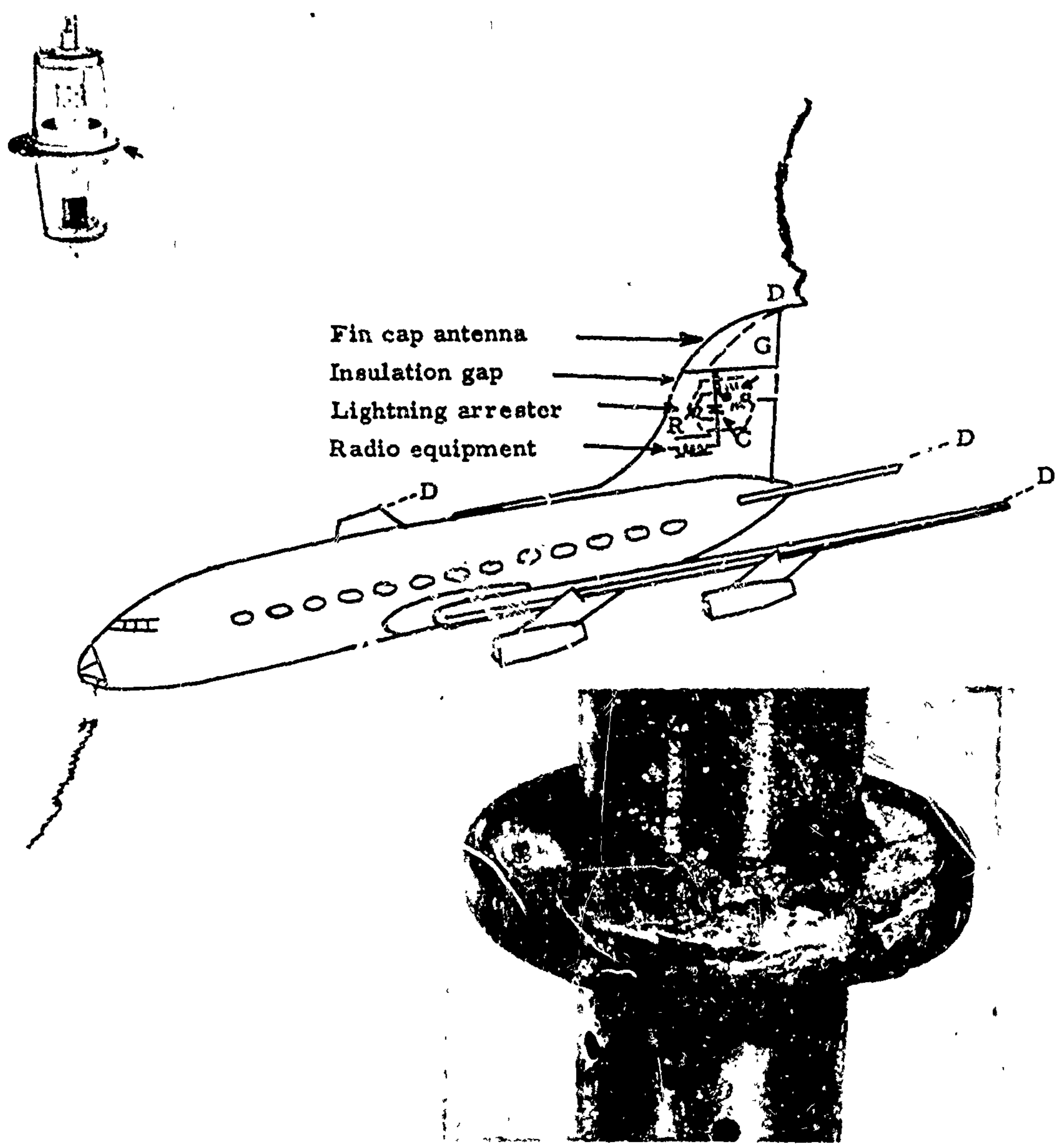



Figure 3.

Diagram illustrates arrester antenna system protection of fin cap. Upper left photo of arrester unit, and lower right shows effect on gap electrode of natural lightning discharge comparable to laboratory reproduced effect of about 500 coulombs.

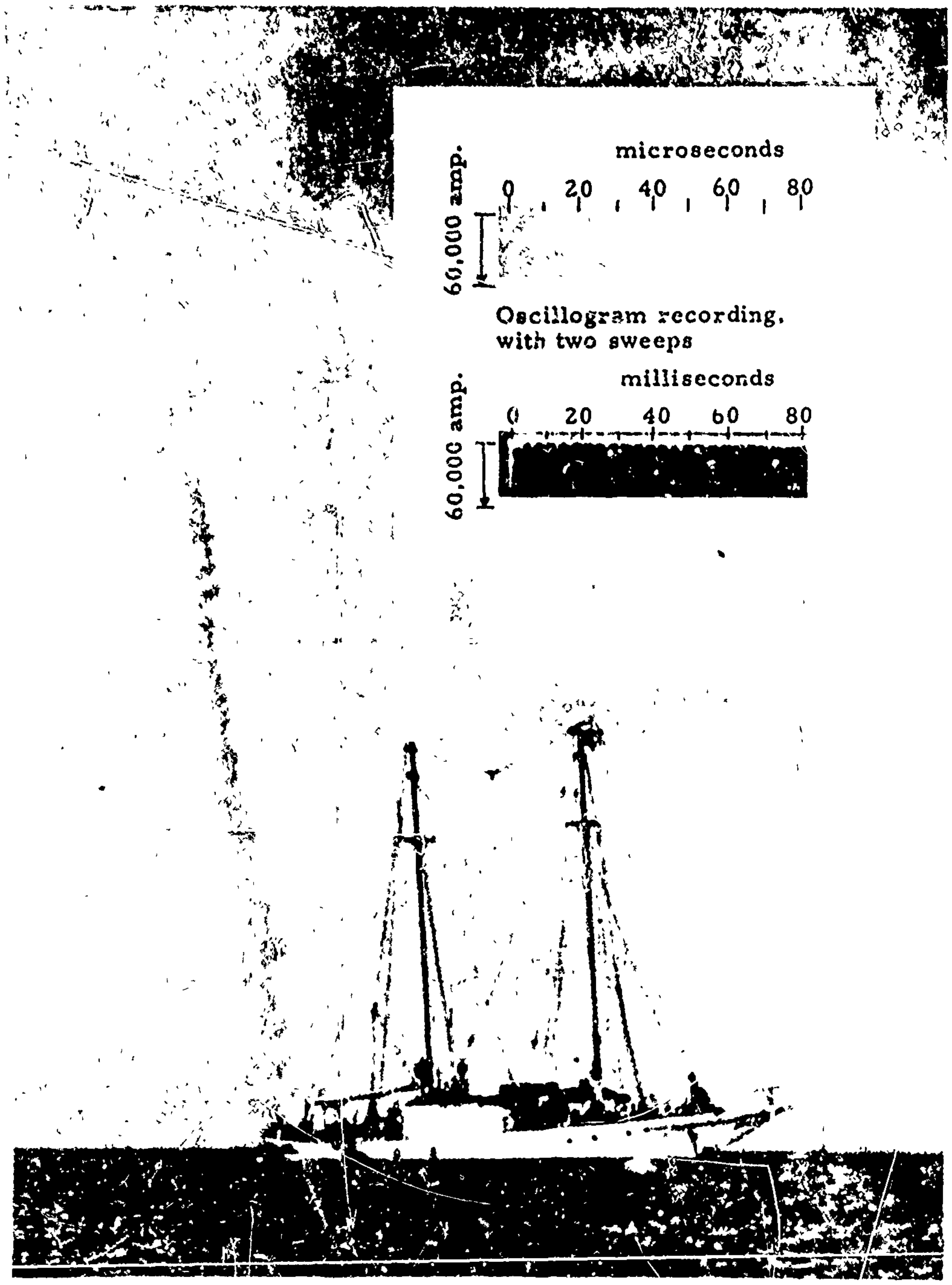



Figure 4. Photograph of rocket launching with trailing wire attached to insulated platform and cables for oscillographic recording triggered lightning. Insert above is the oscillographic record of a triggered discharge with an initial rate of current rise of over 50,000 amperes per microsecond

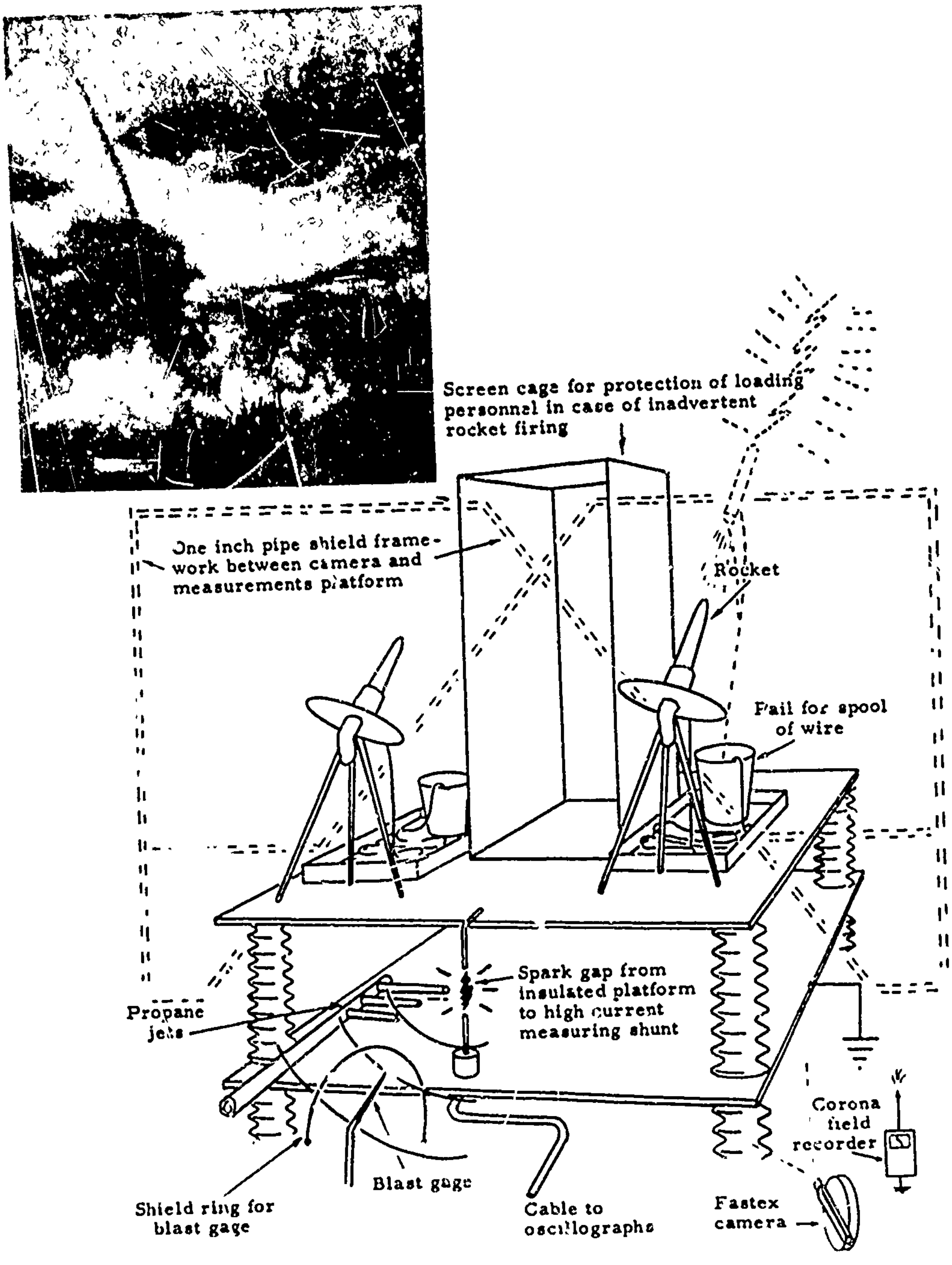



Figure 5. Schematic diagram of measurements platform on top of high voltage generator Location of measurements platform on launch ship is shown in insert photograph, circled area A.

Figure 6 Photograph of triggered lightning discharge to measurements platform. Lower arc to current measuring shunt illuminates blast gage. Photograph taken with Polaroid type 55 positive-negative film, held open 10 seconds after rocket firing, using f16 opening and 10% transmission filter to reduce light intensity.

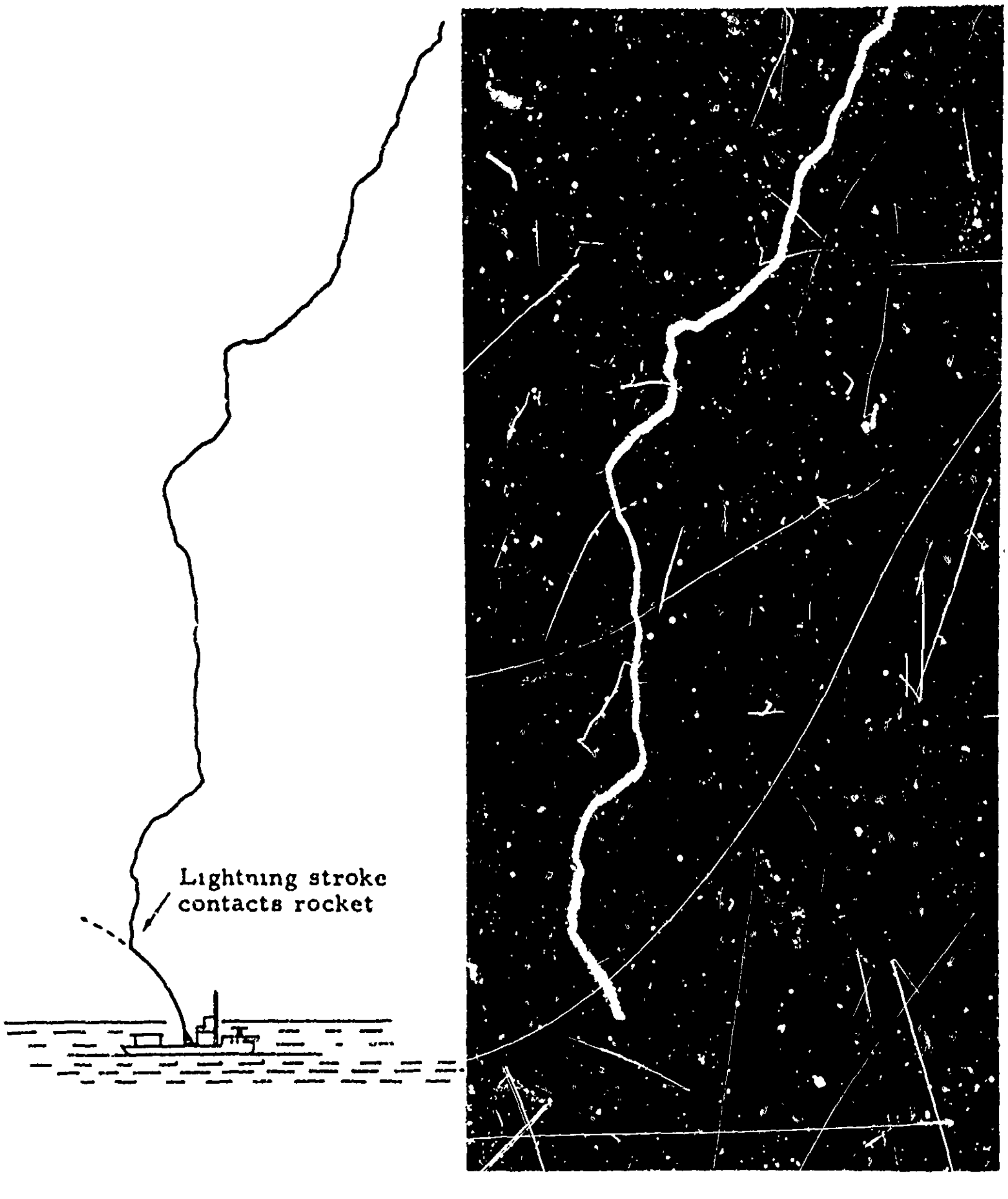



Figure 7. View of lightning discharge channel from distance of two kilometers. Rocket trail of about 100 meters length may be seen above ship in lower left corner of photo. Photograph taken by R. A. McDonald vacationing at nearby St. Petersburg Beach using a Polaroid 100 camera.

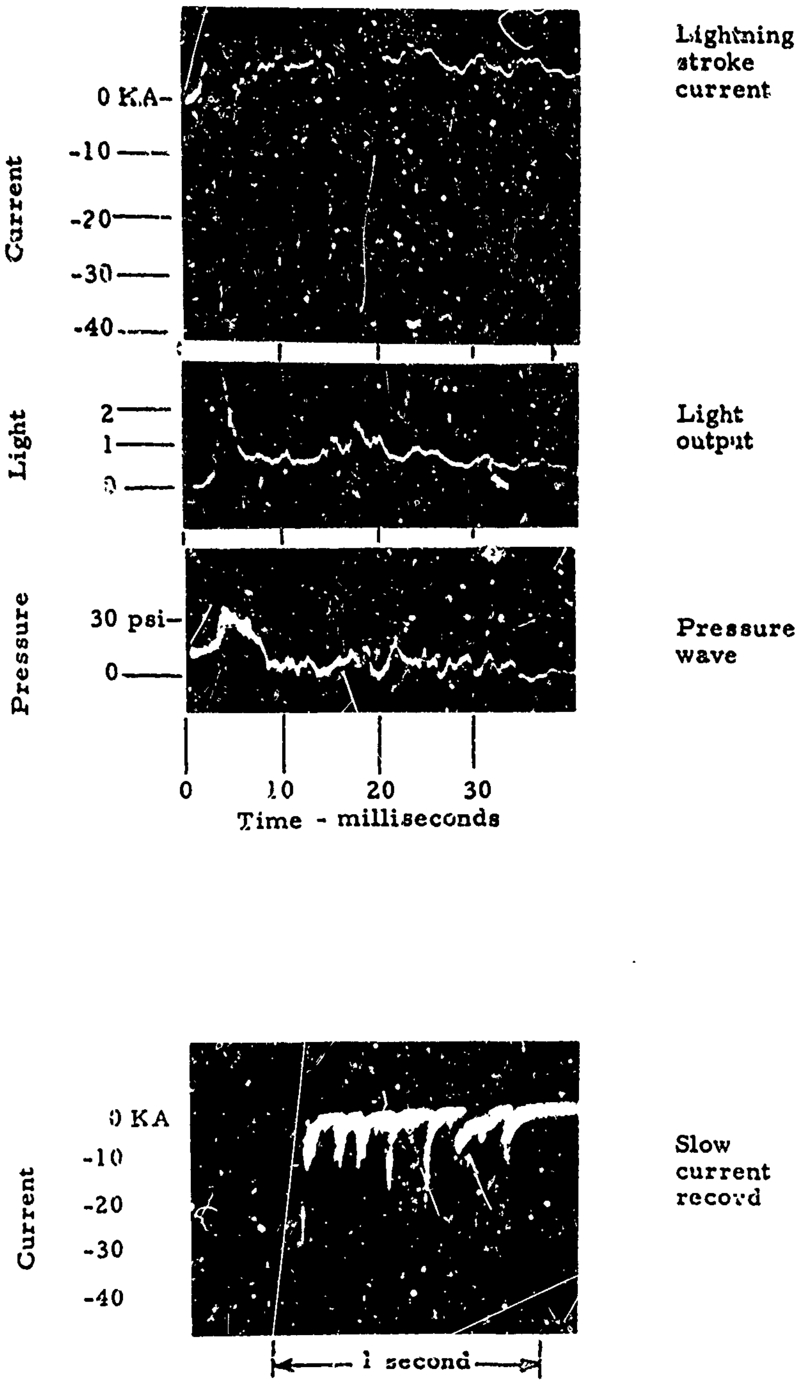



Figure 8 Lightning discharge current oscillograms showing multiple restrikes. Slow current oscillogram may be compared with photographs of multiple stroke channels, Figures 1 and 2.

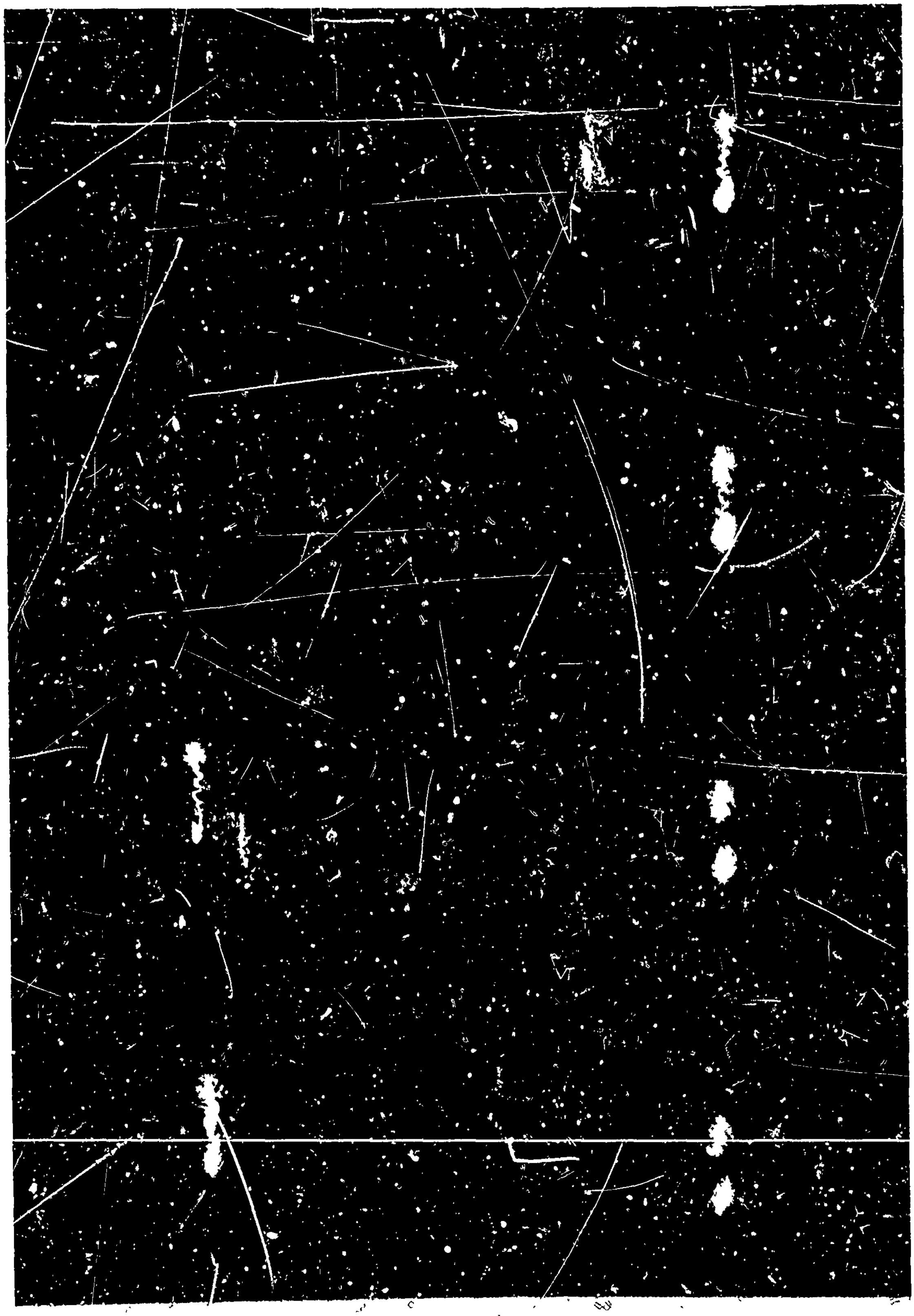


Figure 9 Lower 10 segment of lightning discharge channel to high current shunt recorded at the rate of 1000 frames per second by Fastax camera showing first eight frames

<u>Figure 10.</u> Typical swept stroke punctures of wing tip fuel tanks by movement of aircraft through the relatively stationary lightning discharge ionized channel.

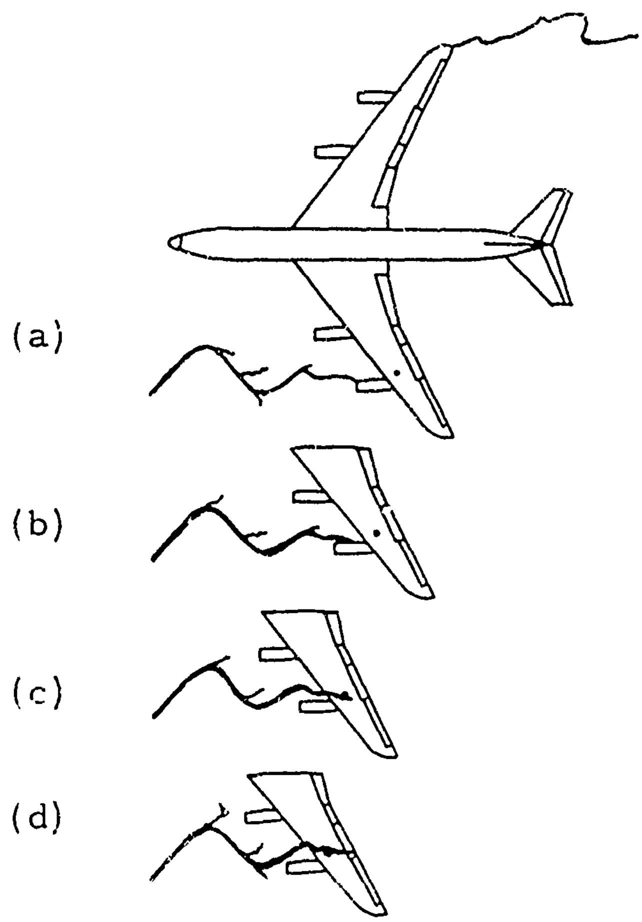
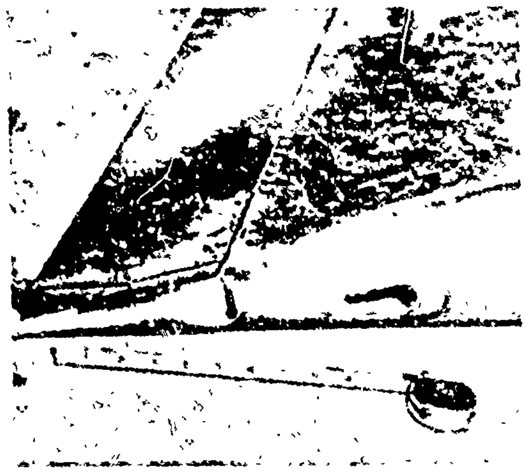



Figure 11. Transfer of lightning ischarge current contact path aircraft flies on through the htning ionized air channel.



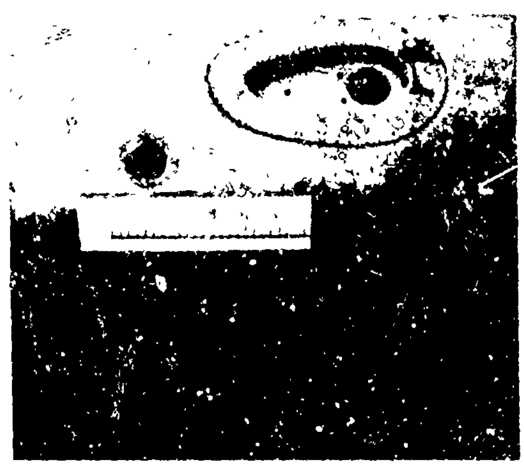


Figure 12. Natural lightning discharge to jet transport wingtip produces larger hole than 300 coulomb discharge in laboratory (lower photograph).

Figure 13 Sequence from Fastax film of swept artificial lightning discharge with approximately five milliseconds separation per photograph and an air speed of 155 mph. Discharge from electrode above contacts skin and hangs-on to forward point while arc is swept nearly the full length of the five foot panel.

and that there is no real difference between a horizontal channel sweeping along the tank and a vertical channel to some extremity on a wing swept across the wing tank surface. We are discussing the swept-stroke situation at some length because the suggested capability of the fuel tank to withstand 500 coulombs is rather difficult to meet, and should include sweeping in tests to 500 coulombs. Yet, since skin burnthrough in the flight damage reports has required over 300 coulombs to duplicate, LTRI considers the direction to take is not to reduce the overall coulomb requirement, but to accelerate work in the direction of reducing the time of "hang-on" by controlling skin surface, filler tank, and access door characteristics to withstand swept strokes of reasonable maximums, applied with operational wind stream in marginal design cases, as discussed further in the next section.

b. Standardization of Test Current Requirements

Aside from the definition of "swept strokes," there are also differences in test specifications between the various regulatory agencies. LTRI's function has been to bring out the background facts and, where they are unknown, to carry out basic researches as in the triggered lightning direct measurements of currents and pressures and by duplicating actual flight effects. By bringing the various parameters to attention of the aerospace Industry and the Government regulatory agencies, test requirements have been set up which are presently considerably different from each other. The first lightning test specification was set up at a joint meeting at LTRI specifically for the antenna system lightning arrester then under development. The test specification included three current components, as illustrated in Figure 14, to duplicate different natural lightning effects, summarized in Table I.

Different types of stroke components, of course, may be used to test different aircraft parts. However, it would probably be safer to test all major components with a natural lightning discharge such as specified in MIL A9094C with a fast rising current component added than to limit the test to one specific aircraft component, since changes in materials are so rapid today that a limited test may be adequate today but be totally inadequate tomorrow.

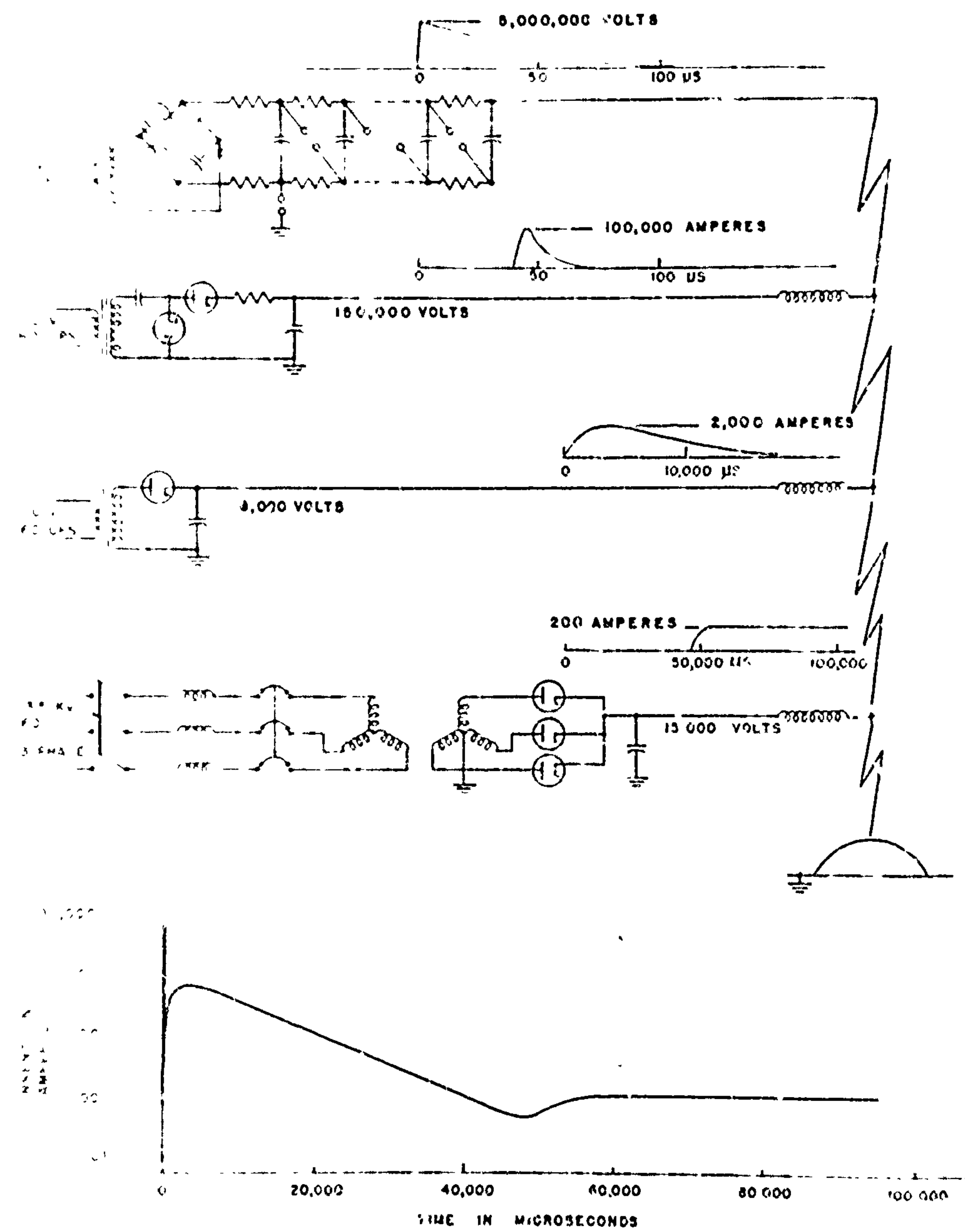



Diagram of multiple generators which produce typical effects of natural lightning strokes, individually or as a composite single discharge. The combined output current waveform is shown below.

TABLE I.

Comparison of peak current, mechanical forces, heating effect and charge transfer of the three types of surge current generators used in testing of the lightning protection systems.

| | High current generator 3.3 µf at 150 kv | Secondary stroke generator 3000 µf at 10 kv | Long duration current generator 200 amp at 12 kv |
|---|---|---|---|
| Peak Current | 100,000 amperes duration=$10^{-5}$ sec to 1/2 value critically damped | 5000 amperes duration=$10^{-2}$ sec to 1/2 value critically damped | 200 amperes duration = 1 sec rectangular wave |
| Relative Mechanical Force (proportional to $I^2$) | 100.0 | 0.25 | 0.0004 |
| Heating Effect $\int I^2 dt$ R | = 69,000 R | = 172,000 R | = 40,000 R |
| Charge Transfer Q | (C) x (E) $3.3 \times 10^{-6} \times 15 \times 10^4$ = 0.5 coulombs | (C) x (E) $3000 \times 10^{-6} \times 10{,}000$ = 30 coulombs | (I) x (t) 200 x 1 = 200 coulombs |

It was found feasible to design lightning arresters to meet the specification of 100,000 amperes decaying to 1/2 value in 10$\mu$sec, with a 2000 ampere intermediate, a 20 coulomb component, and a 200 coulomb total charge transfer. Later flight experience with arresters and radome bonding development have indicated occasional stroke currents of 200,000 amperes and overall charge transfers of 500 coulombs. Accordingly, we recommended that where flight safety is concerned, the specification be doubled, to require 200,000 amperes. The complete test discharge would consist of a current rise to 200,000 with an initial rate of rise of 100,000 amp/ sec decaying to 1/2 value in 50$\mu$sec, with a superposed long-duration 500 coulomb component. While we have calculated such currents to be technically feasible, the expense would be great. Therefore, we suggest a cut-off first 1/2 cycle of a 200,000 amp crest 10 KC partially damped oscillatory discharge wave with a superposed initial pulse of 100,000 ampere/$\mu$sec, followed by a typical long-duration multiple component totaling 500 coulombs. Because of the swept stroke problem, it may be desirable to reinstate the intermediate component but at a sufficiently higher current to transfer about 50 coulombs in the first 10 milliseconds (totaling 500 coulombs overall) and carry out marginal fuel tank designs using swept discharges as illustrated in Figure 15.

Finally, we would comment that the standards should have sufficient leeway, with reasonable plus or minus limits, so other laboratories could use their equipment rather than duplicate LTRI tests exactly. LTRI policy has been to assist research-cooperating companies to set up their own test equipment, as most of the test work can be done with much smaller equipment, to weed out weak designs and reserve LTRI's extensive and high energy (Figure 16) facilities for final checks. Eventually, as time permits, it is LTRI policy (as a non-profit institute for general advancement of the state-of-the-art) to make final definitive recommendations concerning principles of safety and to make them available through publications and meetings such as this Air Force-Navy co-sponsored conference. LTRI has so broad a foundation of safety techniques in this field that the various cooperating research groups, though highly competitive in aircraft and component design, should consider sharing these safety principles freely, in the spirit of this conference.

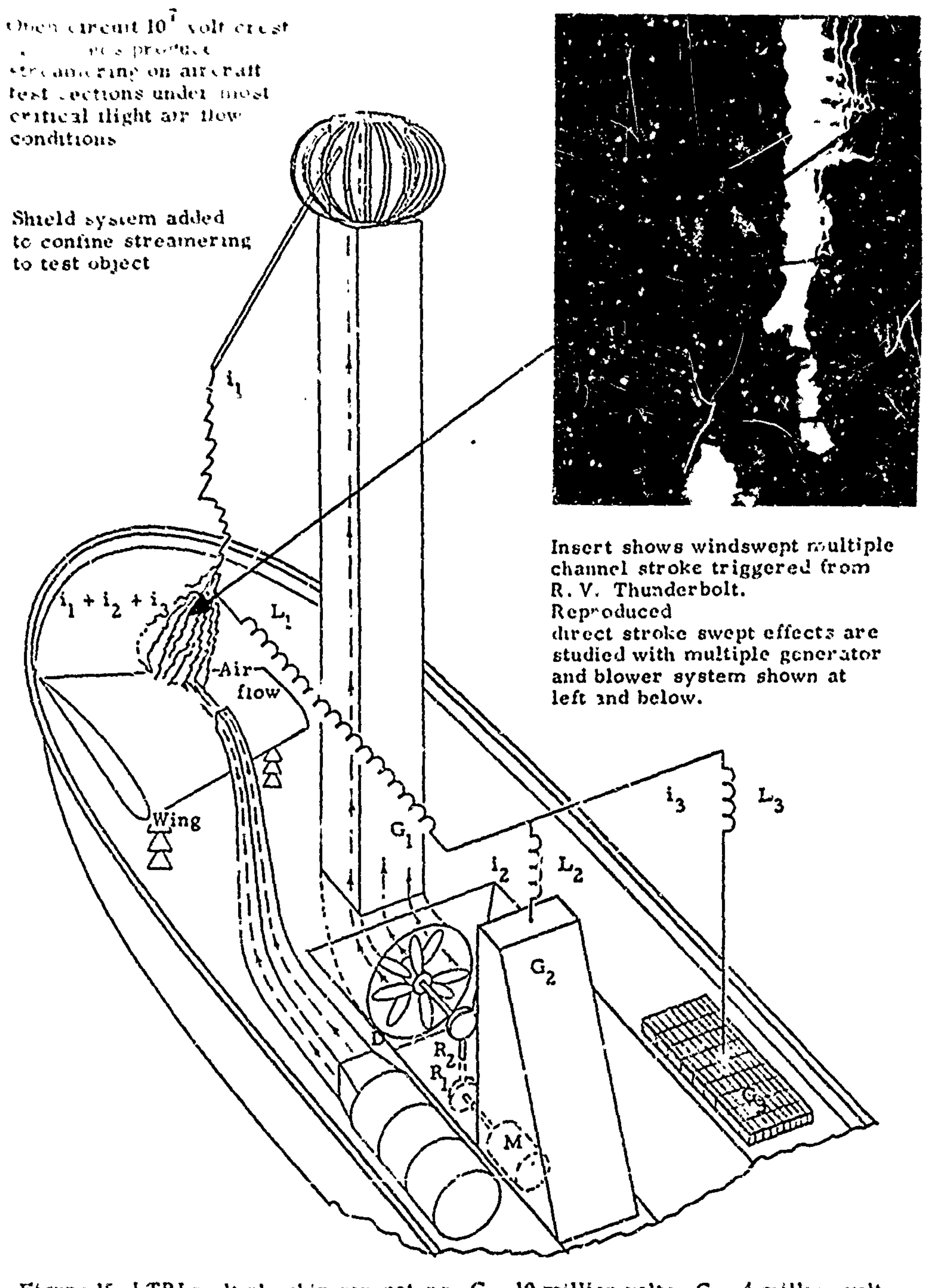



Figure 15. LTRI multiple ship generators, $G_1$, 10 million volts, $G_2$, 4 million volts, $G_3$, 300,000 or, long duration, 1000 amps, coordinated with wind tunnels, to reproduce natural lightning multiple discharges as shown in insert photo

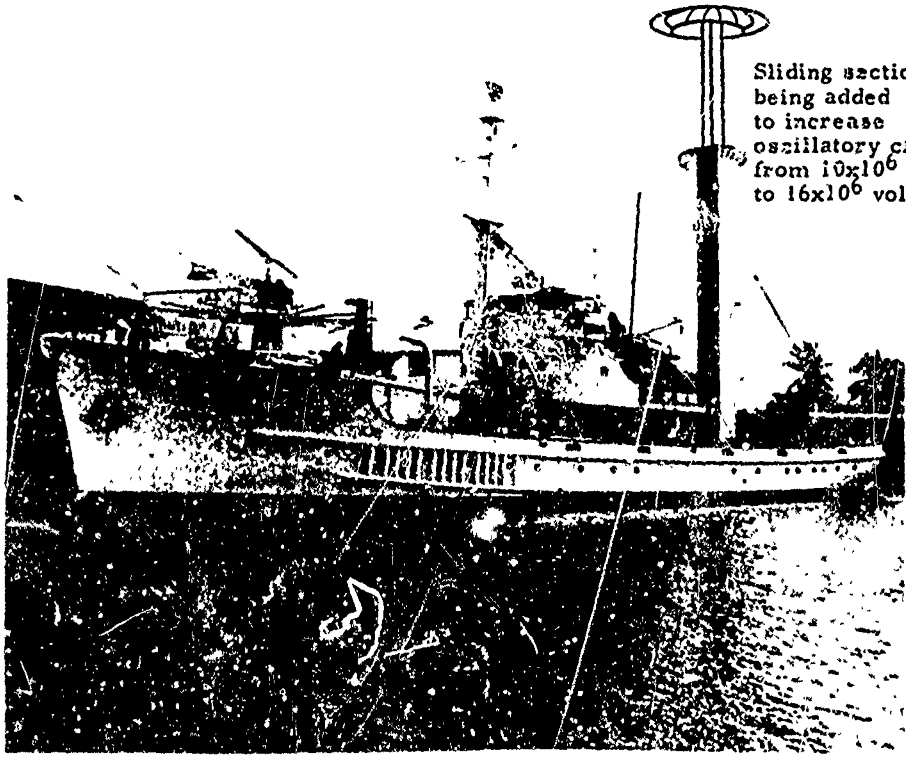



**Figure 16.** LTRI mobile 10 million volt surge crest facility showing increased voltage modifications in progress, to obtain a crest voltage of 16 million volts

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# CANADIAN AIRCRAFT LIGHTNING PROTECTION RESEARCH

H. R. Shaver

Industrial Development Officer

Department of Industry, Canada

In comon with most other aviation-minded people, Canadians have, for the most part, been content to oversimplify the hazards which atmospheric electricity presents to aircraft. In the early days of flying, problems of controllability, power, and lift overshadowed all other considerations. There is no evidence that the "leather helmet and goggles set" ever worried about lightning strikes. As the aircraft evolved to a machine capable of faster flight at higher altitudes, it tended to become a metallic bird, in sharp contrast to the wood and fabric models of earlier years. The evolution of instrumentation and communication and the practice of instrument flying brought aircraft into increasingly frequent contact with clouds and thunderstorm activity. However, World War II brought with it the overriding demands of accelerated production of a quick succession of fighter and bomber aircraft, each with an increment of speed and performance over its predecessor. In this atmosphere of rapid technical development and high risk combat operations, there was little time to worry about lightning strikes; strikes did occur, however, with lasting impressions on aircrews and damage to skin, structure, and communication gear and/or instrumentation.

The post-war years and the cold war brought continuation of a high level of military flying in parallel with a rapid and continuing expansion of general and commercial aviation. As total flying hours grew rapidly, so did the number of lightning strikes and incidents. However, the concern of aviation people in general remained slight, apparently because of the grossly oversimplified reasoning that the metal structure and skin would pass lightning currents without difficulty. As late as the early 1960's it was possible to find otherwise responsible people who refused to take lightning hazards to aircraft seriously.

In the late 50's and early 60's an increasing number of commercial aircraft were equipped with weather radar which enabled them, for the most part, to avoid thunderstorm areas. However, military aircraft frequently are forced to fly through or remain in areas of thunderstorm activity. Furthermore, large-scale operations involving many aircraft will occasionally coincide with poor weather in which electrically charged areas abound. The time is then ripe for a rash of lightning strikes which severely test the lightning protective features of the aircraft. Such an occasion arose for Canada in 1961. A series of strikes caused structural damage as well as other minor damage and loss of facilities to the Argus ASW aircraft, and urgent demands were made for improved protection. A quick search of the literature indicated that the Lightning and Transients Research Institute of Minneapolis had been active in research of this problem for many years.

After a thorough review of the lightning strike problem in relation to the Argus aircraft, the RCAF decided to proceed with full-scale tests. These tests involved simulated lightning strikes to parts of the aircraft which were likely to experience strikes in operation. A number of modifications to improve lightning protection were fitted to the aircraft; in several cases they were evaluated by LTRI prior to the full-scale tests.

The full-scale tests were carried out in July 1963 at the U.S. Naval Air Station at Mayport, Florida. This is an ideal site, where a large aircraft can be positioned in close proximity to the shipborne lightning facility "Thunderbolt," which LTRI was just putting into service (see Figure 1). Areas selected for simulated lightning strike tests were the fin cap antenna, nose radome, nose observers dome, and the MAD boom. A large number of strikes to the selected test areas were carried out to determine vulnerability to lightning strikes (as illustrated in Figures 2 and 3) from a number of different approach angles. Protection measures were agreed upon by LTRI and the RCAF test team. The fin cap antenna was fitted with two diverter rods to minimize skin damage, and an older model arrester and tubular connector inside the vertical stabilizer were tested. An over-specification stroke was fired to the vertical fin cap and the arrester was damaged. The arrester was replaced with a newer model MIL-A-9094C specification arrester; retest confirmed the installation was capable of carrying repeated strike currents without serious damage. The radome protection of foil strips proved satisfactory, as did the conductor array on the nose observer's dome. Professor Newman checked the installation by staying inside the observers'

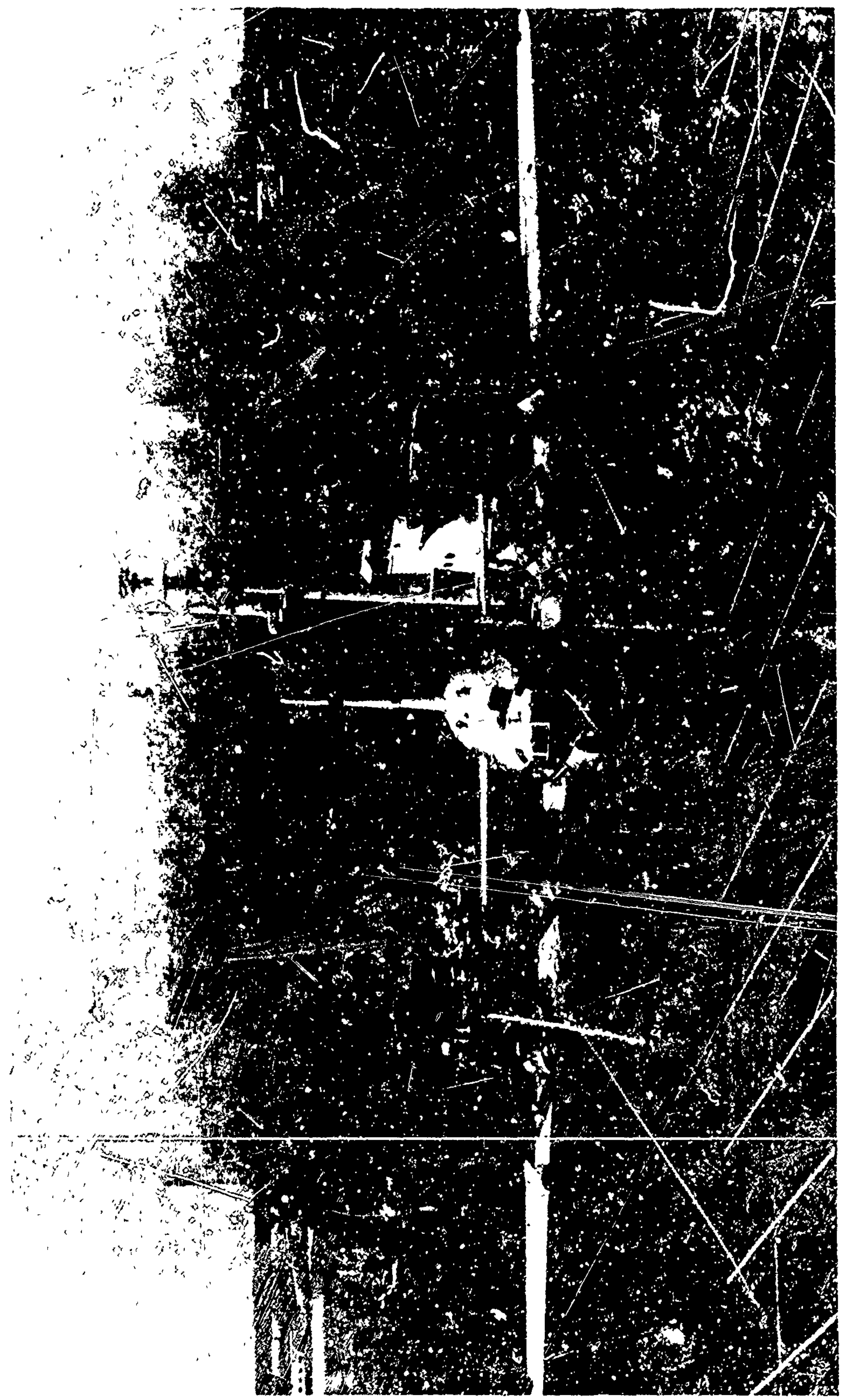

Figure 1. Carrier Pier at Mayport U.S. Naval Air Station With Full-Size Aircraft Alongside LTRI Research Vessel Thunderbolt for Test With Artificial Lightning Generators and Wind Tunnels

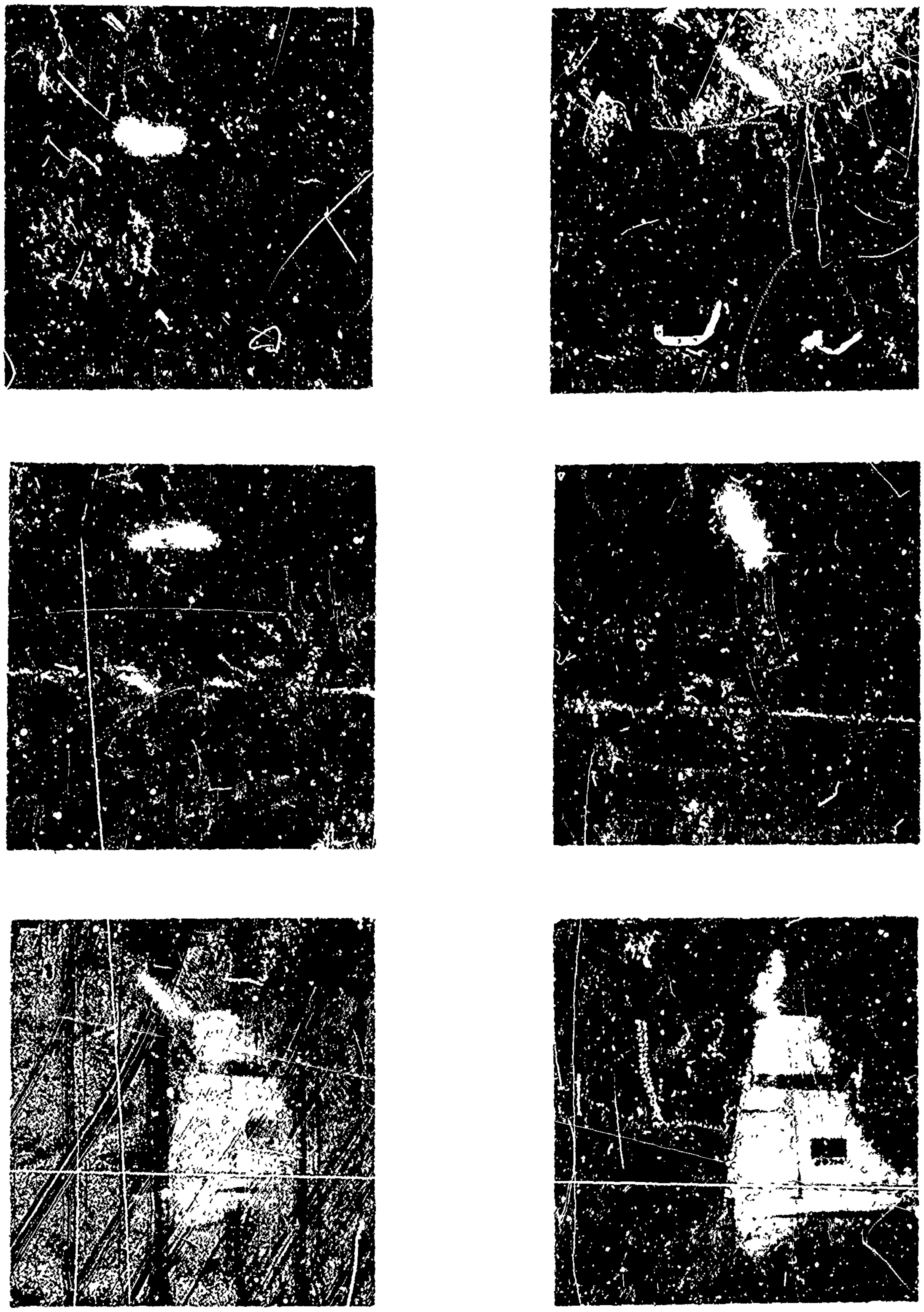

Figure 2. Artificial Lightning Discharge Tests of Graded Resistance Lightning Diverter Rods on Argus Fin Tip


Figure 3. Artificial Lightning Discharge From Above Fin Showing Diverting Action as Stroke Follows Irregular Path to Fin Tip

dome while artificial lightning discharges were applied (Figure 4). Based on this experience, slight rearrangement of the conductors was recommended for the final configuration. The MAD boom was protected by a single diverter trailing from it. Lightning strike current is carried to the airframe by two rods which follow the outside contour of the MAD boom along a horizontal midplane. One of these rods has a 1/8" spark gap, which ensures that a conductive loop will not be present, which might affect the operation of the MAD equipment.

Another aircraft was subjected to full-scale lightning protection tests in February 1966, with CF-104 (Figure 5), which is operated extensively by the RCAF. This aircraft was built by Canadair Limited of Montreal under licence from Lockheed. The areas tested were the nose radome and pitot boom, canopy, wing tip tanks, filler caps, and access doors. LTRI recommended modifications which greatly improved lightning protection of each section. All protection is highly individualistic, although relatively standard techniques are used, such as stroke diversion, conductors over nonconductive surfaces, or arresters. In areas involving fuel systems, design changes were required to prevent internal sparking at the discontinuities presented by filler caps and access doors. Fuel vents also require close study not only as to location so that lightning strike probability will be low but also as to flame arresting should a strike ignite a combustible fuel-air mixture. Although lightning protection features are individualistic when the aircraft is considered as a system, redesign and retrofit can be minimized if individual components such as antennae and fuel filler caps are designed to provide maximum protection. I feel that it is accurate to say that requirements for an adequate standard of lightning protection are now being incorporated in the airworthiness regulations of the United States and Canada.

No technology can be comprehensive and complete unless all environmental factors are systematically and thoroughly explored. The interaction of atmospheric electricity and aerospace vehicles is an interesting and sometimes awe-inspiring study. The reliability of our air transport industry and the effectiveness and operational flexibility of our military air services are dependent on the results of the painstaking work carried out by LTRI's staff and other interested workers. They deserve — and I am sure they will receive — continuing support from aerospace leaders in both Industry and Government.

Figure 4. Artificial Lightning Discharge to Protected Nose Observer Dome With Observer Inside

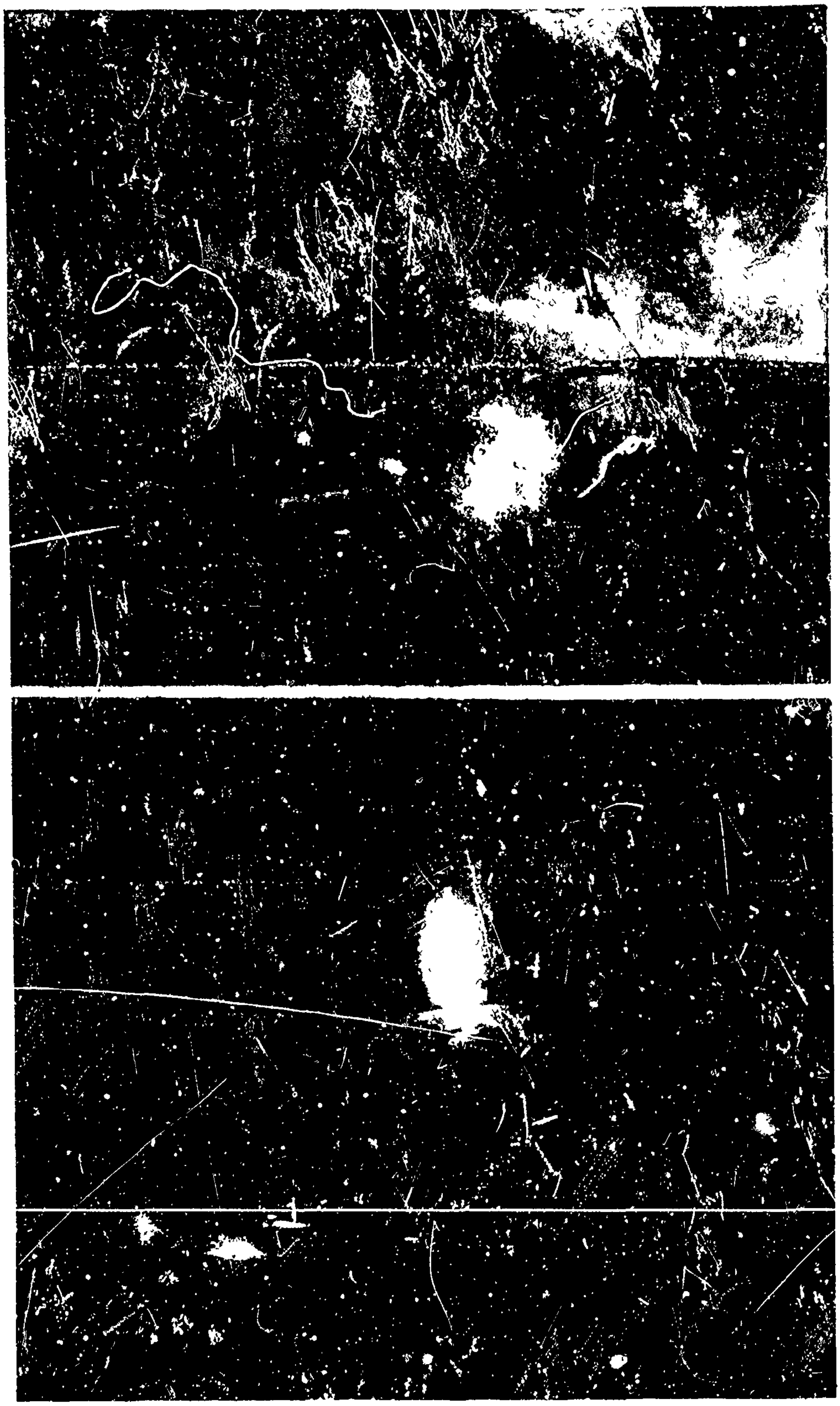

Figure 5. Test Lightning Stroke to CF-104 Fighter Canopy With Pilot Inside

# BRITISH RESEARCHES AND PROTECTIVE RECOMMENDATIONS OF THE BRITISH AIR REGISTRATION BOARD

B. L. Perry

British Air Registration Board

For centuries, the damage caused to buildings and other objects on the ground has been well documented and methods of protecting such buildings have been evolved and developed. With aircraft, however, the rate of development has been such that no time was available to solve such problems, and the rapid growth of airline operations in all parts of the world and under all climatic conditions has meant that more and more aircraft are being exposed to the effects of lightning strikes.

In the late 1950's it was realised within the Air Registration Board that the requirements for the protection of aircraft against damage, and possible disaster, were not adequate. This was confirmed by several cases of severe damage to aircraft which were inadequately protected.

A study of the data available at that time showed a dearth of detailed information on the incidence and effect of lightning strikes on aircraft. Military information suffered from difficulties regarding security and the comparatively limited flying hours, and civil data was made available via aircraft constructors and was thus second or third hand before being presented and consequently suffered from incompleteness and inaccuracy.

With the cooperation of several British airlines, a reporting system was introduced whereby details of all lightning strikes to their aircraft were recorded. This paper will summarise the data gathered and outline the requirements for the protection of British civil aircraft based on these records and other available information.

## ANALYSIS OF LIGHTNING STRIKE REPORTS

The recording system involved the completion of a form by the aircrew and the ground staff. An example of this form for a Trident aircraft is shown in Figures 1 and 2 for the upper and lower surfaces of the aircraft, respectively.

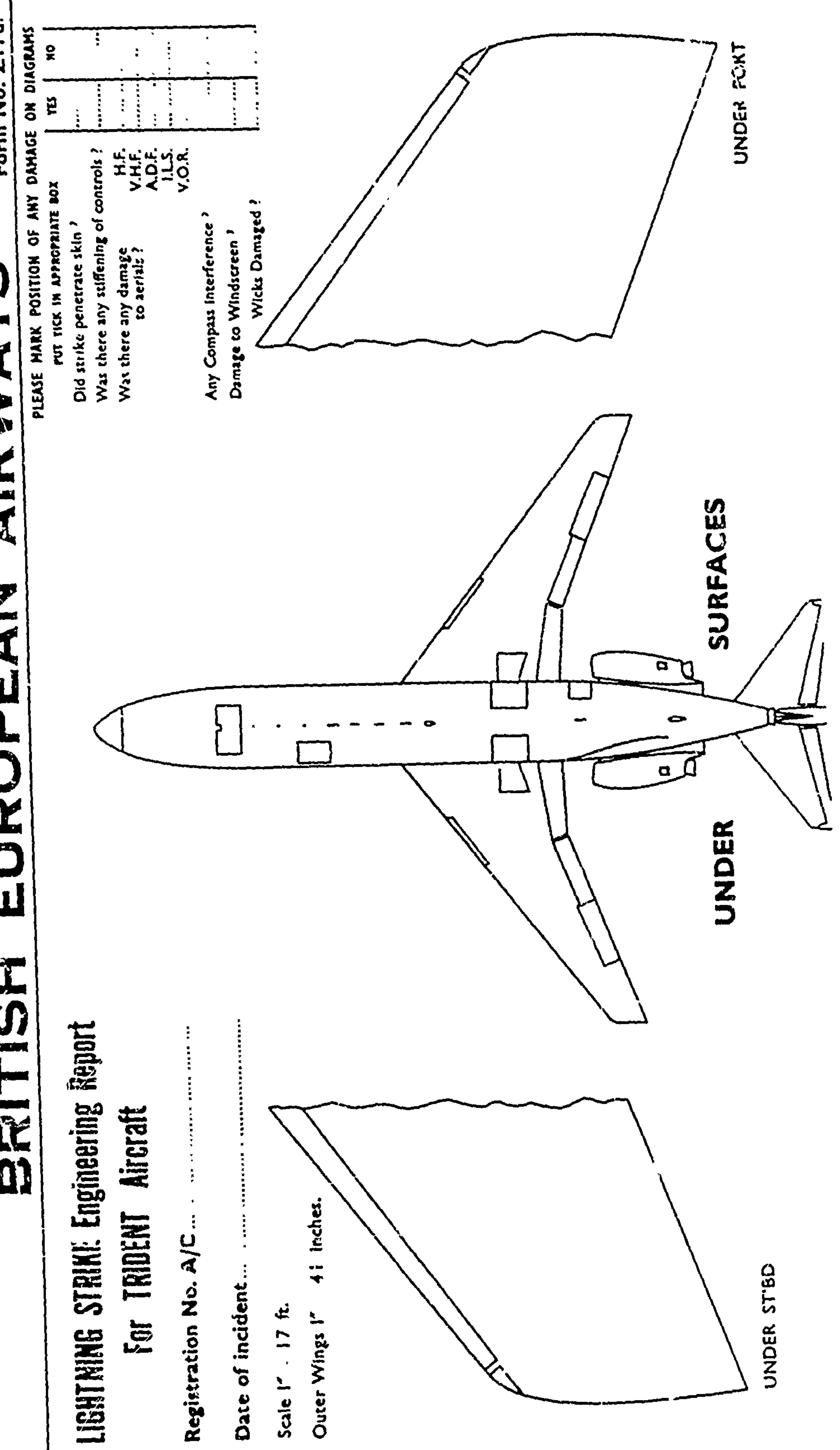


Figure 1. Typical Form for Reporting Lightning Strikes — Front

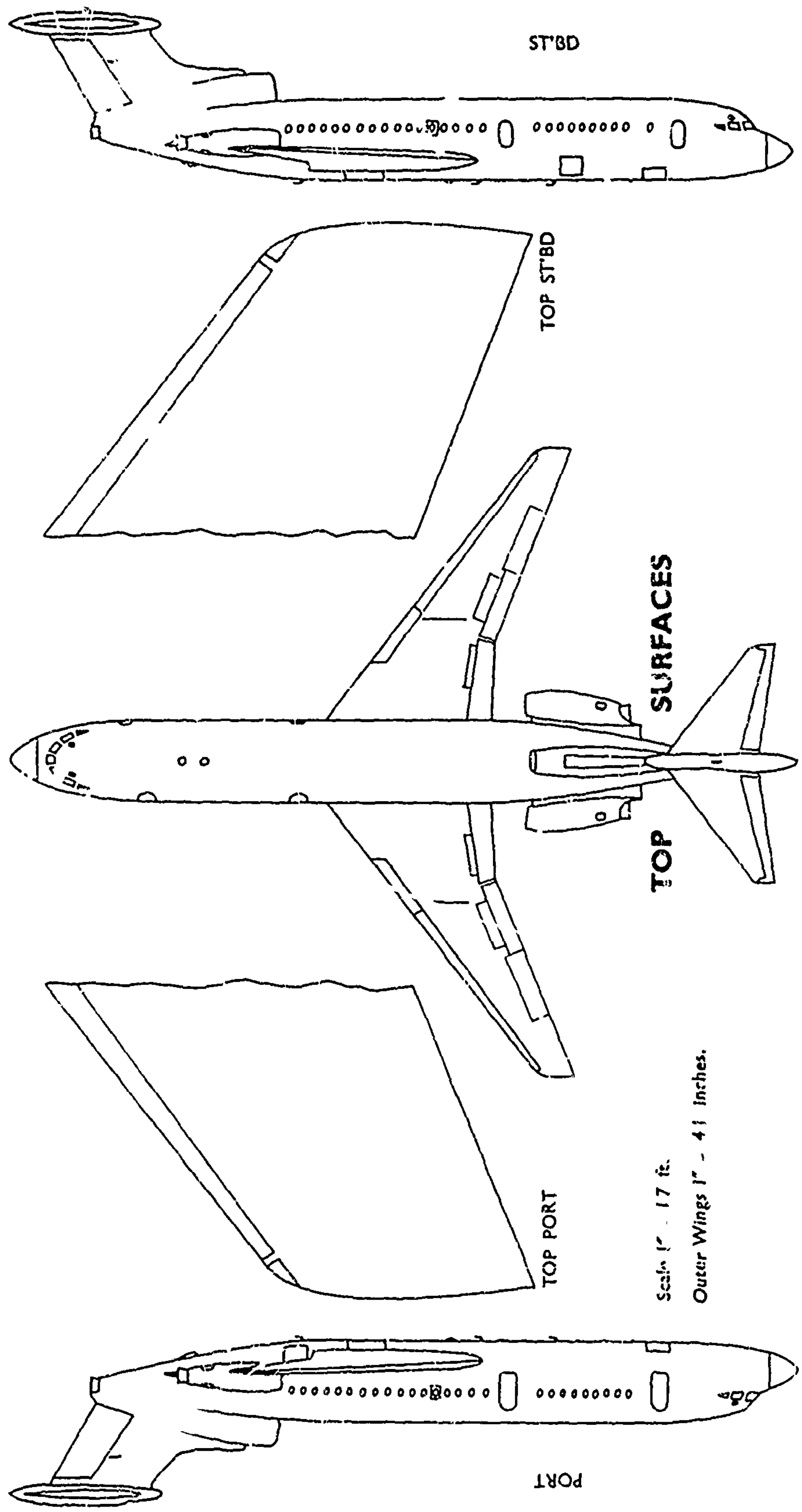


Figure 2. Typical Form for the Reporting of Lightning Strikes — Back

The location and type of damage were recorded and aircraft altitude, outside air temperature, and weather conditions recorded. These records have been kept for varying periods since 1959 for Viscount, Vanguard, Comet 4E, Trident, Britannia, Boeing 707, and VC.10 aircraft. Results are summarised in Table I.

Not all strikes have been recorded, but those recorded indicate that the average incidence rate for aircraft operating in Europe is one strike per 2400 flying hours. The rate for world-wide operation is lower, at one per 6000 flying hours. Since the reports of world-wide operators do not show any more strikes occurring on their aircraft in Europe than in the rest of the world, then it can be assumed that the difference between European and world-wide incidence rates is a function of the percentage of the flying time spent at lower altitudes; this, as will be shown later, presents the highest risk.

Assuming each airplane flies 6-1/2 hours per day, then in Europe each aircraft will be struck, on the average, once per year; at 10 hours per day, each aircraft will be struck once every eight months. While the incidence of strikes per flying hour is lower in world-wide operations, the number of strikes per flight is similar to that in Europe.

Figures 3 and 4 show the percentage of strikes relative to outside air temperature and altitude, respectively, and provide meteorological information regarding the most probable temperatures and altitudes for thunderstorms. Several strikes were recorded at altitudes over 20,000 feet and up to 30,000 feet.

Figure 5 shows, for aircraft operating in Europe, the most probable months of the year for strikes. These percentages have not been corrected to allow for the increased flying hours of the aircraft during the summer months but this correction would not materially affect the overall pattern. This data, therefore, supplies approximate meteorological information on thunderstorm activity over the year.

Consider now the damage caused to the aircraft by these strikes. Data for a total of 456 strikes on six aircraft types is summarised in Table II. Under the heading of "hole in structure or radome" is included all cases of

TABLE I

INCIDENCE OF LIGHTNING STRIKES RELATIVE TO AIRCRAFT TYPE, ZONE OF OPERATION AND FLYING HOURS

| Type of Aircraft | Zone of Operation | Period Covered | No. of Strikes | Total Flying Hours | Incidence of Strikes |
|---|---|---|---|---|---|
| Viscount | Europe | March 1959 to June 1964 | 195 | 567,000 | 1/2900 hrs |
| Vanguard | Europe | May 1961 to June 1966 | 79 | 194,000 | 1/2500 hrs |
| Comet 4B | Europe | June 1960 to June 1966 | 86 | 162,000 | 1/1900 hrs |
| Trident | Europe | May 1964 to June 1968 | 92 | 140,000 | 1/1400 hrs |
| Britannia | Europe | October 1959 to April 1961 | 6 | 115,000 | 1/19,000 hrs |
| Boeing 707 | World-Wide | January 1962 to December 1967 | 103 | 458,000 | 1/4400 hrs |
| VC 10 | World-Wide | August 1964 to May 1968 | 28 | 251,000 | 1/9000 hrs |
| Total | Europe | — | 452 | 1,063,000 | 1/2400 hrs |
| Total | World-Wide | — | 137 | 824,000 | 1/6000 hrs |
| Total | Europe and World-Wide | — | 589 | 1,887,000 | 1/3200 hrs |

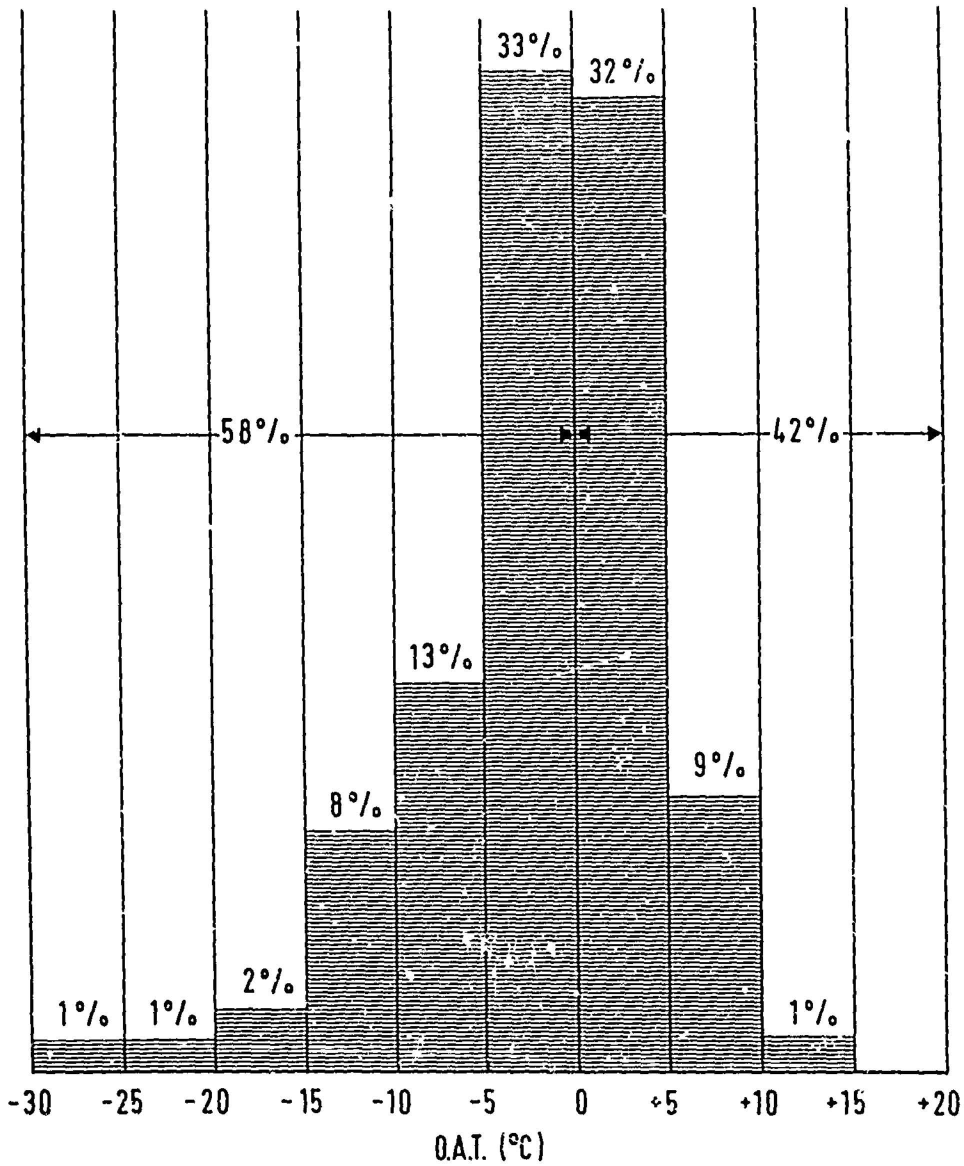


Figure 3. Distribution of Lightning Strikes Relative to Temperature — Viscount, Vanguard, and Comet 4B Aircraft

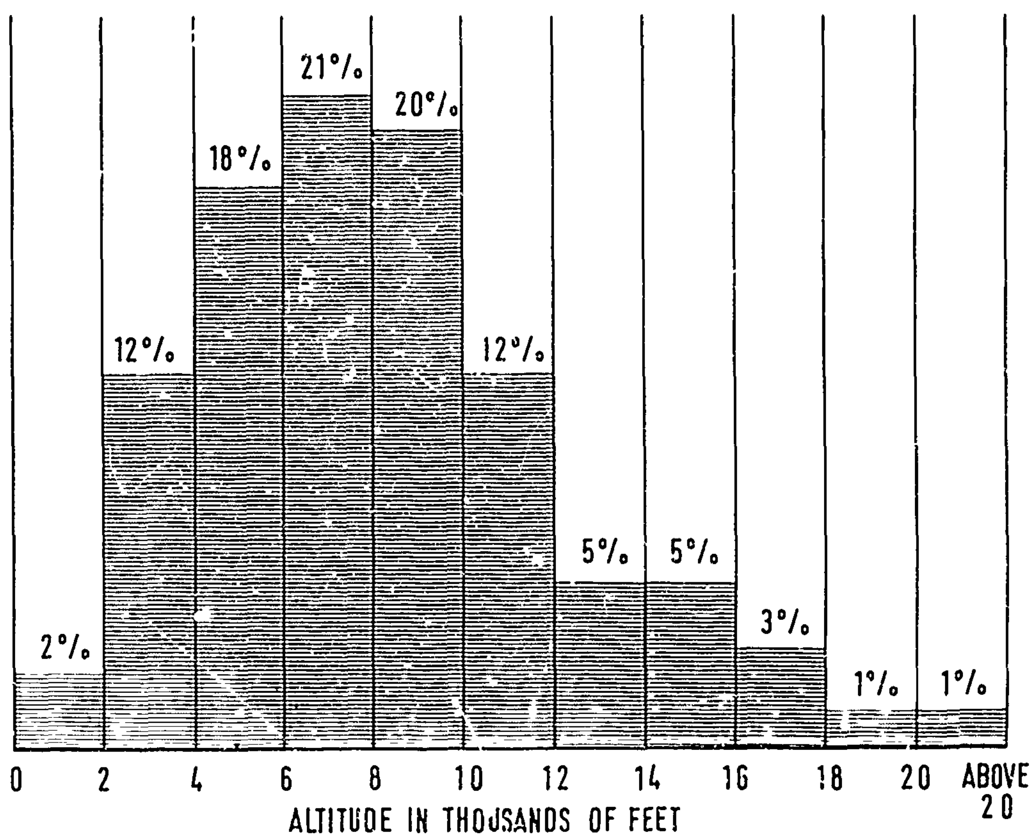


Figure 4. Distribution of Lightning Strikes Relative to Altitude - Viscount, Vanguard, Comet 4B, Trident, VC. 10 and Boeing 707 Aircraft

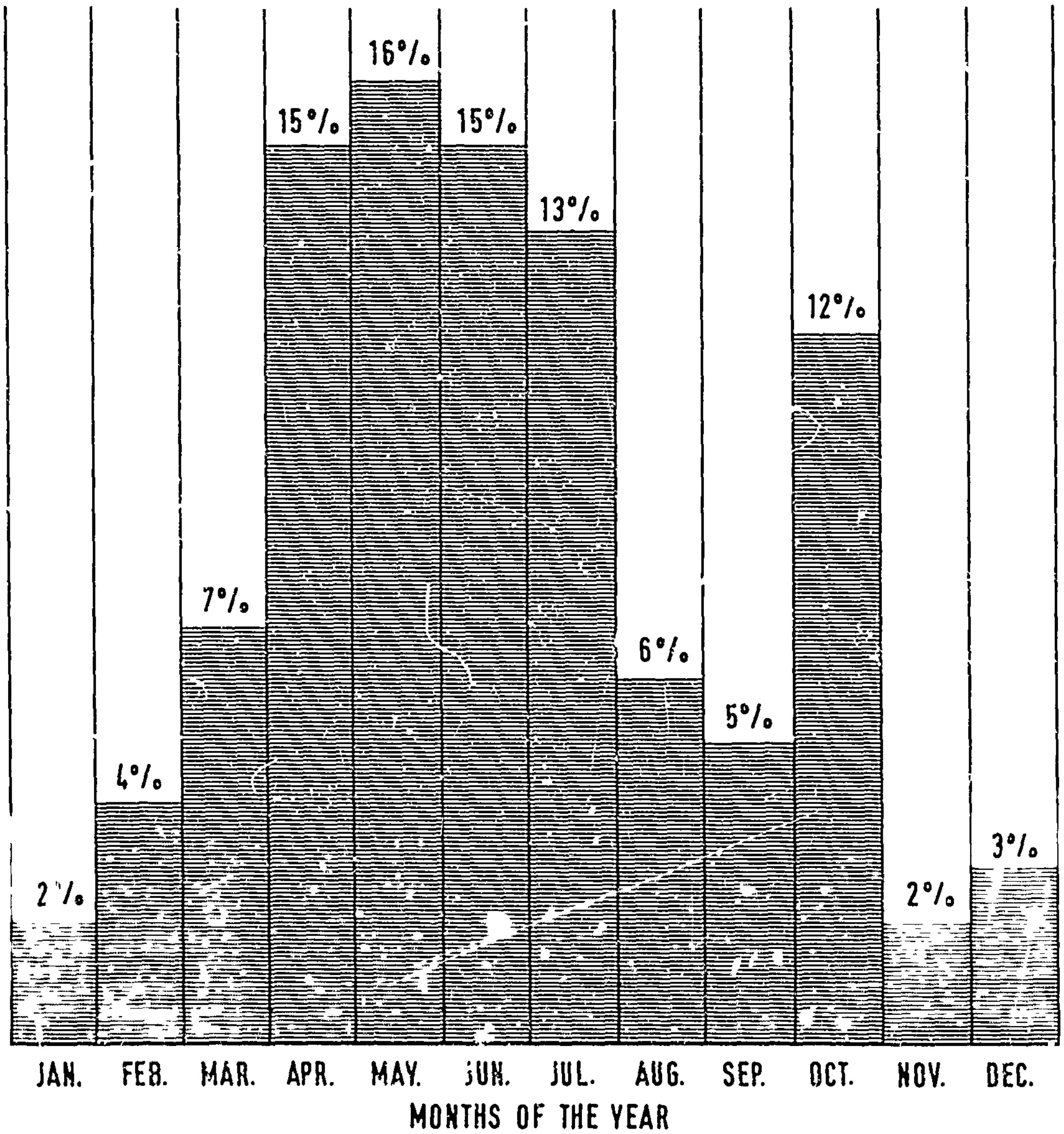


Figure 5. Occurrence of Lightning Strikes Relative to Months, in Europe — Viscount, Vanguard, Comet 4B, and Trident Aircraft

TABLE II

DAMAGE CAUSED TO AIRCRAFT BY LIGHTNING STRIKES RELATIVE TO AIRCRAFT TYPES

| Type of Aircraft | Period Covered | No. of Strikes | Damage Caused: Hole in Structure or Radome % | Slight Damage % | No Damage % |
|---|---|---|---|---|---|
| Viscount | May 1961 to June 1964 | 92 | 39 | 42 | 19 |
| Vanguard | May 1961 to June 1964 | 79 | 30 | 52 | 13 |
| Comet 4B | May 1961 to June 1966 | 72 | 35 | 25 | 40 |
| Trident | May 1964 to June 1968 | 90 | 11 | 57 | 32 |
| Boeing 707 | January 1962 to December 1967 | 96 | 17 | 50 | 33 |
| VC. 10 | August 1964 to May 1968 | 27 | 30 | 44 | 26 |
| Total | — | 456 | 27 | 45 | 28 |

actual penetration of the skin or puncturing of the radome. "Slight damage" covers burns and scorch marks; rivet heads, in particular, were particularly susceptible to this type of damage.

A detailed study of the location of damage from the entry or exit of a strike shows that the areas affected were nose and fuselage; wing tips; aileron, flap, rudder, and elevator trailing edges; fin and tailplane tips and tail cone; and the antenna and static wicks. Damage of the Viscount, Trident, and Boeing 707 aircraft is shown in Figures 6, 7, and 8, respectively; similar data is available for the other aircraft. All data is summarised in Table III.

These data show the swept wing aircraft to be more liable to strikes along their main axes (i.e., nose to tail) than in the wing areas, compared with the straight-wing propeller-driven aircraft.

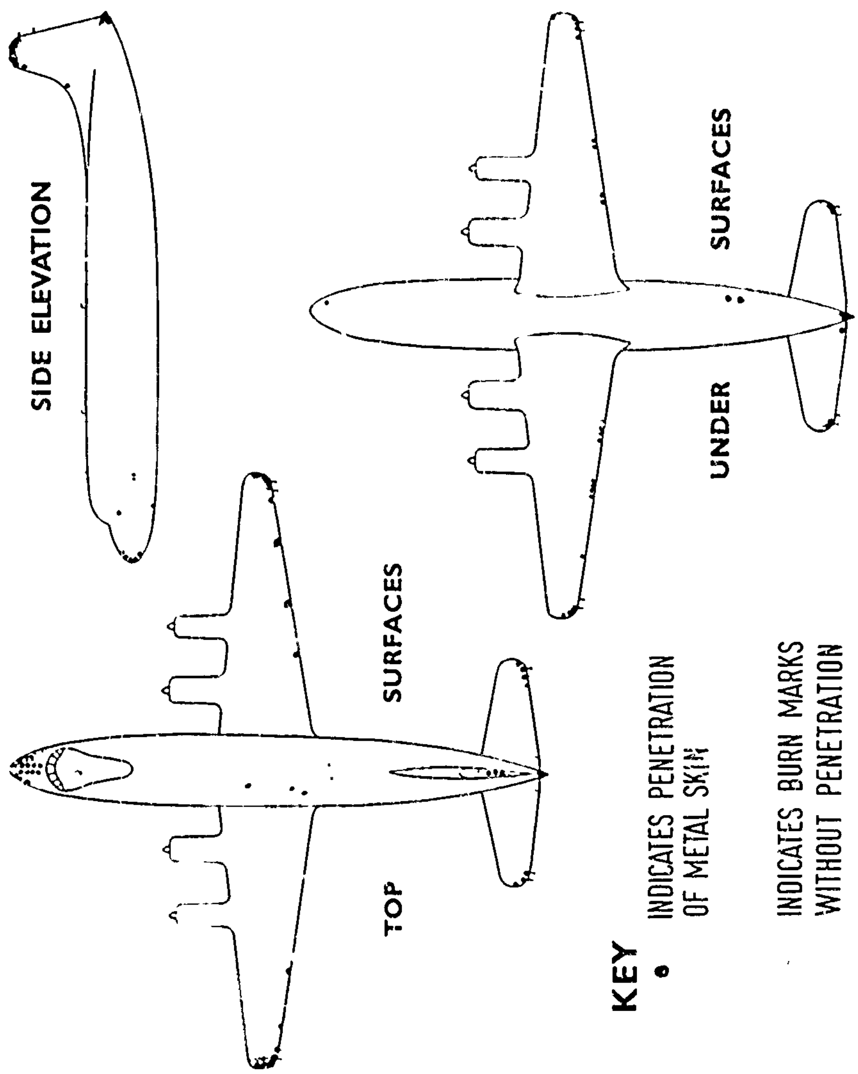


Figure 6. Position of Lightning Strikes on Viscount Aircraft — March 1959 to June 1964

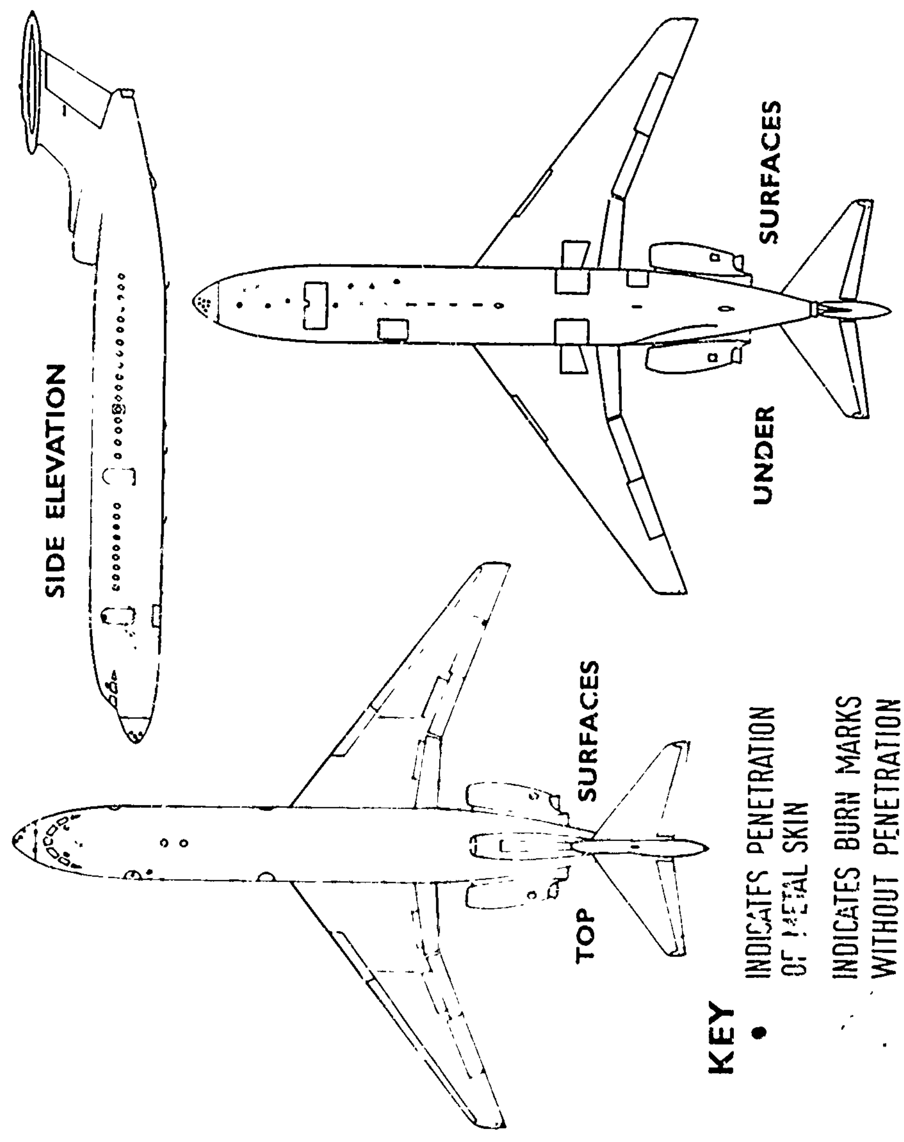


Figure 7. Position of Lightning Strikes on Trident Aircraft — May 1964 to June 1968

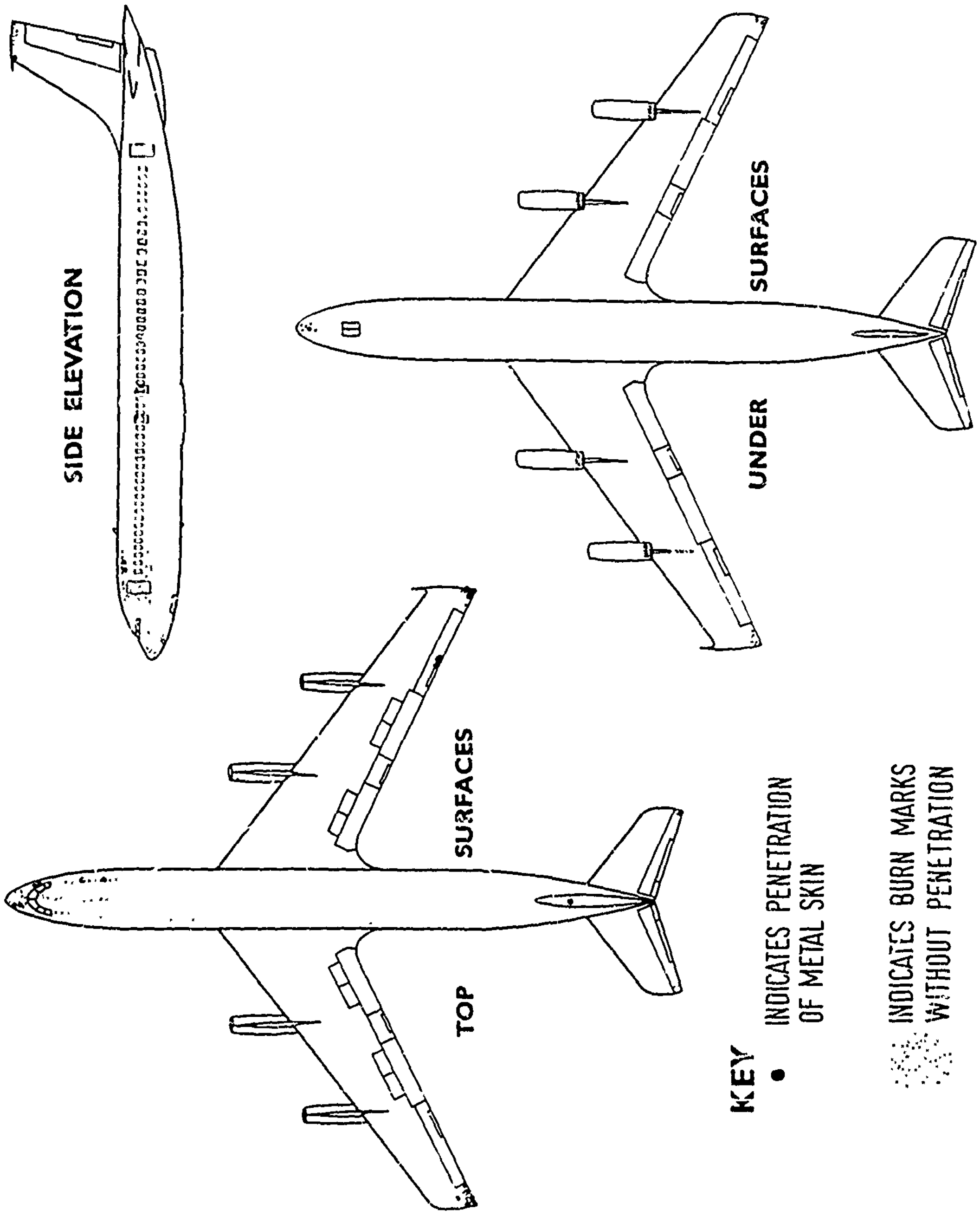


Figure 8. Position of Lightning Strikes on Boeing 707 Aircraft — January 1962 to December 1967

TABLE III

POSITIONS OF LIGHTNING STRIKES ON AIRCRAFT RELATIVE TO AIRCRAFT TYPE

| Type of Aircraft | General Area Struck | |
|---|---|---|
| | Axial - Nose, Fuselage, Tail End, including Fin, Rudder and Elevators | Transverse - Wing Area including Propellers, Nacelles, Flaps and Ailerons |
| | % of Total Recorded Damage | % of Total Recorded Damage |
| Viscount | 65 | 35 |
| Vanguard | 77 | 23 |
| Comet | 80 | 20 |
| Trident | 75 | 20 |
| Britannia | 64 | 34 |
| Boeing 707 | 75 | 25 |
| VC 10 | 80 | 20 |
| Total "Straight Wing" | 69 | 31 |
| Total "Swept Wings" | 78 | 22 |

Study of the reports showed that damage to the antenna was confined mostly to HF aerials and their associated A.T.U.'s. Even direct strikes to antennas, as shown by burn marks, rarely affected the associated radio equipment. As would be expected, the static wicks on all types of aircraft were regularly damaged or destroyed. Compasses, both remote and direct magnetic, were affected; on one type of aircraft, the direct-reading magnetic compass was found to have a large error after approximately 20% of the strikes, indicating the presence of "soft iron" adjacent to the instrument.

No physiological effects on the aircrew have been reported, even on strikes to the nose and windscreen pillars.

HAZARDOUS EFFECTS OF LIGHTNING STRIKES

Since the incidence of strikes on aircraft in civil operation is of the order of one strike per aircraft per year, then it is obviously vital from a safety point of view, that these strikes do not create a hazard to the aircraft or its occupants. Potential risks are considered under four headings; Structural, Fuel, Other Systems, and Personnel. These are interrelated but are considered separately, with some of the possible effects reviewed and illustrated.

Structural Effects

Figure 9 shows the damage caused to the nose of an Ambassador aircraft by a lightning strike in about 1955. The nose, of wood and plywood construction, was so damaged and displaced that the view through the windscreen was obstructed; the aircraft was landed safely by the second pilot looking through the D.V. window. Such materials are now not normally used in the construction of civil aircraft.

Dangers can arise if portions of the airframe are damaged and blown into flying control surfaces or propellers, or ingested into the engines. Such disruption of structure can be caused by rapid expansion of gases within the materials, themselves, or by rapid build-up of pressure within enclosures covered by the parts struck by lightning, which causes damage to the parts themselves or the surrounding structure. Because of the increasing use of fibre glass for aircraft structural purposes, this danger must be fully recognised. Holes or structural damage which occur, unnoticed, during the early stages of a flight may affect performance and handling of high-speed aircraft during later, possibly supersonic, stages of the flight.

This study of actual strikes shows that a fair proportion of strikes entered or left the aircraft at the trailing edges of flying control surfaces, ailerons, flaps, rudders, or elevators. Thus, high currents must have passed across the hinge points of these surfaces. In some aircraft, flexible conductors of sufficient size to carry these currents are connected across the bearings. This may not be essential, however, since tests, carried out in England, with peak currents of up to 100,000 amps through various bearings down to 3/4 inch outside diameter, showed slight damage to the bearings but no cases of bearing seizure


Figure 9. Damage to an Ambassador Aircraft by a Lightning Strike on the Nose Cone

(Reference 1). This is supported by actual aircraft experience; while cases of damage to bearings have been reported, the writer has no knowledge of actual seizure of a control surface bearing due to a lightning strike.

The need to provide a path, not only of low resistance but also of low impedance, is illustrated by the damage caused to the seals of the hydraulic jack operating the control surfaces on the tail of a Tee-tailed aircraft. The jack was shunted by a conductor of adequate cross section to carry lightning strike currents, but difficulties in the physical installation caused this conductor to run in a fairly large loop. In service, strikes to the tail caused current to flow, not through this conductor, but through the jack body and across the seals, resulting in leakage of hydraulic fluid.

The effects of a lightning strike to a glider constructed completely of wood is shown in Figure 10. In this incident, the glider was wrecked and the pilot was killed. The strike entered the aircraft at the port aileron pulley support, then travelled via the aileron cables to leave via the starboard pulley. As can be seen, the cables were disintegrated; it is calculated that a current in excess of 50,000 amps would provide sufficient energy for the cables to boil with some vapourisation, and for temperatures in the order of 3000°C to be reached. The wreckage also indicated that extensive damage of the mainplane had occurred before impact; probably a rapid and intense temperature rise had caused an explosion within the wood. No evidence of burns or other signs of electrocution were found on the pilot's body.

Fuel Hazards

The danger of fuel vapour igniting at the aircraft fuel system vents is well known and the large amount of investigation of this problem is encouraging. The importance of locating these vents and fuel jettison pipes away from likely strike areas cannot be overemphasised. Potentially dangerous areas are shown in Figures 6, 7, and 8; areas for other aircraft are similar.

The incidence of strike damage to aircraft wing tips confirms the potential danger of fitted wing tip fuel tanks. They are exposed to the full energy of direct strikes. If they are constructed of metal, skin thickness must be adequate to

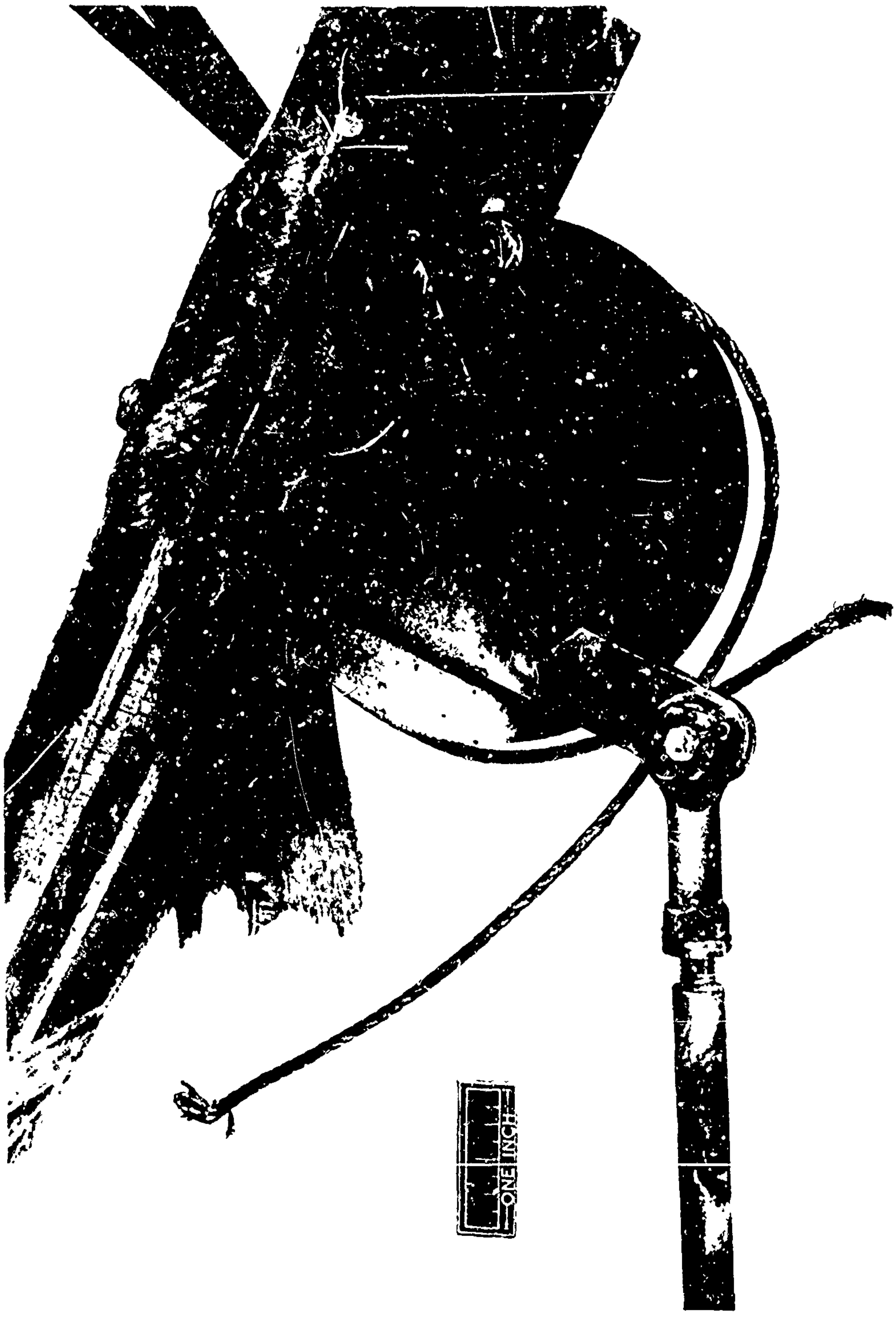


Figure 10. Effect of Lightning Strike Currents Flowing Through the Aileron Cables of a Glider of Wooden Construction

prevent penetration; if nonmetallic, they must be protected with a suitable conducting cage. Many cases have been reported of damage to such wing tip tanks on military aircraft.

Little evidence is available of skin penetration of aircraft wing surfaces resulting in a hazard to integral fuel tanks. This problem and the danger of semi-insulated access panels into these tanks will be considered later. Experience to date has shown no problems due to direct or induced currents from lightning strikes in the electrical circuits of fuel-tank-mounted equipment. This danger must be considered in the design of the aircraft and location of cable runs in the wing areas.

Damage to Other Aircraft Systems

The items most susceptible to direct damage from lightning strikes are external protuberances, such as the antenna. Apart from HF aerials, which sustained considerable damage, present aerials provide adequate protection of the associated radio equipment.

Damage to electrical cables due to lightning currents is shown in Figure 11. The lightning struck the navigation light on one fibre glass wing tip, which exploded and was lost completely, and damaged the adjacent wing ribs. The trailing edges of both ailerons were badly burnt and all the static wicks on the aircraft were damaged. The strike current had apparently entered the aircraft electrical system by way of the navigation light and caused overheating of wiring throughout the aircraft, particularly at the wing root and behind the pilot's dash panel. All navigation light bulbs had exploded with considerable force and the heater igniter coil, stall warning horn, and flap motor were all burnt out. One can only speculate on what the effect would have been on an aircraft fitted with semiconductor equipment.

The possibility of damage on other externally mounted equipment such as pitot-static heads and stall warning vanes must also be considered. A direct strike could not only cause mechanical damage to the equipment itself, but such items may provide a passage for the strike into the aircraft's electrical system. Several such cases have been recorded, but adequate protection is available.

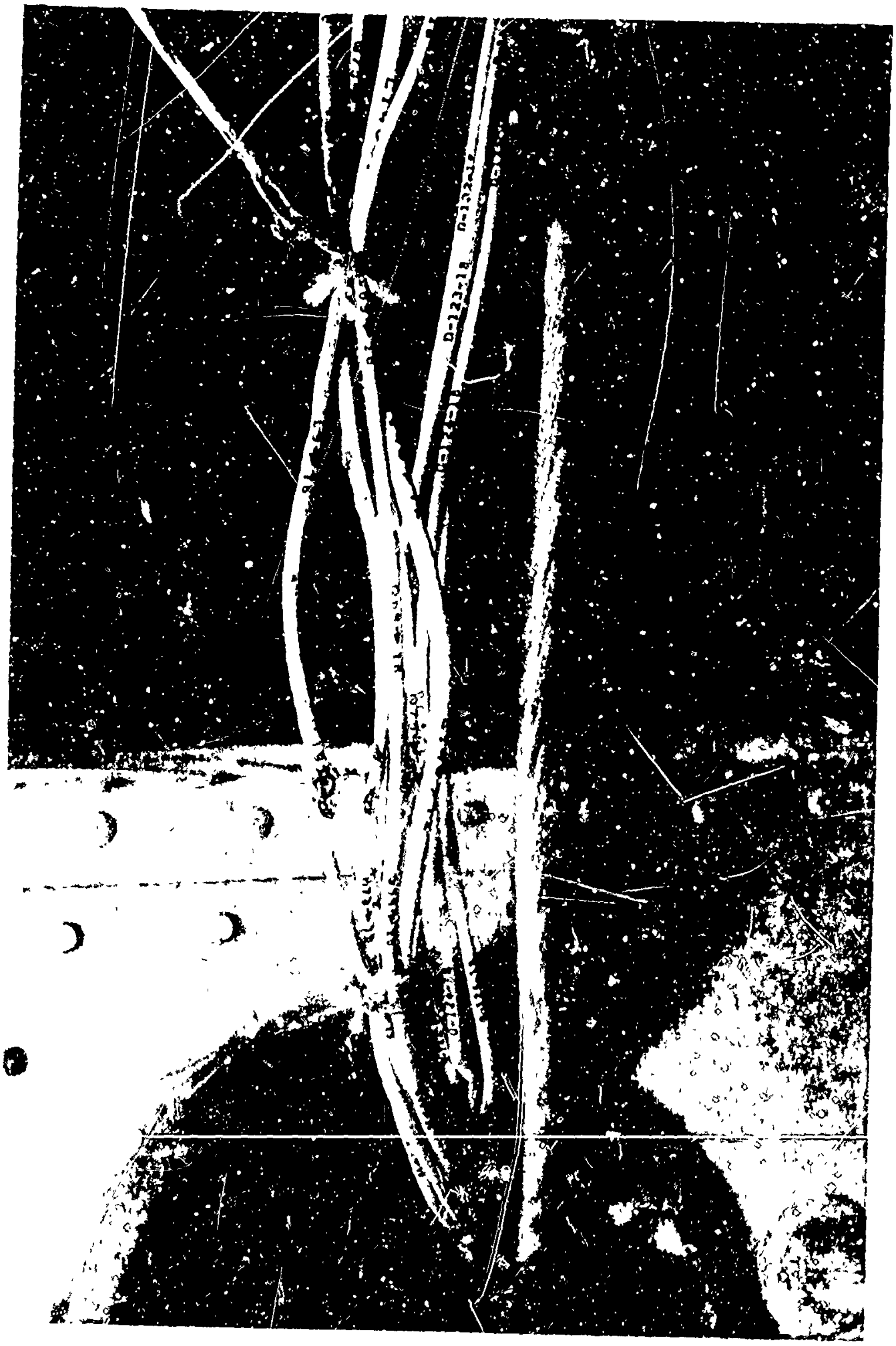

Figure 11. Damage to Electrical Cables on an Aircraft Struck by Lightning

### Personnel Risks

None of the reports covering the 600 strikes tabulated above mentioned any injury or other effects on the aircrew. Reference 2 reported, however, that in two cases of lightning strikes to a Viscount aircraft "both pilots were blinded for 5 to 15 seconds, and partially incapacitated for several minutes." The possible electric shock hazards to the pilot of a glider, both from static electricity and lightning effects, are considered in detail in a Polish report (Reference 3).

## REQUIREMENTS AND RECOMMENDATIONS FOR LIGHTNING STRIKE PROTECTION

In 1963, following a study of the airline reports and a detailed review of all other available data, requirements for protecting British Civil Aircraft against lightning strikes were completely rewritten and published in British Civil Airworthiness Requirements (Reference 4). The latest information has been considered in the Anglo-French Supersonic Transport Aircraft TSS Standard No. 56 (Reference 5). While the TSS Standard specifically concerns supersonic aircraft, it does cover the most recent thinking in Britain for the protection of civil aircraft. The TSS Standard incorporates the requirements of Chapter D4-6 of B.C.A.R.'s, and also takes into account the recommendations of the FAA Advisory Circular AC20-53 (Reference 6).

It is not proposed, in this paper, to repeat all the detailed requirements and recommendations of B.C.A.R.'s or the TSS Standard. Some of the general requirements, however, are outlined below. It should be noted that both sets of standards also cover associated requirements for protection against static electricity hazards and for airframe bonding to provide an adequate return path for the normal aircraft electrical currents in earthed systems.

### Structure

Primary bonds (i.e., a copper conductor of not less than 3 sq. mm. cross-sectional area or the equivalent in other materials) are required to bond all external metallic parts not directly connected to the main aircraft earth system. Unless a strike to the part is improbable or damage to it will not endanger the aircraft, all external nonmetallic parts require a diverter system of primary conductors to carry strike currents into the main aircraft earth system. A metal cage of primary conductors is required for nonmetallic or composite-construction aircraft; this requirement is detailed in B.C.A.R.'s Chapter D4-6.

### Fuel

In general, the recommendations of the FAA Advisory Circular are incorporated in the British requirements. The hazards of tip tanks are further emphasised in Civil Aviation Circular No. 26/1965 (Reference 7). It is felt, however, that the recommended test currents for semi-insulated access doors and filler caps on wing surfaces are unrealistically high. Such items are generally exposed to only swept strokes, and it is doubtful if currents of 200,000 amperes peak will be involved. The same contention applies to swept strokes causing skin penetration and endangering integral fuel tanks. It is agreed that a minimum skin thickness of 0.080 inches for metal fuel tanks in areas exposed to direct strokes is reasonable, but there is little evidence to confirm that such a thickness is necessary for areas not exposed to direct strikes. Further research work is required to determine both the characteristics of swept strokes and the energy required to penetrate various thicknesses of skin on an aircraft and hence the evidence for a rational airworthiness requirement.

### Other Hazards

Experience shows that external protuberances such as the antenna or pitot heads can be designed so as to prevent dangerous consequences to the aircraft as the result of a lightning strike. Semiconductor devices may present problems in the future due to damage by surge voltages. As yet, no specific requirements can be written to cover such an eventuality.

## CONCLUSIONS

Civil aircraft are regularly struck by lightning and this is expected to continue at approximately the present rate. Probably, no major advance in weather radar will enable aircraft to avoid thunderstorm activity completely, and even if possible, the necessary flying restrictions would most probably be commercially unacceptable.

Although the present British accident record relative to the number of strikes is good, we must not become complacent. Two of the examples show how easily aircraft damage could become aircraft disaster. More work is required to find answers to the many unknowns concerning both the nat[illegible] lightning strikes and their effect on the aircraft and its systems. Some [illegible]oblem areas have been mentioned in this paper.

The responsibility to provide a safe aircraft rests w[illegible] the aircraft designers. Dangers from lightning strikes must be the concern of all aircraft designers, particularly the structural, fuel system, electrical, and radio specialties. Protection can be built in, at little or no penalty in weight and cost, if considered early enough in the design. Those of us concerned with airwrothiness must ensure that adequate standards are maintained, and that these standards are both realistic and up to date.

ACKNOWLEDGMENTS

The help and co-operation given by the airlines in the recording of the data on which this paper is based is gratefully acknowledged. The author thanks the Air Registration Board for permission to publish this paper, and his colleagues within A.R.B., particularly Mr. E. Lloyd, for their help in its compilation. Thanks are also due to many members of the British aircraft industry for the provision of other information used in this paper.

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6. Federal Aviation Agency, Advisory Circular No. AC20-53, "Protection of Aircraft Fuel Systems Against Lightning."

7. Civil Aviation Information Circular 26/1965, "Vulnerability of Certain Foreign-Built Light Aircraft to Lightning Strikes."

# LTRI – INDUSTRY COOPERATIVE PROGRAM RESEARCH ON THE CONCORDE SST

S. T. M. Reynolds

British Aircraft Corporation

Filton House, Bristol, England

and

M. M. Newman

Lightning Research Oceanic Laboratory

## SST RADOME LIGHTNING PROTECTION

Model studies with high-voltage discharges have shown that a high percentage of strikes to typical SST's may be expected to the pitot boom on the nose radome, as illustrated in Figure 1. The most important consideration is, therefore, to provide adequate current paths to the airframe to assure that no internal arcs are produced to cause explosive gas pressures. Experience with many supersonic radome designs for fighter aircraft has shown that the most important phase of protection development is the test of the production bonded pitot boom radome assembly. Small changes in protection systems to make them easier to manufacture can completely ineffectuate them. Also, surge protection is required for the pitot heater wiring, which can provide a direct path for coupling lightning strike energy into the aircraft electrical system, as illustrated in Figure 2. Surge coupling can be handled on an individual basis by shielding or using surge protection devices on the wiring passing the forward bulkhead period. These techniques are being applied to the nose radomes on the supersonic aircraft presently under development.

## ANTENNA SYSTEM LIGHTNING PROTECTION

Use of HF shunt-fed antennas simplifies antenna lightning protection by reducing the problem from channeling the total lightning stroke current into the aircraft interior to the much less hazardous problem of inductive surges across the antenna. This produces short high-voltage low-energy pulses in the communication equipment, which can result in secondary power follow damage if not properly

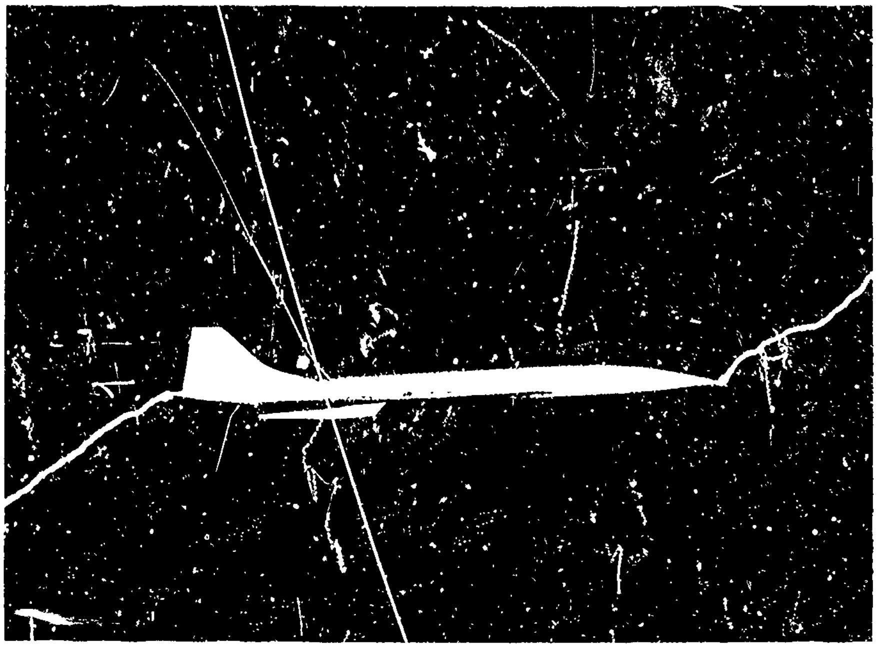

Figure 1. Artificial Lightning Discharges to Model Indicate the Most Probable Point of Lightning Strike Would Involve the Radome

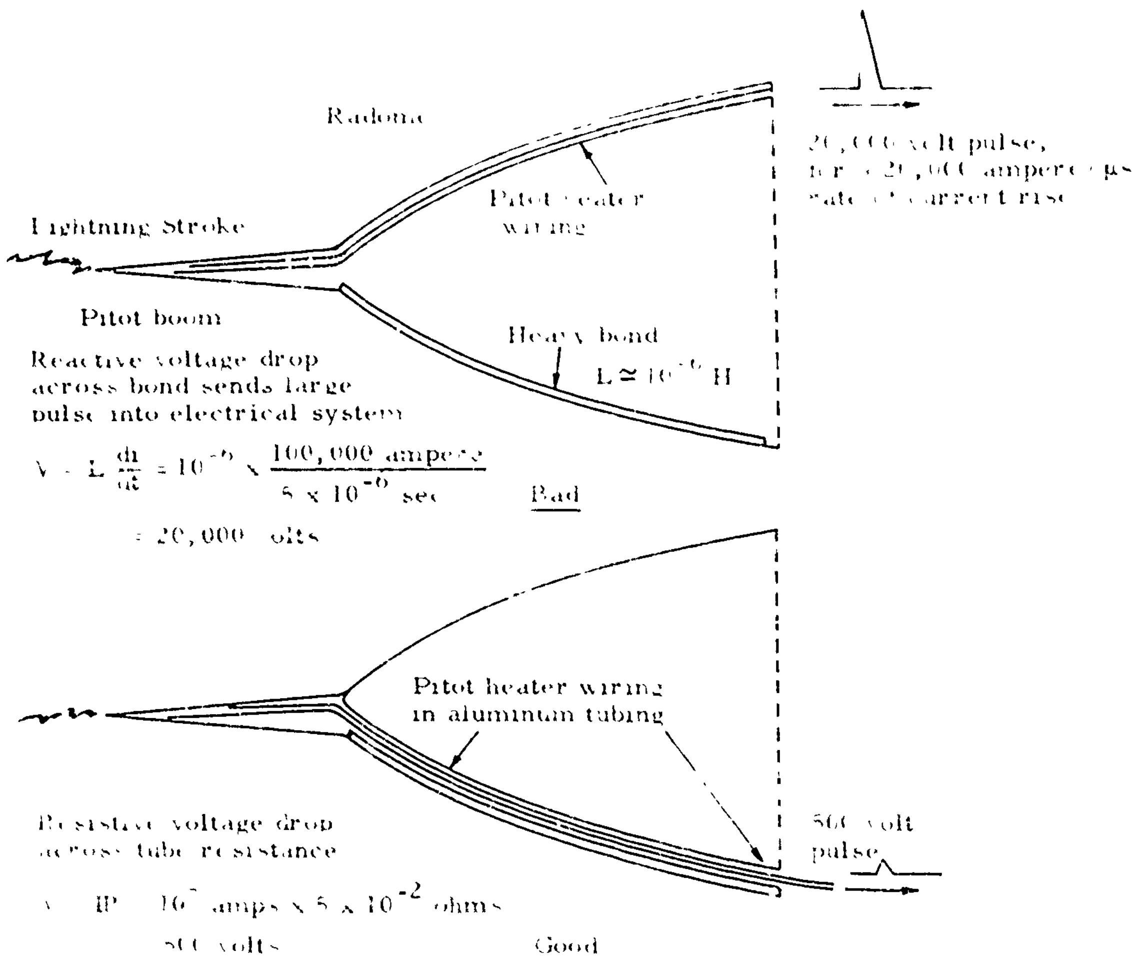


Figure 2. Typical Examples of Surge Coupling Into Aircraft Wiring From Natural Lightning Discharges. (Discharge to pitot boom couples pulses into pitot heater wiring, requiring wiring shielding which can be incorporated inside a main bond tubing conductor.)

handled. At present most lightning protection systems are not designed to prevent the high-voltage low-energy pulses, so these must be handled separately. One of the best ways to handle the problem is simply to include components in the input system to handle these, which can best be determined by operational artificial lightning tests. It has often been found in the past that replacing semiconductors, capacitors, resistors, or relays with components of slightly higher voltage ratings was sufficient to provide adequate surge protection.

## ELECTRONIC EQUIPMENT SHIELDING

Of special concern on all newer aircraft is the surge vulnerability of the electronic systems because of the increased use of integrated circuits and high impedance semiconductor devices (such as field effect transistors), and the surge sensitivity of computer circuits and memories. As pointed out previously, plastic skins leave the wiring exposed to the thunderstorm's electromagnetic field. Theory indicates that little current or magnetic field can penetrate a well-bonded aircraft skin; with very long fuselages, however, the IR voltage drop from the residual current penetrating to the interior may be sufficiently large to cause problems on electronic or electrical circuits.

What is still of particular concern is the wide variance between what seems practical in the way of shielding for wiring, such as radar antenna control wiring under the radome (which is exposed to lightning electromagnetic fields) and what pulse amplitudes the computer and electromagnetic interference control engineers feel is allowable. Present specifications require that sufficient shielding and bonding be provided so that no pulse greater than 500 volts be seen on the wiring with a 200,000 ampere lightning discharge to the radome or radar antenna. For practical installations with roll and pitch stabilization, the addition of sufficient shielding to meet this requirement severely inhibits antenna movement. Conversely, interference control groups often feel that 500 volts, even if it could be met, is much greater than can be permitted on airborne digital computer circuitry. Typical pulse coupling mechanisms are illustrated in Figures 3 and 4. Thus, serious research and development efforts are still required to resolve this incompatibility between component requirements and protection techniques. In the meantime, critical equipment installations are duplicated to provide an operational safety factor.

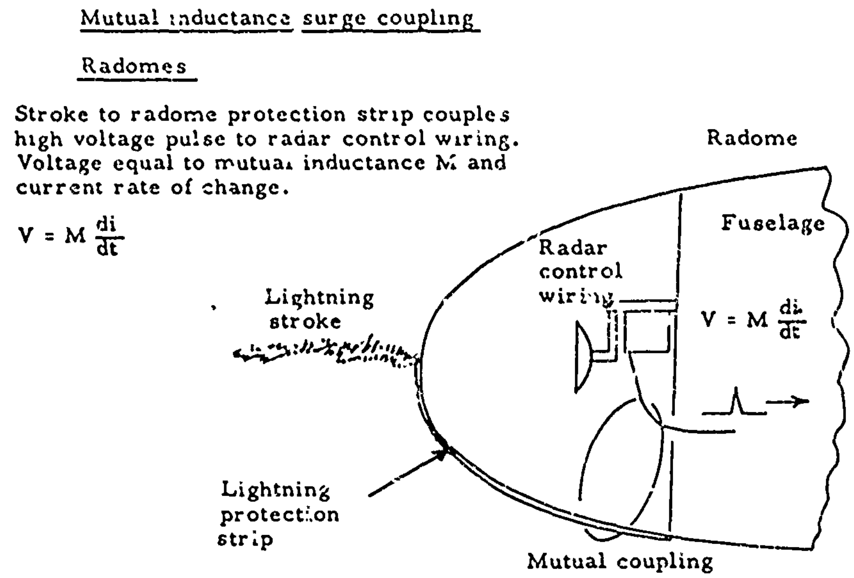


Figure 3. Lightning Discharges to Radome Protection Strips Can Couple Pulses Into Main Electrical System — Requiring Adequate Shielding or Pulse Limiting

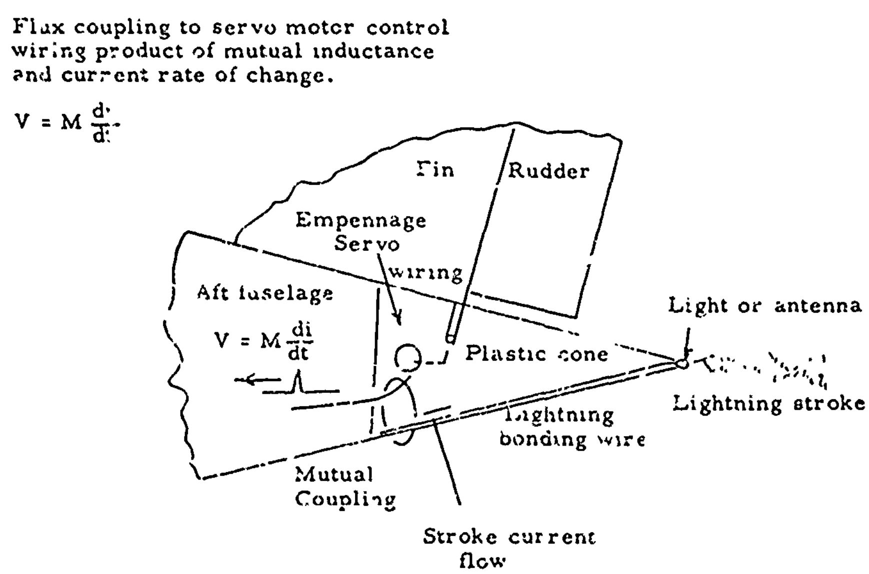


Figure 4. Strokes to Frequently Used Plastic Tail Cover Lights and Bonding Wires Can Couple Pulses Into Empennage Servo Wiring — Requiring Special Attention to Shielding

CONCLUDING DISCUSSION

Most techniques used on present commercial and military jet subsonic transports (References 1, 2, and 3) are applicable to the new SST designs but require reevaluation in relation to new materials (Reference 4). The industry cooperative program at LTRI has been particularly useful in making available the most up-to-date improvements in aircraft lightning protection during the early design stages, drawing from all the other related LTRI programs. For example, in the earlier Canadian researches at LTRI (Reference 5) reported by H. R. Shaver, full-scale experiments with an aircraft on the ground indicated that when using copper braid, the shielding shown in Figure 2 still passed enough surge potential to disable the navigation-communication equipment. In a following test, it was determined that carefully installed shielding aluminum tubing was needed to eliminate that hazard. Consequently, such improvements have been incorporated in the SST designs.

Other phases being worked on include fuel system protection studies. Fuel tank access door and filler cap lightning protection principles for preventing internal sparking were developed earlier, in coordination with the LTRI-Industry Cooperative program. Continuing tests with the LTRI artificial lightning discharge facilities on sections of actual full-scale SST production assemblies, have provided very useful and essential checks on the effectiveness of protective applications. Aside from phases of direct interest to the cooperative program sponsors, about half the grants are applied toward continuation of basic lightning studies for more accurately defining the lightning environmental conditions, particularly for multiple strokes which may be swept along the fuselage as the aircraft flies past the original lightning channel.

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AFAL-TR-68-290
PART II

# AIRLINE COOPERATIVE PROGRAM RESEARCHES WITH THE LIGHTNING & TRANSIENTS RESEARCH INSTITUTE ON AIRCRAFT LIGHTNING PROTECTION

D. S. Little*

ARINC Research Corporation

M. M. Newman

Lightning & Transients Research Institute**

Lightning strokes to aircraft radio antennas have frequently damaged unprotected radio equipment, and in some cases have started fires. Simple spark gap lightning protection has proven inadequate because long duration lightning strokes, with low current rates of rise do not build up sufficient voltage across lead or tuning inductances to flash the gaps. To provide protection against this type of stroke, Lightning and Transients Research Institute (LTRI) originally proposed a lightning arrester utilizing a combination gap and series capacitor. An experimental group of such arresters was constructed, as illustrated in Figure 1, to serve as a combination protection unit and recorder of lightning stroke current and charge. To obtain needed basic data on aircraft-intercepted lightning currents (under a Navy-sponsored program and the cooperation of one of the airlines joining in the cooperative research program) 70 of the units were installed and operated for a two-year period. Whenever the antenna was struck, the protective gap cartridge was replaced and was sent to LTRI for analysis. Natural and laboratory-reproduced lightning effects on various parts of the aircraft and on the gaps were compared, as illustrated in Figure 2. A very few cloud-to-ground strokes were found to be about 100,000 amperes; most of the intercepted discharges were probably cloud-to-cloud of much lower current magnitude (of the order of 5000 amperes) but of relatively long duration and high charge transfer, frequently as high as 200 coulombs.

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*Formerly with American Airlines, Inc.
now consultant to ARINC Research Corporation

**Lightning Research Oceanic Laboratory
Research Vessel Thunderbolt, Miami Beach, Florida

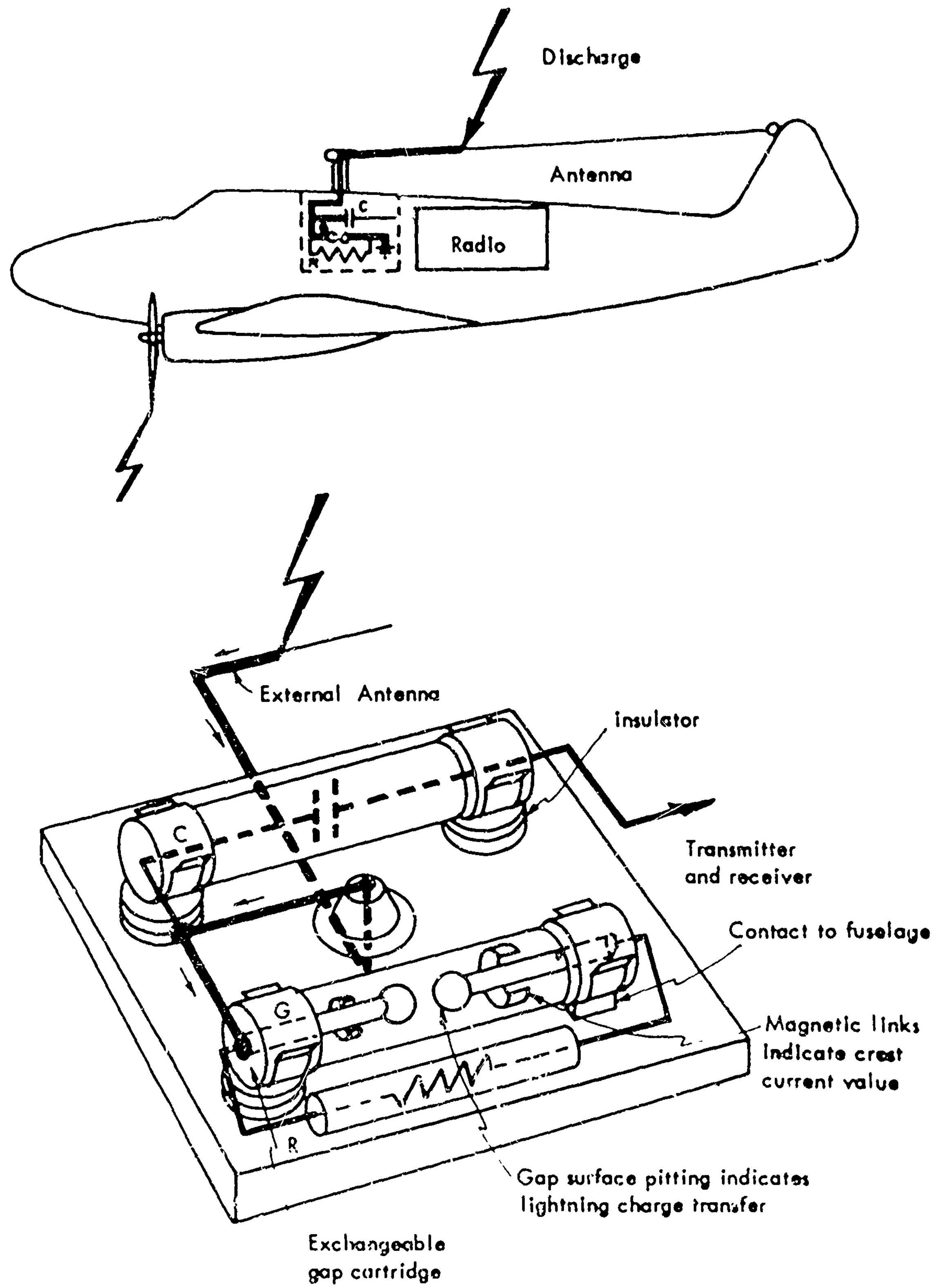


Figure 1. The Protective and Current Recording "Arrester" for Discharges to Aircraft Antenna, in Early Airline Experiments

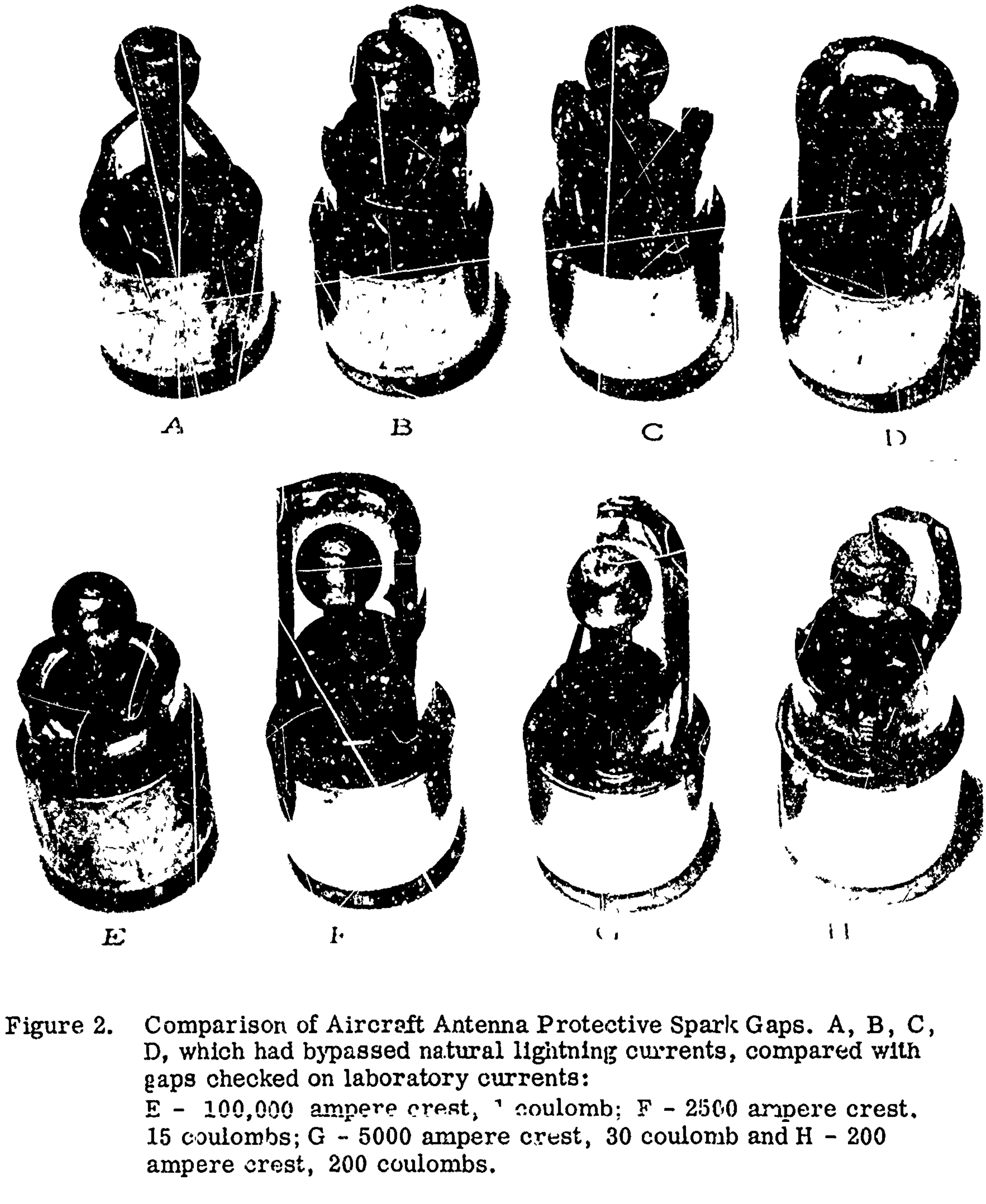


Figure 2. Comparison of Aircraft Antenna Protective Spark Gaps. A, B, C, D, which had bypassed natural lightning currents, compared with gaps checked on laboratory currents:
E - 100,000 ampere crest, 1 coulomb; F - 2500 ampere crest, 15 coulombs; G - 5000 ampere crest, 30 coulomb and H - 200 ampere crest, 200 coulombs.

The gap cartridges forwarded to LTRI for analysis had recorded the major stroke components of the lightning discharge to the aircraft. These were used to set up the LTRI artificial lightning generation facilities, developed through cooperative industry research grants to duplicate the effects of the natural lightning strokes on the arrester gaps. The waveform of the composite test stroke reproducing the mechanical disruption, heating, and pitting effects of natural lightning consisted of a 100,000 ampere wave of about 10 microseconds duration, a second component of 5000 amperes and 10 milliseconds duration, and a DC component of 200 amperes and one second duration, as shown in Figure 3. Thus, arresters must withstand these major lightning current components.

In typical early airline operations, records of 45 lightning strikes damaging the antenna resulted in 38 cases of the radio equipment being damaged. Damage to the radio equipment was completely eliminated by the installation of experimental lightning arresters in the two-year checking period. With later antenna installations using integral insulated aircraft sections, the lightning arrester became more important to flight safety by preventing structural damage as well as maintaining communications. Consequently, the LTRI arrester principles were generally applied in commercial transports, as illustrated in Figure 4; the photograph illustrates a relatively rare case of arrester gap section burning, indicating it had by-passed a relatively high charge transfer stroke, of the order of 500 coulombs.

Thus, search initially conducted through cooperation of a cooperating airline has led to the protective system now in use on most commercial transports. At the same time, airlines have cooperated in filling out questionnaires, as illustrated in Figure 5, to provide valuable related data as to stroke point distributions, types of strokes, and atmospheric data, as illustrated in Figure 6. Stroke point distributions with natural lightning were duplicated in model tests using artificial lightning to within a few percent; thus model tests are a good indicator of stroke point expectancy. However, the lightning channel is essentially stationary, but the motion of the airplane relative to the stroke channel results in a "swept-stroke" effect which might cause a multiple component lightning discharge to terminate in rather dangerous locations, as illustrated in Figures

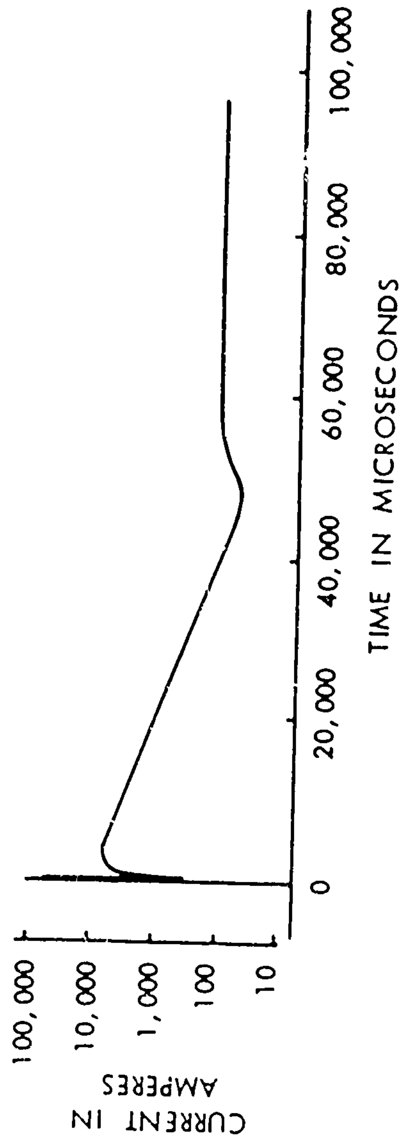


Figure 3. Waveform of Composite Artificial Lightning Discharge Components for Reproducing Natural Lightning Effects

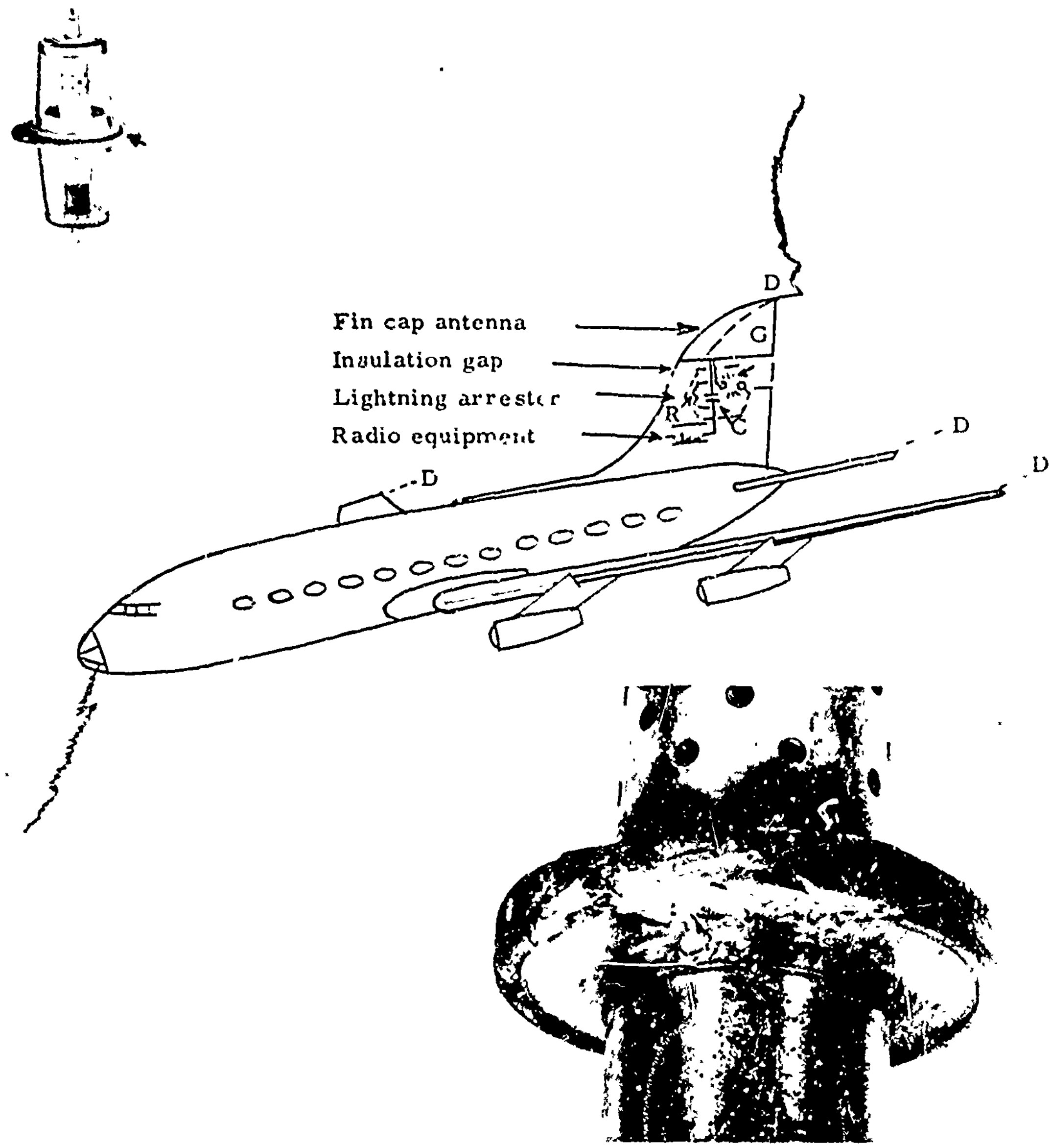


Figure 4.

Diagram illustrates arrester antenna system protection of fin cap. Upper left photo of arrester unit, and lower right shows effect on gap electrode of natural lightning discharge comparable to laboratory reproduced effect of about 500 coulombs.

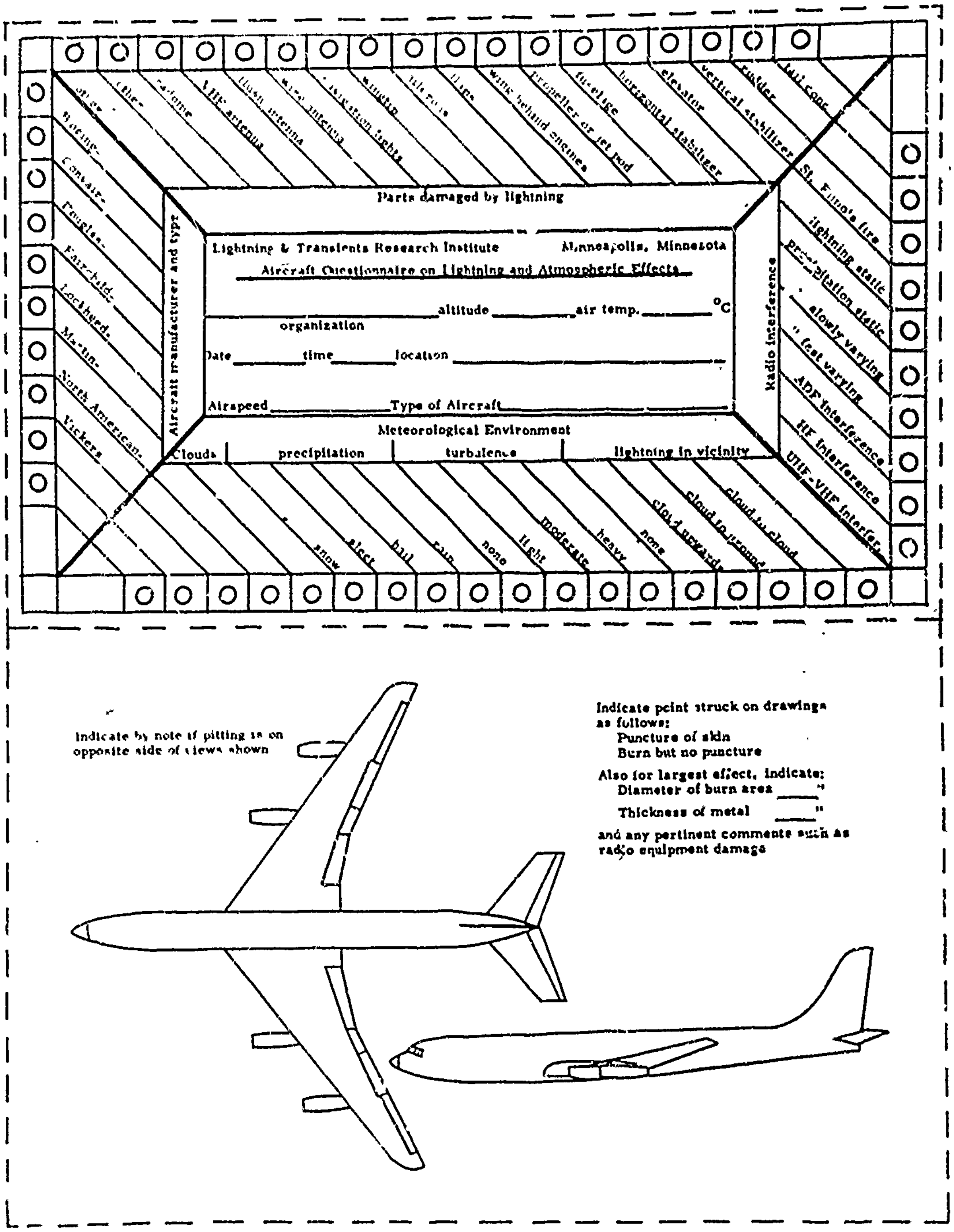


Figure 5 Lightning questionnaire modification for simplified data analysis, with individualized aircraft sketch on which strike points are to be indicated Page to be folded along centerline, sealed and returned for statistical tabulations.

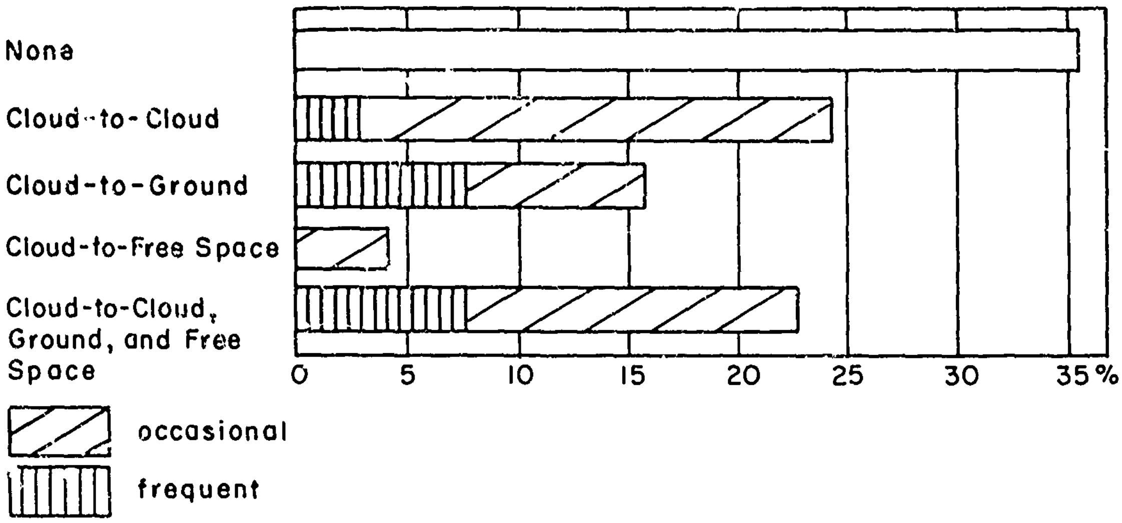


Figure 6a. Type of Lightning Observed vs. Percent of Strikes

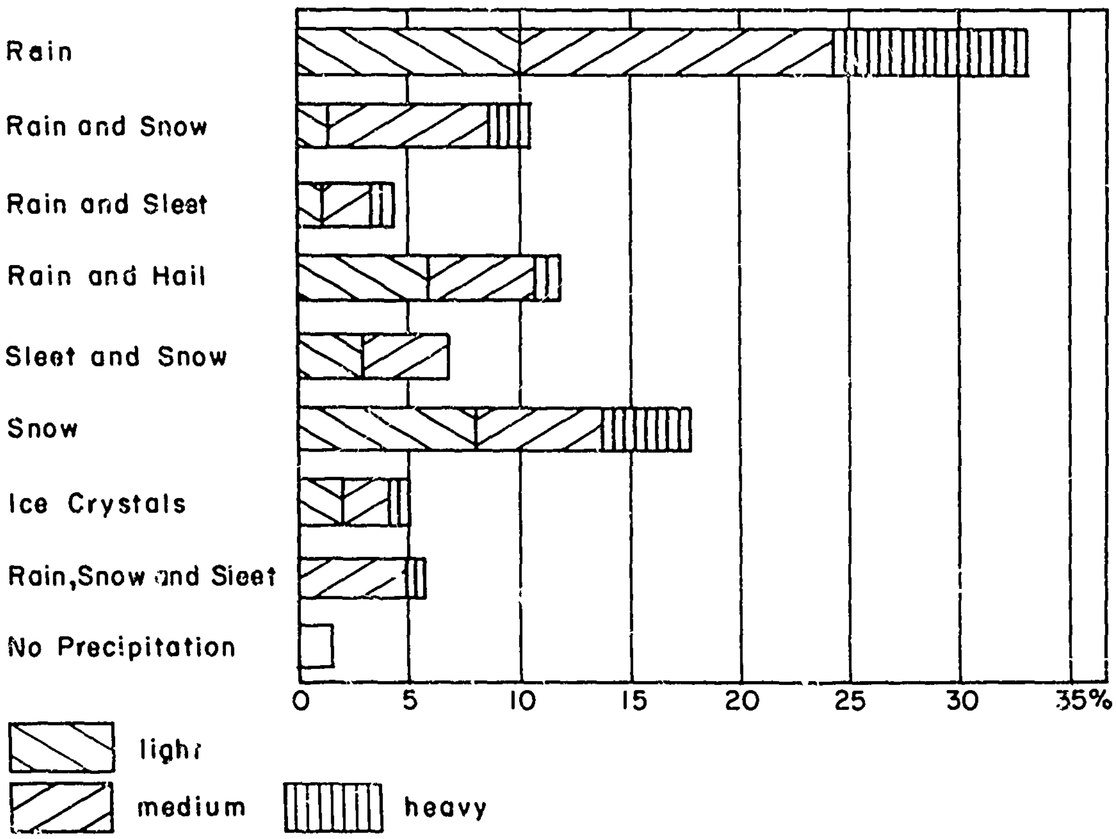


Figure 6b. Type of Precipitation vs. Percent of Strikes

Figure 6. Atmospheric Data Related to Strike Questionnaire

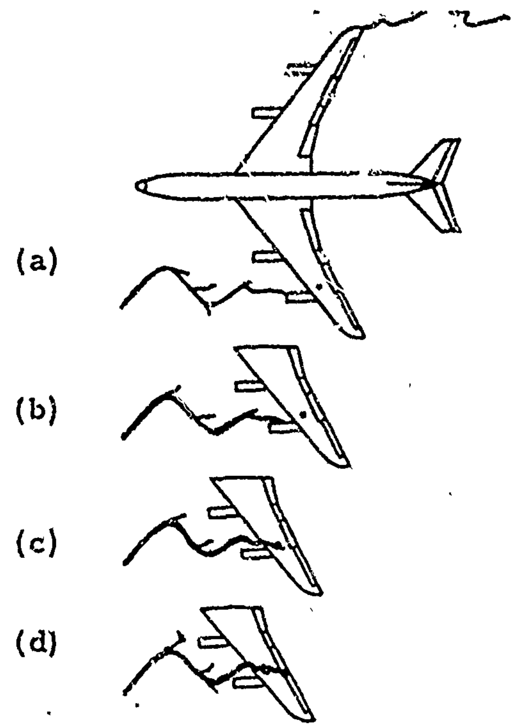


Figure 7a. Transfer of Lightning Discharge Current Contact Path as Aircraft Flies Through Lightning-Ionized Air Channel

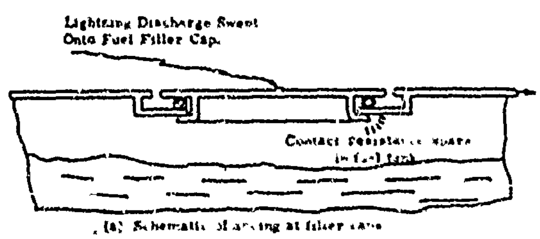


Figure 7b. Schematic of Sparking Inside the Fuel Tank, With Filler Cap Intercepting a Swept Discharge

Figure 7c. Fuel cap in KC-135 wing before direct strike.

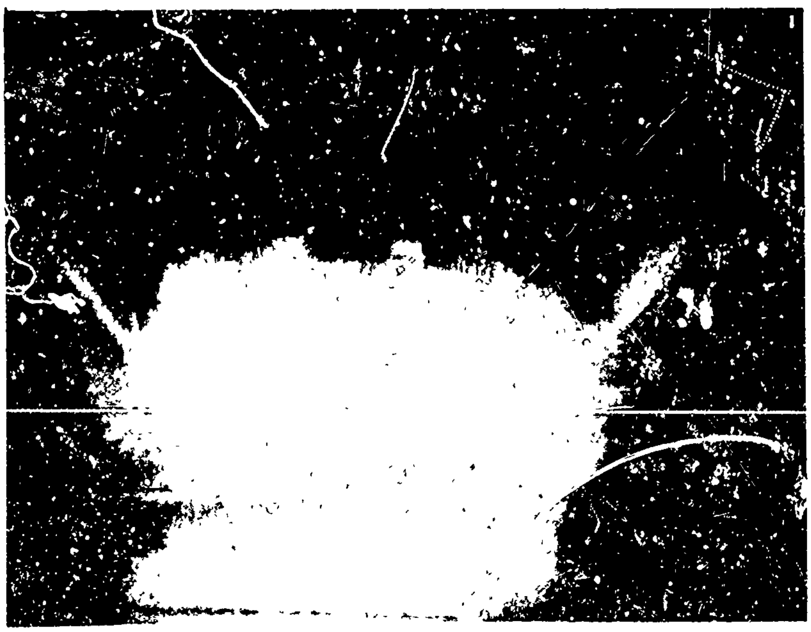

Figure 7d. Fuel cap in KC-135 wing sparking during direct strike.

7a and 7b. In other cooperative research, it was shown that serious sparking as illustrated in Figures 7c and 7d, could result inside the fuel tank. These studies were expanded to consider similar problems with access doors, under a later FAA-sponsored program, in cooperation with aircraft manufacturers and airlines. Airline research, in cooperation with FAA and Air Force-Navy sponsored programs on aircraft skin joints, has greatly helped in defining requirements and feasible improvements toward preventing sparking inside fuel systems.

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AFAL-TR-68-290
PART II

# USAF FLIGHT LIGHTNING RESEARCH

Donald R. Fitzgerald

Air Force Cambridge Research Laboratories

## AIRCRAFT STRIKE EXPERIENCE

There appears to be two major categories of electrical phenomena that may be encountered: (1) the normal or "triggered" lightning stroke associated with thunderstorm activity, and (2) an aircraft-initiated discharge of regions of electrical charge storage in snow and rain shower clouds. The snow and rain electrification problem in flight operations has been recognized for many years. Most aircraft incidents reported as "static discharges" are associated with the discharge of local charge centers in precipitating clouds. Essentially no research has been completed as yet to establish the details of cloud size, form of precipitation, turbulence structure, etc., that are required to generate charges sufficient to cause aircraft damage. Our group believes that the charge generation processes in the larger shower clouds and thunderstorms are generally similar, but the rate of formation, size, altitude, and complexity of structure of the charge centers are different.

A few examples of discharge phenomena from smaller clouds are given in Table I as an indication of the type of data available. Electrical incidents occur in the world-wide USAF operations at the rate of about 50 per year. The domestic air carriers experience about 750 incidents per year, or 1 per 2000 hours of flight. Most of these are considered to be "static discharges," however, rather than true lightning strikes.

## RESEARCH AIRCRAFT LIGHTNING TESTS

AFCRL, in cooperation with Aeronautical Systems Division, AFSC, Sandia Laboratory, and the FAA Aircraft Development Service, conducted a multiaircraft lightning research program in 1964-1966. C-130, F-100F, and U-2 aircraft were instrumented and used to study the structure of Florida thunderstorms. The LTRI vessel THUNDERBOLT was employed in an attempt to induce lightning

TABLE I

SELECTED NON-THUNDERSTORM ELECTROSTATIC INCIDENTS

| Location | Altitude | Weather | Vehicle | Damage | Reference |
|---|---|---|---|---|---|
| 60 NE of Sault Ste. Marie | Descending 3500 to 2500 MSL | Small snow showers tops visually 4000 to 4500 ft. faint precipitation return on radar | B-52H | Center of radome arcing and damage along a 2 ft. strip | Ltr from Det. 28, 26th Wea. Sqdn. Wurtsmith AFB, Michigan |
| Los Angeles Basin and just out to sea | Descent to Los Angeles International Airport | Rain showers and multiple layers to 15,000 ft. bases 1500. Faint radar echoes. Temp. near or above freezing. | Prop and jet comm | Six strikes within a few hours. Quite a bit of damage to radomes, etc. All strikes over water. | UAL Met. Circular No. 57 |
| Seattle, vicinity of Boeing Field | Take off climbing through 3700 ft. | IFR in moderate rain shower 8°C. All tops below 8,000 ft. No build ups. | Comm. prop | Very loud explosion sound, burned nose, wing tip, shattered taxi light, gnd. HF transmitter disabled and South Seattle power interruption | NACA ltr. 12 July 1955 |

strikes to the F-100F aircraft during low-altitude passes over the ship. Fifty-five strikes were obtained during in-storm passes at altitudes of from 10,000 to 30,000 ft. Transient currents, RF waveforms, electrostatic fields, precipitation, turbulence, photographs, and radar measurements were obtained. Detailed analysis of individual storms is continuing. Some of the more general statistical results are presented in this paper.

The peak currents and rise and fall times for surges recorded on high-frequency oscilloscopes in 1968 are presented in Table II. These strikes are believed representative of in-cloud or intracloud discharges and not the more extreme values in a cloud-ground channel. The 1965 data was related to power line experience and plotted in Figure 1; the scale on the left refers to the Lewis and Foust results and to the F-100F currents as shown by the two histograms. The cumulative distributions are given by the continuous curves, with the percentage values on the right. For the Lewis and Foust curve, 50 percent of the currents are below 15 kiloamp, and 10 percent exceed 40 kiloamp. The much smaller sample of flight measurements indicate 50 percent of the currents were below 2 kiloamp and 10 percent were above 7 kiloamp. The largest current measured in-flight was 22 kiloamp. A feature of the in-cloud strikes was a high repetition rate; restrikes occurred at intervals of a millisecond or two in some areas, in contrast to the 20-50 msec intervals associated with cloud-ground strokes. This feature may be important in some protective circuit design problems.

TABLE II

PEAK CURRENTS, RISE AND FALL TIMES, 1965-1966

| Sensor Location | Number of Measurements | Peak Current-Amperes | Peak Rise Time-A/$\mu$s | Peak Fall Time-A/$\mu$s |
|---|---|---|---|---|
| Nose Boom | 30 | 5,800 | 2,000 | 7,000 |
| Right Wing Tip | 18 | 14,500 | 5,000 | 2,200 |
| Left Wing Tip | 4 | 2,800 | --- | --- |
| Vertical Stabilizer | 23 | 22,000 | 2,000 | 210 |

Electrostatic fields associated with the storm and with the surface charge on the aircraft were measured with a system of induction field meters. The

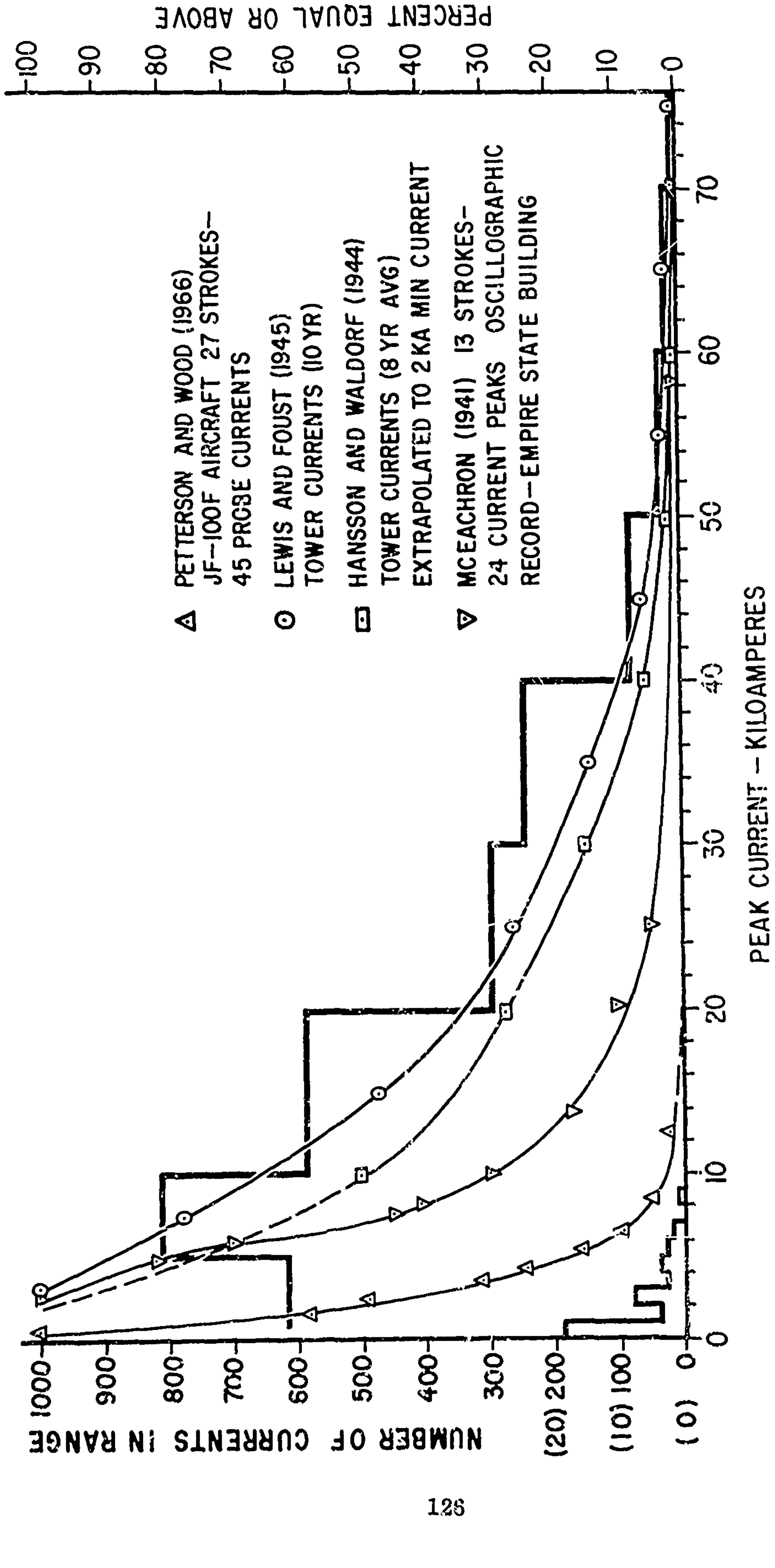


Figure 1. Lightning Current Distributions

maximum field in a storm was about 390,000 volts per meter. Sixty-nine percent of the strikes occurred when the field was less than 130,000 volts per meter. The general relationships of field strength, aircraft charge, and lightning incidents are displayed in Figure 2. The data indicates that very high fields or strong aircraft charging are not required for lightning strikes to occur.

During the test period it was noticed that different storms produced different lightning strike frequencies. An analysis of these has been published by Fitzgerald (1967). It was found that the average probability of a strike to the airplane was 0.021, based on the ratio of aircraft strikes to total strikes counted during the penetration intervals. On two exceptional days, this probability increased to 1.0 (three strikes in three trials), and to 0.5 (one strike in two trials). These storms were in an early dissipating stage of their life cycle. The observations are in agreement with a suggestion made by L. P. Harrison in 1946 that an aircraft may act to initiate streamers and a lightning discharge by suddenly augmenting the field in a localized region of the storm. The effect is believed most likely to occur shortly after the storm activity has diminished to the point where natural streamer formation is difficult. It can occur in regions of little turbulence and no distinctive pattern on a typical Air Traffic Control radar. In normal IFR flight operations in regions with thunderstorms combined with showers and cloud decks, aircraft routinely avoid the most active radar echo areas, which can easily lead to flight through a decaying storm and an isolated lightning incident to the aircraft.

## COMPARISON OF NORMAL AND THUNDERSTORM FLIGHT DISCHARGE EVENTS

Considerable statistical data has been accumulated over the years on the relation of electrical discharges to flight altitudes and precipitation conditions. For example, Figure 3 represents a recent summary of data as prepared by the Field Service Digest, Lockheed California Co., in March 1964. It indicates the usual maximum frequency near the freezing level, but a line of X's show percentages that appear to be independent of altitude or temperature. Our research flight data in storms has been used to prepare a plot of the lightning events near the aircraft as a function of flight altitude in the storm. This plot is given in Figure 4. In contrast with the long-term data, this curve indicates maximum lightning

2 AUGUST— 25 AUGUST 1965

| FIELD COMPONENT VOLTS PER CM | | NO. OF STRIKES |
|---|---|---|
| HIGH | 2600-3900 | 1 |
| MED | 1300-2590 | 9 |
| LOW | 0-1290 | 17 |

| AIRCRAFT CHARGE | NO. OF STRIKES |
|---|---|
| HIGH | 1 |
| MED | 7 |
| LOW | 18 |

**MAXIMUM FIELDS**
VOLTS PER CM

$E_Q^* = 3400$

$E_Y^* = 3900$

$E_Z^* = 3700$

**FIELD DEFINITIONS**

$E_Q^* = C_0 E_Q$

$E_Y^* = A\,E_Y$

$E_Z^* = [B_1 E_Z - D_1 E_X + C_1 E_Q]$

Figure 2. Summary Electrostatic Field Conditions for In-Storm Lightning Incidents

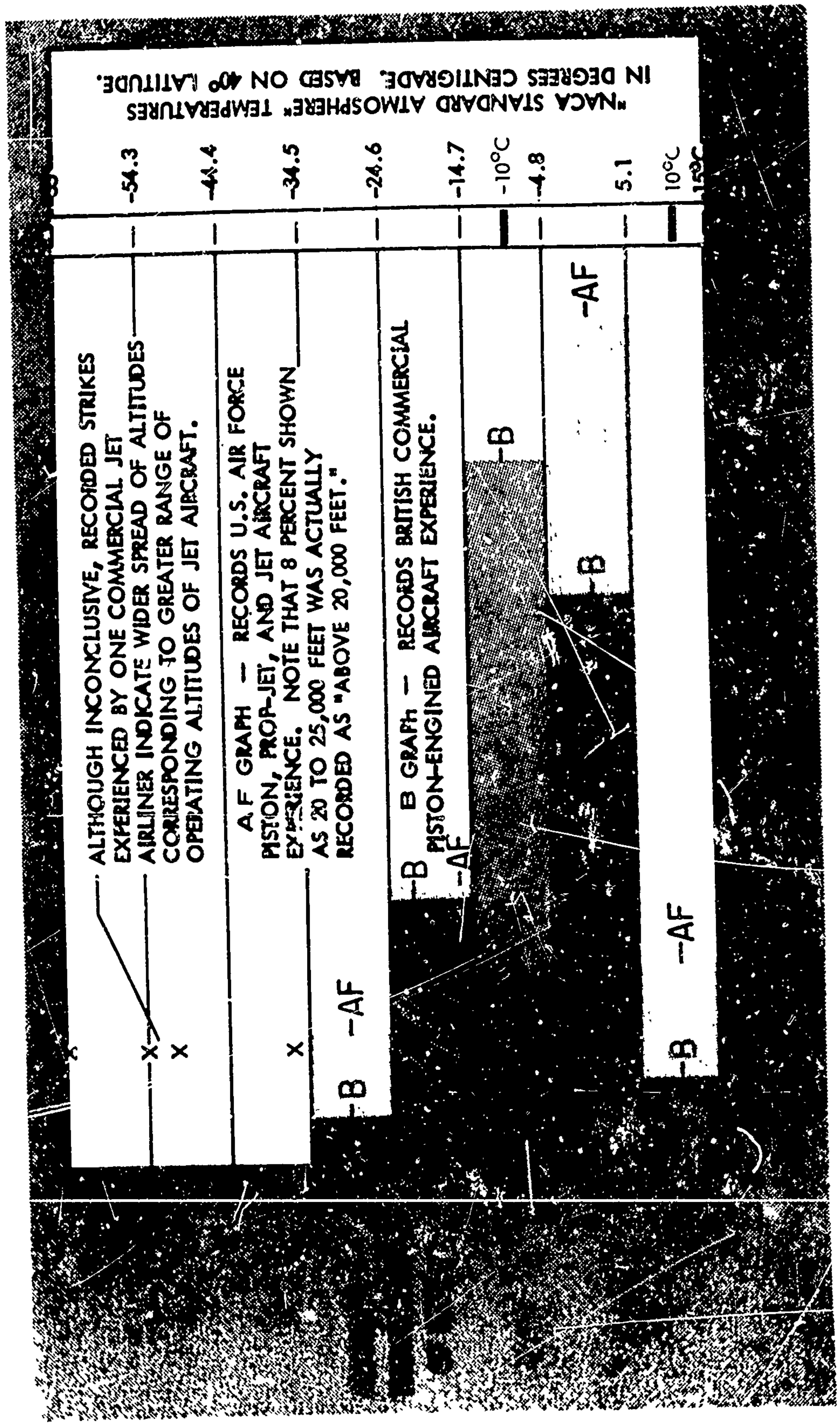


Figure 3. Normal Aircraft Lightning Statistics

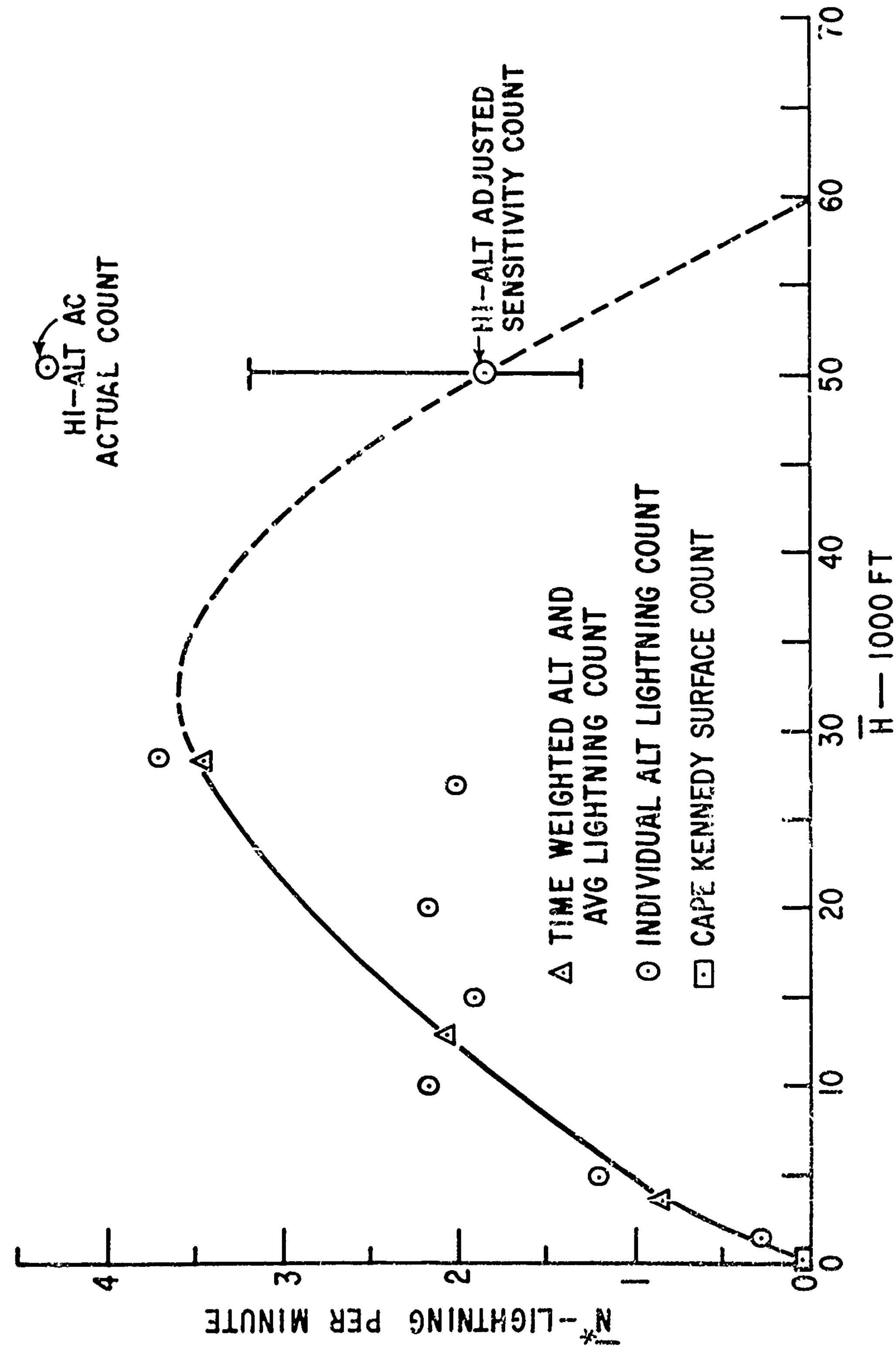


Figure 4. In-Storm Lightning Frequency Vs. Altitude

frequencies in the vicinity of 30,000 ft, or near -40 deg C. This is the region of major storm charge separation activity, with many localized pockets of positive and negative charge, so it is not surprising that considerable lightning is generated.

The different altitude distribution in the two curves is probably due to several factors. Our data is normalized for the time spent at different altitudes and only applies to flight in thunderstorms with known lightning activity present. The other data is not normalized for exposure at different altitudes and includes a much greater number of cases of discharges from non-thunderstorm clouds, which are frequently of limited height. The FAA statistics on air carrier IFR enroute altitudes were summarized for six ARTC centers in an attempt to normalize the exposure factor for altitude. The averaged assignments are shown as a histogram and as a cumulative percentage distribution in Figure 5. It is interesting to note that some 60 percent of the assignments were to altitudes below 16,000 ft. This, coupled with the fact that a number of discharge events occur to aircraft in low-altitude holding patterns near the terminals, suggests that normalization of the distribution in Figure 3 would lead to a curve more independent of height and lessen the significance of the 0 degree region as an indicator of the physical conditions required for discharge phenomena.

The dashed portion of Figure 4 is the estimated trend in lightning activity above the penetration aircraft flight altitudes used. Data was obtained above the storms with a U-2 aircraft operating near the storm tops which averaged 50,000 ft in Florida. The actual lightning count rate seemed to increase with altitude, but it is much easier to distinguish lightning field changes outside the storm than within it. To maintain the same counting efficiency for all portions of the curve, simultaneous counts were made from the F-100F and U-2 records, and the high altitude data adjusted downward as shown by the adjusted sensitivity count data point in the figure. Considerable lightning activity remains in the upper portions of the storms studied, but the strike probability at 60,000 ft for a 50,000 ft storm top is estimated to be somewhat less than at the ground. In addition, the stroke characteristics should be less severe. Estimates of this nature may be of value in considering SST operations in the vicinity of thunderstorms.

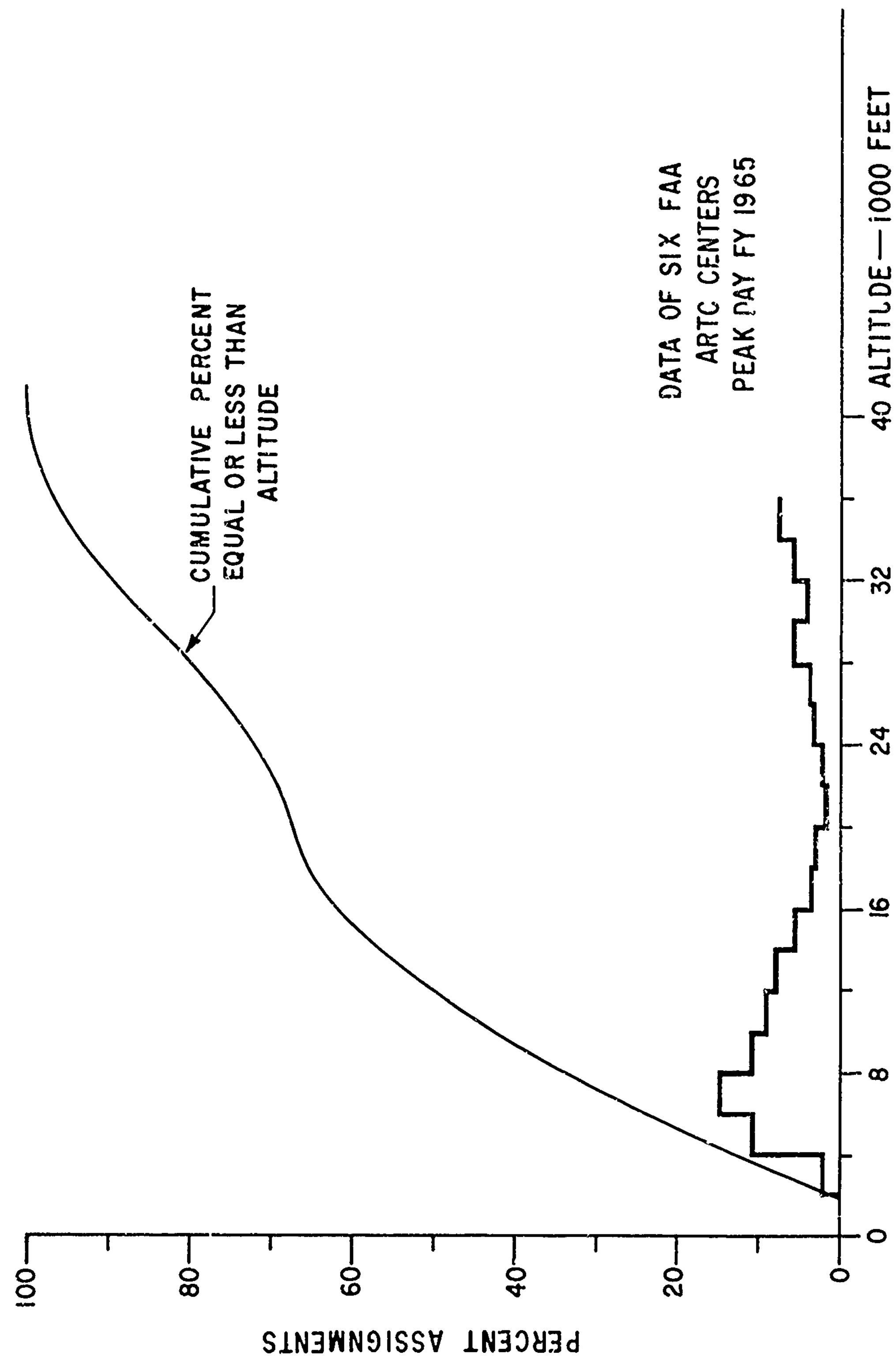


Figure 5. Air Carrier Averaged Enroute IFR Altitude Assignments

## CONCLUSIONS

Two categories of electrical phenomena may be encountered in IFR flight. The first is a normal or "triggered" lightning stroke from an active or dissipating thunderstorm. The second is a discharge from an electrical center in rain or snow shower clouds. These centers may be found in "cells" embedded in generally stratiform-appearing clouds, as well as in distinct convective showers. World-wide USAF operations encounter such electrical incidents at the rate of about 50 per year. Domestic air carriers experience about 750 incidents per year, or about 1 per 2000 hours of flight. Long term statistical experience indicates most of these events are encountered near 0°C in flights in mixed snow and rain conditions. In active thunderstorms, maximum lightning activity occurs near 30,000 ft at about -40°C. The differences are found to be related to the duration of lower altitude holding patterns in bad terminal weather, frequency of IFR altitude assignments, and attempt to avoid obvious thunderstorm areas in normal operations.

In a sample of 55 measured strikes to an F-100F aircraft, the peak lightning current was 22,000 Å, the peak rise time 5000 Å/μsec, and the peak fall time 7000 Å/μsec. Electrostatic field components of up to 390,000 V/m were measured in the storm. The data is compared with long term surface lightning stroke measurements to indicate ranges of values that may be encountered. The suggestion that aircraft entering a dissipating thunderstorm may "trigger" a lightning discharge was analyzed and is considered probable.

## ACKNOWLEDGEMENTS

The research flight program involved the enthusiastic cooperation of many individuals from AFCRL, Aeronautical Systems Division, Sandia Laboratory, AFFTC Edwards AFB, The Federal Aviation Administration, and the Lightning Transients Research Institute. Principal contributions were made by Drs. R. Cunningham and G. Kinzer, B. Petterson and R. Wood, Capt. E. Miller, and the F-100 fighter test pilots, Majors Arquilla, Hambleton, and Vandenheuvel. The tests were conducted under Office of Aerospace Research Project 8620 with financial assistance to Sandia Laboratory and LTRI by the Federal Aviation Administration.

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# TRIGGERED NATURAL LIGHTNING NEAR AN F-100 AIRCRAFT

James R. Stahmann

Lightning & Transients Research Institute

For Natural Lightning Triggering required for this study, the Research Vessel THUNDERBOLT, Figure 1, was equipped with line-throwing rockets, which were fired toward charged clouds while they remained electrically connected to the ship by a thin trailing stainless steel wire. The rockets were fired when electric field measurements showed a gradient high enough so that a stroke to the rocket was about a 50% possibility. A specially instrumented F-100 aircraft flew over the ship at the time the rocket was fired to intercept any strokes triggered to the ship. Data concerning the lightning current and associated pressure waves, streamers, and electric fields were collected on both the ship and the aircraft. It was also hoped that data concerning the probability of aircraft contacting nearby strokes would be obtained.

The basic instrumentation and setup is shown in Figure 2. The triggering wire was vaporized, but the ionization was maintained by the lightning discharge and the current continued to flow as in a natural stroke. The wire was of 8-mil-diameter stainless steel and was pulled off the end of a 5000 foot reel as in a spinning reel. The maximum rocket altitude was about 1200 feet, so that a spool could be used several times. Pitting or melting of parts of the reel gave an indication of the charge transferred in the stroke. The lightning current was recorded oscillographically using a high current shunt. The pressure waves at various distances from the lightning channel were measured with blast wave transducers, and the lightning waveform spectrum could be analyzed using a multifrequency receiver and recorder. The lightning was forced to jump across a gap placed between the rocket platform and a lower grounded platform to fix the channel location. To measure the extent of plasma ignition of explosive fuel vent mixtures, propane-air jets were placed at various distances from the stroke channel. The electric field was monitored by corona field meters. Polaroid and other cameras were used to record the strokes from various angles, and a Fastax high-speed motion picture camera was used to photograph the stroke

Figure 1. LTRI Mobile Lightning Generator Facility (A 10-million-volt generator is lowered at stern during storm to launch a rocket trailing a thin 8-mil diameter wire toward charged clouds for natural lightning triggering.)

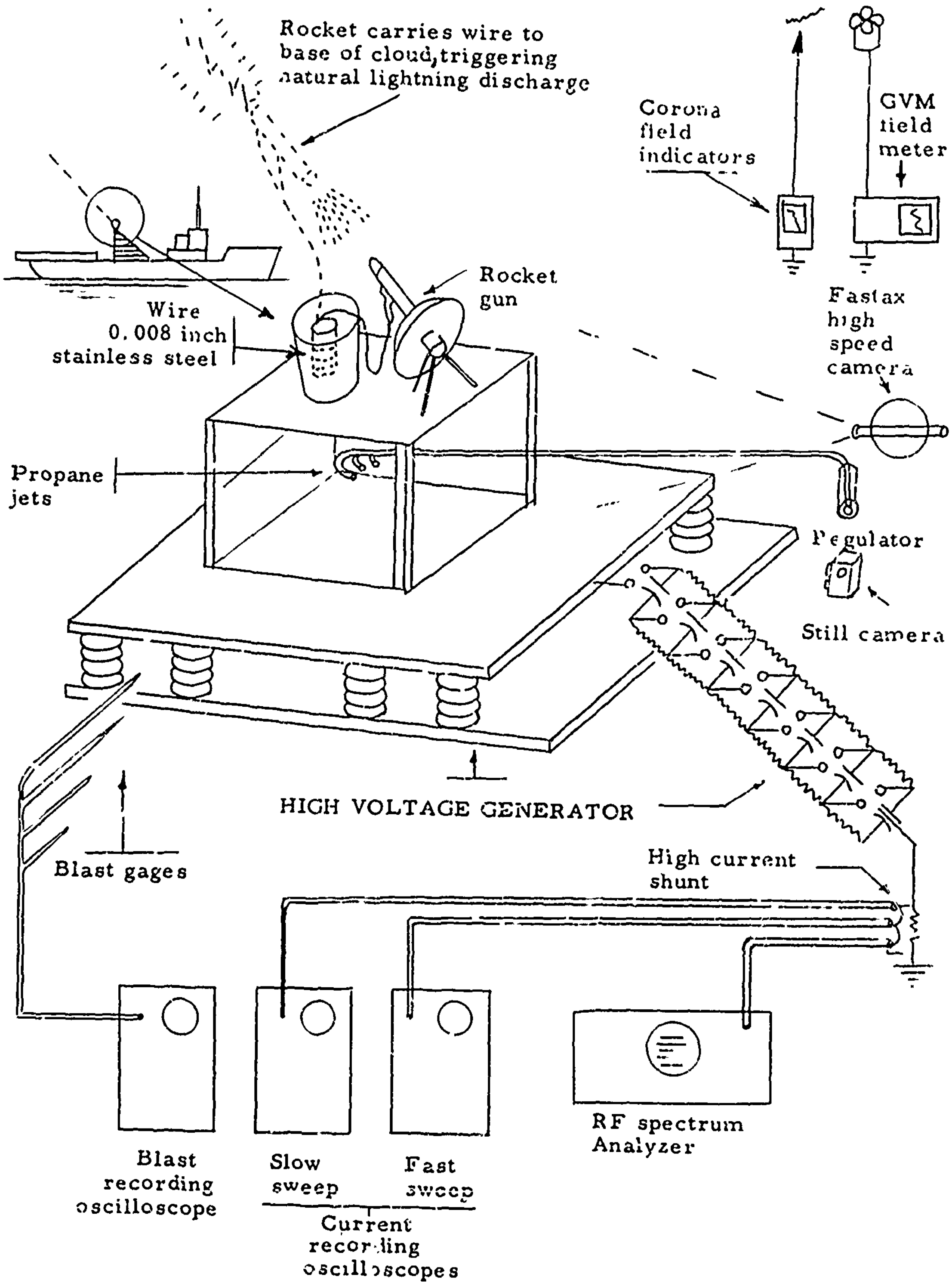



Figure 2. Measurement Equipment for Studying Natural Lightning Discharge Triggered From Ship; (Generator area shown at upper left)

plasma on the launch platform at a rate of about 1000 frames per second. A high voltage generator could be placed in series with the stroke and used to pulse the natural channel in a controlled fashion to measure channel properties such as length, branching, characteristic impedance, etc. However, to simplify this operation the generator was grounded.

A rocket launch is shown in Figure 3, and the F-100 flying over the rocket is shown in Figure 4. As shown in Figure 5, the stroke may contact the rocket any time its altitude exceeds about 500 feet. The actual time of the stroke and its direction could not be known in advance. On two occasions, a triggered stroke missed the F-100 an estimated distance of less than 1000 feet. The F-100 overpass was timed so that the aircraft was directly over the ship when the rocket reached maximum altitude.

A Coast Guard Loran receiver was used to obtain the ship position, which was translated to bearing and distance from the TACAN station at Patrick Air Force Base, and radioed to the F-100. Most of this program was carried out along the coast of Florida near Port Canaveral and Cape Kennedy.

## LIGHTNING PROTECTION FOR THE F-100

Evaluation of the lightning protection of the F-100 revealed two primary hazardous areas:

1. Induced streamering inside the canopy and possible canopy puncturing, as shown in Figures 6 and 7, which expose the pilot and observer to electric shock.

2. A direct stroke or streamering in the vent tube region on the vertical stabilizer may ignite the fuel-air efflux mixture, with possible flame propagation into the vent tube system.

The F-100 used for this program was specially protected by an aluminum strip placed on the surface of the canopy to prevent streamers or canopy puncture, and a nitrogen inerting system was used to prevent efflux ignition. A possible alternative consisting of a diverter to keep the stroke away from the vent is shown in Figure 8.

Figure 3. Rocket Leaving Launch Platform

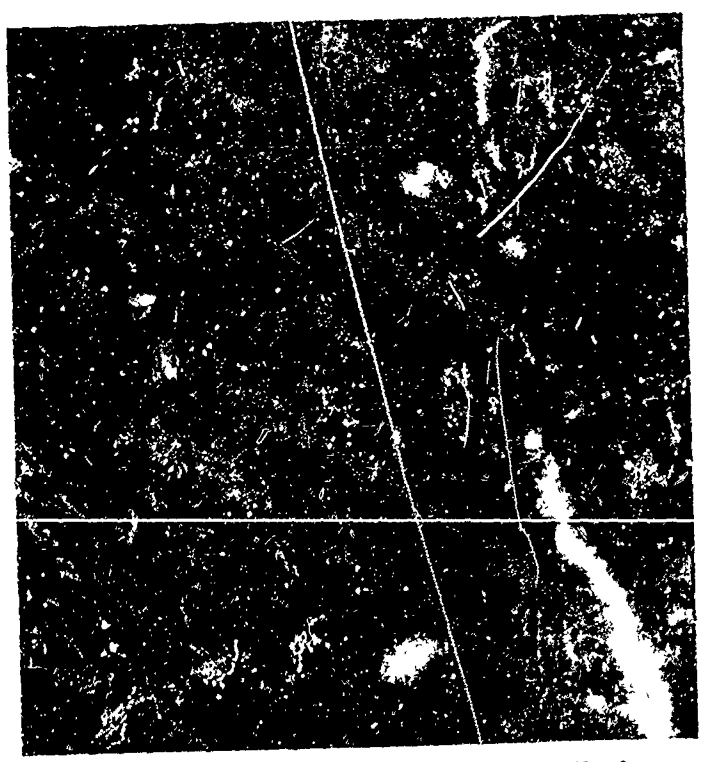


Figure 4. F-100 Flying Over Rocket Path

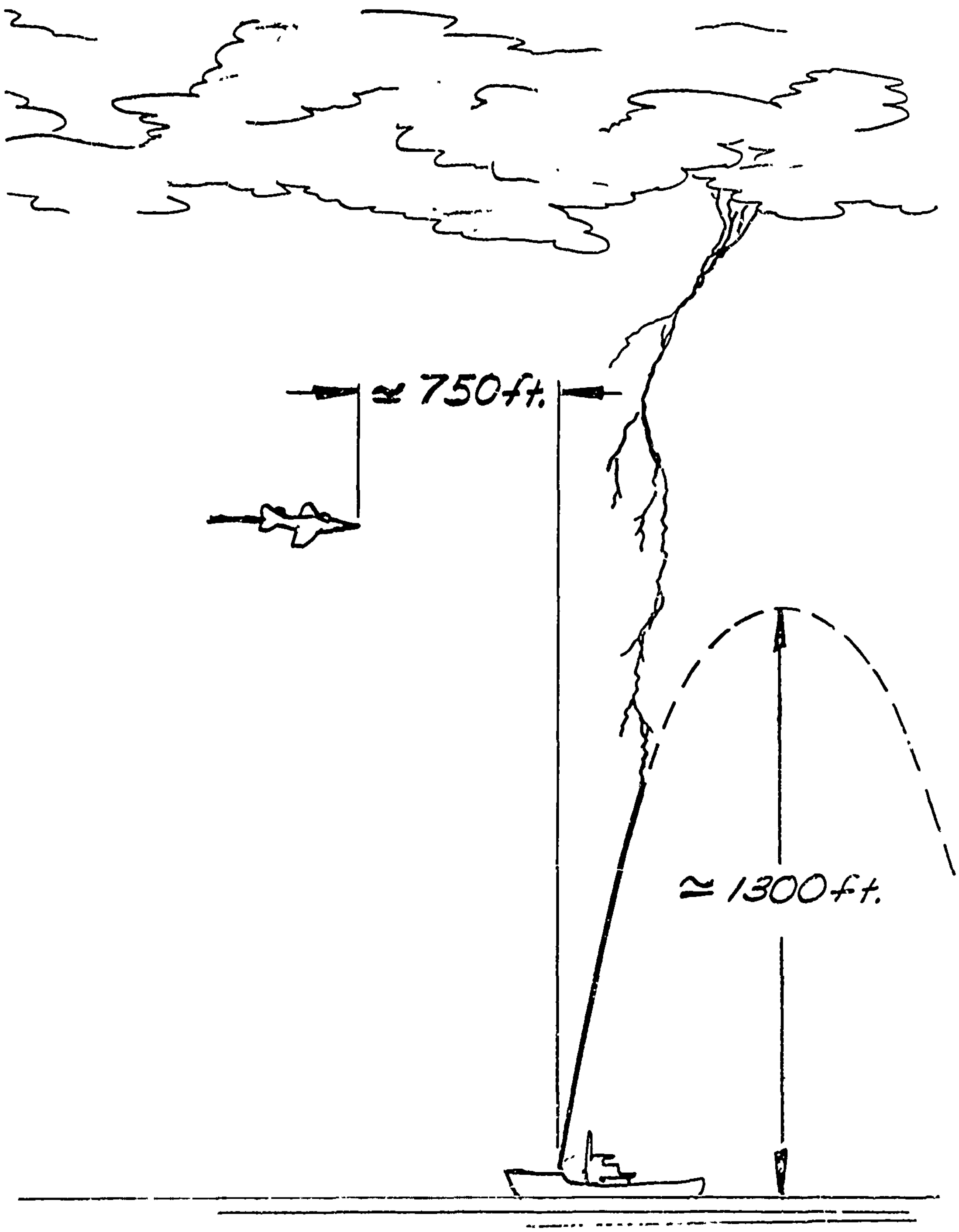



Figure 5. Triggered Lightning Stroke Within 1000 Feet of F-100 Aircraft. (Stroke did not contact the F-100, but large field changes were recorded on aircraft instrumentation.)

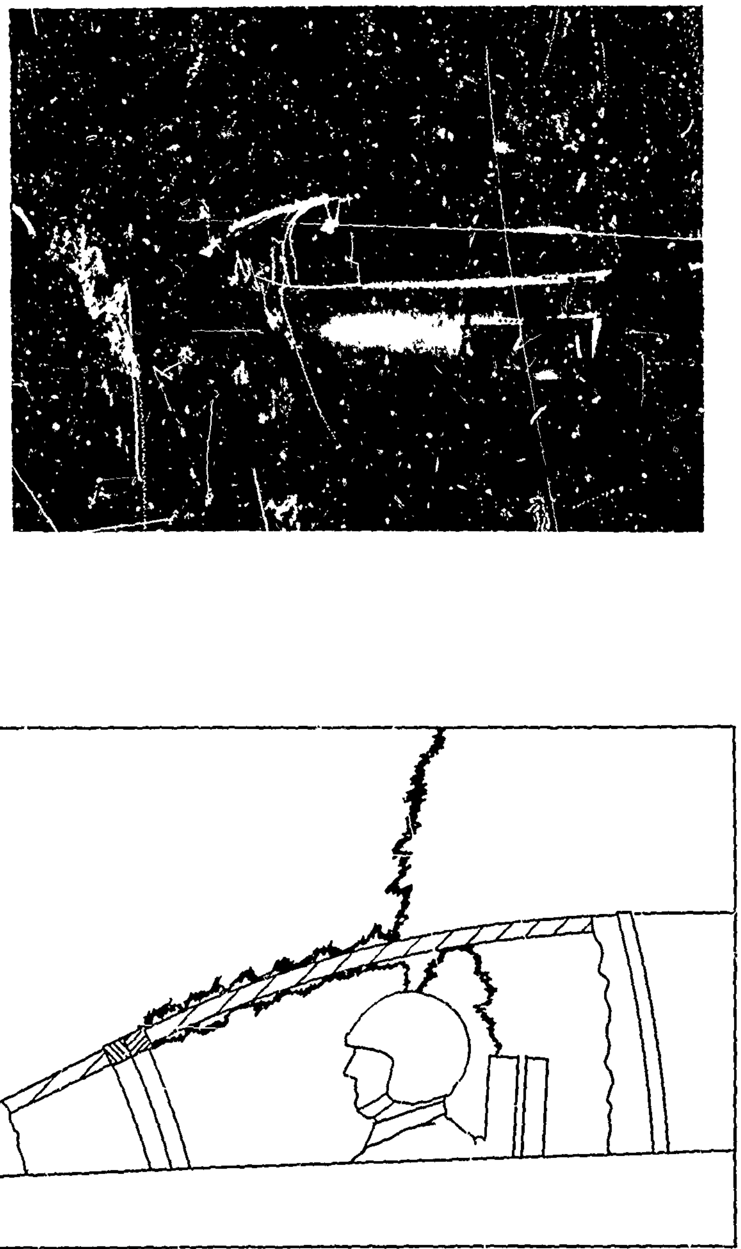
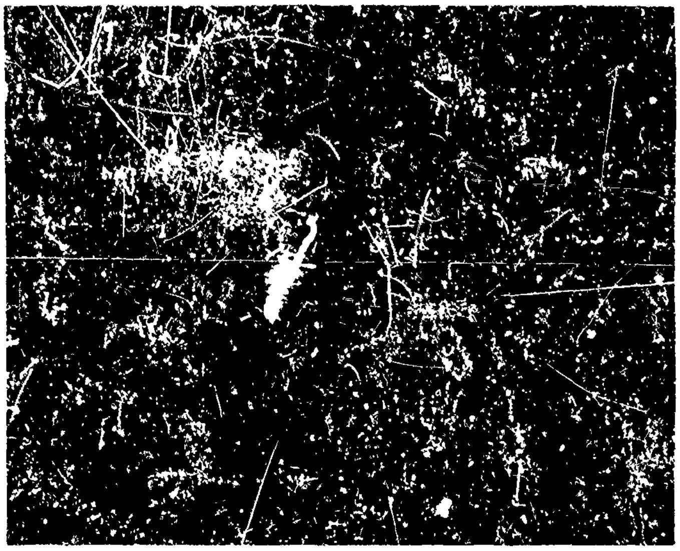


Figure 6. Streamering Off Pilot's Head May Be Induced by a Lightning Stroke



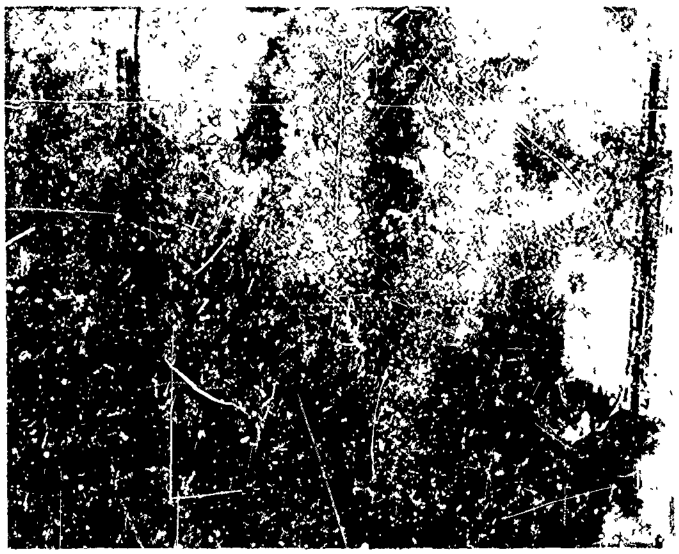


Figure 7. Canopy Puncture Due to a Simulated Lightning Stroke

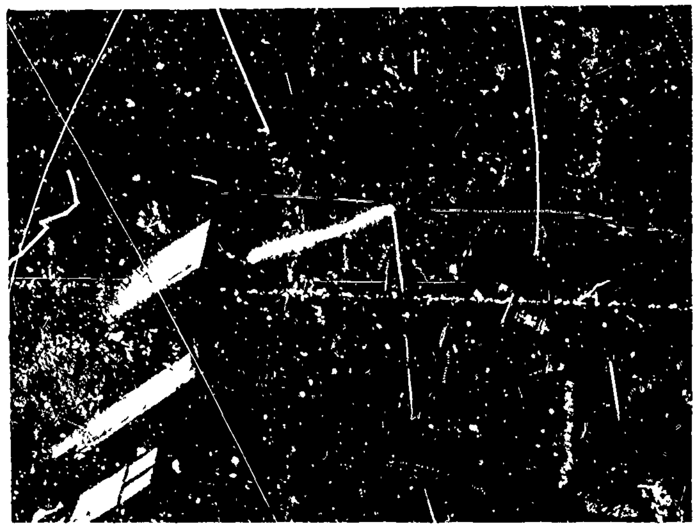


Figure 8. Demonstration of a Diverter on the F-100 Vertical Stabilizer as a Possible Fuel Vent Protective Measure

Final proof demonstration of the adequacy of the protective measures on the actual full-scale aircraft is shown in Figure 9 under simulated lightning conditions.

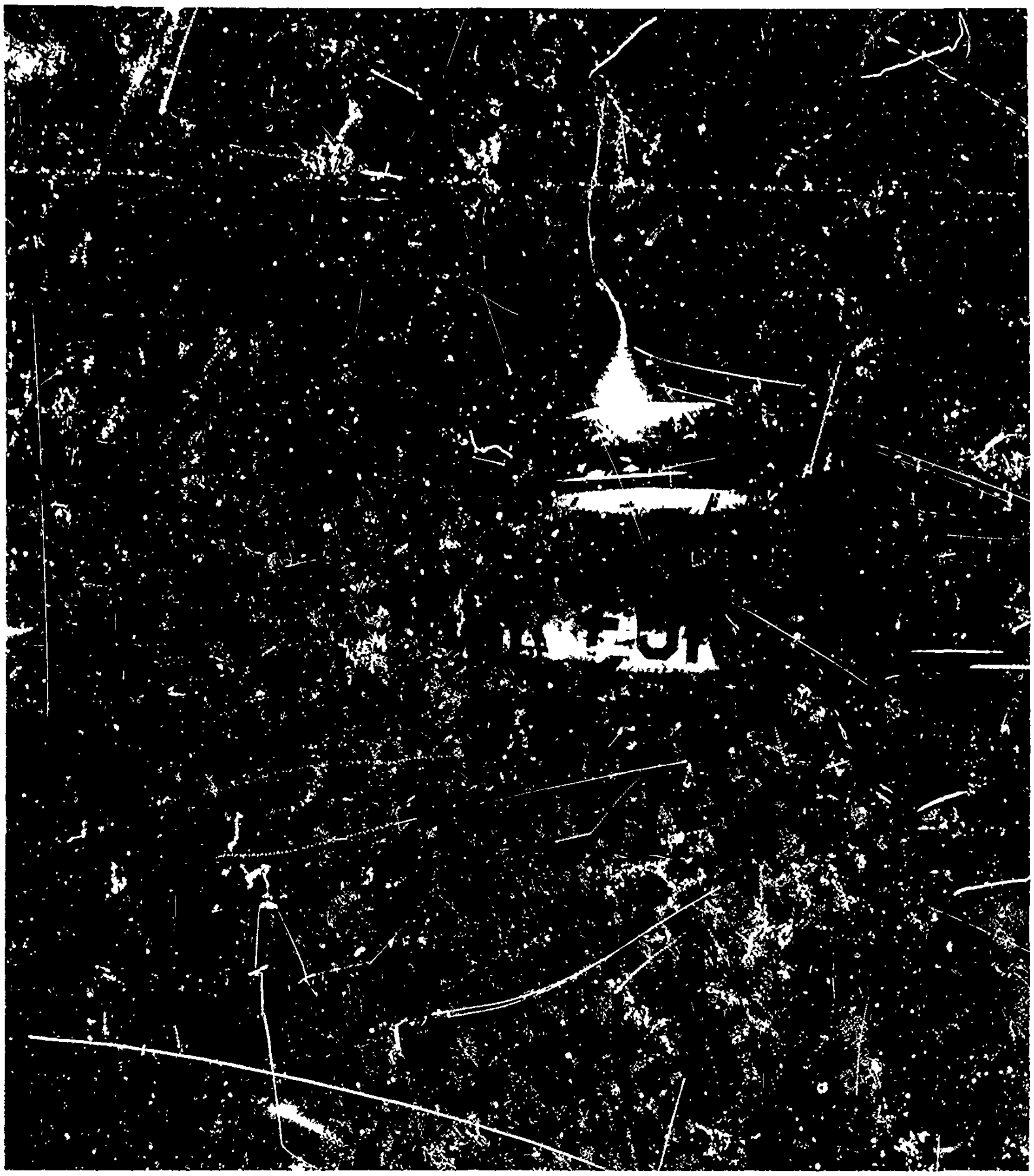


Figure 9. Final Demonstration of the Adequacy of Canopy Protective Strip, Full-Scale With an Observer in the Cockpit

AFAL-TR-68-290
PART II

# TRANSIENT PENETRATION EFFECTS ON AEROSPACE VEHICLE ELECTRONICS AND FUEL SYSTEMS

J. D. Robb

Lightning & Transients Research Institute

Electrical transients from natural lightning discharges to aircraft date back to the first use of aircraft radio equipment. The trailing wire antennas used for long range overseas flights were particularly susceptible to lightning strikes and damage and, to a lessor degree, were the HF long wire antennas. LTRI records show occasional reports where nearly all the radio equipment and much of the electrical equipment on aircraft were damaged or disabled. With the greater utilization of modern aircraft, the greater exposure through use of nonshielding plastic external skin sections, the greater vulnerability of semiconductor equipment, and the greater importance attached to the functions of electronic equipment such as completely automatic approach couplers, controlling transient penetration becomes of primary importance. A brief review is presented of the principal mechanisms by which transient surges from lightning discharges penetrate present aircraft. Also illustrated are the newer problems associated with nonmetallic skins.

## SURGE PENETRATION MECHANISMS FOR VEHICLES WITH METAL SKINS

Lightning protection for earlier propeller-driven aircraft was initiated with the development of lightning arresters for aircraft HF (high frequency) radio systems. Because of the high voltage insulation required for the antenna lead-ins, lightning strikes to the long wire antennas were channeled into the aircraft and severely damaged electrical equipment. In some cases, nearly all of the electronics equipment and much of the electrical system was lost, leaving the aircraft essentially with just flight and engine controls. With the introduction of the capacitor-spark gap type lightning arrester in which the lightning energy was blocked from the radio equipment by a series capacitor and shunted to the air frame by a spark gap, the record has been fairly good. Only where improper HF lightning arrester installations have been made has serious structural damage been produced. However, the arrester does not prevent the entrance of front-of-wave-voltage spikes into the radio equipment, and some provision must be made for these transients.

Until very recently, little lightning protection has been available for UHF-VHF antenna systems. Because of the much lower insulation levels required for the VHF antennas, less lightning energy can penetrate past the antenna into the aircraft's interior. However, the antennas are often damaged; protection techniques are available to prevent this damage. UHF and VHF grounded type antennas are available which are inherently lightning resistant and also offer a degree of protection to the associated radio equipment. Therefore, there is no reason to use the older voltage-fed or high-impedance type antennas that channel the lightning stroke energy into the aircraft. Schematic diagrams of HF and VHF antennas of both types are illustrated in Figure 1.

The use of HF shunt-fed antennas is highly desirable from a lightning protection standpoint, since the problem is thus reduced from protection against the total lightning stroke current to the much less hazardous one of inductive surges across the antenna inductance. These short high-voltage low-energy surges in the communication equipment can result in power follow arcs or semiconductor damage. Thus, lightning arresters and shunt-fed HF antennas prevent most of the lightning energy from entering the aircraft, but not these high-voltage low-energy transients.

One of the best ways of handling the high voltage pulses passed into the HF couplers past the lightning arrester is to provide the coupler system with components to handle these pulses; the necessary components can best be determined by artificial lightning tests with the equipment operating. LTRI strongly recommends tests of HF systems, since tests in the long run are much more economical than retrofits.

Another important category of components susceptible to surge penetration includes the navigation and anticollision lights. Many of the small older type lights were not too susceptible to lightning damage, but serious incidents of lightning strikes of newer type lights have included explosions of the light wiring and extensive structural damage in the vicinity of the light, as illustrated in Figure 2. Protection for lights can be provided by use of grounded metal

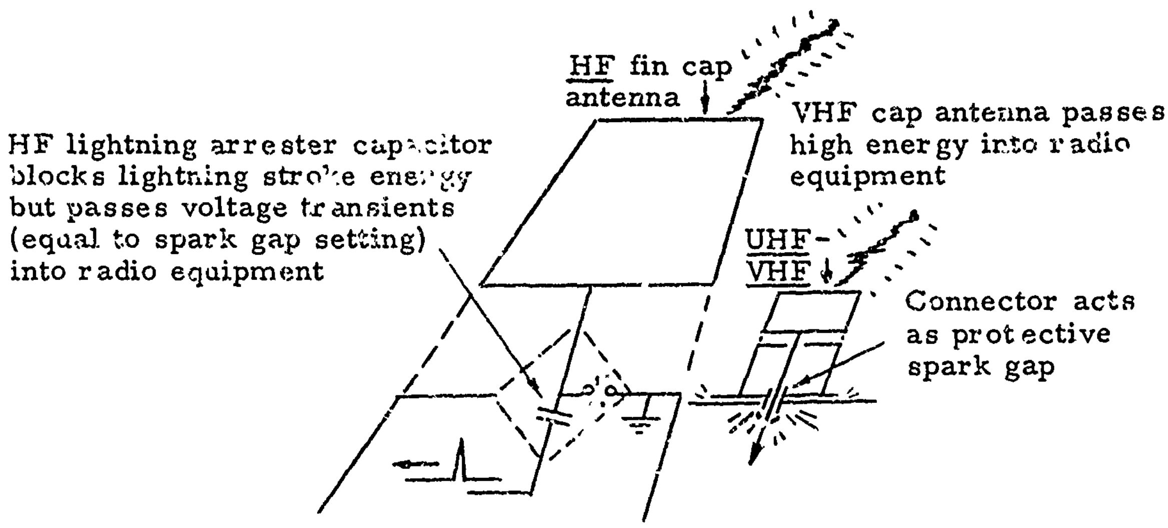



Figure 1a. Voltage-Fed High-Impedance Cap-Type Antennas

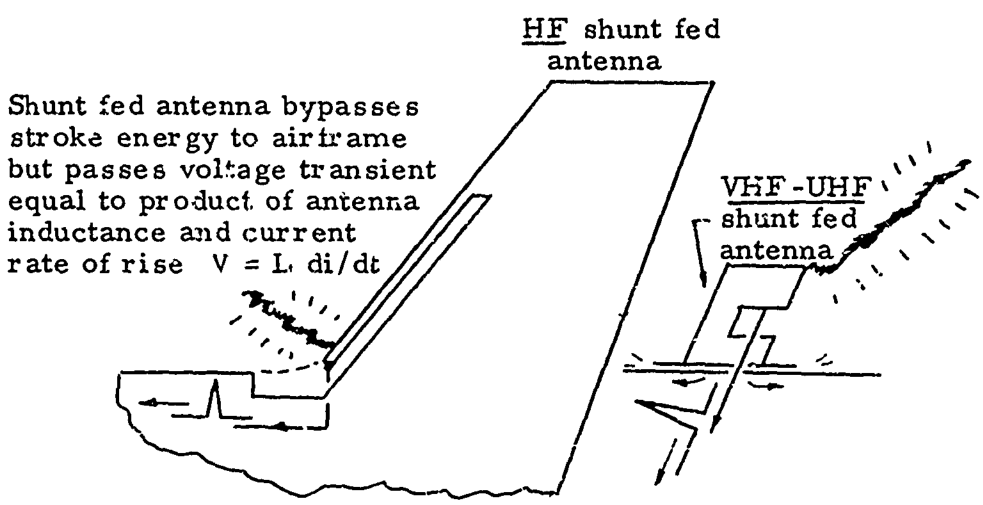



Figure 1b. Shunt-Fed HF and VHF Antennas

straps over the lights and, in some cases, by simply relocating the retainer screw for the glass lens or cover. The simplest procedure is to design the lights on the basis of artificial lightning discharge tests to pass the stroke currents without producing explosive damage or excessive transient penetration into the system wiring. The problems may quickly be recognized by test.

Another area of concern is transient penetration on wiring connected to access doors in fuel tanks. Lightning strikes swept onto an access door can produce impedance drop voltages across the door boundaries, which are directly coupled into wiring connected to components mounted on the inside of the door, as illustrated in Figure 3. Transient voltages of several hundred volts may be acceptable on light wiring, but voltages of this magnitude on fuel tank wiring may present a hazard; therefore, all covers on fuel tanks which have connected wiring (as, for example, covers over fuel quantity probes or transfer pumps) must be suitably bonded to prevent voltage coupling into this wiring and possible sparking. The use of simple bonding jumpers as in the past is totally inadequate for this purpose; even lightning discharges to other parts of the vehicle may produce sparking on access doors which are improperly bonded because of inductive drop potential effects along the direction of current flow. Thus, complete low impedance peripheral bonding is required for access door covers to ensure that the voltages coupled into the internal wiring will be below a possibly hazardous level.

## SURGE PENETRATION OF NONMETALLIC SKINS

Changes taking place in aircraft structures through use of fiber glass and composite structures such as boron-epoxy and graphite fibers introduces a serious potential hazard to aerospace vehicles through reducing the electromagnetic shielding. From a shielding point of view, this change is nearly as great as the change from wooden aircraft to metal. These materials are not only explosive through resistive heating of the conducting fibers, which produces gas vapors and extensive structural damage, but they greatly reduce shielding of electronic and fuel system components within the aircraft. Thus, the change from metal to fiber glass or composite structures must be accompanied by a careful evaluation of the possible hazards introduced and suitable protection.

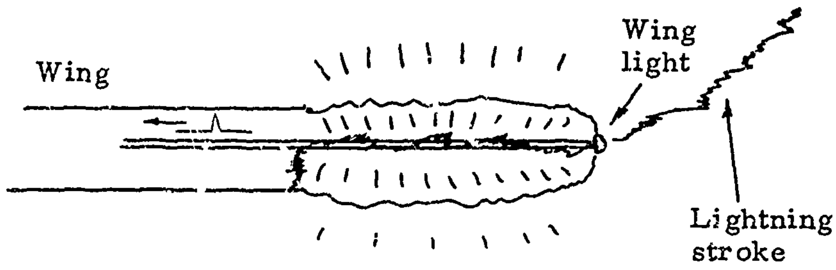



Figure 2. Navigation Lights or Rotating Beacons Which Channel Stroke Into Aircraft Interior

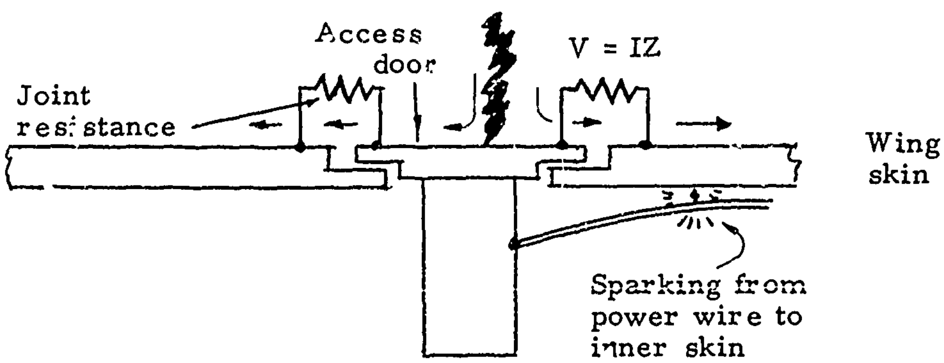



Figure 3. Lightning Stroke Swept Onto Fuel Tank Access Door (Hazardous Voltages on Wiring of Internal Components if not Adequately Bonded)

The degree of shielding provided by the older all-metal aircraft is indicated by the measurements LTRI personnel routinely made inside mock-up sections of aircraft or missiles while firing 200,000 ampere currents to the outside skin. The external fields are huge, but the internal fields are so small that sensitive instruments are required for even measuring them. Thus, skins providing a nearly perfect Faraday electromagnetic shield are rapidly being replaced by little or no electromagnetic shielding capability.

In the natural thunderstorm environment, an aircraft is subject to almost continuous, intense electromagnetic fields due to charge redistributions (lightning strokes) in adjacent areas. These electromagnetic fields occur every time the aircraft is in the general area of a thunderstorm, whereas lightning strikes of the average airliner occur only once or twice per year. Thus, exposure to electromagnetic fields is greater by many orders of magnitude. These electromagnetic fields have almost no effect on internal electrical and electronic systems in metal-skin aircraft, but may affect the aircraft interior in varying degrees with nonmetallic skins. The electric and the magnetic field components produce entirely different effects on circuitry.

Plastic skin panels can result in induced streamering from thunderstorm <u>electric fields</u> of 10 to 100 amperes on electrical components inside the aircraft, even though the skin is not punctured, as illustrated in Figure 4. These currents are more than sufficient to damage semiconductor components, introduce errors in computer systems, and ignite flammable fuel vapors. Thus, a vehicle with extensive sections of external plastic or composite skin might be described as an electrical wiring system and open fuel tanks exposed to thunderstorm cross fields which produce continuous visible streamering and sparking. The hazard that this presents to aircraft electronic and fuel systems hardly needs to be pointed out. It should be noted that external electrical activity in the form of corona streamering from the aircraft nose and windshield (for example) is rather common to pilots flying regularly in a thunderstorm environment.

A second type of coupling phenomena for electrical circuitry enclosed in nonmetallic skins is due to magnetic fields from lightning discharges to or near the vehicle. The skin effect causes the high-frequency high-current components of the lightning discharge to flow almost exclusively in the outer surface of the

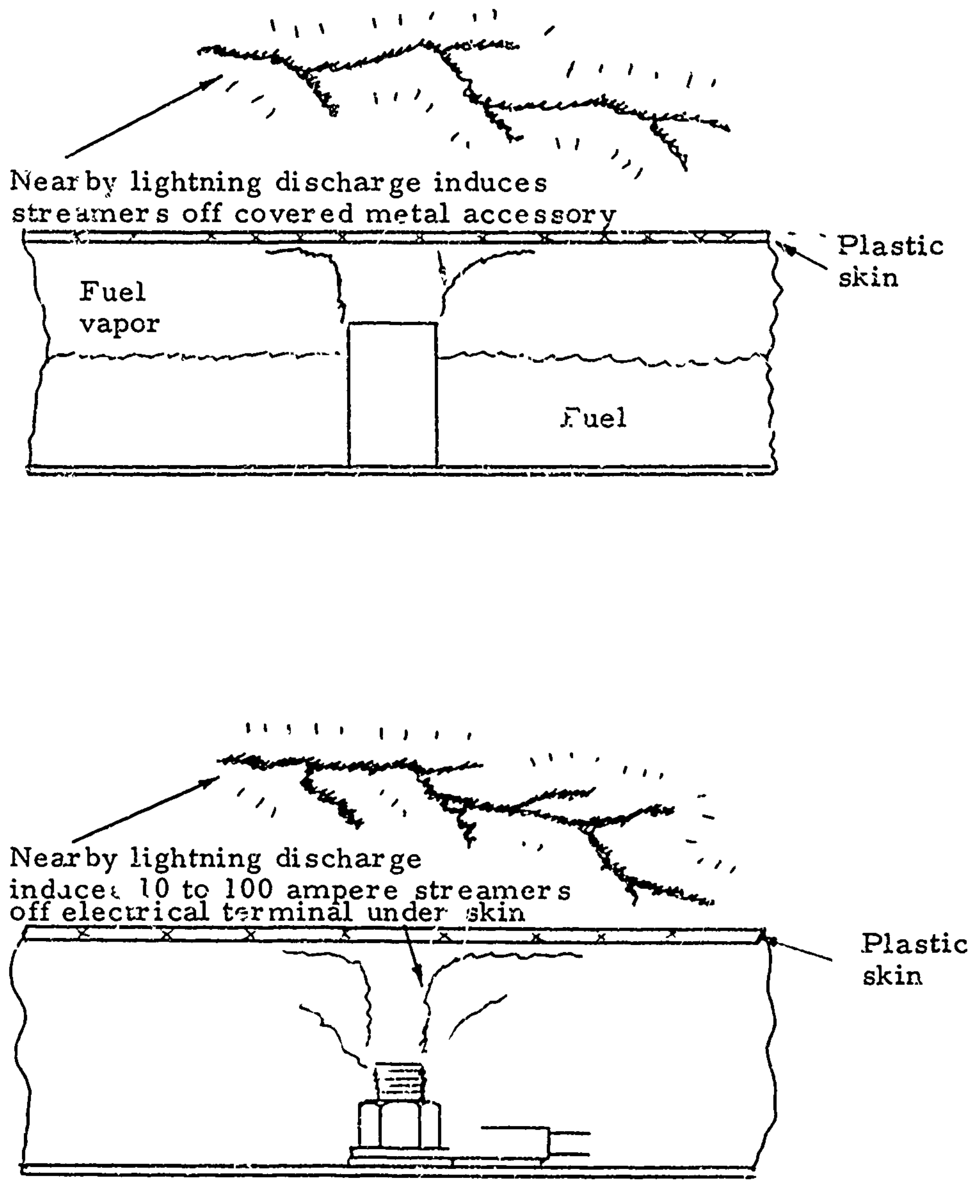



Figure 4. Induced Streamer Inside Plastic Outer Skins Produced by Thunderstorm Electric Fields

skin; wiring buried inside all-metal skin aircraft, therefore, is relatively immune to magnetic field coupling and induced surges from electric fields. Not so with wiring inside plastic-skin panels; the high current densities in the external skin result in strong magnetic fields, which couple substantial voltages into the enclosed wiring. For example, consider a 20 meter section of wire running along the spar of a wing section enclosed with a plastic leading edge, as illustrated in Figure 5. Assume that a wire is located four inches (0.1 meter) forward of the leading edge spar and that the current density in this vicinity is 20,000 amperes per meter. (This would be a reasonable value of skin current for a moderately severe lightning current of 100,000 amperes rising in one microsecond to crest.) The "H" field along the leading edge spar would be approximately 10,000 ampere turns per meter, and the resultant voltage coupled into the loop, as shown by the simplified calculation below, would be approximately 25,000 volts.

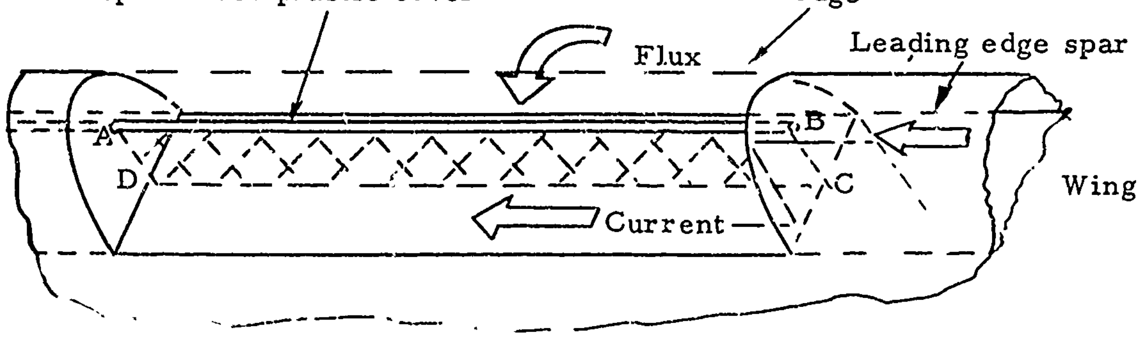



Figure 5. Voltage Coupled Into Unshielded Wiring Running Under Plastic Leading Edge From Magnetic Fields of Direct Stroke or Nearby Stroke Currents

Voltage about loop ABCD = rate of change of flux which links the loop:

$$V = \frac{\partial \phi}{\partial t} = \text{Area} \times \frac{\partial B}{\partial t} = \text{Area} \times \frac{\partial \mu H}{\partial t}$$

$$= \text{Area} \times \mu \times \frac{\partial H}{\partial t}$$

where:

V = voltage induced loop area

$\phi$ = total magnetic flux

H = magnetic field intensity

B = magnetic flux density

$\mu$ = permeability of space

$\partial t$ = current rise time

A = area of loop

Using simplifying approximations

$$V \cong A \times \mu \Delta H / \Delta t = 4\pi \times 10^{-7} \times 2 \times 10{,}000 / 10^{-6}$$

$$\approx 25{,}200 \text{ volts}$$

This voltage would be induced in all wiring exposed for the 20 meter length along the leading edge spar under a plastic leading edge.

Thus, large voltages can be coupled into the wiring enclosed under plastic sections. From the standpoint of electromagnetic fields, the wires are outside the vehicle even though it actually lies inside the plastic skin of the vehicle. The degree of shielding provided by composite structures remains to be fully determined. Regardless of their shielding effectiveness, composite structures present the problem of sparking at joints, which is common to all adhesively bonded joints.

The obvious answer to the problem of surge penetration through plastic skins is to provide shielding by using conduit or foil wrap cable shielding. This technique is highly effective against electric fields, but its effectiveness against magnetic fields in actual installations is still being determined. Initial tests of these new special shielding techniques when used in conjunction with optimum low field location of the wiring are encouraging. Other techniques being considered include the use of aluminum plasma flame spray coating or thin aluminum foil on the outside of the plastic sections. These coatings provide some protection from direct strokes and effectively shield from thunderstorm cross field effects, but problems remain in providing coatings which will not require excessive maintenance and which will be practical and economical in both weight and cost.

## CONCLUSIONS

Techniques are available for protection of metal-skinned vehicles from lightning induced surge penetration transients, but tests are required as the final criteria. The principal problem for new vehicles with all-metal skins would be to detect changes in construction techniques which could permit entry of transients.

The trend toward the use of plastic materials and composite structures in new aircraft because of urgent requirements for weight reduction will require some major efforts to provide low-cost lightweight lightning protection. The problem does not appear to be insurmountable, particularly if approached during early stages of aircraft design and development when the appropriate techniques can be much more easily applied. What is immediately required, therefore, are programs to obtain the basic data on the effectiveness of techniques being proposed for the protection of these materials.

# SESSION II

# STATIC ELECTRICITY

Chairman - J. E. Nanevicz

Stanford Research Institute

# STATIC ELECTRICITY ON FLIGHT VEHICLES

J. E. Nanevicz and E. F. Vance
Stanford Research Institute

Various aspects of static electricity manifest themselves in connection with the operation of flight vehicles. Since some of the effects of static electrification can compromise the operation of flight vehicles, it is essential to have a good understanding of the basic processes involved, the problems they create, and the steps that can be taken to minimize the seriousness of the problems. It is particularly important that all persons involved with the design of flight vehicles have some familiarity with the problem, or at least an awareness of the fact that static electricity can cause problems, because the design of one system or structure on the aircraft often affects the static susceptibility of other systems. Recently, the increased use of plastic materials for aircraft structure has greatly increased the possibility of static noise problems, particularly if designers are not fully aware of possible consequences of design changes. In this paper we will discuss in a general way the manner in which static electricity is generated on flight vehicles, the problems that can result from static electrification, and methods by which the problems can be minimized.

## CHARGING PROCESSES

The various ways in which static electrification of a flight vehicle can occur are illustrated in Figure 1. Figure 1(a) illustrates frictional electrification; as uncharged precipitation particles strike the aircraft, they acquire a positive charge, leaving an equal and opposite negative charge on the aircraft (References 1, 5). Charging occurs both on the metal structure of the aircraft and on plastic surfaces such as the windshield (References 4, 9). The figure also illustrates the charging that can occur when fluids such as fuel or hydraulic fluid flow through pipes. Engine charging, illustrated in Figure 1(b), occurs when flight vehicles are operated at low altitudes (References 6, 7). Processes as yet incompletely understood occur within the engine combustion chamber and cause a predominantly positive charge to be expelled with the engine exhaust, which causes an equal and opposite (negative) charge to be imparted to the flight



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vehicle. Exogenous charging, illustrated in Figure 1(c), occurs when the vehicle flies in a region of electric field, such as that generated between oppositely charged regions of clouds (Reference 3); this field can cause discharges to occur from the extremities of the vehicle.

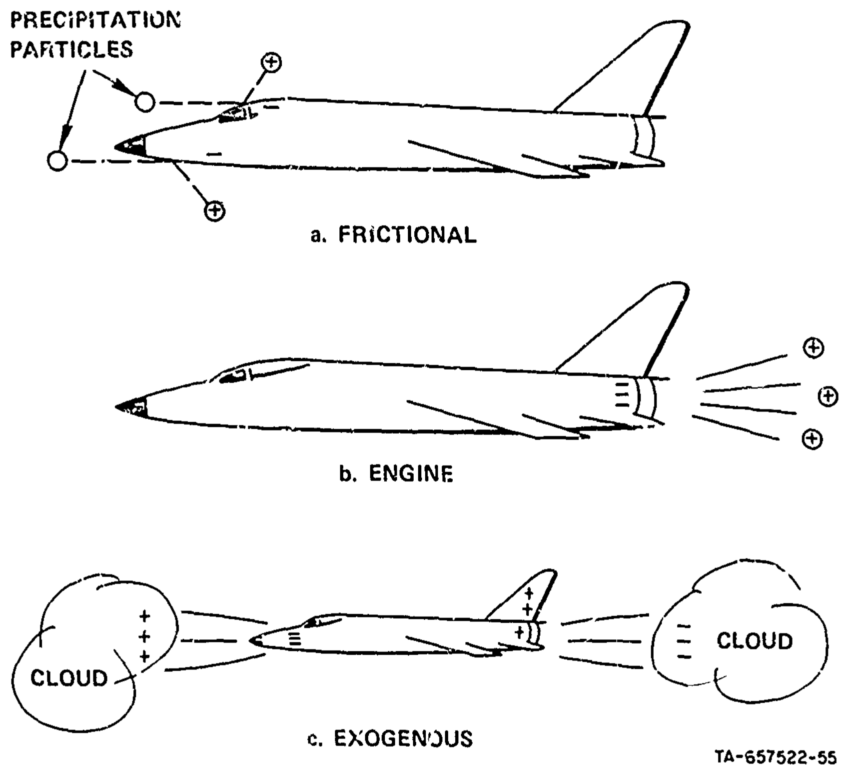


Figure 1. Charging Processes

The operational conditions under which static electrification can occur depend somewhat on the class of vehicle. Airplanes encounter severe charging during operation in clouds in horizontal flight; thus electrification can continue for considerable periods of time on all-weather missions (Reference 7). On jet aircraft operating at low altitude, engine charging can be an additional source of long-term electrification. Helicopters also become charged while flying through naturally occurring clouds (References 10, 11). In addition, a hovering helicopter stirs up snow or dust and thereby generates its own cloud of particles to produce frictional electrification. Thus, helicopters encounter static problems in regions where conventional aircraft do not.

Since rockets fly vertically, they have only short encounters with natural clouds; thus, this form of electrification persists for only short periods. The rocket engines are important in charging the vehicle at low altitudes, but the exhaust plumes serve as dischargers or potential limiters at the higher altitudes (References 12-18).

PROBLEMS CREATED

The charging process itself produces virtually no difficulty, but vehicle voltages and electric fields become so high after a period of time that electrical discharges can occur. It is the discharge of the accumulated static electricity that produces the harmful effects.

An important result of static electricity is electrical noise. The various noise mechanisms that have been identified are shown in Figure 2. As the airplane becomes charged, the electric fields at the extremeties of the vehicle become sufficiently high to cause corona breakdown of the air. At the operating altitude of airplanes, this breakdown occurs as a series of very short pulses containing energy in the radio frequency spectrum (References 6-9). These noise pulses can couple into communication, navigation, or digital circuitry to produce interference.

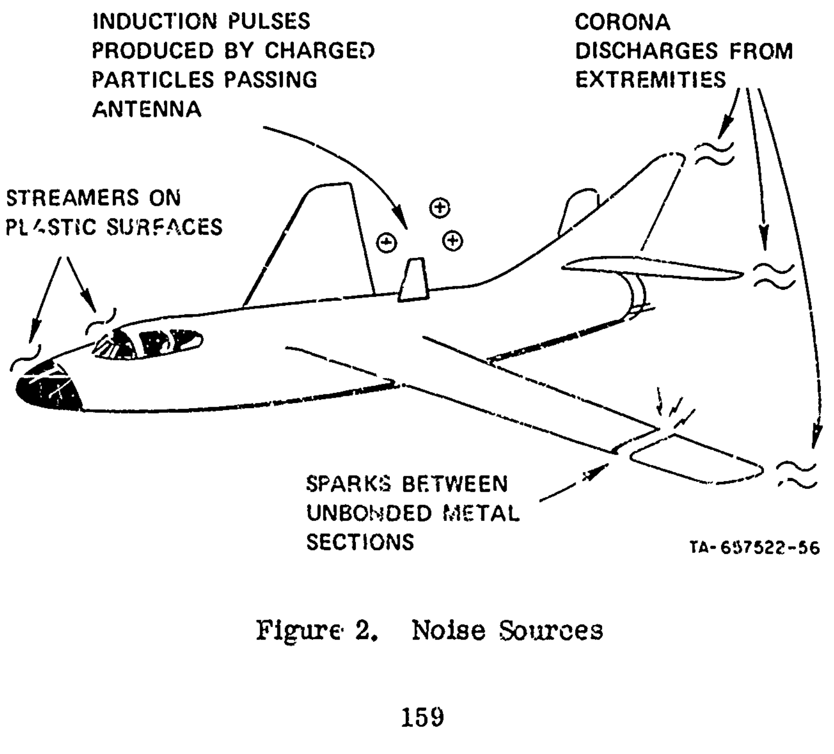


Figure 2. Noise Sources

Another source of noise occurs when plastic surfaces on the front of the airplane, such as the windshield and radome, are exposed to frictional charging, as illustrated in Figure 2. These surfaces also become charged by impinging particles. Since these materials are insulators, however, the charge is bound at the place where it was deposited and cannot be discharged until sufficient electric charge has accumulated to produce a streamer (a spark-like discharge) across the plastic surface to the metal airframe (References 4, 5). The streamer discharges are very short in duration but involve a large quantity of charge; they also produce radio frequency interference which can couple into susceptible systems on the aircraft. As is indicated in Reference 19, the streamering on a square inch of surface is often sufficient to disable systems.

A third source of interference which often occurs inadvertently on airplanes is associated with sparking between unbonded adjacent metal sections of the aircraft. For example, consider Figure 2, which shows a break in the wing; charging processes on the airframe will raise the potential of the inboard section with respect to the outboard section until a spark occurs in the gap. The spark produces a short current pulse, which is a source of noise (Reference 6). In flight, corona discharges occur from the wing tip, as shown in Figure 2; the current required for these discharges is supplied from the remainder of the airplane. Thus, although the potential of an isolated outboard part of the wing is sufficiently below that of the airplane to produce sparks in the gap, it is still sufficiently high for corona to occur from the tip.

Finally, slowly varying induction pulses can be produced in antennas by the passage of charged particles. This noise is of importance only at VLF or ELF and does not pose much of a problem on conventional communication and navigation equipment. With the advent of systems operating at frequencies of the order of 10 kHz, however, induction noise should be considered (References 4, 5, 19).

The second category of problems associated with vehicle electrification is the ignition of fuels or explosives (References 15, 20). The various ways in which electrostatic energy can initiate an electro-explosive device on an aerospace vehicle are illustrated in Figures 3, 4, and 5. Figure 3 illustrates how

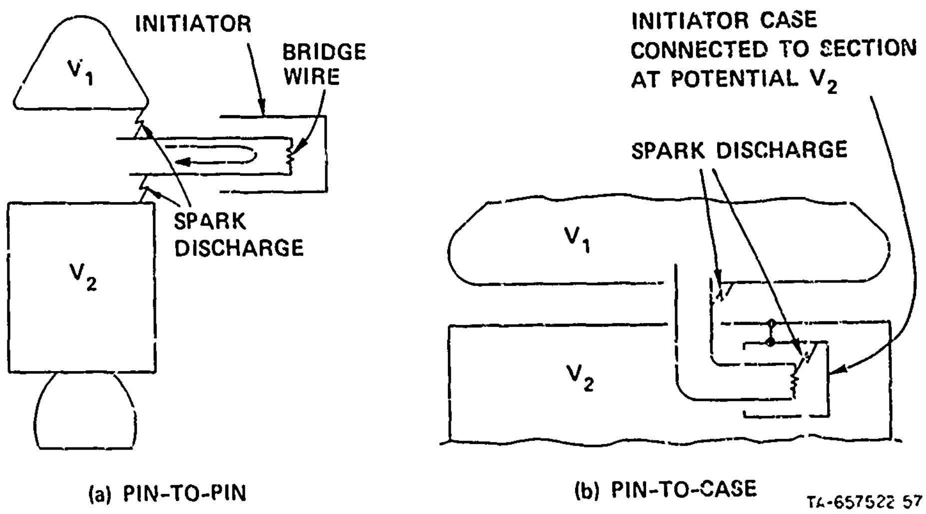


Figure 3. Squib Initiation by Sparks Between Sections

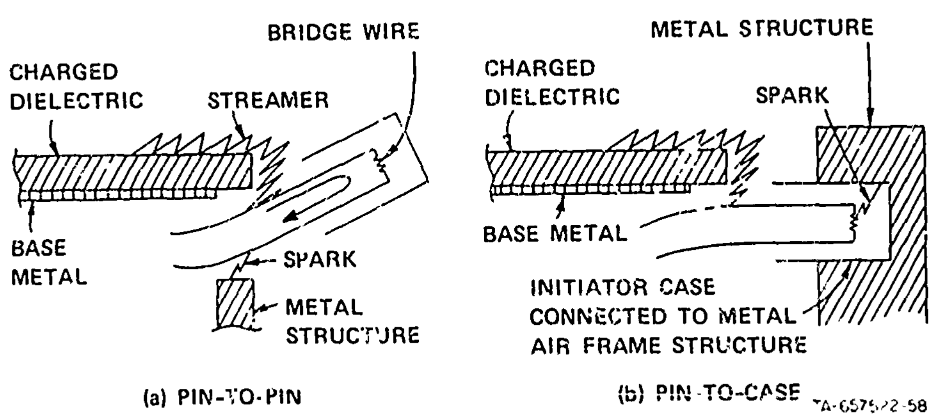


Figure 4. Squib Initiation by Streamer Discharge

the potential difference existing between unbonded sections of the vehicle can produce a spark which may initiate the squib in the pin-to-pin or pin-to-case mode. The ways in which streamer discharges from a plastic surface might initiate a squib are illustrated in Figure 4. If squib wiring is exposed to high electric fields, as indicated in Figure 5, corona discharges occurring from the wiring can cause a spark between bridge-wire and case to initiate the electro-explosive device.

Static electricity generated by the flow of fuel through pipes has been known to cause fires and explosions. In large storage tanks, the charge stored on the fuel is sufficient occasionally to produce a spark and an explosion. Research is currently under way to study the effects of fuel charging of plastic lines to determine if this produces a potential ignition hazard. In the use of helicopters to carry fuel or explosives, the charge on the helicopter itself becomes a potential hazard. The helicopter can assume potentials of tens of thousands of volts, which is sufficient to produce sparks capable of igniting either fuel or explosive devices contacted by the cargo hook (References 10, 11).

A further problem produced by electrification of flight vehicles is that of electric shock. For example, ground handling personnel often experience shocks when they touch cargo hooks of hovering helicopters. The shock is generally not lethal, but it can cause the individual to momentarily lose his balance and fall. A shock can cause an individual being rescued from water to let go of the cable and fall into the water and drown.

Modern windshields are constructed with an electrically conductive layer between the two layers of glass to achieve windshield de-icing. This film acts as one plate of a capacitor and permits considerable charge to be stored on the outer surface. Fighter pilots have been shocked by touching the windshield as they leave the aircraft; here again, the shock is not lethal but can cause a dangerous fall. This high charge stored on the windshield has also led to the damage of solid state devices controlling the windshield heater. Personnel can also be shocked, of course, when working near an ungrounded aircraft in the presence of a high external field. This shock hazard is perhaps more properly classed in the near-lightning category, however, because the normal fair weather field is not sufficient to charge the airplane to a sufficient potential to produce shock.

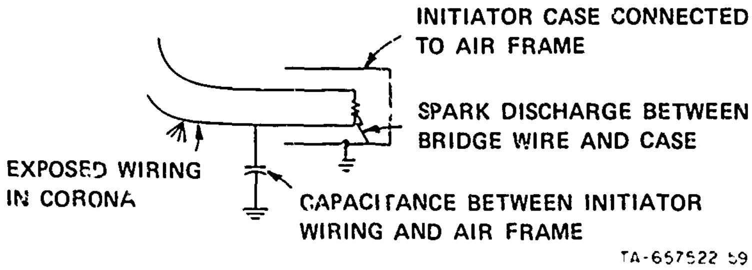


Figure 5. Squib Initiation by Corona Discharge

Finally, static charging of dielectric surfaces can result in the deterioration of the dielectric. For example, if a large radome is placed over a radar dish on the nose of the aircraft, the outer surface will become charged. The charge can return to the airframe either by streamers over the outside surface or by penetrating the radome and sparking to the radar dish. When the radome is punctured, moisture collects in the holes and causes delamination of the material and, eventually complete deterioration of the radome.

PRECAUTIONARY MEASURES

Insofar as static electricity effects are concerned, the ideal aerospace vehicle would be a completely bonded metal structure equipped with noise reducing dischargers at appropriate locations. Careful bonding of the adjacent metal parts prevents sparking from section to section. Since the vehicle is made completely of metal, streamer discharges cannot occur. Thus, the only electrical breakdown possible is corona discharge from the extremeties, and the use of noise-reducing dischargers eliminates any problems associated with corona.

Such a structure, of course, is not possible for various reasons. First, to see out of the vehicle and to communicate from it requires an antenna and a wind shield; these use dielectric materials, some of which must be located on frontal surfaces where they can become charged. Metal corrosion is forcing direct metal-to-metal contact at structural joints to be reexamined, which may lead to poor electrical bonding of the structure. Finally, weight considerations are forcing more and more of the aircraft to be fabricated of plastic. Practical aircraft, therefore, will occasionally have imperfect bonds and plastic surfaces will be included in the structure. Insofar as static electricity is concerned, plastic surfaces cause no trouble provided they are not exposed to impinging precipitation particles. Thus, plastic for trailing edges and perhaps for body sections should generate very little radio frequency noise, although it complicates the installation of corona dischargers and aggravates the lightning protection problem. If plastic materials must be located on frontal surfaces of the aircraft, they should be shielded from impinging precipitation particles by a particle deflector, the outer surface of the plastic should be made conductive to drain away the charge, or the entire plastic piece should be conductive. Such a treatment will prevent discharges (thereby eliminating noise) and will prevent charge build-up and puncture which damages the dielectric itself.

Under some circumstances, the static electricity problem cannot be eliminated merely by using passive devices on the vehicle; these cases require an active discharger system. For example, in the operation of helicopters it is desirable to reduce their potential to roughly 1,000 volts. The corona threshold potential of active dischargers does not permit attaining such low voltages. To hold the vehicle at such low voltages, some sort of active discharger system is needed, capable of sensing the vehicle potential and adjusting a power supply voltage to discharge current of the appropriate sign to maintain the potential near zero (References 21-14).

## SUMMARY

In summary, we have observed various forms of static electrification that are possible on airborne vehicles. The charging processes cannot be eliminated, but their deleterious effects can be minimized through proper vehicle structural and systems design. Design changes that are likely to modify the electrical bonding of aircraft structure should be carefully reviewed to make certain that they do not cause problems to the system. When plastic materials are to be used on the vehicle, use of the material should be examined to ensure that it will not cause problems in any of the pyrotechnic or communication and navigation systems. Finally, if plastic material is used on a frontal surface, it should be treated to ensure that static electrical charges do not cause rapid deterioration.

Most of the decisions regarding the structural form and the materials used in fabricating an aircraft, of course, are not made by people well versed in static electricity and its effects. It is hoped that this brief review will provide some insight into the problems that can result from relatively trivial changes in structure if static electrical effects are not borne in mind.

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# PRECIPITATION STATIC NOISE PROBLEMS ON OPERATIONAL AIRCRAFT

K. A. Moore

Continental Airlines

Perhaps one of the most elusive problems in the operation of airborne navigation and communications systems since the first airline radio-telephone was installed in the early '30's is "P-Static." Precipitation static or "P-Static" was the bugaboo of low-frequency ranges and homing beacons in the early days. Today it interferes with more sophisticated systems such as Loran C, Decca, or even satellite communications.

Too often flight crews are willing to accept P-static as a natural phenomenon to be reckoned with and complained about, but which little can be done about. The causes of P-static have been recognized since the late '30's: ice crystals, snow, charged water droplets, dust, or flying close to thunderheads. Most of these are related to precipitation, so the term "P-Static" was coined and has remained with us in spite of the more scientific definitions such as triboelectric charging, particle impingement and charge transfer, and external electric fields.

Although the causes of P-static are readily identified, the electrical charging mechanism and the way noise is generated and coupled to antenna systems has been fully understood only in the last dozen years. Much of this knowledge has been gained through funded USAF studies, which have been instrumental in reducing P-static in NAV-COM systems on civil and military aircraft. Today, jet transports properly treated against the electrical charging phenomenon can operate with quiet radios in P-static environments that would be completely useless in an untreated aircraft. The reduction in noise coupling to the aircraft's antenna system is of the order of 40 to 50 dB.

In the early days of airline operation, electrical bonding of the various metal parts of the airplane was found to be necessary for quiet radio reception, as well as for flight safety. Once the aircraft's ignition system was shielded (after Western Electric installed the first radio telephone in a "tin goose") and the metal parts were bonded, the radio system was found to work well in clear air but to be ineffective under some precipitation conditions, usually when it was most needed.

One of the first devices used to combat P-static was the "anti-static" loop, which was widely used from the late '30's through the '40's. Flight crews discovered that the electrostatically shielded loop of the direction finder would often allow reception during P-static. This, of course, was due to greater decoupling of the loop from the electromagnetic field of the airframe.

No doubt many will recall the 18 inch fixed antistatic loop mounted under the nose of most airline aircraft until the early '50's, when the LF ranges were replaced by the VHF omni-ranges (VOR). These new VHF NAV and COM facilities were said to be "static-free" because of their good propagation within line-of-sight distances to the aircraft. "Eureka! No more P-static!" Well, not quite!

By the close of World War II the trailing wick discharger had been developed (Reference 1). This device allowed the electrical charge to leak off the aircraft through resistively impregnated cotton fibers, and discharge from fine points at the end of the wick. This electrically isolated the discharge from the aircraft structure, at the frequencies involved, and reduced noise coupling to the radio antennas considerably. Although these "static-wicks" were reasonably effective, they were short-lived in the flight environment. Later carbon impregnated nylon was used to produce a wick with a somewhat longer life.

As jet transport aircraft flew higher and faster, P-static once again reared its ugly head. It seriously affected Loran, ADF, and HF systems, and was a limiting factor in the high altitude performance of VHF NAV COM systems, which were now being used over much greater distances.

In the late '50's, Stanford Research Institute developed the ortho-decoupled P-static discharger, which provided 50 to 60 dB decoupling of the electrical discharged noise from the aircraft's antenna systems (Reference 2). Jet airframes were clean and reasonably free of sharp protuberances, and usually incorporated flush-mounted antenna systems. By contrast, propeller-driven aircraft had propeller tips and external antenna hardware, which served as low-threshold voltage corona discharge points, and hence highly coupled noise sources, during electrical charging conditions. Jet aircraft provided an opportunity to fit stable P-static dischargers to a clean airframe and effectively reduce, if not solve, the P-static problem. If this were possible, there would be little justification for this paper, but it is not so simple.

Have you ever listened to P-static? Let us describe what a crew member hears in the headphones. As an unprotected aircraft enters precipitation, such as ice crystals at high altitude, it begins to accumulate charge; the corona threshold will be reached almost instantaneously. Discharge will become evident in the form of popping sounds, with the pulse rate a little irregular at first, but increasing as the degree of charging increases and sh ..tly becoming a steady roar that masks all but the strongest signals. As new points on the aircraft break into corona, noises like a buzz saw or a fire siren are heard. When the aircraft breaks out of the precipitation, the radio is immediately quiet. The noises are easily duplicated in the laboratory by putting points of various diameters into corona discharge in the coupling field of an antenna.

In the following paragraphs, I will attempt to describe the real world of P-static problems and how state-of-the-art technology is being applied to solve these problems (Reference 3).

## THE P-STATIC DESIGN PROBLEM ON NEW AIRCRAFT

An airframe manufacturer should first conduct model studies to obtain the DC electric field pattern of a new aircraft (Reference 4). Thus, the parts of the aircraft that will first go into corona can be predicted and the charging rate calculated from information on frontal area and speed. This information correlated with airflow patterns will allow P-static dischargers to be developed before conducting functional and reliability testing.

The extremities of aircraft subject to turbulent airflow and low-pressure vortices are probably low threshold electrical discharge points and must be considered in the discharger configuration. Such discharger configurations often become a part of FAA certification of transport aircraft.

Electrical bonding is now a matter of electrical specifications. Most manufacturers of civil transport aircraft have specifications with standards at least equivalent to those of the Military Specifications. Radomes and external covers for the flush antenna should be given resistive coatings sufficiently dense as to allow electrical charges to leak off to the surrounding structure without streamering, and yet not be so conductive as to interfere with radar or radio performance. A resistivity of between 10 and 100 megohms per square is felt to be adequate. With thorough design treatment, the probability of operational problems related to P-static will be lessened. The techniques for such design are fully described in available documents.

## OPERATIONAL PROBLEM NO. 1 — LIGHTNING DIVERTER STRIPS

An airline began to experience P-static of sufficient intensity to adversely affect VOR navigation and VHF communications. Minute cracks were found in the lightning diverter strips on the nose radome due to weathering of the thin aluminum foil, as illustrated in Figure 1. Whenever precipitation charging was experienced, charge deposited on the radome surface would flow to the isolated portion of the diverter strip and accumulate until the voltage across the gap became high enough to initiate a spark. Each spark discharges the isolated strip completely, so considerable charge is transferred; therefore, each spark is an extremely energetic noise source. The processes involved in spark formation are such that the noise spectrum contains appreciable energy at frequencies extending from LF through the VHF range. The dimensions of the diverter strips, furthermore, are such that at VHF they are a large fraction of a wavelength long; thus, at VHF a defective diverter system degenerates into an efficient antenna system driven by a spark noise source. As a matter of fact, interference from the diverter strips on one aircraft might couple into the VHF antennas of adjacent aircraft when flown in close formation. This diverter strip problem was experienced on DC-8, CV-880, B-707, B-720, C-135, and C-141 aircraft.

The problem was solved by replacing the diverter strips with an improved design that does not crack and break in the flight environment.

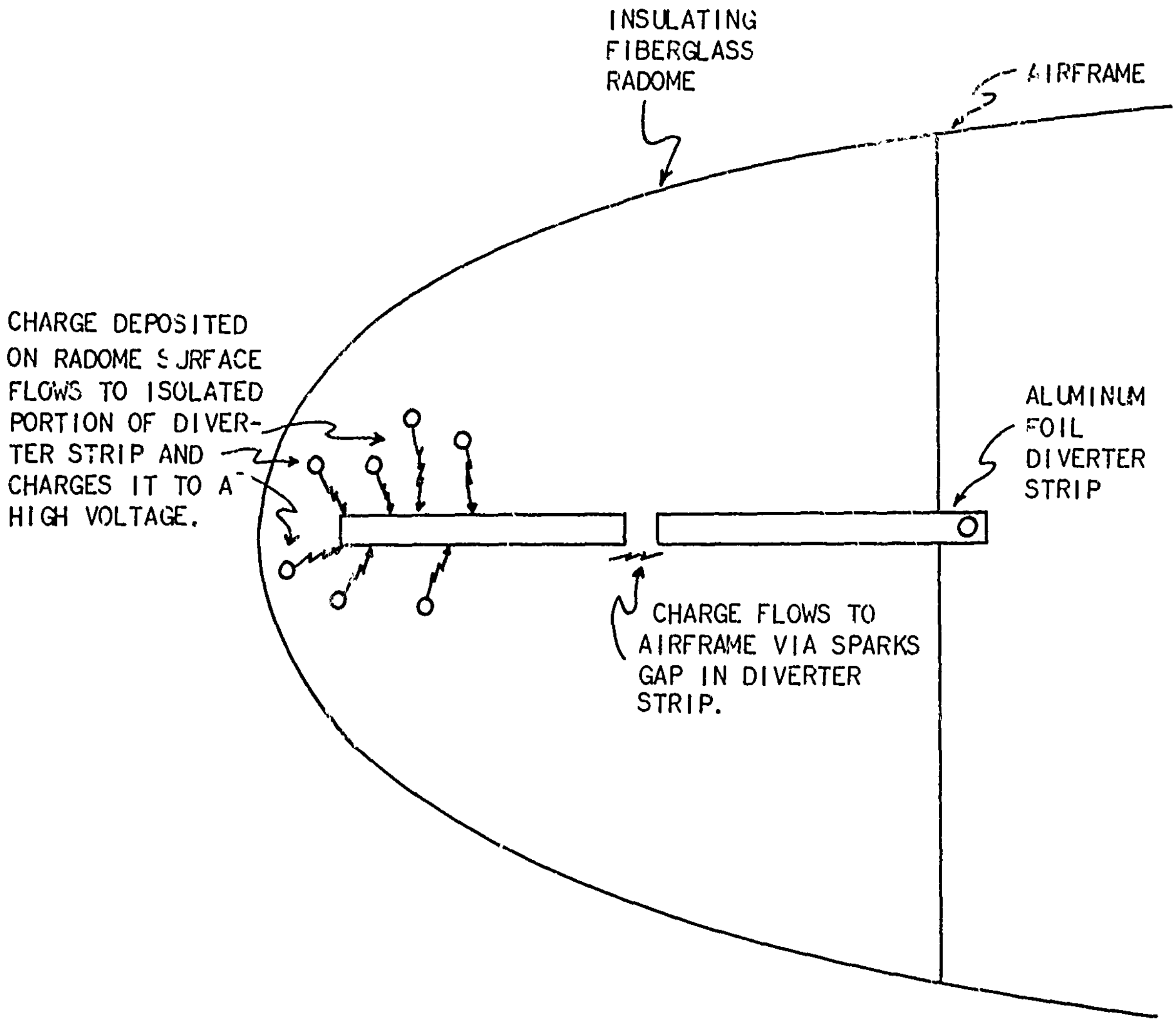


Figure 1. Broken Lightning Diverter Strip

OPERATIONAL PROBLEM NO. 2 — BONDING

A particular B-707 aircraft of an international airline experienced severe P-static on its radio. The manufacturer was contacted, and all efforts to solve the problem failed. This problem threatened to ground the aircraft as unsafe for long range flight. Finally, an outboard wing inspection panel was discovered to have been repainted during a previous inspection; when it was reinstalled on the aircraft, the mounting screws had not penetrated the paint, which left an isolated section, as illustrated in Figure 2. Precipitation charging of the isolated panel charged it to a high potential with respect to the airframe so that sparking

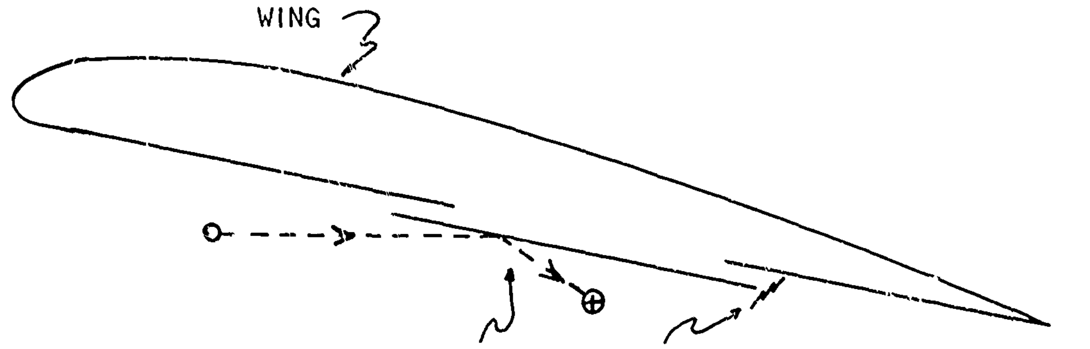


Figure 2. Isolated Wing Inspection Panel

occurred across the insulating gap. Thus, the insulated panel was, in effect, an electric dipole antenna driven by a spark noise source. Removing the paint around the screw heads solved the problem.

## OPERATIONAL PROBLEM NO. 3 — BONDING

In the early application of the ortho-decoupled dischargers to jet transports, two airlines flying DC-8 aircraft reported bad P-static on discharger-equipped aircraft. The design of the DC-8 vertical fin is such that a bonding strap must be used to bridge from the metal main structure of the forward part of the fin to the dischargers mounted on the insulating plastic trailing edge. Damage to the electrical bond was found, as illustrated in Figure 3, so that current flowing from the airframe to the discharger produced sparks across the gap in the discharger bonding system. This resulted in a spark-excited radiating system on the vertical fin of the aircraft. Improving the mechanical integrity of the bond solved the problem.

## OPERATIONAL PROBLEM NO. 4 — "THE CASE OF THE MISSING DISCHARGERS"

The flight crew of a C-135 aircraft on an Alaska to U.S.A. flight noted that the ADF's would point to one of the aircraft's wing tips whenever ice crystal

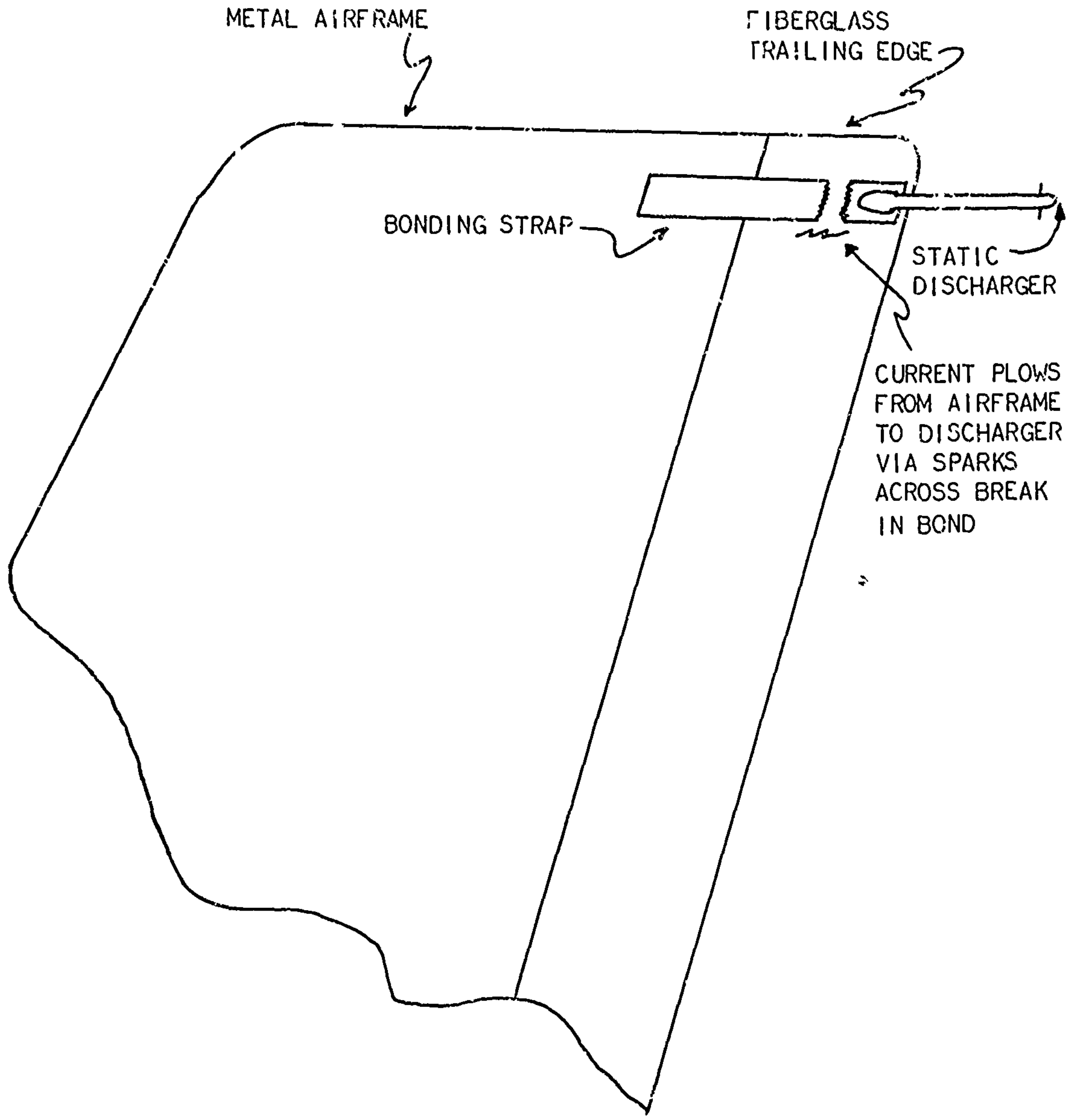


Figure 3. Broken Electrical Bond to Discharger on Vertical Fin

static charging was encountered. Inspection revealed that damage to the wing tip had removed the P-static dischargers, as indicated in Figure 4. Passive static dischargers are always installed in those locations where static electric fields are high and discharges to the surrounding atmosphere normally occur without the dischargers. Thus, removal of some of the dischargers resulted in noise-producing discharges from the airframe. The problem was quickly overcome by reinstalling the dischargers. This item demonstrates the importance of proper planning and maintenance of the P-static discharger installations.

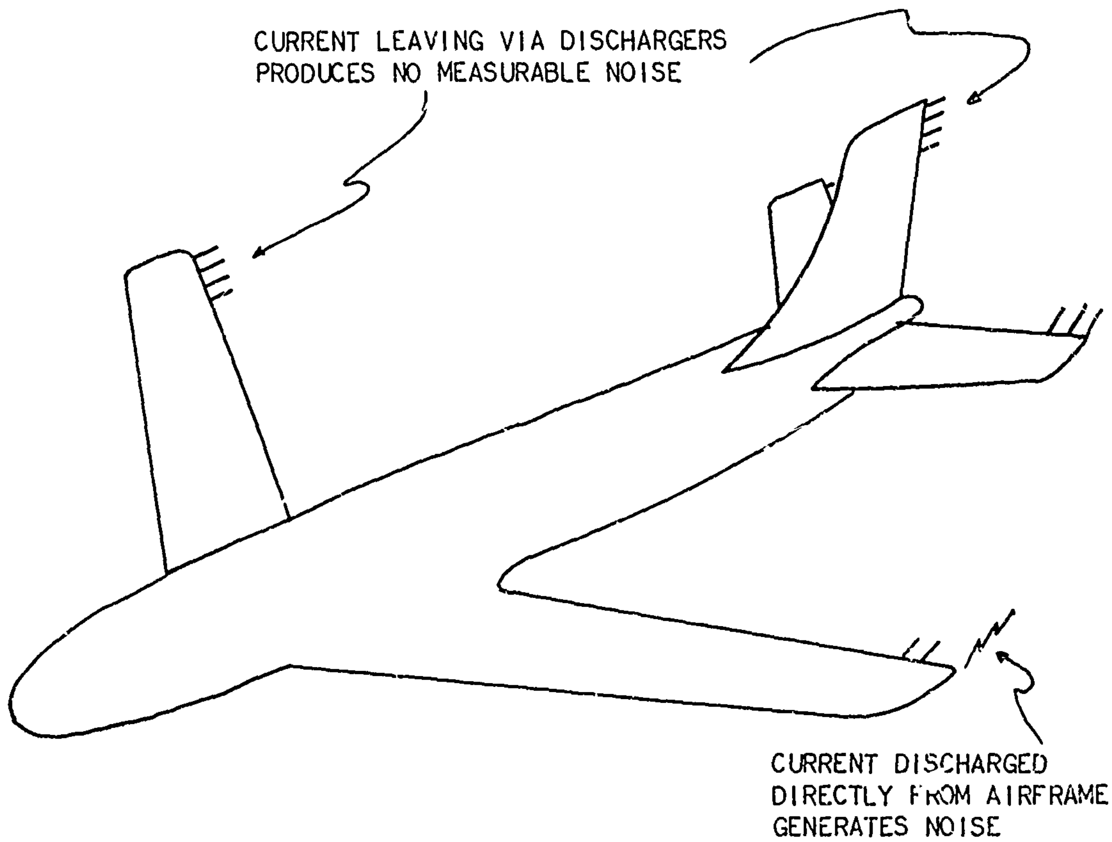


Figure 4. Dischargers Missing From Wing Tip's Trailing Edge

OPERATIONAL PROBLEM NO. 5 — VHF ANTENNA DISCHARGE

An airline noted that the VHF communications system appeared to be desensitized while in P-static conditions. The DC-8 aircraft had not yet been equipped with P-static dischargers. Tests verified that the VHF blade was going into electrical discharge (corona) under precipitation charging conditions. Fitting the aircraft with P-static dischargers lowered the discharge threshold voltage sufficiently so that, at normal charging rates, no further discharge took place directly from the VHF antenna.

This problem can generally be alleviated by locating the antennas at places such as the middle of the fuselage where static electric fields are normally low, avoiding sharp edges in the antenna design, and reducing aircraft potential to the minimum possible value with static dischargers. If the problem still persists, static dischargers can be installed directly on the affected antenna.

OPERATIONAL PROBLEM NO. 6 — FOR VOR OPERATION

An airline operating B-727 aircraft experienced poor VOR reception (reduced range) consistently on one aircraft. Checks of the antenna system VSWR of the antenna calble, etc., failed to show any equipment discrepancy, but the P-static dischargers on the horizontal tail plane showed that a lightning strike had damaged an outboard trailing discharger. The damaged discharger as well as adjacent dischargers were replaced as a precautionary measure, which cured the problem. It appears that the location of the horizontal tail plane immediately above the VOR slot antenna allowed the corona interference generated by the defective dischargers to couple to the VOR system.

OPERATION PROBLEM NO. 7 — HF P-STATIC

Quite recently, an airline operator of B-727 aircraft equipped with HF systems noted that whenever the slightest precipitation at altitude (ice crystals) was experienced, the HF receiving capability was almost useless. Tests showed P-static in the ADF, Loran, and HF, but no serious noise in VHF, eliminating the probability of electrical bonding discrepancies. The aircraft was equipped with a normal complement of P-static dischargers in good condition.

The HF antenna on the B-727 is a probe extending about 6 feet aft from the rear bullet (center of horizontal tail), as illustrated in Figure 5. The probe is about 2-1/2 inches in diameter, and the tip is smooth and well rounded. The manufacturer noted no previous reports of P-static interference with this HF system, but conceded that a low-pressure turbulent area would be present behind the rounded tip, thus forming a low threshold corona discharge point. The tip of the HF probe was fitted with P-static dischargers of the ortho-decoupled type, which solved the problem, and quiet HF operation was obtained during moderate precipitation charging conditions.

SUMMARY

The foregoing are typical of problems being experienced and solved daily by the aviation industry. P-static is indeed a problem in the all-weather operation of aircraft, but there are effective ways to combat it. Engineers involved in the design of airborne NAV-COM systems should take heed and carefully consider all possible methods of reducing the effects of P-static so as to provide the best possible performance in the flight environment.

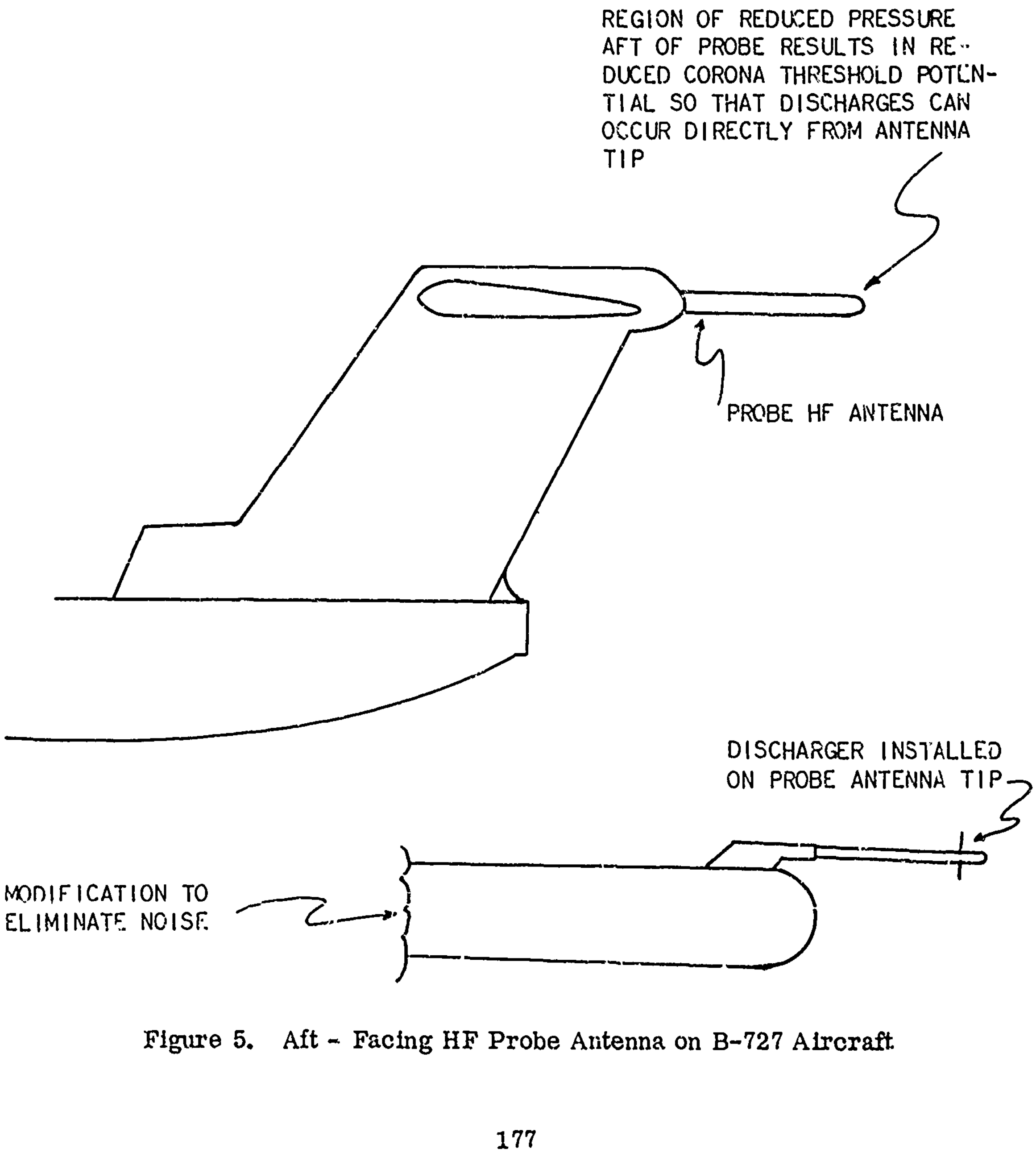


Figure 5. Aft - Facing HF Probe Antenna on B-727 Aircraft

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PART II

# THE ELECTROSTATIC CHARGING AND DISCHARGING PHENOMENA OF HELICOPTERS IN FLIGHT

G. Born, E. Sharkoff, and E. J. Durbin

Princeton University

and

R. Creed

U. S. Army Electronics Command

## BACKGROUND

The expanding role of the helicopter in cargo and rescue operations in all environments has created new and perhaps unexpected problems, such as the electrostatic charging and discharging phenomena experienced by helicopters in flight. In forward flight, the presence of static charges on the airframe presents no particular problems. Some radio and navigational equipment interference may be experienced during natural corona discharge, but passive dischargers can diminish this effect.

A cargo helicopter is required to hover in the vicinity of the ground during external cargo loading and unloading operations under many environmental conditions. Under certain conditions, large electrostatic potentials on the airframe with respect to ground present a serious hazard to personnel and/or cargo.

The problem can be divided into three basic areas: (1) injury to ground handling personnel performing the cargo hook-up; (2) possible damage to the cargo as a result of discharge current passing through it; and (3) ignition of fuel-air mixtures or premature detonation of flammable or combustible cargo in the vicinity of the arcs produced on ground contact.

## THE CHARGING AND DISCHARGING PHENOMENA

The main parameters involved in the charging and discharging phenomena are:

a. The electrostatic properties of the helicopter
b. The atmospheric conductivity

c. The charging and discharging processes.

## 1. THE ELECTRICAL PROPERTIES OF THE HELICOPTER IN THE ATMOSPHERE (References 1 & 2)

The capacitance of a helicopter, assumed as a perfect conductor in an infinite atmosphere, depends only on its shape and its dimensions. The complex shape of a helicopter can be approximated by a sphere having a radius of 1/5 to 1/6 of the helicopter length. The capacitance of a sphere is given by

$$C = 4\pi\epsilon a = 4\pi\epsilon\sqrt{\frac{S}{4\pi}} \tag{1}$$

where c is the capacity in farads
a is the radius in meters
s is the area of the sphere ($m^2$)
$\epsilon$ is the dielectric const  nt

The capacitance of a helicopter approaching earth is equal to the sum of its capacitance in an infinite atmosphere an l that with respect to earth.

Calculations and measurements, as plotted in Figure 1, indicate that the above approximation of a sphere in an infinite atmosphere can be used for altitudes higher than 10 meters (30 feet). The measured capacity of a helicopter $C_H$ is equivalent to that of a sphere with the same capacity or

$$a = \frac{C_H}{4\pi\epsilon} = (m). \tag{2}$$

where $C_H$ is the helicopter capacitance in farads

The potential of a sphere is defined as the potential difference between the sphere and infinity (or earth defined as zero potential). The potential function of a charged sphere is given by

$$V = \frac{Q}{4\pi\epsilon}\left(\frac{1}{r}\right) \tag{3}$$

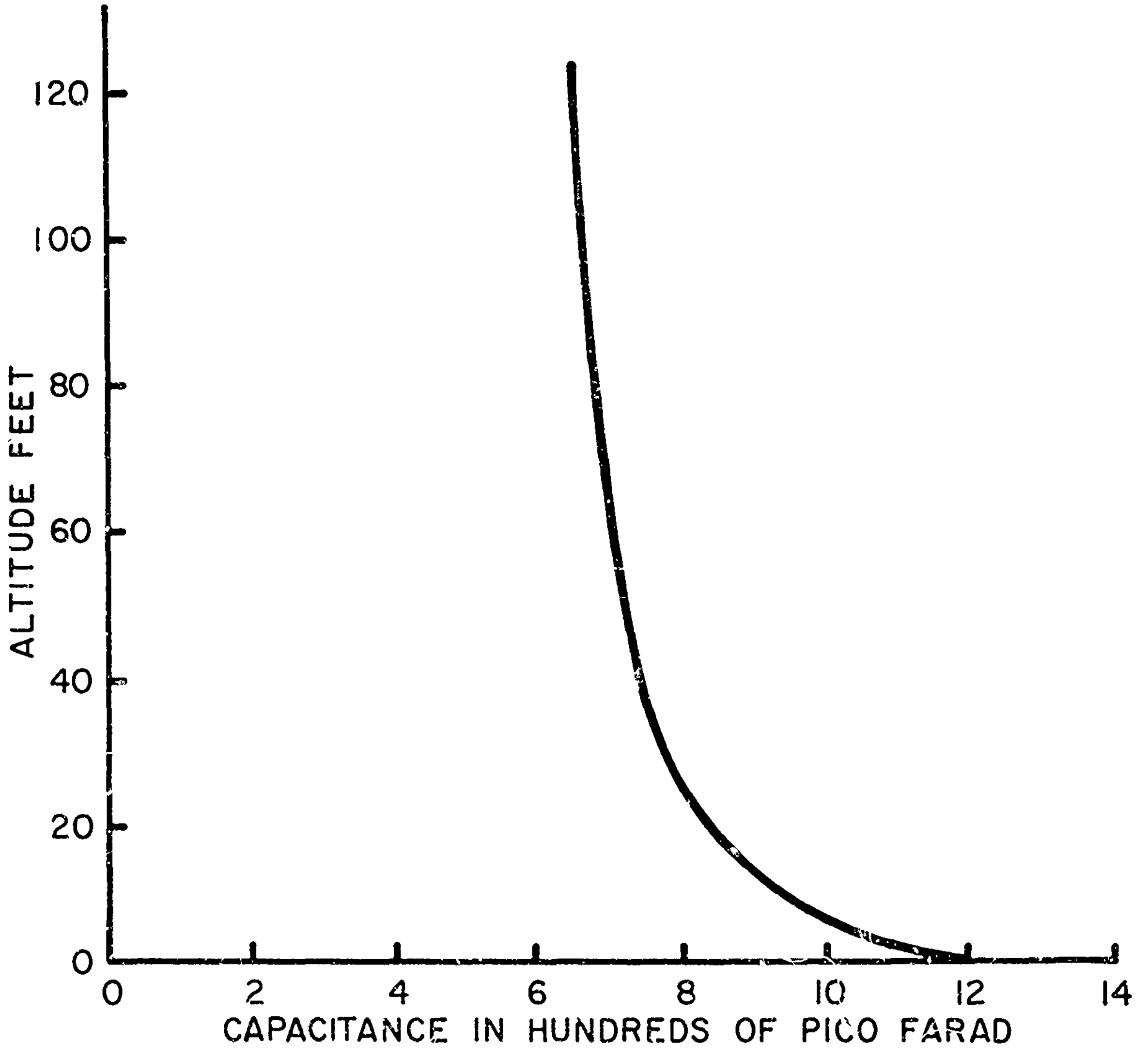


Figure 1. Helicopter Capacitance as a Function of Altitude, CH-47

where V is the potential function, in volts
Q is charge of the sphere, in coulombs
r is the distance from the center of the sphere, in meters.

At the surface of a sphere with radius a, the potential is

$$V_a = \frac{Q}{4\pi\epsilon}\left(\frac{1}{a}\right) = \frac{Q}{C} \qquad (4)$$

where $V_a$ is the potential of the sphere (V).

The electric field in the r direction, $E_r$ is

$$E_r = -\frac{dV}{dr} = \frac{Q}{4\pi\epsilon}\left(\frac{1}{r^2}\right) \qquad (5)$$

where $\frac{dV}{dr}$ is potential gradient in the r direction (V/m).

Note that the potential gradient is equal but with opposite sign to the electrostatic field. The relationship between sphere potential and surface field (Equations 4 and 5) is given by

$$V_a = aE_a \qquad (6)$$

where $E_a$ is electric field at surface of the sphere (V/m).

When an uncharged sphere is exposed to a homogeneous electric field, F, the potential function and the charge density on the sphere surface are as given by Equations 7a and 7b and shown in Figure 2.

$$V = E_e r \cos\Phi \left(1 - \frac{a^3}{r^3}\right) + E_e h \qquad (7a)$$

$$q = -3\epsilon E_e \cos\Phi \qquad (7b)$$

where $E_e$ is the atmospheric field strength (V/m)
$r, \Phi$ are polar coordinates (m, radians)
h is height above ground (m)
q is the charge density on the sphere.

When the earth is present, the errors in Equations 7a and b are less than 20% when the distance between the sphere and the earth is more than one sphere

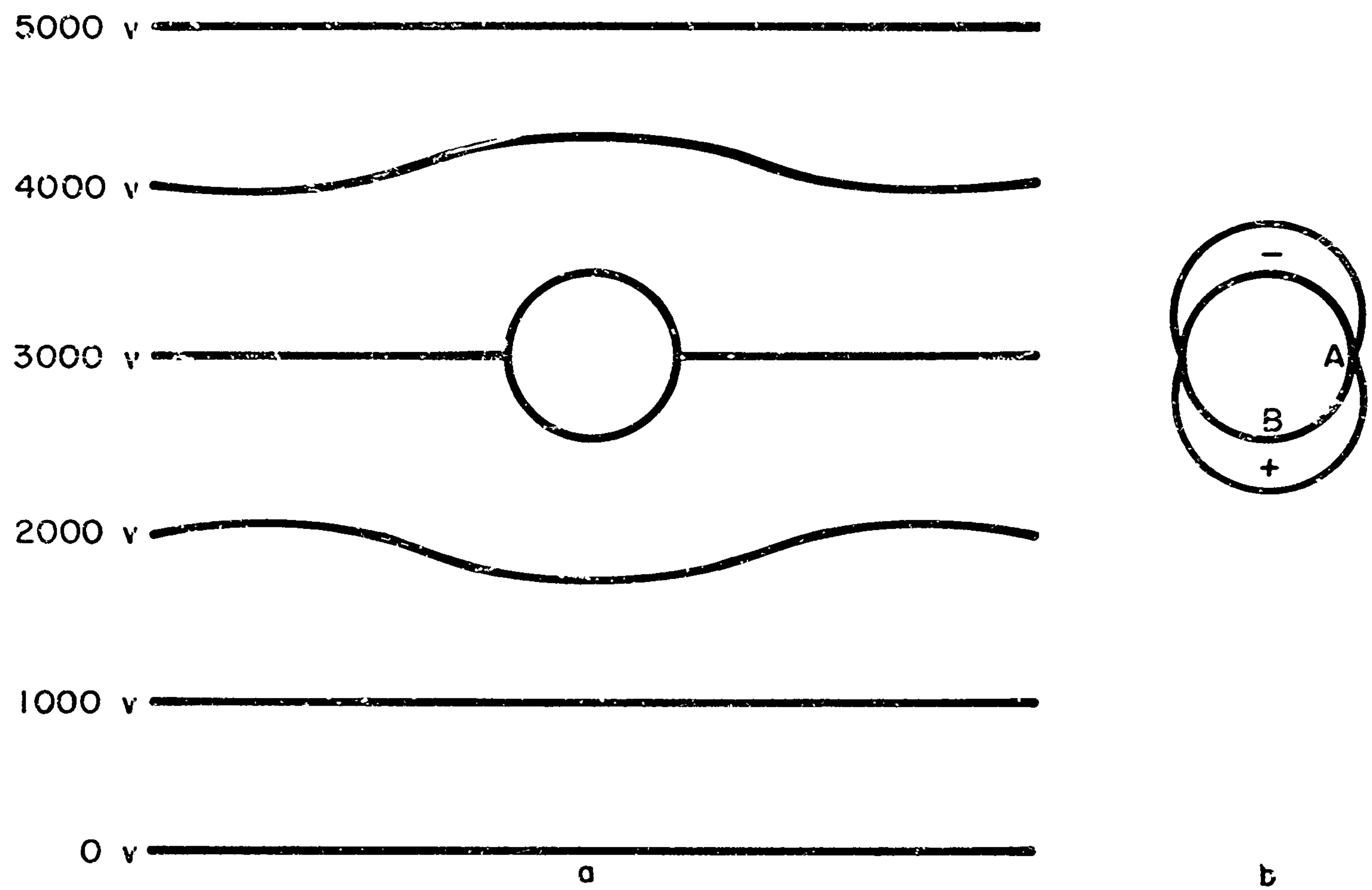


Figure 2. a. Equipotential Lines of Uncharged Sphere in a Homogeneous Electric Field

b. Charge Distribution of Uncharged Sphere in a Homogeneous Electric Field diameter. Equation 7 shows that when $\cos \Phi = 0$, the potential of the sphere is equal to the product of potential gradient and height. This corresponds to the basic law of electrostatics, that an uncharged body is always at the potential of its location (instantaneous). The earth's field also causes a charge separation so that the upper and lower halves of the sphere acquire an influence charge opposite to the potential gradient of the field.

When a charged sphere is exposed in a homogeneous field, the potential distribution and charge can be obtained from superimposing the above cases. The potential function is

$$V = E_e r \cos \Phi \left(1 - \frac{a^3}{r^3}\right) - \frac{3a^2 E_e}{r} + E_e h \qquad (8a)$$

and

$$q = 3\epsilon E_e (\cos \Phi - 1) \qquad (8b)$$

Charged and uncharged spheres in a homogeneous electric field are illustrated in Figures 3 and 4. Measurements indicate that the field close to earth is often homogeneous; see, for example, Figure 5 and Reference 1.

2. ATMOSPHERIC CONDUCTIVITY OF THE LOWER ATMOSPHERE (References 2, 3)

The conductivity of the air is proportional to the concentration of charge carriers (electrons, whether or not attached to molecules and ions), the charge on them, and their mobility. In general, the concentration of charge carriers increases with altitude due to higher cosmic ray intensity, but the conductivity can be assumed to be constant at altitudes where helicopters normally hover. The mobility of the charge carriers is inversely proportional to the air density. The ions attach themselves to impurities such as smoke and fog particles when present, thereby forming "large ions" with reduced mobility. In the lower atmosphere the mobility and ion concentration depends upon the purity of the air; this explains the large variations in measured air conductivity, since the atmospheric resistivity is the reciprocal of the atmospheric conductivity. At sea level, the atmospheric resistivity is the order of .1 to 1 x $10^{14}$ ohms-meter. The resistivity of the air in the vicinity of a propeller or rotor system is

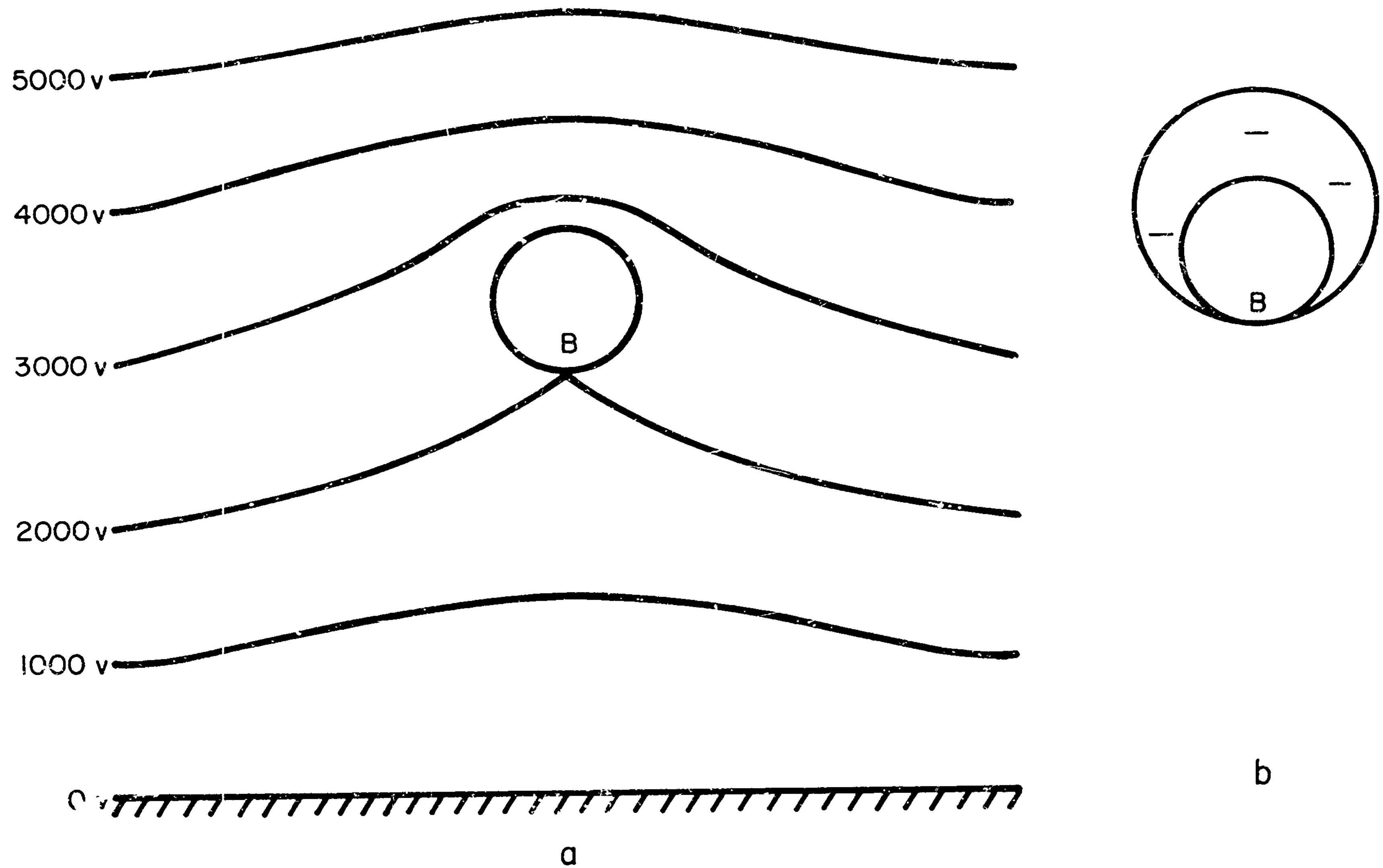


Figure 3. a. Equipotential Lines of a Negatively Charged Sphere in a Homogeneous Electric Field

b. Charge Distribution of a Negatively Charged Sphere in a Homogeneous Electric Field

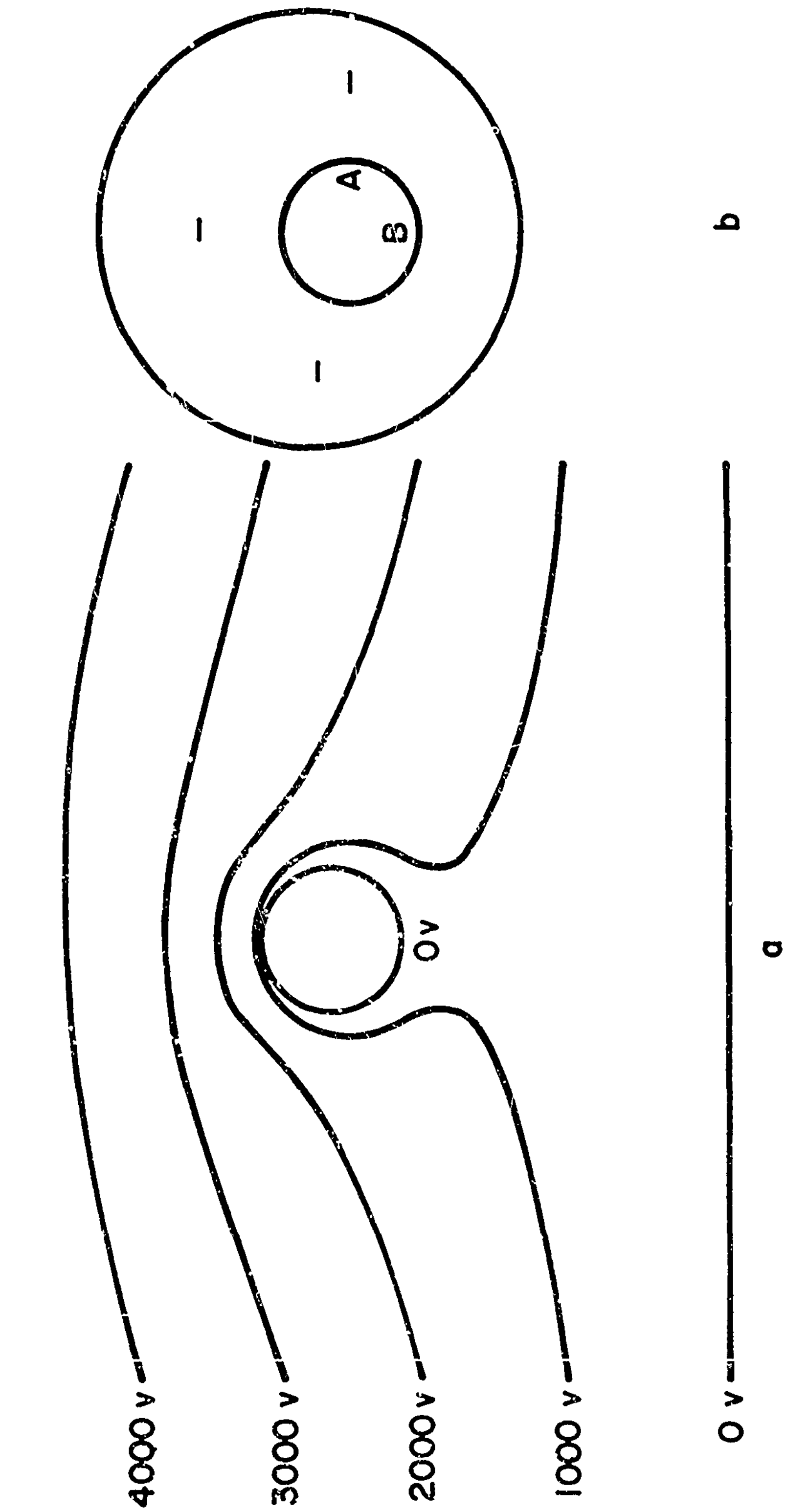


Figure 4. a. Equipotential Lines of a Sphere with Zero Potential in a Homogeneous Electric Field

b. Charge Distribution of a Sphere with Zero Potential in a Homogeneous Electric Field

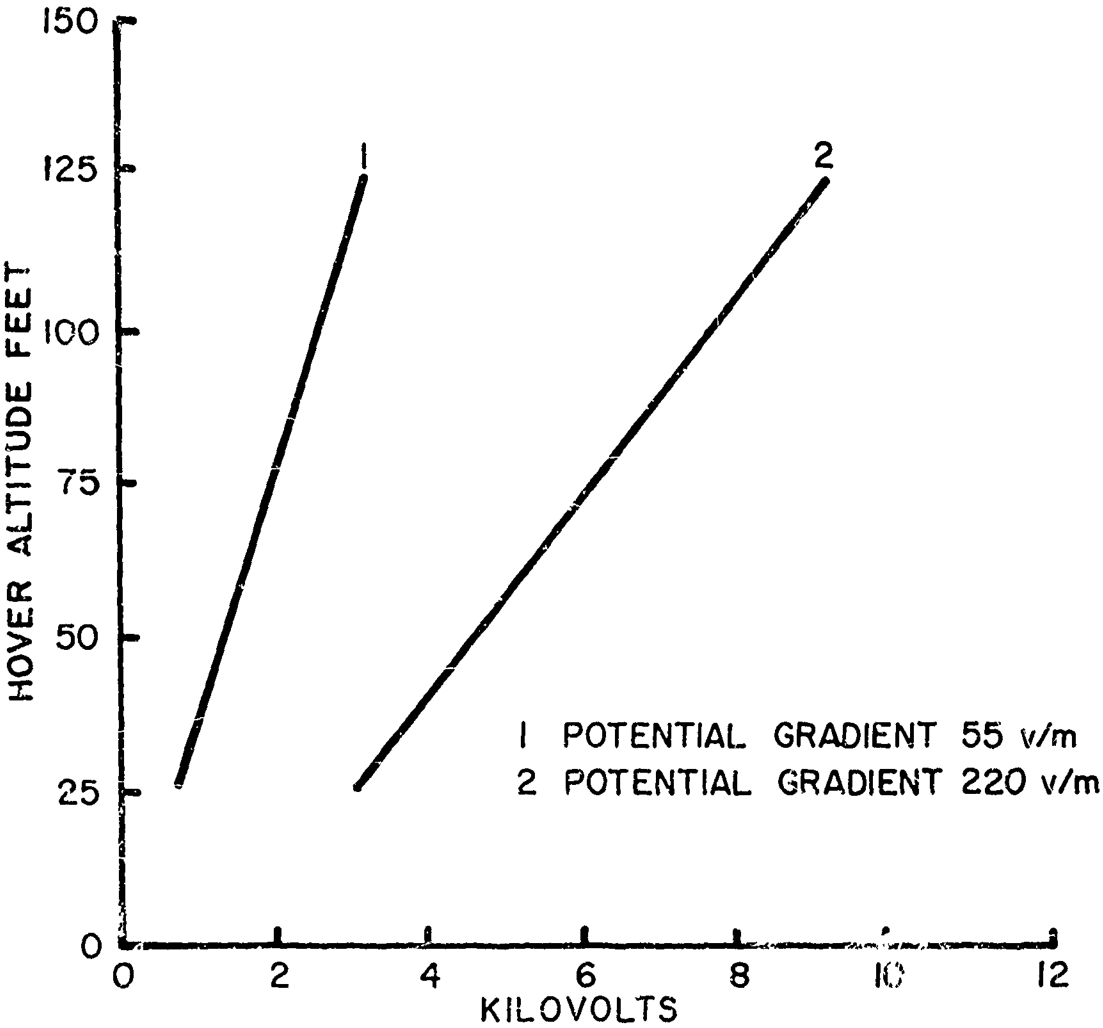


Figure 5. Natural Helicopter Potential Accumulation as a Function of Altitude, CH-47

decreased over that of still air because the propeller or rotor system imparts velocity to the air, and the effective mobility of the ions is increased. Since the resistivity is a function of both the number of ions and their mobility, the effect of this velocity is to reduce resistivity.

The effective resistance of a helicopter without the engine running to the surrounding air is in the order of $1 \times 10^{12}$ ohms. With a capacitance of 1000 picofarads, the helicopter charge and discharge time constant is in the order of 1000 seconds. Hot engine exhaust fumes produce ionization, however, which reduces the effective helicopter resistance, $R_H$, to the order of $10^{10}$ ohms. Therefore, the helicopter charge and discharge time constant is reduced to the order of 10 seconds. Measured values of a few seconds to 30 seconds have been recorded, depending on helicopter size and atmospheric conditions.

3. THE CHARGING AND DISCHARGING PROCESSES

When the processes described below increase or decrease the absolute potential of the helicopter with respect to the earth's potential, they are called, respectively, charging or discharging processes.

a. Atmospheric Electric Field Charging Processes. A helicopter flying in the atmosphere, in the absence of all other charging processes, will assume a potential equal to the atmospheric potential that exists for that altitude. The atmospheric potential at an altitude h is in the order of the product of altitude and potential gradient. In fair weather, the potential gradient is usually positive and can be in the order of several hundred volts per meter. In electrically disturbed weather, fog or rain for example, the potential gradient is usually negative and can be in the order of several kilovolts per meter. Under conditions of an approaching thunderstorm, much higher potential gradients have been measured, up to several hundred kilovolts per meter prior to lightning strikes.

b. Nonatmospheric Field Charging Processes. Many effects have been described and measured. The most important ones that yield high charging currents are described as follows:

(1) Triboelectric charging. Triboelectric or frictional charging results when dissimilar materials come in contact with one another. When particles normally found in a helicopter environment, such as dust, sand, snow, rain, etc.,

strike the helicopter, the aircraft charges. The charge rate or current depends upon the materials, mass, total amount of particles intercepted, and velocity with which the particles strike the helicopter. On the average, these currents are less than +50 microamperes, but under extreme conditions, currents of several times this number have been reported (References 5, 7). Some evidence exists that the effective helicopter resistance is lowered in conditions of high triboelectric charging ($R_H$ in the order of $10^8$ ohms).

(2) Precipitation. Heavy rain, especially from cumulonimbus clouds, snow, etc., can be charged and produce positive or negative charging currents in the order of 100 microamperes or more. Charging times to the maximum voltage of .6 seconds have been measured during hover at altitudes of 25 feet (References 6, 7).

(3) The self-generating charging process. The main source reported for the self-generating charging process is the engine, which can produce ions of one predominant polarity. Also, the ions generated are probably a function of fuel and air composition, as well as engine condition. The aircraft is charged to the opposite polarity of the ions. The order of magnitude reported for engine exhaust is in the order of a few microamperes (References 6, 7).

(4) Corona discharge phenomenon. The basic mechanism of a corona discharge consists of electrons accelerated by the strong electric field around any sharp point. The field strength around the sharp point is directly proportional to the potential at the point and inversely proportional to the radius of curvature of the point. Thus, when the radius is very small (sharp point) the field strength can be quite large. Locally accelerated electrons ionize the air; these ions (space charge) are removed from the vicinity of the point by the motion of the air or by the electrical field.

The discharge current from a corona point in still air with no other charges present can be represented by

$$i = KV(V - V_c) \tag{9}$$

where $i$ is the discharge current in amperes

$K$ is a constant related to the mobility of ions, the permittivity of free space, the composition of the air, the sharpness of the point, and the air gap.

$V_c$ is the voltage at which the corona current starts, which depends on the atmospheric condition and the dimensions of the corona point.

$V$ is the applied voltage at the corona point.

A discharge current in the order of tenths of a microampere occurs if the voltage on the point exceeds a certain predetermined value $V_c$. One of the most important points concerning the magnitude of a corona discharge is that the corona discharge current is space charge limited. The method of increasing the corona current consists of removing the space charge (ions) by:

- Larger voltages on the corona point (or increased field strength so that the ions will have greater velocity).

- "Blowing" away the space charge results in faster traveling ions. The corona current with wind along the corona point is equal to the corona current in still air, multiplied by a factor $(1 + \frac{W}{U})$ where W is the wind velocity and U the effective ion velocity in m/sec in still air.

When corona discharge occurs from a charged helicopter, an ion cloud can be formed. The time constant for establishing this ion cloud is in the order of 20 to 30 seconds, as determined experimentally from the ground for a CH-47 hovering at 25 feet with a corona discharge current of 100$\mu$a. As a result of this ion cloud, electric field measurements made from the helicopter did not indicate the correct helicopter potential.

Figure 6 shows the measured potential of a helicopter as a function of corona current as measured by a field mill located on an electrically grounded CH-47 hovering at 25 feet.

The effective helicopter resistance to ground is also decreased as a result of ion cloud formation, as shown in Figure 7. When the corona discharge was terminated, the ion cloud decayed with a time constant in the order of 0.1 to 4

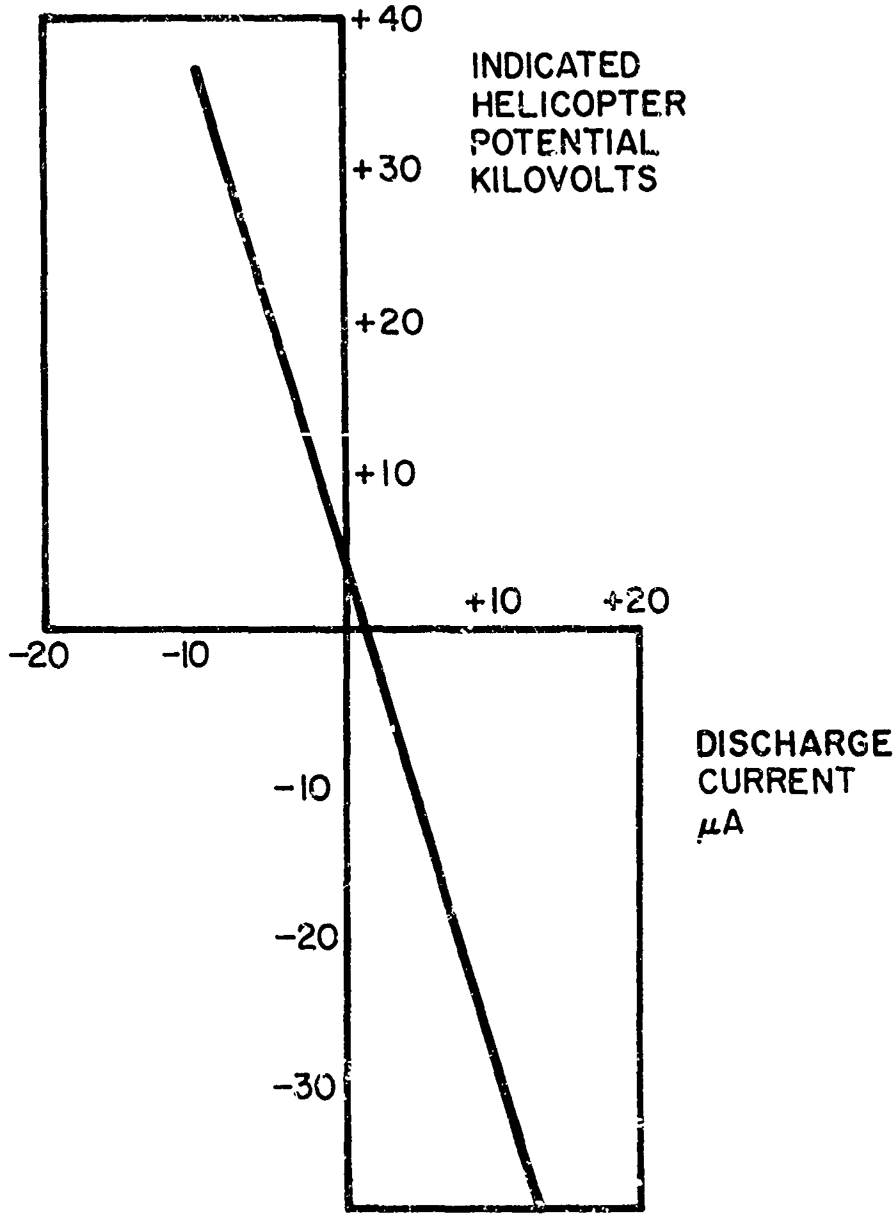


Figure 6. Indicated Helicopter Potential by a Fieldmeter as a Function of Discharge Current for an Electrically Grounded Helicopter

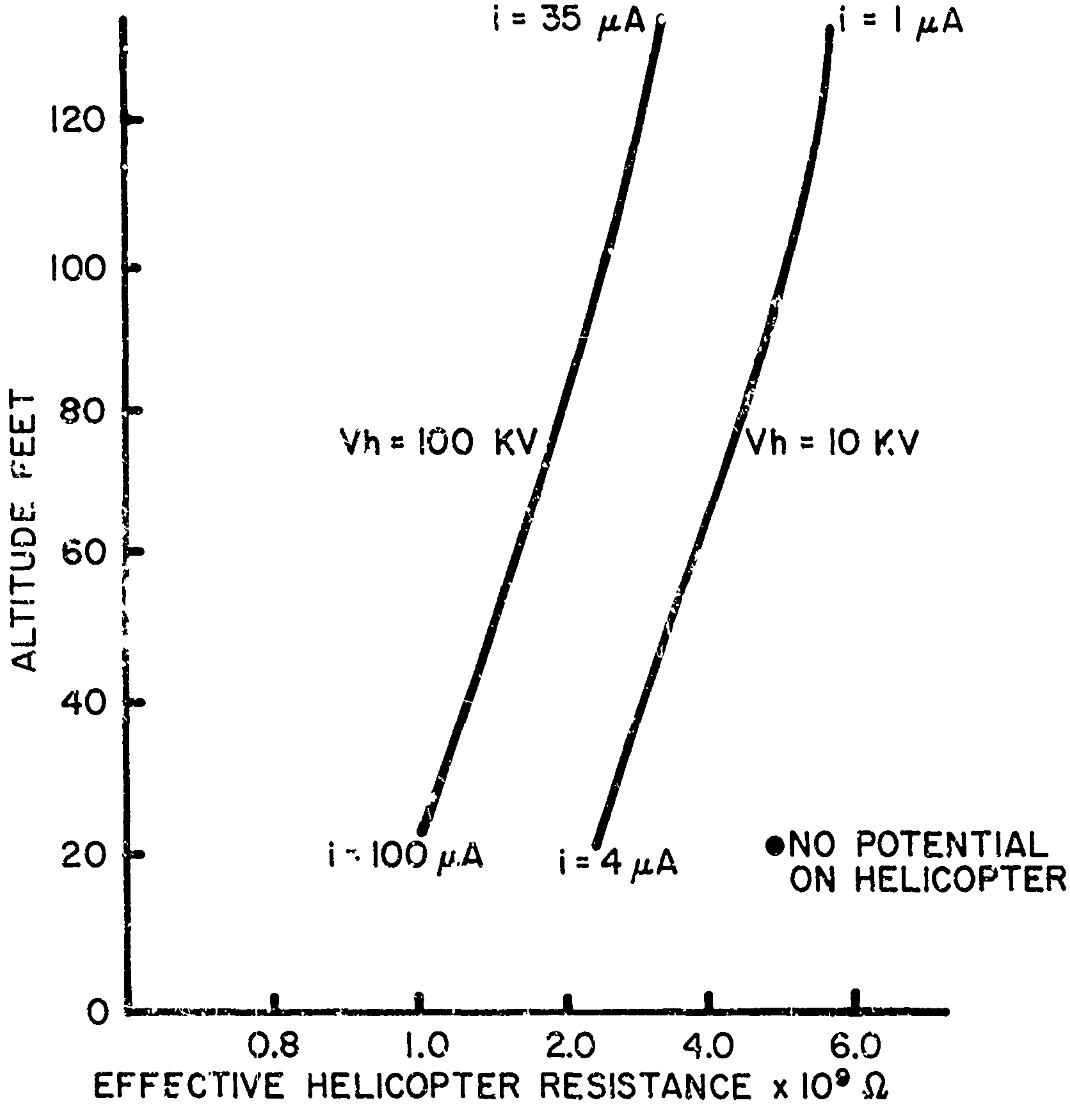


Figure 7. Effective Helicopter Resistance Vs Altitude CH-47

minutes. The time constant and effect of the ion cloud are strongly affected by the air flow pattern around the helicopter.

When the helicopter is charged by triboelectric charging, charge separation occurs. The helicopter and particles have opposite polarity. When corona discharge occurs, two "ion clouds" of opposite polarity are present. These clouds will neutralize each other if, and only if, both clouds are present at the same place at the same time. This condition is difficult to satisfy, as can be seen from the flow patterns around the helicopter.

## THE POTENTIAL ACCUMULATION OF A HELICOPTER IN FLIGHT

Assume an uncharged airborne helicopter that becomes charged because of one or more of the charging processes, as discussed in the preceding pages. As the potential of the helicopter rises above the potential of the surrounding air, the helicopter starts discharging towards the atmosphere, with a time constant in the order of 10 seconds. This discharge time constant is normally long compared with the charge rates of the triboelectric, precipitation, and self-generating charging processes (few seconds or less). The potential of the helicopter rises to a value of

$$V = \frac{Q}{C} = \frac{\int i\,dt}{C} \qquad (10)$$

where V is the voltage in volts
Q is the net charge in coulombs
C is the capacitance of the aircraft in farads
i is the charging current in amperes
t is the time in seconds

When the potential rises, the corona discharge process starts; this is mainly corona current from sharp points. The potential of the helicopter rises until a state of equilibrium is reached such that charging current equals the discharging current. The net effect of all the corona points on the airplane gives a total effective resistance of the helicopter.

PROBLEMS AND SAFETY LIMITS OF HAZARDS OF A CHARGED HELICOPTER

If the helicopter capacitance is charged to a high potential and discharges through personnel or cargo, hazardous conditions can occur. A large number of "safe" limits for discharges have been determined and reported in the literature. The problem can be divided into three parts, personnel safety, cargo safety, and radio frequency interference.

## 1. PERSONNEL SAFETY

For a capacitor type of discharge through the human body (resistance on the order of a hundred to several thousand ohms), the sensation threshold level is an energy of the order of 1 millijoule. (Note: 1 millijoule in 1000 Pf, corresponding to the approximate helicopter capacity, corresponds to 1400V.) For a continuous current type discharge through the human body, the sensation threshold level is in the order of 1 ma. Due to the relatively short discharge time, the peak current is higher in the capacitance discharge for the same sensation. Laboratory tests have confirmed that when a high resistance is placed in series with the human body, the discharge time of a capacitor increases and the sensation level is then determined by the continuous current sensation threshold which is the order 1 ma. This observation can be the basis of a simple solution to the personnel safety problem.

## 2. CARGO SAFETY (References 9, 10, 11)

Some examples of published safety limits of a capacitive type of discharge for cargo operations are as follows:

a. Explosives (commercial initiators): $10^{-3}$ millijoule
b. Explosives (secondary): several millijoules to 0.5 joule
c. Ignition of fuel-air-gas mixtures: 0.5 - 1 millijoule; expressed as a discharge current this would be:
   (1) 180-200 $\mu a$ for constant corona current,
   (2) 1 ma for a duration of 100 millisec,
   (3) 100 ma for a duration of 0.01 millisec.

3. RADIO FREQUENCY INTERFERENCE

When the helicopter is charged to a high potential, erratic corona discharges occur which produce RFI. This electrical noise source can severely hinder and even saturate the communication and navigation equipment.

SOLUTIONS TO THE PROBLEM

A satisfactory helicopter discharge system should provide a solution to the above problems and provide protection for personnel and all kinds of cargo.

There exist basically three approaches to reduce the electrical potential of a helicopter in flight.

1. PREVENTING THE HELICOPTER FROM ACQUIRING A CHARGE

Let us consider the ideal case where only one specified charging mechanism is present. Theoretically, a suitable coating can be provided to accommodate this condition, causing no charge to be developed. If this specific charging mechanism is, for example, frictional electricity, then the material with the highest dielectric constant will be charged positively. If the materials have the same dielectric constant, no charge develops. It is apparent that a coating can be developed for a specific case of frictional electricity if the materials are known. By using a coating consisting of two equal parts each with extremes of dielectric constants, the total charge current can, in principle, be reduced. Experiments have shown that the sign of triboelectric charging can be altered by coatings. However, the coatings used were sensitive to contamination and their effect could not be relied upon.

2. PREVENTING DANGEROUS DISCHARGE AT THE LOADING OR HANDLING POINT

When the discharge occurs via a resistive link, the power in the discharge path is determined by the helicopter potential and the resistance of the link.

In practical configurations, a resistive link can be placed between the airframe and the cargo hook, or a resistive link with hook can be hanging from the airframe cargo hook. The first approach is operationally more desirable.

Laboratory bench tests were conducted with a resistive link between the airframe and cargo hook assembly. The resistance was $10^8$ ohms at 100 kv and the capacitance between the cargo hook and airframe was in the order 40 picofarads. Satisfactory tests with potentials up to the order 70 kv were made. This corresponds to light shock. The second approach is electrically much more attractive but has the problem that for each cargo operation a new link must be placed on the airframe hook by the helicopter personnel. This setup has been tested satisfactorily to 70 kv (limit of the generator).

## 3. REMOVING THE HELICOPTER CHARGE BY A DISCHARGE METHOD

a. Grounding the Helicopter. Properly, grounding the helicopter (e.g., through a resistive link) and keeping the helicopter grounded solves the personnel problem. However, under certain conditions, the discharge energy can ignite fuel-air mixtures and explosives.

Experiments have shown that almost any contact between a conductor and any type of ground surface (sand, asphalt, grass) is sufficient to bring the potentials to safe levels. For example wire or nylon rope ($R \approx 10^9 \Omega$) being dragged along the ground from the helicopter works well.

b. Passive Dischargers on the Rotor Blades. Properly designed passive systems (corona points) on the rotor blades reduce the effective helicopter resistance $R_H$. Hence the helicopter potential with respect to the surrounding air is lower for a given charging current.

Examination of test results using good corona points to enhance discharge current (Figures 8, 9) show that the lowered helicopter potential is still hazardous for personnel and cargo operations.

It has been clearly established that the Decca Navigator System is given additional protection during corona discharge with the use of properly designed Passive Discharges on the Main Rotor Blades.

c. Active Discharge System. A closed loop active system should contain a sensor which measures a quantity proportional to the potential of the helicopter

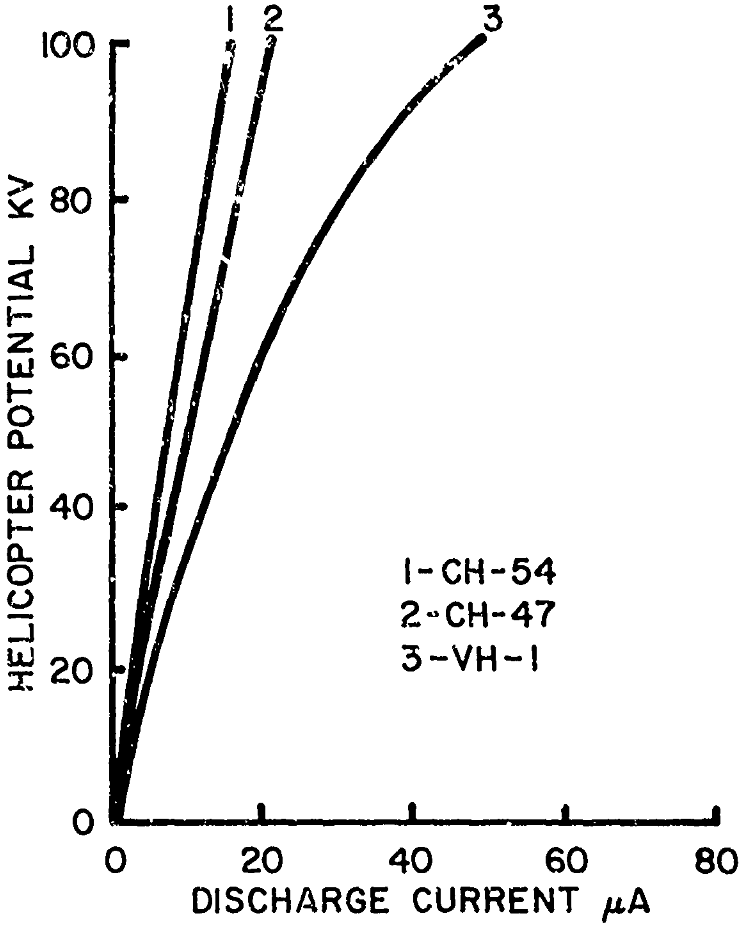


Figure 8. Helicopter Discharge Current as a Function of Helicopter Potential Natural Conditions

with respect to ground. The sensor output should control the discharge system to discharge the helicopter to ground potential during loading operations. Figure 10 shows a block diagram of a typical active system. An active system should, in principle, be "fail safe." It should provide for elimination of malfunctions by which the active system can charge the helicopter.

Available active system sensors have the deficiency that they measure the potential difference between the helicopter and the atmosphere in the vicinity of the sensor, not the ground potential. Consequently, if the atmosphere around the sensor is not at ground potential and/or charged molecules (ions) are present in the atmosphere, the sensor does not determine the helicopter's potential

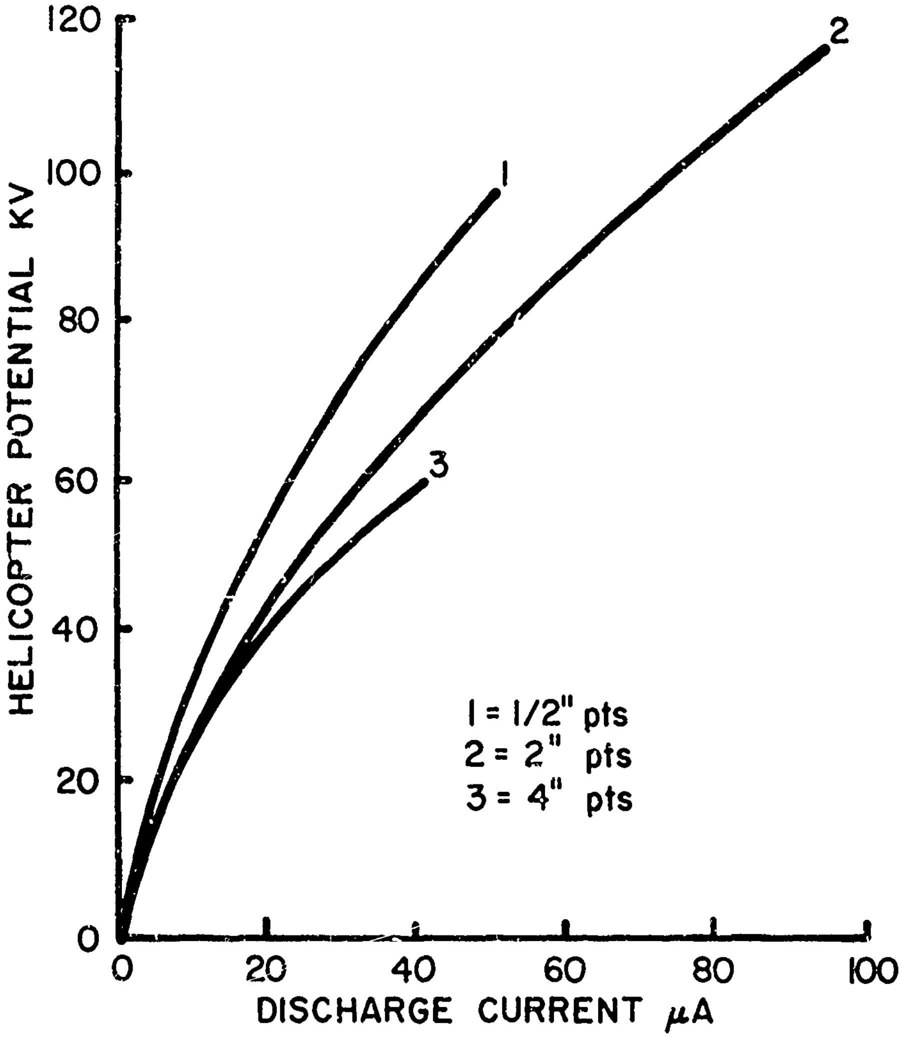


Figure 9. Passive Discharge System on Main Rotor Blades CH-47

with respect to ground. Hence, an active system when working properly brings the helicopter potential to approximately the potential of the local atmosphere as indicated by the sensor. Local atmospheric potential can be quite different from ground potential, for example, in disturbed weather. If, under such conditions, earth contact is made, the helicopter will discharge from the local atmospheric potential to ground potential through the earth contact and can exceed safety limits for capacitor type discharge: then after the helicopter has been discharged to earth potential, the current through the ground contact can be equal to the active system maximum discharge current. In most systems, the latter is below the human sensitivity level when the active discharger is properly current limited.

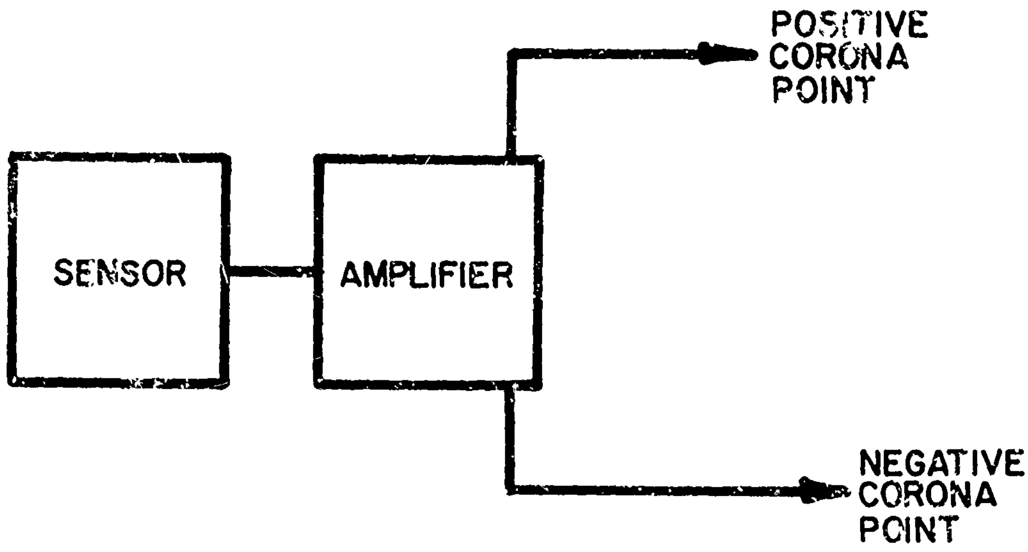


Figure 10. Electrostatic Discharge Block Diagram

Another problem of an active system is that part of the discharge current is recirculated back to the helicopter. Problems associated with this recirculation are dependent on probe and sensor location. When a large recirculation is present, it has been found that a closed loop active system can become unstable and charge the helicopter.

In summary, the concept of an active discharge system is good, and an active discharge system when designed properly may solve the electrostatic charge problem.

CONCLUSIONS

1. The goal for the safety limits should be set by the fuel-air mixture safety requirements.

2. Properly designed passive and active systems can reduce the helicopter potential.

3. Resistive materials can be used to limit the discharge currents and discharge energy.

4. A resistive link in the discharge path, together with a system to limit the extreme potentials on the helicopter, will be sufficient for personnel safety during cargo loading (with the exception of thunderstorm conditions).

5. The best helicopter discharge method to date, under all conditions, is a proper grounding technique. All aircraft should be grounded and continuously held at ground potential when fuels, explosives, or similar dangerous cargo are loaded or unloaded. If the operation takes place where fuel-air mixtures are present, it is advisable that the first earth contact be made via a resistive link of the order of $10^8$ ohms to reduce the danger of fuel-air ignition (e.g., contact is made by a resistive link hanging from the helicopter).

6. It is advisable that, whenever it is possible, the helicopter be electrically grounded during all cargo and personnel operations.

1. PERSONNEL
   - A. 1 millijoule energy
   - B. 1 ma continuous current threshold sensation

2. CARGO
   - A. Explosives (commercial initiators): $10^{-3}$ millijoule
   - B. Explosives (secondary): several millijoules to 0.5 joule
   - C. Ignition of fuel-air-gas mixtures: 0.5–1 millijoule

     180–200 $\mu a$ for constant corona current

     1 ma for a duration of 100 millisec

     100 ma for a duration of ≈ 0.01 millisec

Figure 11. Published Safety Limits

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PART II

# AN ELECTROSTATIC CHARGE PHENOMENON ASSOCIATED WITH MINUTEMAN MISSILE FLIGHTS

J. C. Axtell and T. C. Oakberg

The Boeing Company

This paper describes a system problem associated with the accumulation of electrostatic charge by the Minuteman missile. The paper discusses the circumstances of two missile flight failures and the results of the analysis which was conducted to determine the cause of the failures. A series of rocket motor diagnostic tests is described, and finally the corrective measures taken to prevent recurrence of the failures are detailed.

## HISTORY OF MINUTEMAN FLIGHT FAILURES

The first two test flights of the operational test version of the Minuteman missile experienced catastrophic failures before the completion of the first stage motor burn. FTM 502, flown from the Air Force Western Test Range at Vandenberg, California, destroyed itself at an altitude of 7.6 kilometers. FTM 503, also flown from the Western Test Range, destroyed itself at an altitude of 21.8 kilometers.

Telemetered data monitoring the operation of the missile's electrical and electronic data showed single bit anomalies in certain data channels just prior to missile breakup. Since these data anomalies were similar for both flights, the possibility of both failures being random was ruled out early in the investigation.

The two incidents were particularly puzzling, since the test program of Minuteman R&D missile flights (flown from the Air Force Eastern Test Range) had been highly successful. Furthermore, no failures resembling these had occurred in the Eastern Test Range series of tests.

## THE FAILURE ANALYSIS

An investigation of the failures was begun by the Air Force, in cooperation with Boeing and other contractors for the Minuteman system. The analysis of the missile configuration revealed that the operational version lacked structural electrical continuity. The heat transfer control concept used resulted in the electrical isolation of the reentry vehicle structure from the rest of the missile (Figure 1). On the R&D version of the missile, electrical continuity had been maintained across the reentry vehicle/missile interface by a ground connection associated with the instrumentation carried by the missile.

In ground simulation, it was demonstrated that a spark drawn between the electrically isolated sections would generate interference duplicating the data anomalies observed on the two test flights in question. Further investigation revealed that this interference was sufficient to cause a malfunction of the missiles' guidance and control system.

From the evidence gathered in the ground simulation of the test flights, together with the data from the flights themselves, it was concluded that the missile failures were the result of malfunctions of the guidance and control system caused by electromagnetic interference, probably from an electric discharge occuring between the unbonded sections of the missile. The next question to be answered was: Could this electric spark be generated by an in-flight missile?

## ROCKET MOTOR CHARGING TESTS

The accumulation of electrostatic charge by metal-structured aircraft has long presented an electromagnetic interference problem in the aircraft's radio communications systems. The interference is generated primarily by corona discharges which occur at the aircraft's extremities and sharp edges and points on its outer surface (Reference 1).

The most important mechanism for the generation of a net electrostatic charge on aircraft is "precipitation charging," or triboelectric charging. This results when atmospheric particles (snow, ice, rain, dust) strike the metal surface of the aircraft. Charge transfer between the particles and the aircraft surface takes place, leaving a net electrostatic charge of one on the aircraft and of the opposite sign on the particles. The net charge on the aircraft, and therefore the aircraft's electrical potential, increases until a corona discharge occurs to bleed off excess charge to the surrounding atmosphere.

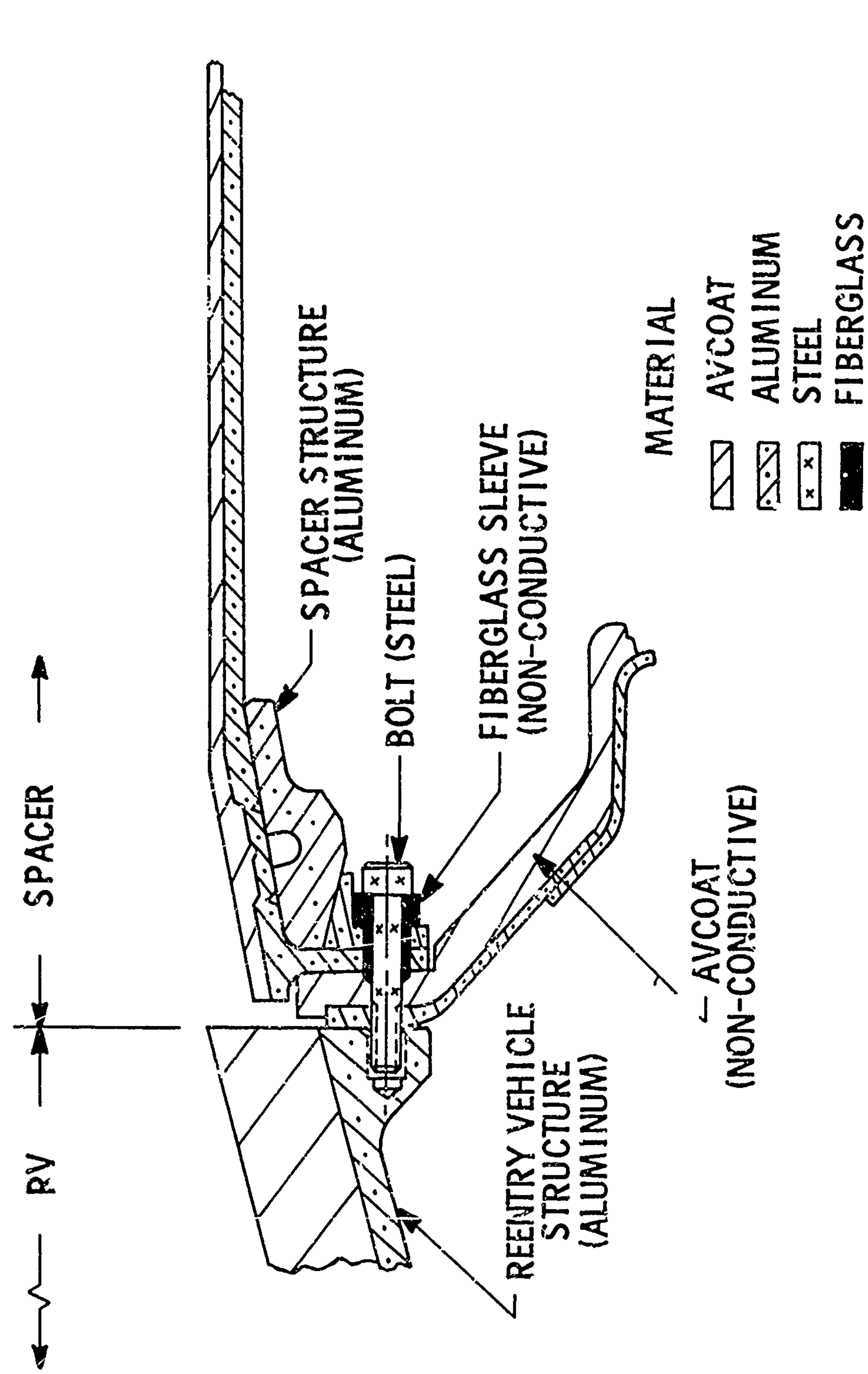


Figure 1. Detail of the Joint Between the Reentry Vehicle and Spacer Sections

Precipitation charging was the most obvious mechanism that could account for the presence of a charge on the vehicle. However, meteorological conditions at the times of the two missile flights indicated that precipitation charging was unlikely.

Earlier papers had suggested that operation of a rocket motor could perhaps generate an electrical current, and that this could lead to the accumulation of a net electrostatic charge by a rocket-powered vehicle (References 2, 3). To verify the existence of rocket motor charging currents and to arrive at some understanding of the charging process, a series of tests was conducted by the Boeing Company at its Tulalip rocket motor test facility near Everett, Washington. The initial tests consisted of burning small-scale rocket motors (approximately 350 pounds thrust) and measuring the current flow between the steel motor case and ground. The nozzle was steel except for the throat, which was graphite.

Each motor was wrapped with fiberglass tape and mounted inside a steel environmental chamber (30 feet long by 19 feet in diameter) for shielding purposes. To decrease any leakage of charge from the motor through moist air, a sheet-metal shroud was provided which allowed dry nitrogen gas to be passed over the motor case during a run. A block diagram indicating the electrical measurement method is shown in Figure 2.

The motors used for this series of tests had a burn time of about 2 seconds. The measurements showed that the three motors charged negatively, with an amplitude of 1.3 microamperes ±50%. The signal was quite noisy but clearly showed a negative current.

In order to establish some measure of the dependence of the rocket motor charging current on engine parameters such as thrust and nozzle surface area, an additional series of rocket motor experiments was performed. Three sizes of motors were used, with design data as given in Table I.

The motors used for these tests were charged with grains approximating the Minuteman first-stage formulation. The mountings ensured that the motors were insulated from ground and a sheet-metal shroud around the motor allowed dry nitrogen to be blown over it to minimize leakage of charge to the surroundings. This entire assembly was installed in a galvanized steel grain silo, with ends and bottom covered with bronze mesh to provide electrical screening (Figures 3 and 4).

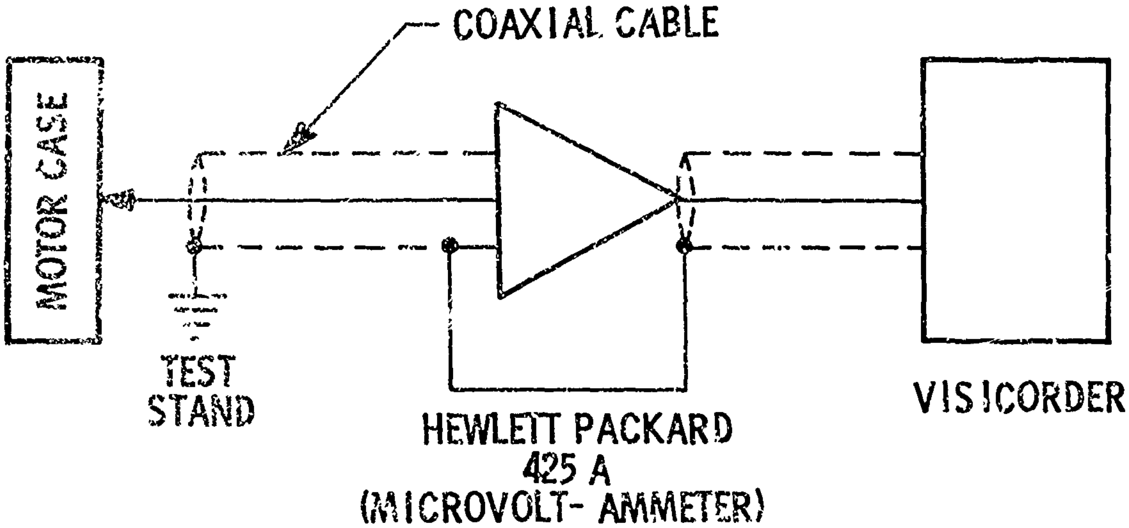


Figure 2. Electrical Schematic for Initial Charging Tests

TABLE I

MOTOR DESIGN DATA

| Design Thrust (pounds) | Measured Peak Thrust | Nozzle Throat Diameter (inches) | Nozzle Expansion Ratio | Nozzle Divergent Half Angle | Nozzle Inside Surface Area (sq. in.) |
|---|---|---|---|---|---|
| 500 | 600 | 0.950 | 10 | 12.0° | 5[illegible].1 |
| 350 | 400 | 0.794 | 10 | 16.0° | 40.2 |
| 200 | 260 | 0.600 | 10 | 18.0° | 22.5 |

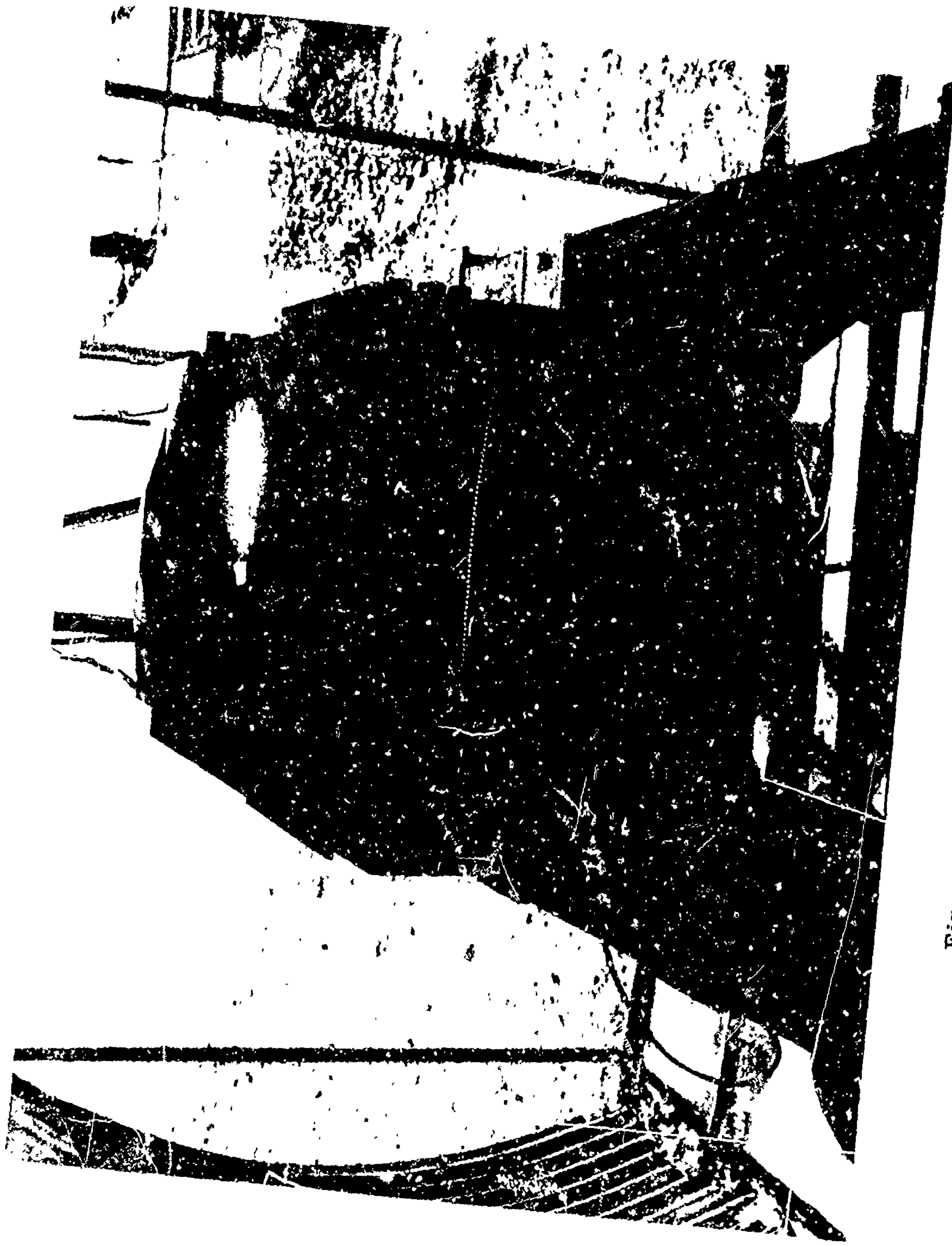

Figure 3. Rocket Motor With Conductive Shield

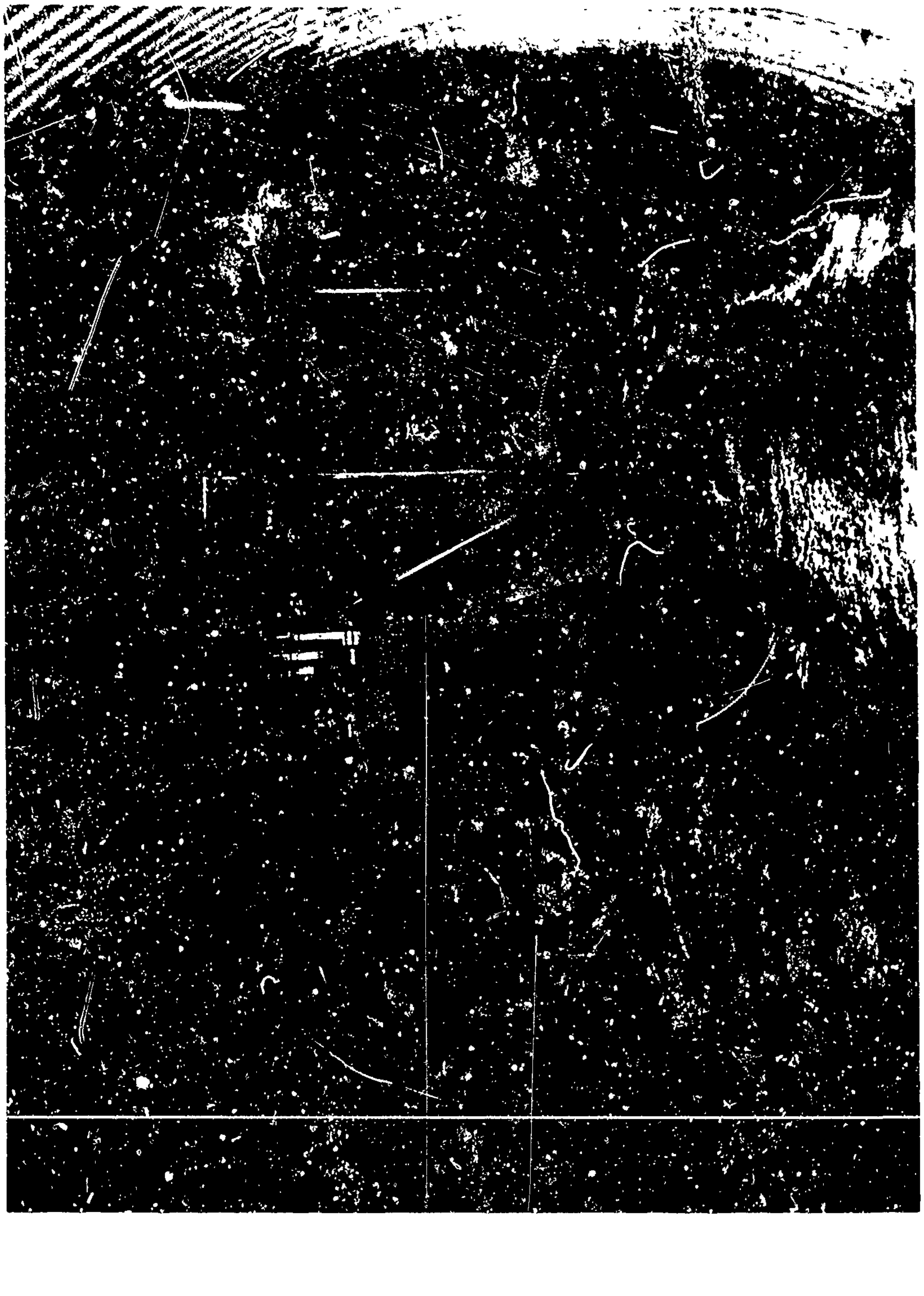

Figure 4. Rocket Motor Test Installation

Measured thrust levels were slightly higher than the design values but were well reproduced from one burn to the next. The motors were started by electrically igniting about 10 inches of solid grain stick placed inside the center hole of the grain. This technique minimized pickup of large electrical transients observed when the motor was ignited directly with an electrical device.

A block diagram of the instrumentation is shown in Figure 5. The three circuits measured charging current, optical plume emission, and thrust. Three each of the 200-pound and 350-pound thrust motors yielded data, and four 500-pound thrust motors burned successfully. Figure 6 shows representative thrust records. Peak thrust was reproducible within ±10%.

The charging current records showed a large number of spikes, many of which coincide with the occurrence of spikes on the optical emission signal. This suggests that part of the current fluctuation may be due to relatively large or hot particles (e.g., alumina) in the exhaust.

Smoothed data from the electrostatic charge measurements are shown in Figures 7 and 8 for the 350-pound and 500-pound thrust motors, respectively. The 200-pound thrust motors showed less than 0.1 microampere charging current, but tests showed that the thrust-time curves were not satisfactory; therefore, results from these motors have been disregarded.

Examination of Figures 7 and 8 and Table I shows that the charging current is not simply proportional to the surface area of the nozzle in contact with the hot gases, but obeys some more complicated relationship. Test results were also analyzed in terms of thrust as shown in Figure 9. These curves were determined by taking the maximum and minimum peak charging currents observed for the 350-pound and 500-pound motors.

The series of tests described above were useful for designing charge dissipation devices for inclusion in the Minuteman missile system. However, the tests give only a partial picture of the rocket motor charging process. All of the tests were "short-circuit" current measurements, with the potential of the test motor maintained close to ground potential. No examination was made of the relationship between charging current and the electrical potential of the rocket

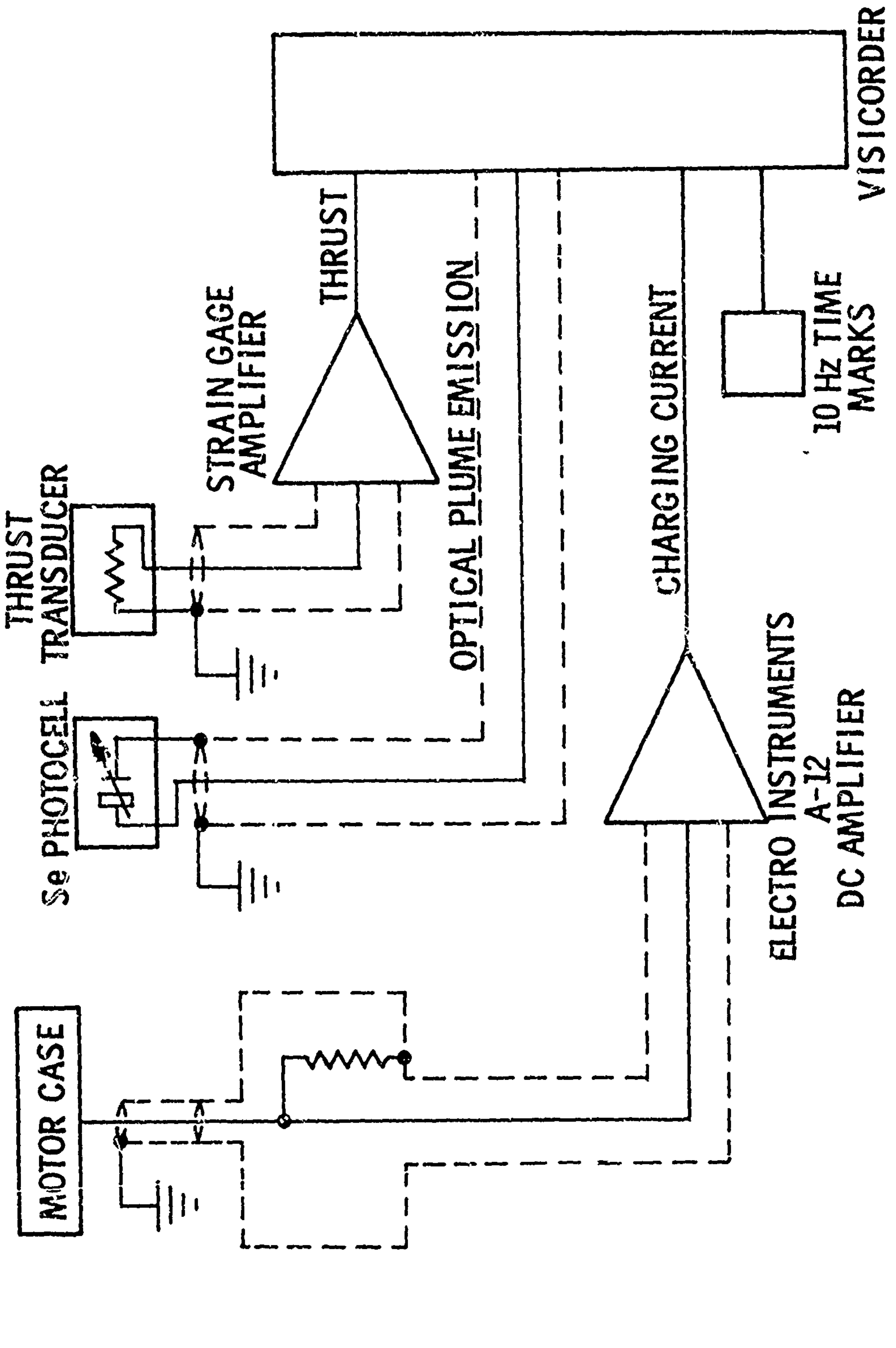


Figure 5. Instrumentation Diagram for the Later Series of Rocket Motor Tests

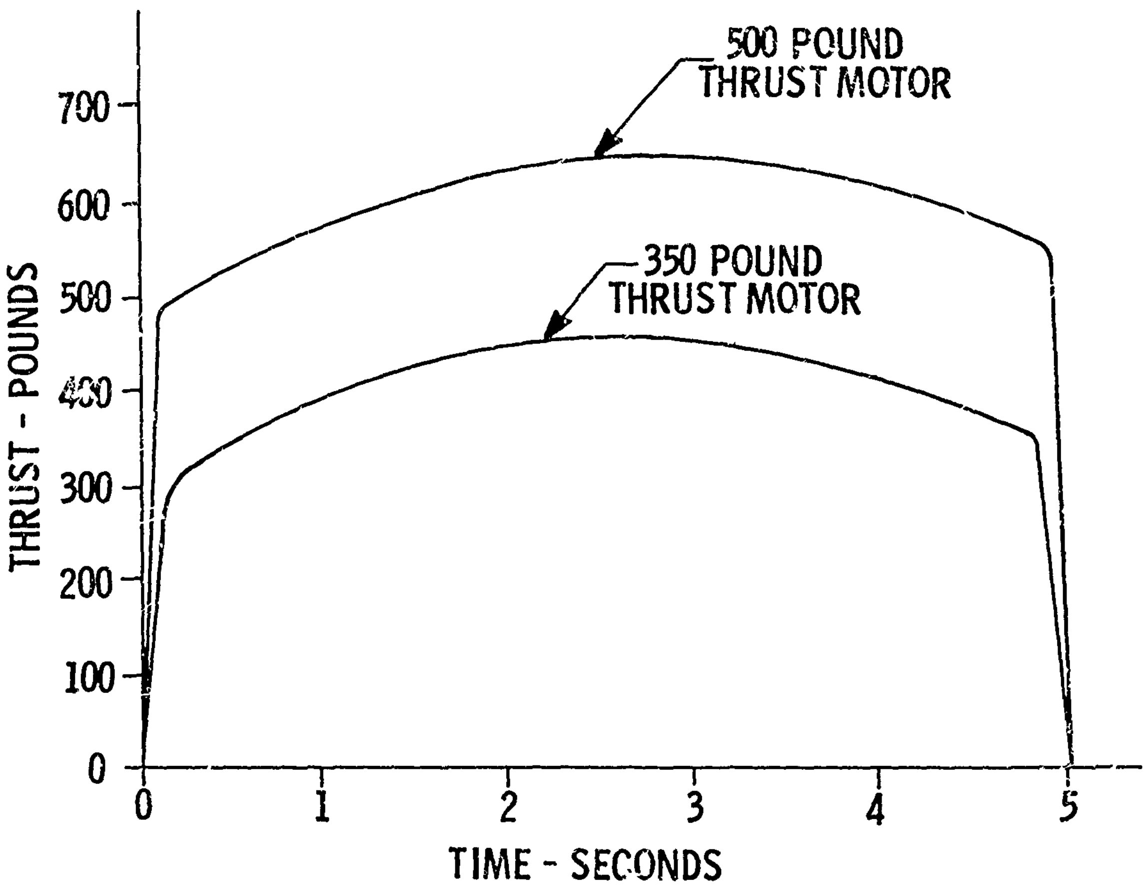


Figure 6. Typical Thrust Curves

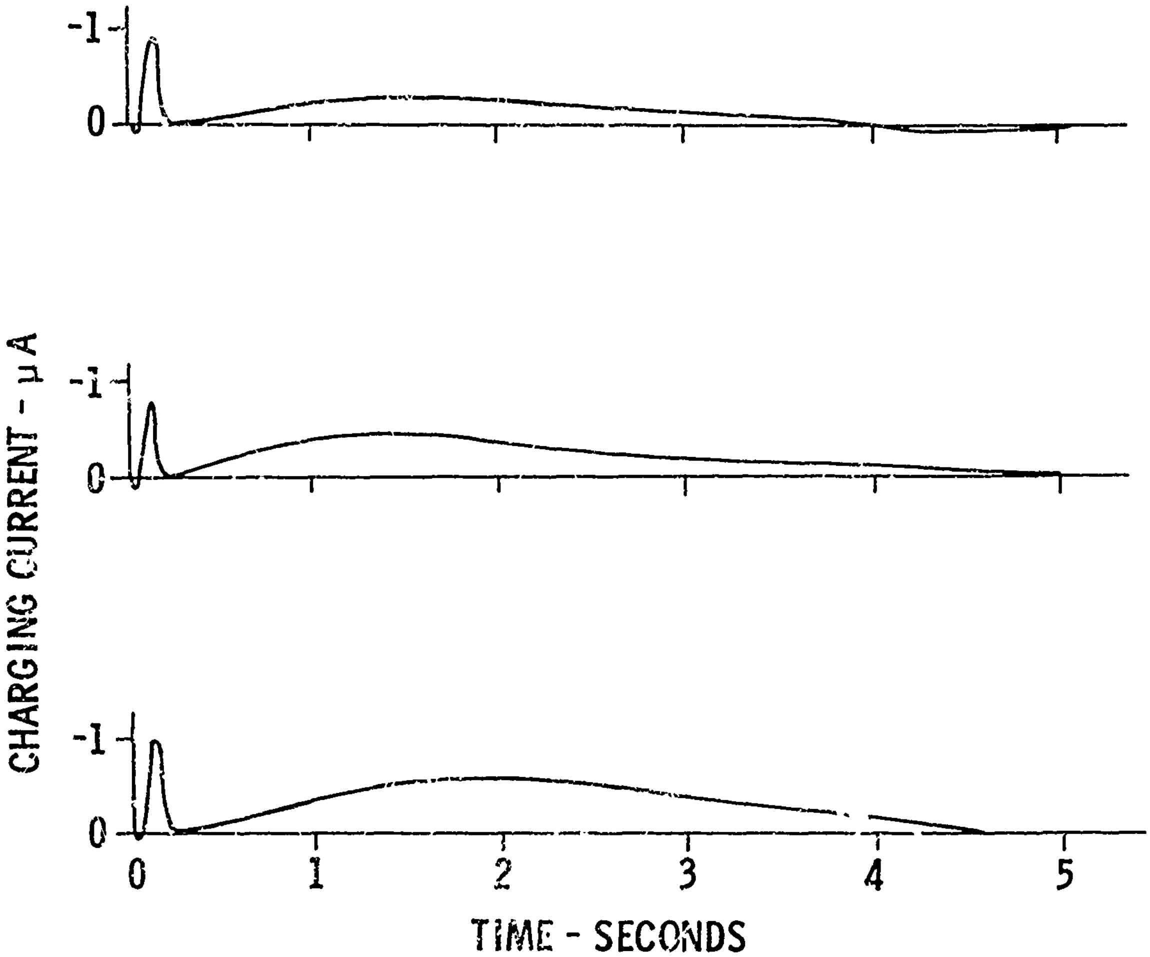


Figure 7. Charging Current Vs. Time for the 350-Pound Thrust Motors

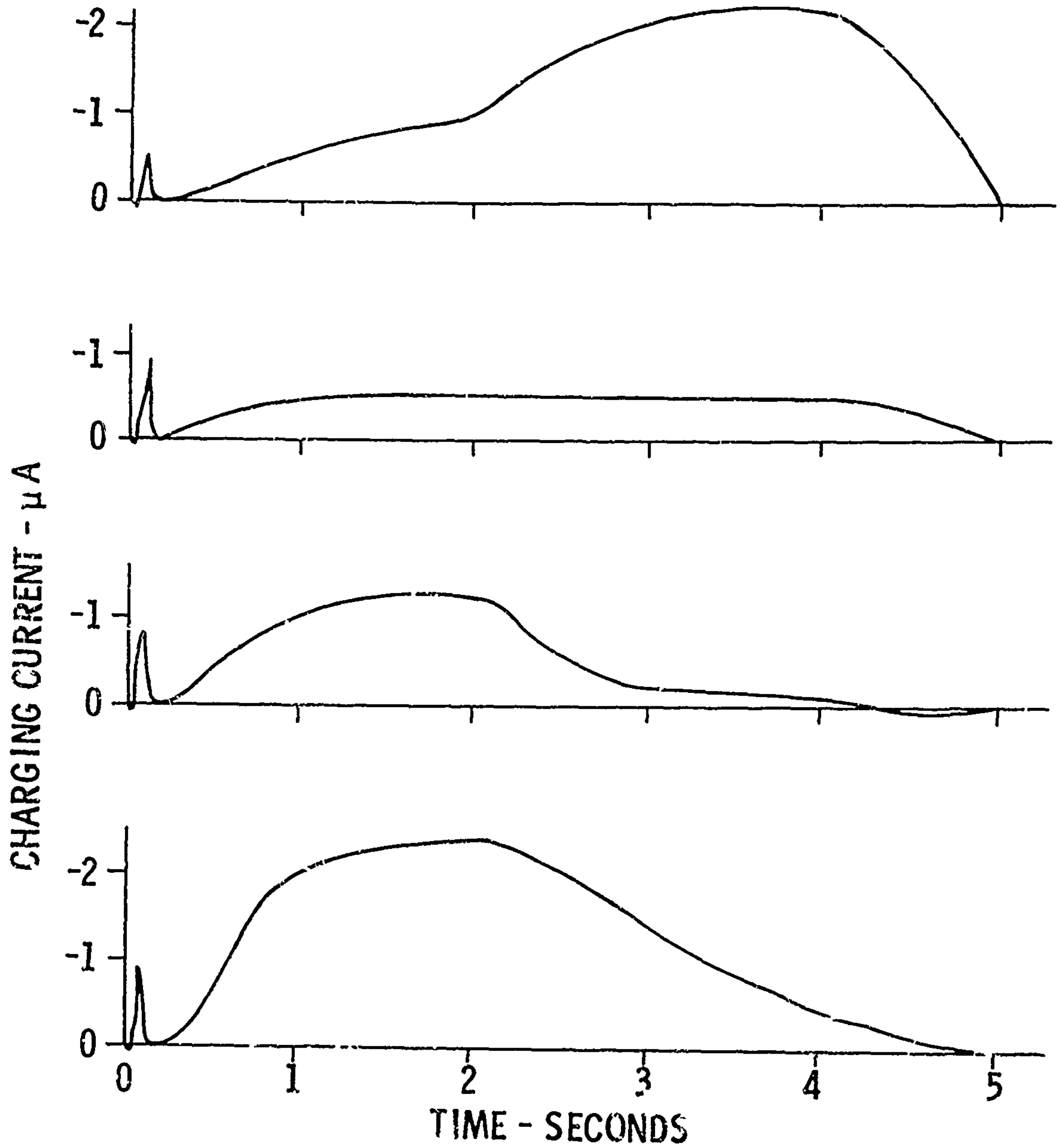


Figure 8. Charging Current Vs. Time for the 500-Pound Thrust Motors

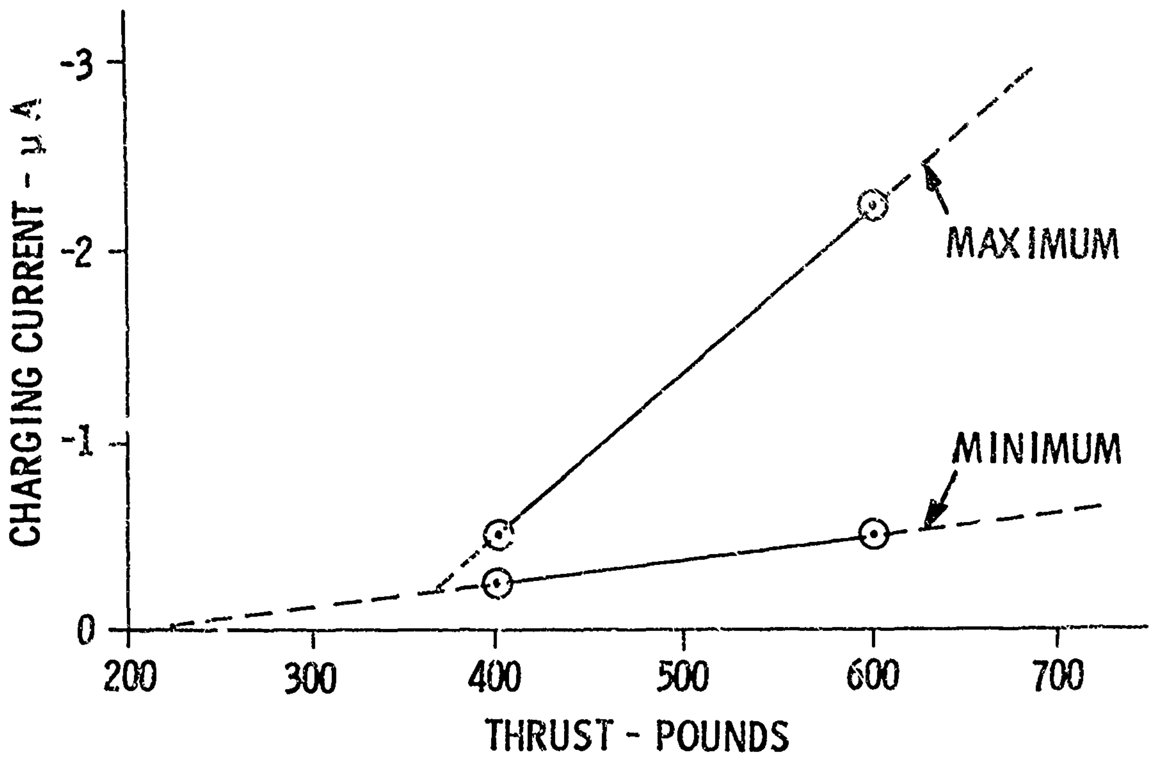


Figure 9. Charging Current Dependence on Thrust

motor. Unfortunately, it appears difficult if not impossible to design a meaningful ground-based laboratory experiment to determine this charging current-potential relationship.

In the past few years, a series of rocket motor charging experiments have been conducted by the Stanford Research Institute, under contract with the Air Force Cambridge Research Laboratories (Reference 4). These experiments used Nike-Cajun research rockets equipped with electrostatic field meters which were used to determine the effective electrostatic potential of the vehicles. The potential vs. altitude relationship is shown in Figure 10 for the second firing of this series. The rocket reached a maximum potential of 28,000 volts before Nike stage burnout. These tests served to demonstrate that high potentials can indeed be generated by rocket motors.

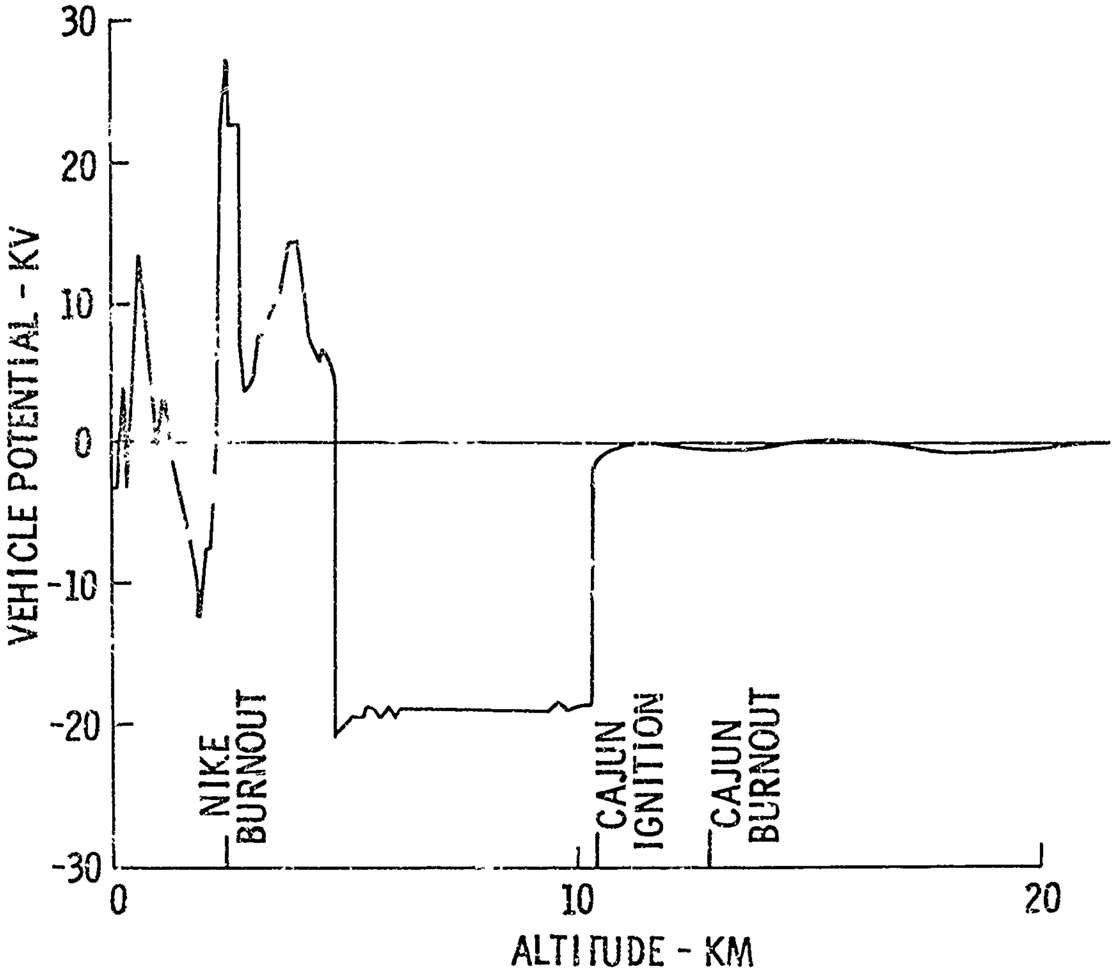


Figure 10. Vehicle Potential Vs. Altitude, Nike-Cajun Electrostatic Charging Experiment

RESULTANT DESIGN CHANGES

A number of design changes were made in succeeding flight test missiles flown from AFWTR to prevent recurrence of these failures with FTM 502 and FTM 503. Some of these changes which were later incorporated into the production missile design are described as follows.

(1) An electrical bond assured electrical continuity between the reentry vehicle structure and the downstage missile structure. Bond integrity is now required on all missiles, established by in-silo measurement of the resistance across the reentry vehicle/guidance section interface.

(2) Certain electronic circuits in the missile's guidance and control system were reworked to reduce sensitivity to electromagnetic interference. In addition, shielding of signal-carrying cables was enhanced. These changes greatly decreased the system's sensitivity to external electric discharges and virtually eliminated the possibility of an interstructural spark discharge.

(3) In order to provide even greater assurance against system failure, steps were taken to reduce the electrostatic charge accumulated by the missile. The object was to prevent the occurrence of an "uncontrolled" electrical discharge from a random location on the missile's exterior surface.

(4) Experimental devices designed to dissipate electrostatic charge were flown on the four succeeding test missiles. Three of these were corona discharge devices. A fourth missile carried tantalum wire thermionic emission devices mounted on the base heat deflectors of the first and second stage engines. All missiles in this sequence exhibited normal flight control throughout first and second stage engine operation.

Following the successful completion of this series of tests, corona discharge devices were designed and incorporated into the Minuteman missile configuration. The primary design requirements were:

(1) The corona discharge devices (static dissipators) must be adequate in number and individual performance to dissipate the charging current expected from the first and second stage rocket motor operation.

(2) The devices must be installed with minimum alteration of the existing missile design.

(3) The dissipators must be able to survive the extreme thermal and shock environment encountered in a silo missile launch.

Figure 11 shows the design of the static dissipators installed on the Minuteman missile. The dissipators were constructed of tantalum wire, since ductility was considered an advantage over the brittleness of tungsten material. Forty-five of these devices were installed on the base of the first stage rocket motor, and 21 on the base of the second stage motor. The dissipators were held in place, in electrical contact with the vehicle structure, by structural bolts already on the missile.

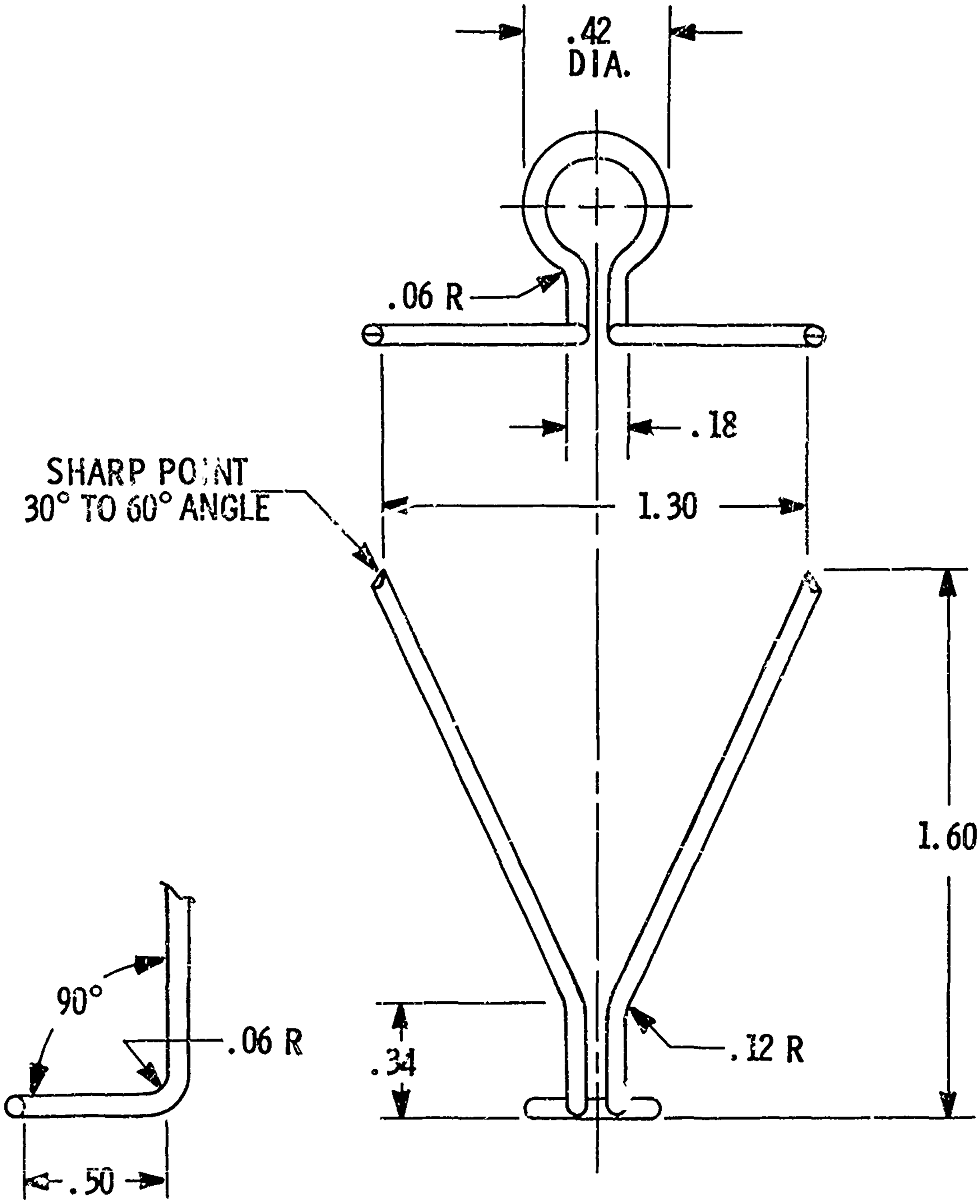


Figure 11. Minuteman I Corona Dissipators

The design changes that accompanied the Minuteman II missile led to a review of the design of the static dissipators. The injury hazard which the sharp corona discharge points presented to personnel working on and around the missile served as an incentive. Several incidents of puncture wounds caused by the dissipators were recorded.

The dissipators incorporated into Minuteman II differed from those on Minuteman I. A new "personnel safe" design was developed, as shown in Figure 12. Since this design had slightly lower current dissipation characteristics than that on Minuteman I, the number was increased to 58 and all were mounted at the base of the first stage motor.

Further, it was found that on Minuteman I, stage separation produced a ragged edge on the portion of the joint remaining with the second stage. Tests revealed that the corona current capacity of the edge exceeded that of the second stage dissipators; therefore, second stage dissipators were removed from the design of Minuteman II.

Design changes were incorporated in the missile configuration for Minuteman II. No failures of the type experienced by FTM 502 and FTM 503 have since been recorded.

## CONCLUSIONS

In-flight failures of Minuteman missiles demonstrated the necessity for giving proper attention to the possible effects of electrostatic charge accumulated by rocket powered vehicles. A series of tests verified that rocket motors could generate a net electrostatic charge on the vehicle.

In the case of the Minuteman missile, corrective measures included the following:

(1) Electrical continuity throughout the missile structure.

(2) Design changes to reduce the sensitivity of critical electrical/electronic circuits in the missile.

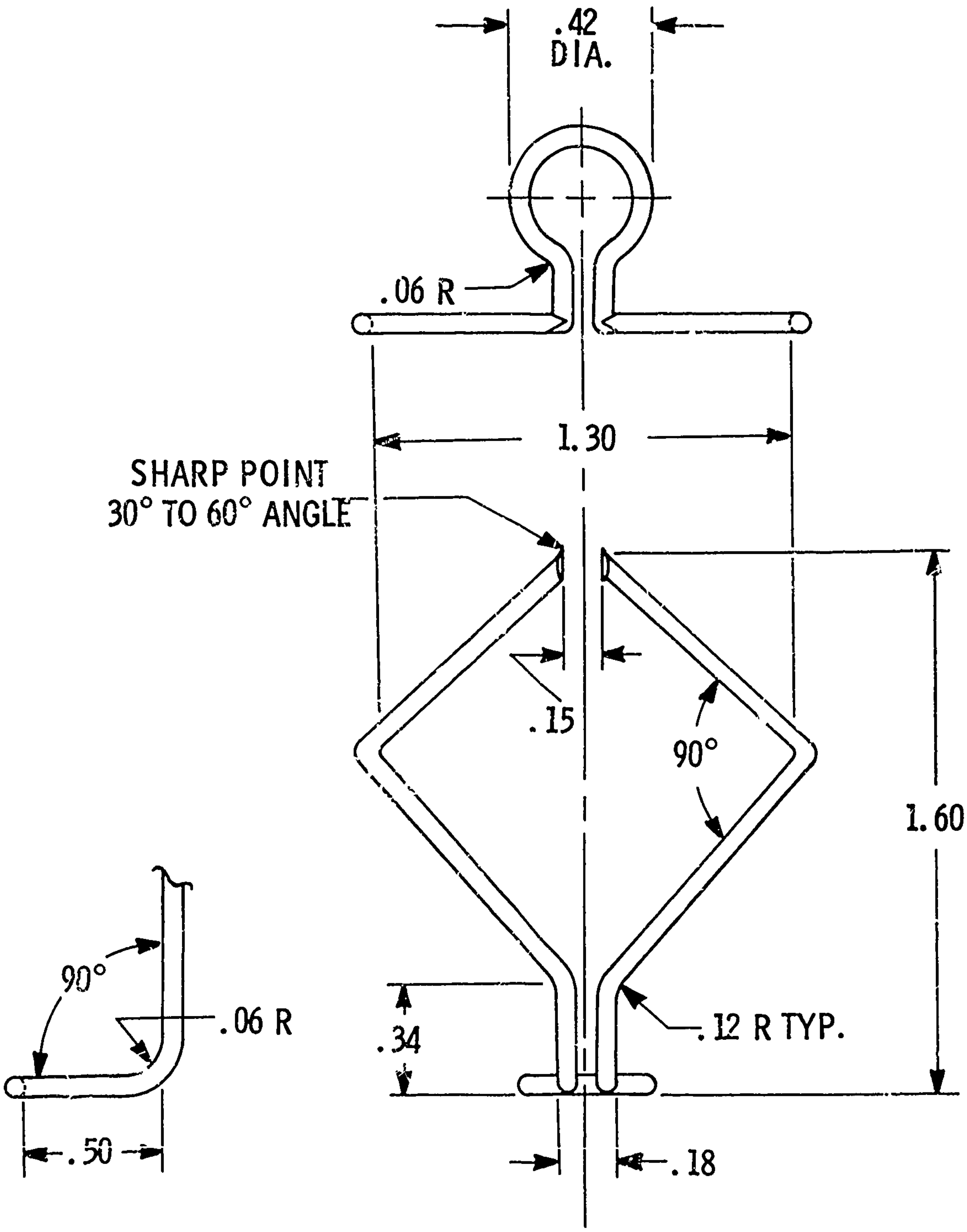


Figure 12. Minuteman II Corona Dissipators

(3) Installation of static dissipator devices on the structure to prevent uncontrolled external electrical discharges.

The measures taken to prevent recurrence of electrostatic discharge-induced failures have been successful.

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AFAL-TR-68-290
PART II

# TECHNICAL NOTE

# OBSERVATIONS OF THE FISSION OF CHARGED DROPLETS

Michael Kearney

Washington High School

Washington, Pennsylvania

This paper was presented at the Pennsylvania Junior Science and Humanities Symposium, November 4, 1968, at Pennsylvania State University, State College, Pennsylvania. It was not presented at the Lightning and Static Electricity Symposium, but the information is deemed relevant and therefore is appended here as a Technical Note.

## INTRODUCTION

An electrical charge is one of the basic forces of nature. One of the most obvious forms of an electrical charge is static electricity. It acts on particles which are free to move about in a manner similar to magnetism (like charges repel, unlike attract). In everday life one can see this force in action in thunderstorms and electrostatic air cleaners. In the field of science it is used in the acceleration of particles and the study of nuclear physics. In fact, the Nobel Prize winning physicist, Niels Bohr, and his associates developed an analogy between the splitting of a charged water droplet and the fission of an uranium atom (Reference 1).

An induced charge splits the droplet in my experiments. Because like charges repel, the droplet elongates; then, if the charge is strong enough, the surface tension, instead of drawing the droplet back together, splits it. Figure 1 is a series of photographs illustrating two cases - that in which the charge is adequate to split the droplet and that in which the charge is too small.

In a theoretical study of droplet fission made by Stanley Cohen and Wladyslaw Swiatecki, published in 1963 (Reference 2), they stated that, "In particular, we would like to stress explicitly that at the present time we do not know how an

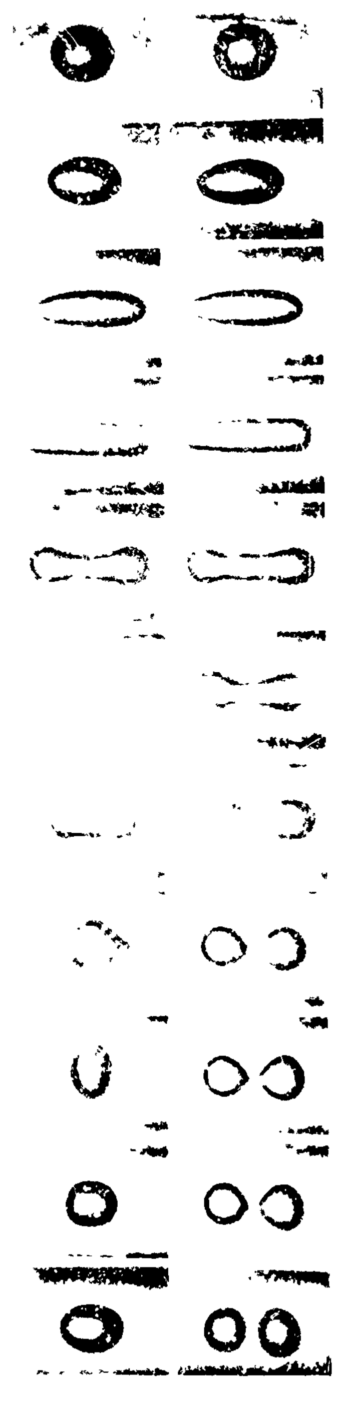

Figure 1. Division of Droplets

idealized drop would divide, whether into two or more parts. In cases when the division might be into two parts, we do not know whether it would be into equal or unequal parts."

In my previous work (Reference 3), I studied a number of variables involved in the fission process. During these experiments I noticed that often the droplet split into unequal parts. Finding that no study concerning the size ratio of the droplets resulting from fission had been made, I made this the object of this project.

## DESCRIPTION OF EXPERIMENT

### A. EQUIPMENT DESCRIPTION

I built a Van de Graff Static Generator for this project. This machine produced two types of arcing. One was a long feathery arc which could reach a distance of about 24 inches. The second arc was brighter, less feathery, and could reach a distance of about 7 inches. (References estimate the charge as about 75,000 volts/inch. This indicated the range of the machine was from 500,000 to 600,000 volts during the tests.)

The test cell was made of 3-1/2" I.D. x 12" Lucite tubing covered and sealed at one end (see Figures 2, 3). The electrodes were number 6 gauge copper wire and spaced so that a droplet could be placed between them (approximately 3/4"). The cell was filled with pharmaceutical grade mineral oil (Squibb).

A wooden jig for taking pictures of the fission process was constructed so that good pictures could be consistently taken. A light fixture holding a 100 watt incandescent bulb was placed behind the cell and the half of the cell facing the light was covered with several sheets of wax paper to reduce the glare and provide a good background for photographing the dark droplets. The charge conducting wires were secured to insulators to ensure stable electrical conditions. A camera was mounted on the opposite end of the jig and focused at the droplet 12 inches away.

During the experiments an Argus C-3 camera with a Plus-2 closeup lens was used. The f. aperture was 16 and the shutter speed was 1/100 of a second.

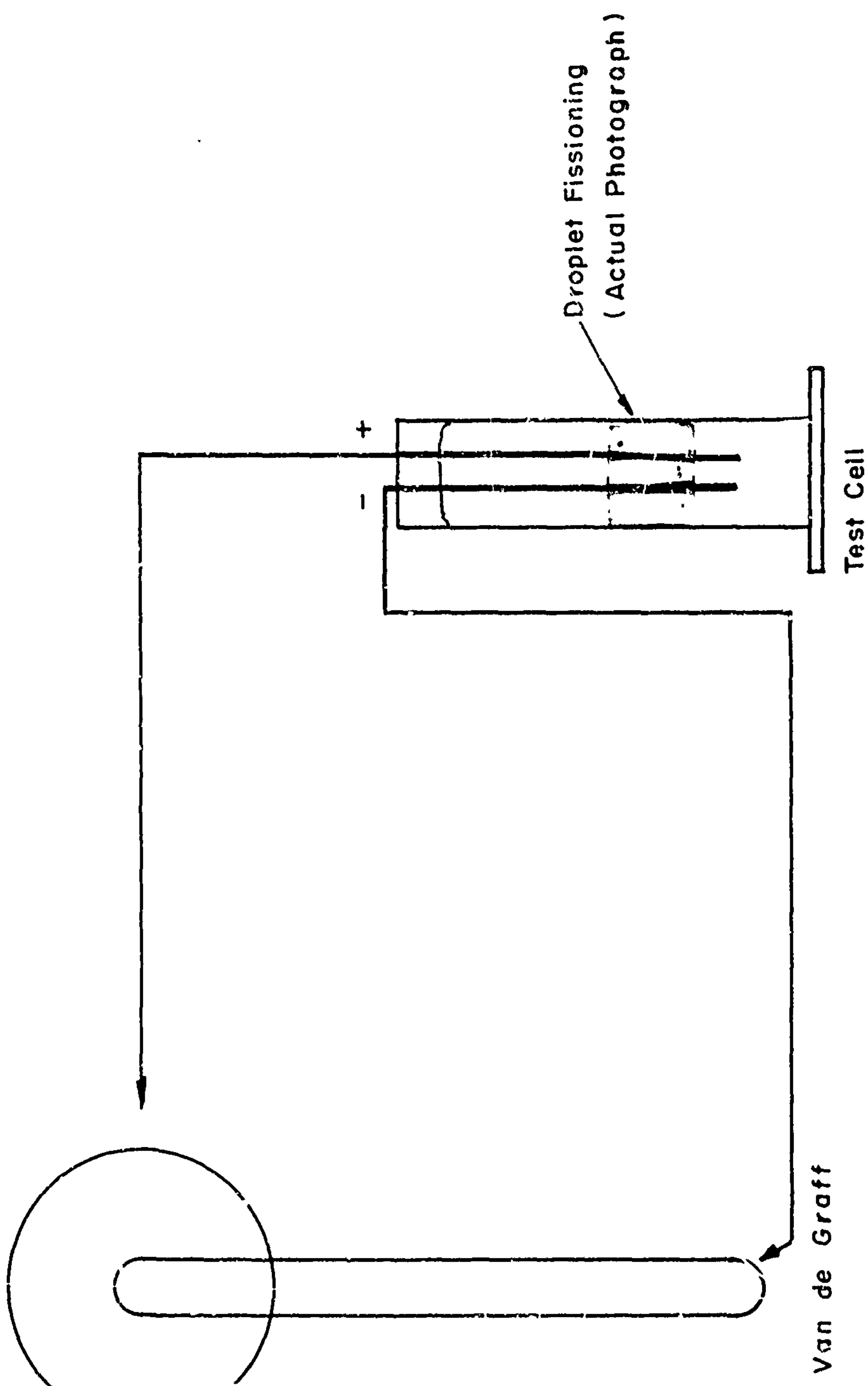


Figure 2. Schematic of Equipment Used in Experiments

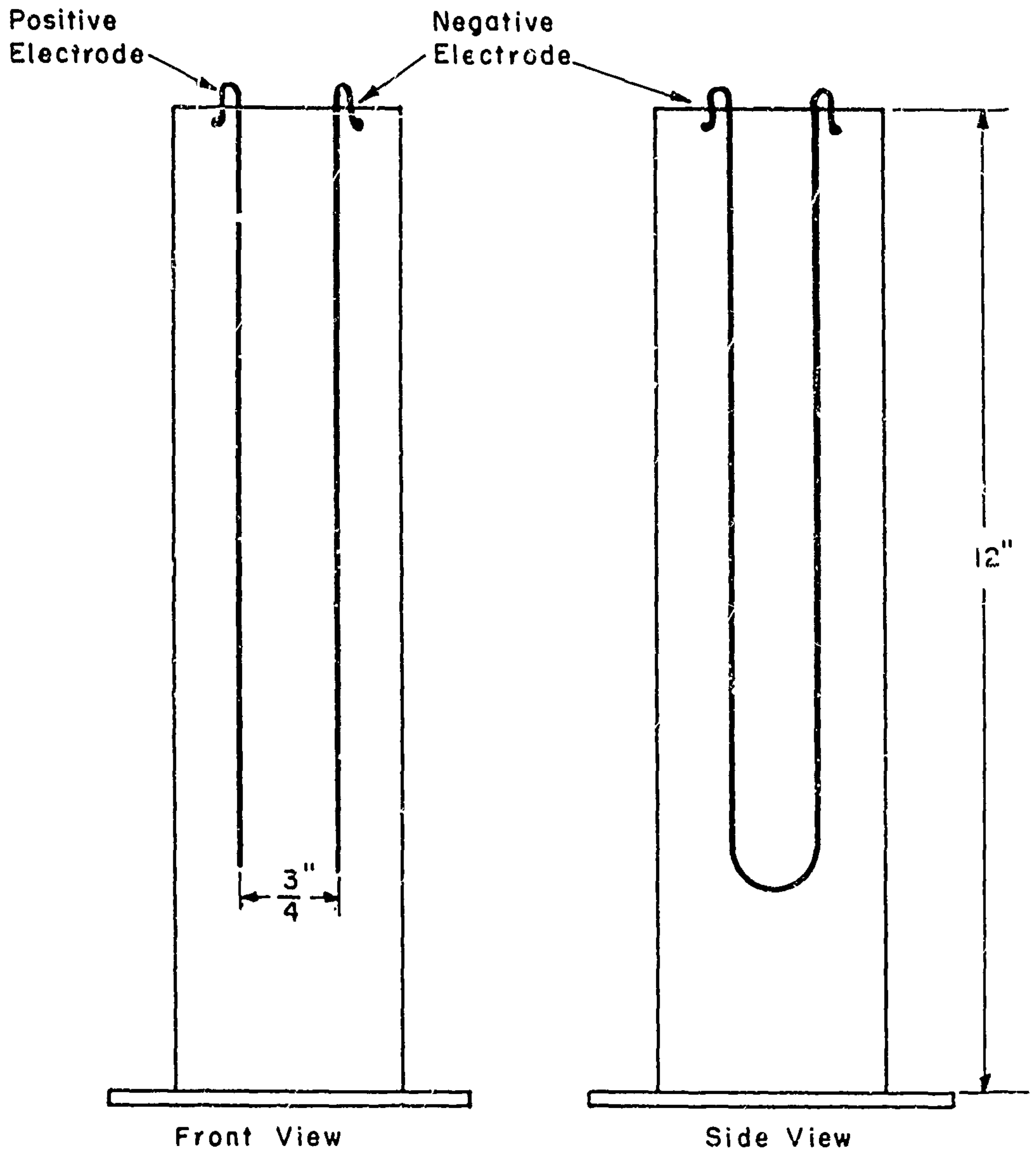


Figure 3. Detail of Test Cell

The camera was masked so that only one quarter of a frame was exposed at a time, thus making it possible to have a maximum of 80 to 144 pictures per roll of 20 to 36 frames, respectively. Kodak Plus-X Panchromatic film of 125 ASA was used in the tests.

## B. PROCEDURE

Each experiment was conducted in the following manner:

1. The equipment for the experiment was set up as described in The Equipment Section.

   a. The Van de Graff electrostatic charge output was tested to see if it was sufficient for fission to occur (the generator operated continuously throughout all tests).

   b. The camera was focused on the area to be included in the picture.

2. The test solutions were prepared:

   a. Distilled water
   b. Tap water
   c. One normal NaCl solution
   d. Detergent solution
   e. For photographic purposes, each solution was dyed with a small amount of food coloring.

3. The light behind the test cell was turned on.

4. A droplet of the test fluid, approximately 2 minims (1 minim equals .06 cc), was placed in the oil between the two electrodes.

5. When the droplet came into the area where the camera was focused and fission occurred, a picture was taken and the film rolled ahead one half of a frame.

6. Steps 4. and 5. were repeated until both halves of the film had been completely exposed. Then the film was developed.

7. The pictures were projected on a wall at a distance of about 12 feet.

8. The enlarged pictures of fission were then studied and any abnormalities were noticed and recorded in the laboratory notebook.

9. If the pictures showed a completed fission, as they often did, the size ratio of the larger drop to the smaller drop was measured (ratio of each droplet's average diameter) and recorded in my laboratory notebook.

10. A distribution graph of results for each solution was made comparing the size ratio of the larger droplet to the smaller droplet to its frequency of occurrence. These graphs are superimposed on Figure 4 for easy comparison of the results.

    (No chart was made showing the results of detergent solution because a droplet of this solution tended to completely break up into many smaller droplets which would have interfered when a number of tests were run consecutively.)

## C. RESULTS

A study of the results of the experiments showed that distilled water had a greater tendency to fission equally than the three solutions. To make certain that this was the case and not merely an unusual coincidence, the results were subjected to the statistical F test (Test for homogeneity of variance, Reference 4) as shown in Appendix I. The F value under my set of conditions was found to be 21. The critical value for this point, however, was 4.6 (at 1%); below this point no difference can be said to exist. With the high value of 21, it was proven that the differences shown on the graphs were significant and highly probable.

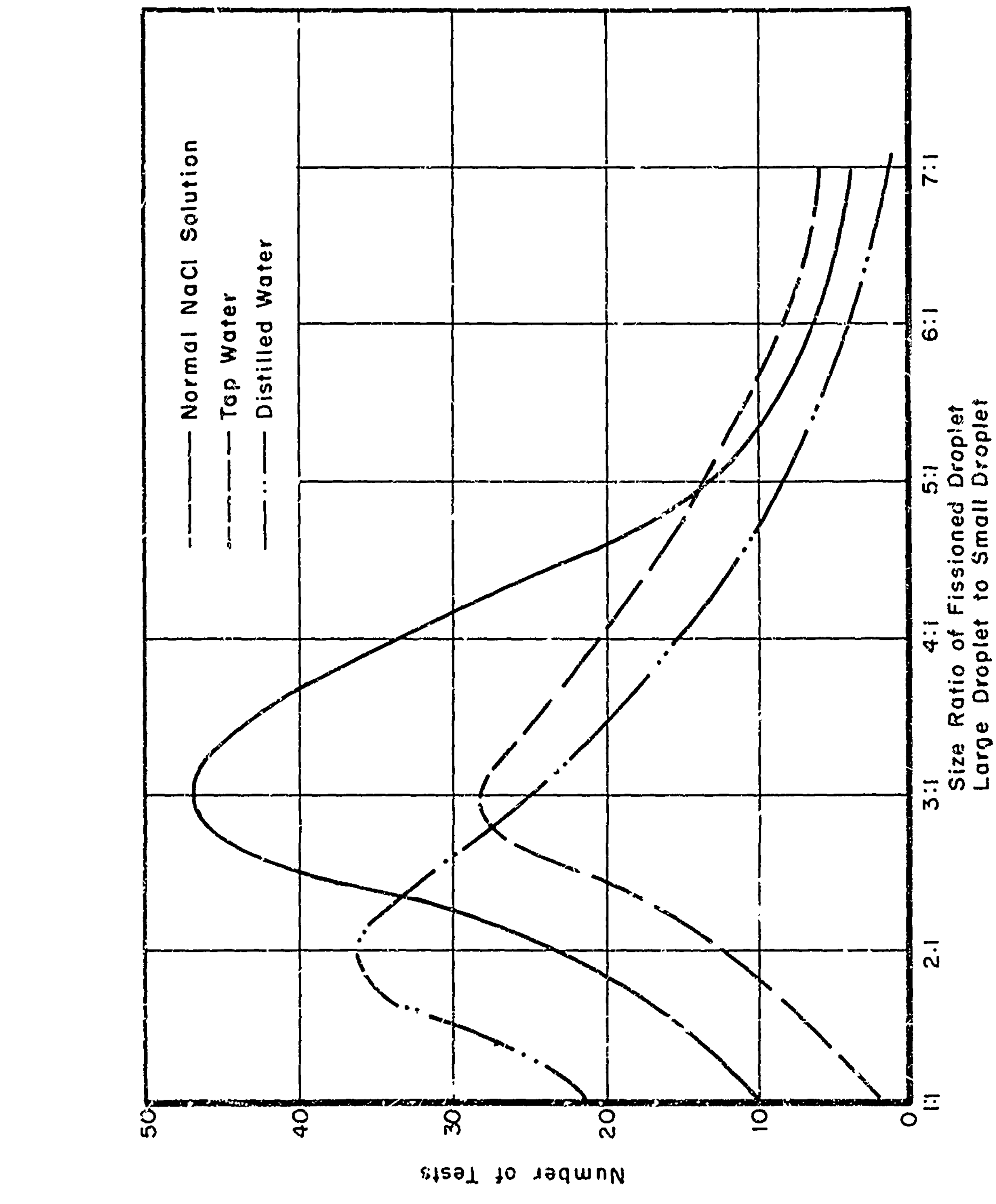


Figure 4. Graphed Results of Experiments of Size Ratio vs. Frequency

With the test results of these solutions, I could only surmise as to the cause of the difference. A logical clue concerning the difference may be found in the tests in which detergent solution was used. It is known that the detergent drastically reduces the surface tension of water. When a droplet of this solution was put under charge, a stream of very small droplets flowed from the droplet to the electrodes. It is also known that distilled water has greater surface tension than any simple water solution. Therefore, it may be concluded that as impurities were added to distilled water its surface tension decreased, thus accounting for the difference in the size distribution curves and also, possibly, for the fact that the photographs of distilled water rarely showed an equal fission, although it occurred more frequently in distilled water than in the solutions, the dye in the water must also be considered an impurity.

It would be desirable to experimentally relate actual surface tension measurements to the size distribution ratio of fissioning droplets. However, more sophisticated equipment, well beyond the realm of that used, would be required to do this.

D. CONCLUSIONS

1. The ratio of large to small droplets formed by the fission or splitting of a single droplet of test solution varied according to the test solution used.

| SOLUTION | RATIO OF LARGE TO SMALL DROPLETS |
|---|---|
| Distilled Water | 2.78 : 1 |
| 1 Normal NaCl Solution | 3.41 : 1 |
| Tap Water | 3.80 : 1 |

2. As the surface tension of a droplet decreases, the probability that an unequal size ratio will occur in the fission becomes greater. The surface tension of the droplet appears to be the primary factor controlling the size ratio distribution of the fission of a droplet of water.

E. RECOMMENDATIONS

My work up to now has dealt with charged water droplets suspended in oil. Because oil is relatively more viscous than other common liquids, it provided a means for studying the fission of a water droplet in slow motion. With this work as background, I would now like to study charged droplets in air with possible application toward a better understanding of weather conditions. To do the experimental and laboratory work, I am building a second Van de Graff generator of an opposite potential with the idea that in the electrostatic field developed between the two Van de Graffs I could easily study the total effects of a charge on many droplets. (Originally, I had planned to do preliminary work on this project this past summer; however, I became involved in the research proceedings at Carnegie-Mellon University concerning fog formation.)

I hope to relate my lab work with actual field studies of weather. In particular, in the mountains of Northern New Mexico (Boy Scout Ranch at Philmont), thunderstorms occur with remarkable regularity each afternoon. Often, when on top of a mountain there, one can actually feel a potential charge though there is no storm activity. If it is possible to get some financial backing, I would like to study the developing thunderstorm and the storm itself using special photographic techniques and relate this to electrostatic charge measurements made on mountain tops in the area at the same time.

I believe that such a comparison of experimental and field work could prove interesting and may add something to the understanding of weather.

In the experiments I made studying detergent water solutions, I noticed an interesting phenomenon. The many small droplets which had broken off the larger test droplet revolved in a vortex-like manner between the electrodes. This vortex spun rapidly despite the viscosity of the oil, suggesting that the droplets drove the oil.

Physicist Vernon Rossow of Ames Research Center in California produced a vortex in air in a similar manner as my experiments in oil. With this study, he suggested that the motion of charged droplets could possibly be a mechanism in causing tornadoes. I believe that this hypothesis can be taken one step further to explain the tremendous air currents in thunderstorms. Differences in the arrangement of the electrostatic field could account for the difference in the weather phenomena, as well as the fact that tornadoes sometimes develop from thunder clouds.

Another interesting area of weather causes and effects to think about concerns the reasons why our most violent storms -- thunderstorms, tornadoes, and hurricanes -- occur frequently during the summer but rarely in the winter. The accepted explanation relates the direct rays of the sun in summer to more heat absorbed by the earth and atmosphere. This heat, in turn, causes a sequence of events resulting in occasional storm activity. If, in fact, staticly charged fields existed between the earth and sun, this field would also be most direct and, thus, most intense in the summer. The absence of this direct field during the winter could explain the absence of the storms.

I believe that electrostatics play a far more important part in weather conditions and disturbances than most give credit to. I am not rejecting theories using air currents as the explanation, for they have been proven to be important. I think, however, that more research concerning electrostatics in thunderstorms would aid in weather prediction and control.

PROPOSED STUDY PLAN

1. Background study and research of literature and electrostatics.
2. Statistical study relating the frequency of "electrical" storms (thunderstorms, tornadoes, hurricanes) to geographical location and time of the year.
3. Experimental study of vortex in air similar to the study of Dr. Rossow but on a larger scale so that variables such as air temperature and pressure could also be studied. (My two Van de Graff generators would be used in this study.)
4. Field study in New Mexico mountains to relate actual daily thunderstorm activity to electrostatic charge measurements (in cooperation with the Boy Scouts of America - Philmont Scout Ranch).

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# APPENDIX

# F-TEST FOR HOMOGENEITY OF VARIANCE

## F-TEST FOR RESULTING STATISTICS FOR EXPERIMENTS

| | Exp. 1 y, x | Exp. 1 y x x | Exp. 1 y x x² | Exp. 2 y, x | Exp. 2 y x x | Exp. 2 y x x² | Exp. 3 y, x | Exp. 3 y x x | Exp. 3 y x x² |
|---|---|---|---|---|---|---|---|---|---|
| A. Tabulation of the Results of Tests | 3 1 | 3 | 3 | 22 1 | 22 | 22 | 10 1 | 10 | 10 |
| | 11 2 | 22 | 44 | 36 2 | 72 | 144 | 21 2 | 42 | 84 |
| | 27 3 | 81 | 243 | 22 3 | 66 | 198 | 47 3 | 141 | 423 |
| | 17 4 | 68 | 272 | 18 4 | 72 | 288 | 33 4 | 132 | 528 |
| | 15 5 | 75 | 375 | 8 5 | 40 | 200 | 11 5 | 55 | 275 |
| | 9 6 | 54 | 325 | 3 6 | 18 | 108 | 7 6 | 42 | 252 |
| | 2 7 | 14 | 98 | 3 7 | 21 | 147 | 5 7 | 35 | 245 |
| B. Total y x x | | 317 | | | 311 | | | 457 | |
| C. Total $y \times x^2$ | | | 1360 | | | 1102 | | | 1187 |
| D. Total y | 84 | | | 112 | | | 134 | | |
| E. $\frac{(\text{Total } y \times x)^2}{\text{Total } y}$ | | | 1196 | | | 866 | | | 1559 |
| F. $(\text{Total } y \times x) - \frac{(\text{Total } y \times x)^2}{\text{Total } y}$ | | | 164 | | | 241 | | | 258 |

G. Total of no. in F 663

H. (84-1) + (112-1) + (134-1) = 327 D. F. (Degrees of Freedom)

I. 663 ÷ 327 = 2. (Variance)

| | Experiment 1 | Experiment 2 | Experiment 3 |
|---|---|---|---|
| J. Average value for x | 3.80 | 2.78 | 3.41 |
| K. (Total y x) | 100489 | 97021 | 206849 |
| L. $\frac{(\text{Total } y\ x)}{\text{Total } y}$ | 1196 | 866 | 1559 |
| M. Sum of Total y x x | 1985 | | |
| N. Square to Total y x x | 1177225 | | |

O. Sum of Total y 330

P. $\frac{1177225}{\text{Total } y} = 3537$

Q. Sum of results in part L 3621

R. $\frac{\text{No. in part Q.} - \text{No. in part P.}}{(\text{No. of Exp.}) - 1} = \frac{3621-3537}{3-1} = 42$ (2D. F.)

S. $\frac{42}{\text{Variance}} = 21$

# SESSION III

## FUELS AND FLUIDS

Chairman, T. G. Horeff,

Federal Aviation Agency

# AIRFLOW VELOCITY EFFECTS ON LIGHTNING IGNITION OF AIRCRAFT FUEL VENT EFFLUX

James R. Stahmann

Lightning & Transients Research Institute

The addition of airflow over the surface of an aircraft wingtip permits a more realistic determination of the probability of lightning ignition of an explosive fuel vent efflux.

In order to produce a high velocity air flow over the surface of an aircraft wingtip, with the wingtip at high voltage, a mockup of the wingtip was mounted on top of the ten million volt generator located aboard the Research Vessel THUNDERBOLT, as shown in the two views of Figure 1. The high voltage generator was modified to also serve as the main duct of the wind tunnel. Normally there are two columns of capacitors in the generator with an aisle in between. One capacitor column was removed so that, along with the aisle space, adequate room was provided for the relatively low velocity air flow at large cross section. The tunnel tapered to a 12 x 20 inch rectangular opening at the top of the tunnel to direct high velocity air over the vent scoop. A curved plexiglas wall was installed to serve as a cover for the capacitor column and to break up any boundary layer which might develop and tend to reduce the effective cross sectional area of the tunnel. The air velocity was produced by an aircraft propeller located at the base of the high voltage generator. The propeller was driven by one of the ship's main propulsion motors through two right angle gear boxes. The maximum power available to drive the propeller system was about 600 horsepower.

With this setup, air flow velocities of the order of 90 knots made ignition, under conditions of most vulnerable fuel vent efflux mixture and flow rates, very difficult. Air flow was thus established as a major factor in the ignition probability. Basically, a mixture which will allow flame propagation in the vent may be leaned out so much by the air flow that it cannot ignite at the vent outlet. On the other hand, if the mixture is made rich enough to ignite, it may not be able to support propagation into the vent tube. A typical stroke to the edge of the

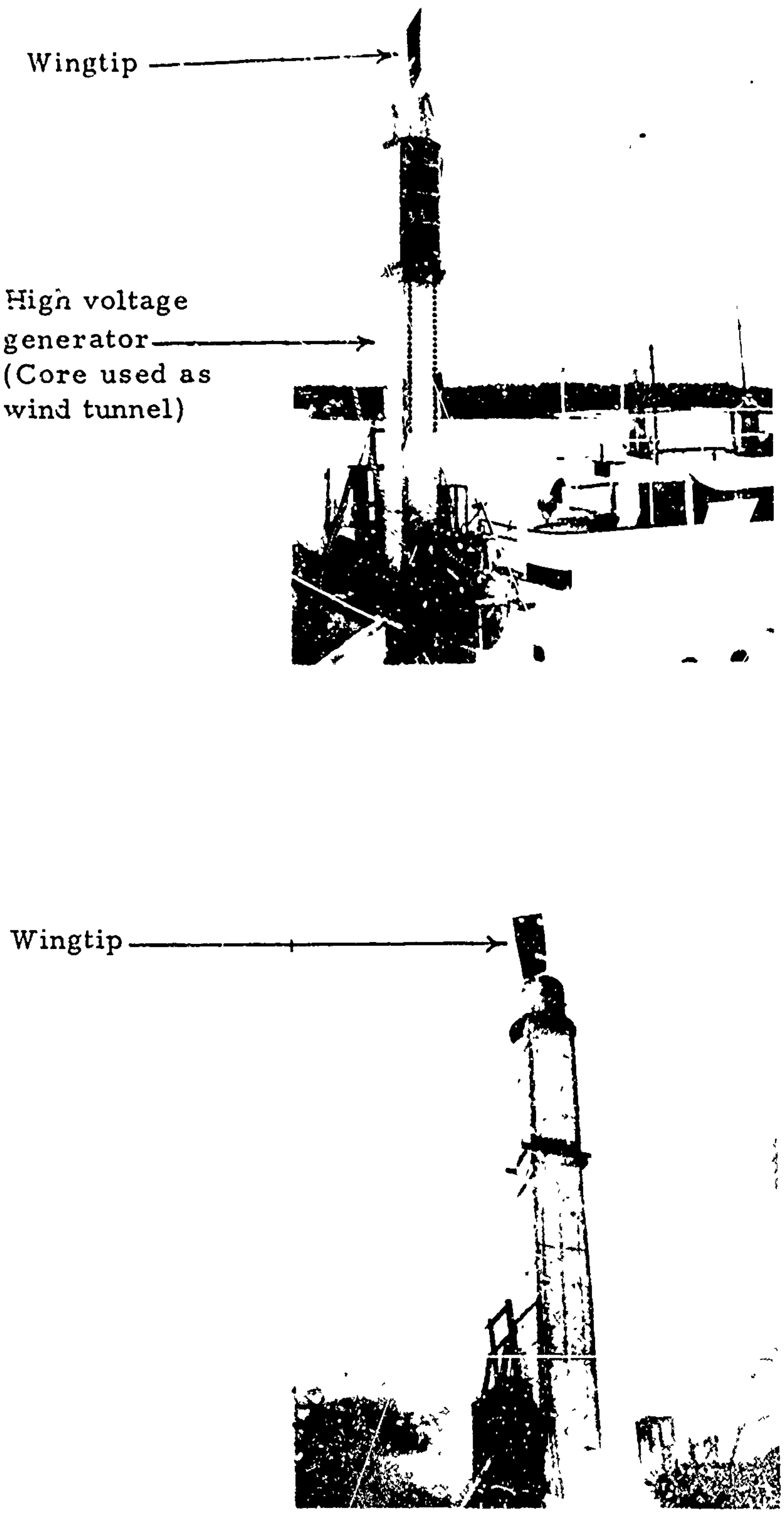


Figure 1. Two Views of Combination High Voltage Generator-Wind Tunnel in Erected Position

vent scoop of a recessed type vent is shown in Figure 2 with the current waveform shown in Figure 3. The oscillatory waveform was used to obtain a maximum peak current.

## FUEL VENT EFFLUX IGNITION MEASUREMENTS

Flowmeters were used to establish known fuel-air ratios and various flow rates. As pointed out in Reference 2, "The fuels used in jet aircraft are usually paraffinic hydrocarbons. The nature of the combustion process in the vapor of these compounds is such that their pertinent combustion properties are nearly independent of their molecular weight. For example, the heat of combustion per gram, flame temperature, quenching distance, burning rate, and lean limit of combustion for the various paraffins are nearly equal. For this reason, it is possible to employ any of the higher homologs of the paraffin series." Thus, propane ($C_3H_8$), which is readily available and easily manipulated in the gaseous state, was used in this program. JP-4 and kerosene were also used to a limited extent with a special carburetor for mixing with air. A blowout bag was provided on the pipe connected to the vent surge tank to help relieve the pressure of a sudden explosion. The vent outflow velocities selected were 1, 2, 5, and 10 fps, and the mixtures used were proportioned for 0.5, 1.0, and 1.5 stoichiometric by volume. The airflow velocity was measured with a pitot tube.

The measurement results are summarized in Table I. For control studies to check the mixture validity, the mixtures were first ignited at the selected flow rates by a spark discharge placed in the vent outlet. The results with airflows of 0 and 50 knots were tabulated in the first column of Table I. Most reliable ignition was obtained at 1.0 or 1.5 stoichiometric with outflows of 5 to 10 fps.

Simulated strokes to the edge of the vent scoop showed that the most vulnerable mixture and flow rate were 1.5 stoich and 5 fps. Combining the 1.0 and 1.5 stoich mixtures at the 50 knot airflow, one ignition out of 33 shots, was obtained at an efflux rate of 5 fps, giving a sample probability of 3%. No ignition was obtained at first with a 90 knot airflow and 5 fps, 1.0 to 1.5 stoich, but later the first shot under these conditions caused an explosion in the vent and surge tank which severely damaged the vent tube outlet box, as shown in Figures 4 and 5; Figure 4

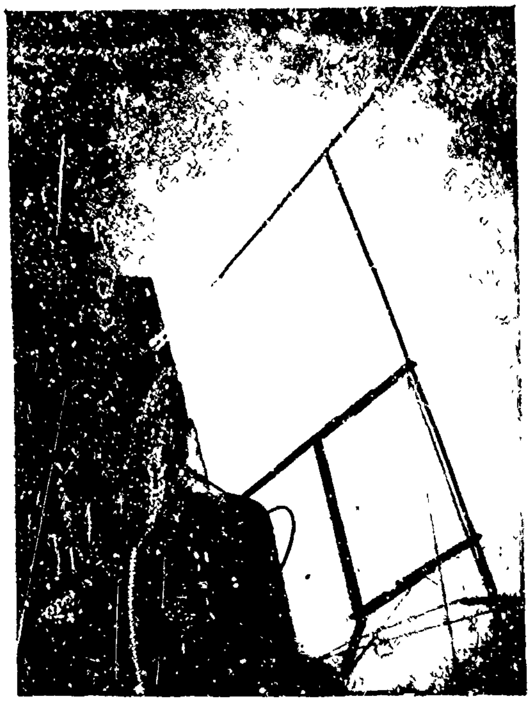

Figure 2. Stroke Directed to the Trailing Edge of a Vent Scoop

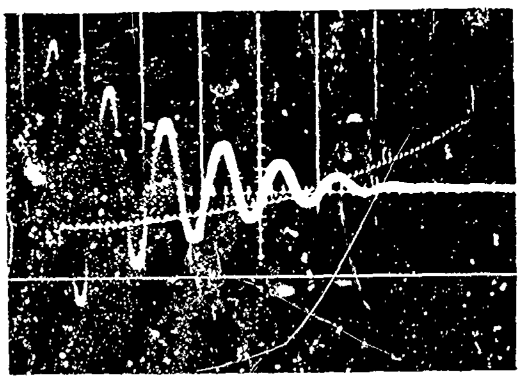

Figure 3. Typical Current Oscillogram of Current at the Top of the Generator. (Scale is 20 kiloamperes and 5 microseconds per large division.)

TABLE I

SUMMARY OF DISCHARGE TEST RESULTS

| Air Flow (knots) | Mixture (Stoich by volume) | Vent Outflow (fps) | No. of Shots | No. of Ignitions | Notes |
|---|---|---|---|---|---|
| 0 | 0.5 | 1 | 1 | 1 | Spark discharge in vent outlet |
| | | 2 | 7 | 1 | |
| | | 5 | 1 | 1 | |
| | | 10 | 4 | 0 | |
| | 1.0 | 1 | 3 | 1 | |
| | | 2 | 3 | 2 | |
| | | 5 | 2 | 2 | |
| | | 10 | 2 | 2 | |
| | 1.5 | 1 | 6 | 1 | |
| | | 2 | 8 | 2 | |
| | | 5 | 1 | 1 | |
| | | 10 | | | |
| 50 | 1.0 | 1 | 2 | 0 | " |
| | | 2 | 2 | 0 | |
| | | 5 | 2 | 2 | |
| | | 10 | 2 | 2 | |
| 0 | 0.5 | 1 | 1 | 0 | Simulated stroke to trailing edge of fuel vent scoop. |
| | | 2 | 1 | 0 | |
| | | 5 | 1 | 0 | |
| | | 10 | 1 | 0 | |
| | 1.0 | 1 | 4 | 0 | |
| | | 2 | 7 | 1 | |
| | | 5 | 11 | 5 | |
| | | 10 | 5 | 0 | |
| | 1.5 | 1 | 1 | 0 | |
| | | 2 | 1 | 0 | |
| | | 5 | 12 | 12* | |
| 50 | 0.5 | 1 | 1 | 0 | " |
| | | 2 | 1 | 0 | |
| | | 5 | 1 | 0 | |
| | | 10 | 1 | 0 | |
| | 1.0 | 1 | 1 | 0 | |
| | | 2 | 1 | 0 | |
| | | 5 | 16 | 1 | |
| | | 10 | 1 | 0 | |
| | 1.5 | 1 | 1 | 0 | |
| | | 2 | 1 | 0 | |
| | | 5 | 17 | 0 | |

*Most vulnerable condition.

TABLE I (CONTD)

| Air Flow (knots) | Mixture (Stoich by volume) | Vent Outflow (fps) | No. of Shots | No. of Ignitions | Notes |
|---|---|---|---|---|---|
| 90 | 0.5 | 1 | 1 | 0 | Simulated strokes to trailing edge of fuel vent scoop |
| | | 2 | 1 | 0 | |
| | | 5 | 1 | 0 | |
| | | 10 | 1 | 0 | |
| | 1.0 | 5 | 15 | 0 | |
| | 1.5 | 5 | 16 | 1** | |
| 200 | 1.15 | 5 | 60 | 0 | " |
| 200 | 1.5 | 5 | 140 | 0 | |
| 250 | 1.15 | 5 | 5 | 0 | |
| 250 | 1.5 | 5 | 5 | 0 | |
| 90 | 1.15 | 5 | 7 | 4 | Swept strokes |
| 90 | 1.5 | 5 | 8 | 7 | |
| 100 | 1.15 | 5 | 5 | 0 | |
| 100 | 1.5 | 5 | 8 | 6 | |
| 125 | 1.15 | 5 | 5 | 0 | |
| 125 | 1.5 | 5 | 5 | 1 | |
| 150 | 1.15 | 5 | 6 | 0 | |
| 150 | 1.5 | 5 | 5 | 0 | " |
| 200 | 1.15 | 5 | 28 | 1 | |
| 200 | 1.5 | 5 | 18 | 1 | |
| 250 | 1.5 | 5 | 2 | 0 | |
| 0 | 1.5 | 5 | 10 | 0* | Streamers off wing-tip and vent at $5 \times 10^6$ volts. |

* Most vulnerable condition.
** Severe explosion.

shows the undamaged box for reference. It was not possible to repeat this type of explosion during our first program, but a similar explosion occurred during a later program, where the vent outlet box was again damaged and an inspection cover was blown off the simulated surge tank. The measured flame speed was in excess of 1000 fps. Such a high flame speed may be due to an interaction of the flame front with the pressure and turbulence of the lightning blast wave. The wave may come from the pressure wave of the stroke or from turbulence or "organ pipe effect" (Reference 2) set up by the high speed airflow over the vent

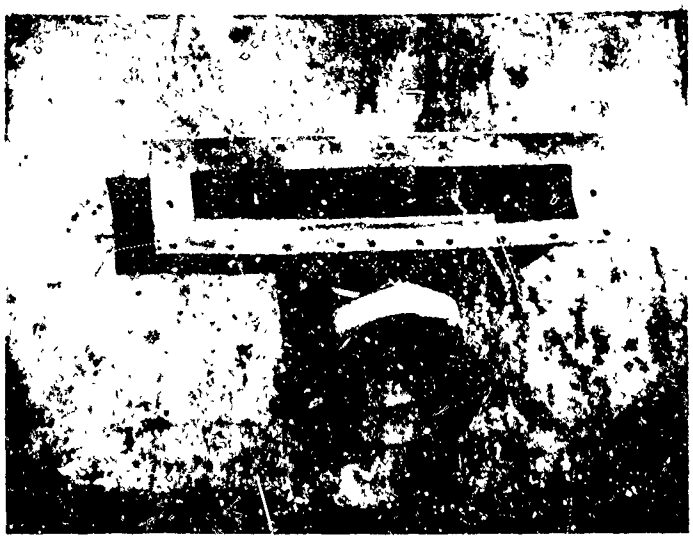

Figure 4. Normal Vent Outlet Before Explosion

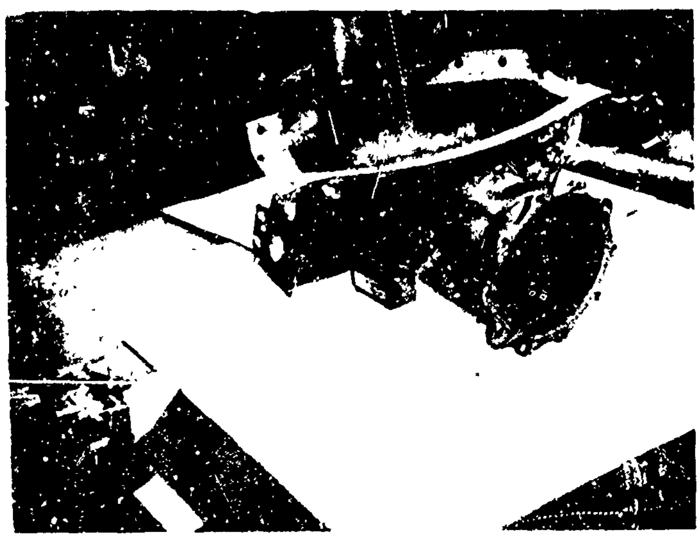

Figure 5. Vent Tube Outlet Slot After Explosion

scoop. More study in this area is indicated to determine the mechanisms involved and to reproduce the high flame front velocities. Vent tube modifications or other measures required to reduce the maximum possible flame speed in the vent tube could then be worked out. The maximum flame speed should be low enough to be easily stopped by available flame arresters or detector-suppressors. With simulated strokes to the trailing edge of the fuel vent scoop, no ignition was obtained at the higher airflow velocities of 200 and 250 knots.

To obtain airflow velocities up to 275 knots, corresponding to the landing speeds of many aircraft, a special wind tunnel was constructed on the Research Vessel THUNDERBOLT, as shown in Figures 6 and 7. The wind tunnel consists of three blowers in series driven by three 250-horsepower motors. Airflow velocities up to 275 knots have been produced with a rectangular output opening of 5 x 33 inches.

When an aircraft is struck by a long duration lightning flash, some of which last for almost a second, the aircraft essentially flies through the lightning channel. An aircraft flying at 600 mph (or 880 fps) can be contacted by a stroke, and the stroke swept along the entire length of the aircraft would leave a series of pit marks. By using the wind tunnel airflow to sweep an artificial lightning stroke over a fuel vent, it is possible to duplicate and study such "swept stroke" effects in the laboratory. The results of one such study are also tabulated in Table I. Ignition was possible with an airflow of 200 knots under swept stroke conditions. To obtain swept strokes, the duration of the current pulse was increased from 50 microseconds to about 2 milliseconds. A typical short duration stroke is shown in Figure 8 and the swept stroke is shown in Figure 9. Preliminary fuel vent investigations, using the current waveform shown in Figure 10, showed that two ignitions could be obtained out of 46 strokes in an airflow velocity of 200 knots. From the swept stroke photograph (Figure 9), it is evident that the plasma is swept across the fuel vent slot. The ionization is produced in the moving air so that the same relative velocities are produced as in the in-flight case where the aircraft is moving through the ionization.

Special sensors were installed in the fuel vent tube to measure flame front velocities. One of the measurements, at a 200 knot wind-stream velocity, showed

Figure 6. Control Panels for Three 2300 Volt, 250 Horsepower Pan Am Test Cell Blowers Used in Series for a 250 Knot Wind Tunnel.

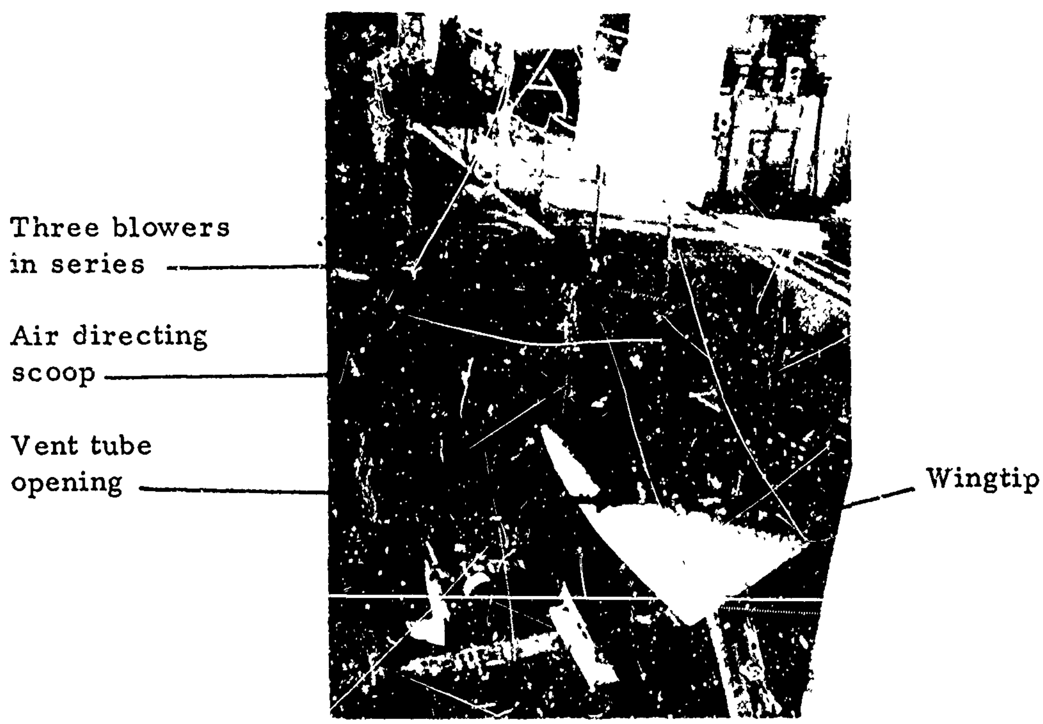


Figure 7. Wingtip Section Containing a Recessed Type Fuel Vent Installed at the Mouth of the Wind Tunnel. (Airflow velocities up to 275 knots were obtained.)

Figure 8. Typical Short Duration Stroke to the Edge of the Vent Scoop

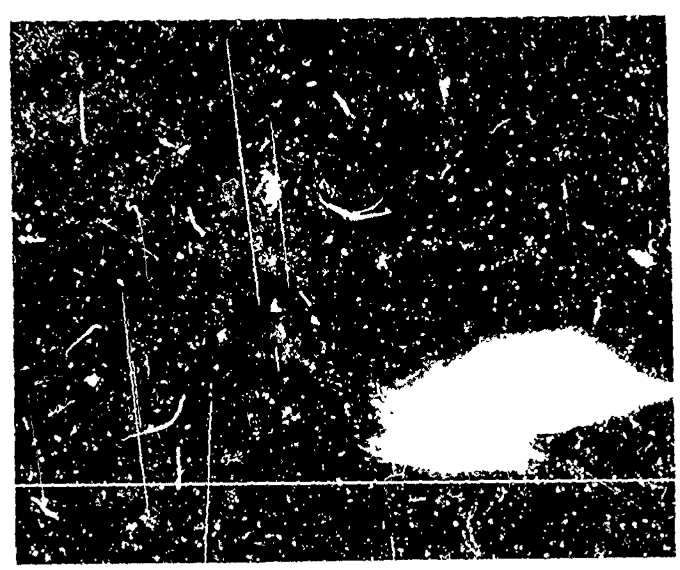

Figure 9. Typical Stroke, of About Two Milliseconds Duration, Striking Ahead of the Vent Slot and Being Swept Across It by a 200 Knot Airflow

an initial flame front velocity of about 200 fps for the first foot inside the tube, in contrast to the 2 to 10 fps velocities usually obtained without airflow. The flame speed measurement oscillograms are shown in Figure 11.

The main reason for applying the high voltage to an insulated wingtip at the top of the combination generator-wind tunnel was to check the possibility of ignition by streamers forming on or near the vent scoop when the wingtip was placed at high voltage. In Figure 12, 5 million volts produced streamering off the wingtip. The fuel mixture was adjusted to 1.5 stoich at an outflow of 5 fps, the most vulnerable condition found earlier. Using simulated lightning discharges with no airflow, we had obtained 12 ignitions out of 12 strokes to the vent scoop. Under streamering conditions, we obtained no ignitions in 10 applications of 5 million volts.

To check effects of the angle-of-attack of the airstream, a shallow climb was simulated by using a deflector, Figure 13, to deflect the airflow. Ignition was first obtained without airflow at 1.5 stoich and 5 fps to verify the mixture; when airflow was added, no ignition was obtained in 5 shots each with an airflow of 50 amd 90 knots.

## CONCLUSIONS

1. The results showed that windstream velocities greatly affect the probability of ignition of the fuel vent efflux. For strokes to the edge of the vent scoop of a recessed vent, the probability of ignition was about 3% with airflows to 90 knots, as compared to 100% with no airflow. No ignitions were obtained in 200 shots with a 200 knot airflow and in 10 shots with an airflow of 250 knots. However, for swept strokes across the vent slot, a high probability of ignition was obtained at 100 knots, decreasing to the order of 4% for 46 shots with a 200 knot airflow.

2. The most vulnerable fuel vent fuel-air ratio was found to be 1.0 to 1.5 stoichiometric, with an efflux velocity of about 5 feet per second.

3. Streamering off the wingtip, produced by the application of a 5 million volt surge, did not cause ignition under the most vulnerable conditions of mixture and efflux velocity, even with no airflow.

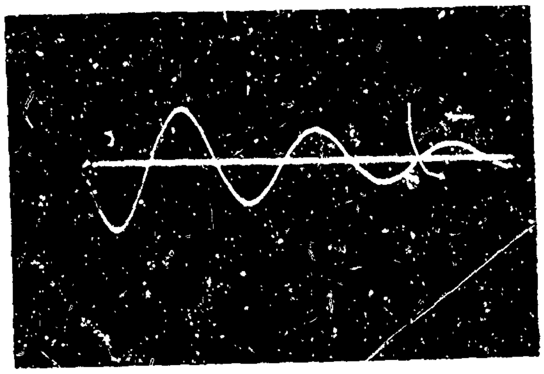

Figure 10. Typical Current Waveform Used in Swept Stroke Studies Across a Fuel Vent. (Scale is 50 kiloamperes and 200 microseconds per large division.)

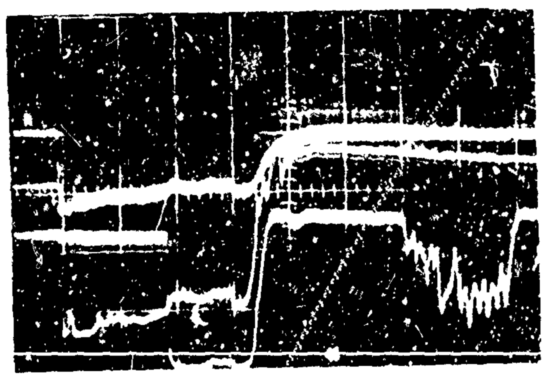

Figure 11. Oscillograms From Three Ionization Probes Located One Foot Apart in the Vent Tube. (Ignition was due to a swept stroke in a 200 knot wind. The time scale is 0.1 second per large division and the first time interval appears to be about 5 milliseconds corresponding to a flame front velocity of about 200 fps.)

Figure 12. Streamers Off Wingtip Mockup Did Not Ignite the Fuel Vent Efflux Under the Most Vulnerable Conditions of Mixture, Airflow, and Vapor Flow From the Vent

Figure 13. A Deflector Added to the Wind Tunnel Plexiglas Chute To Vary the Angle-of-Attack Over the Vent

4. Very high flame speeds have been measured, probably due to an interaction between the lightning blast wave and the flame front. These very high flame speeds, of the order of 1,000 feet per second, would pass through most flame arresters and other explosion suppression devices into the fuel tank. It is therefore advisable to reproduce these fast fronts in the laboratory and find means of limiting the maximum velocity.

5. Changing the angle-of-attack of the airstream hitting the vent scoop of the mockup did not increase the probability of ignition.

## RECOMMENDATIONS

1. Basic studies of natural lightning plasma width and ionization characteristics, acoustic waves, and likely swept stroke paths, applicable to strokes which could contact the aircraft near the fuel vent, should continue.

2. Assuming that swept strokes near the vent are possible, the ignition probabilities and flame speeds of fronts generated by this mechanism should be studied further.

3. In order to reduce the maximum possible vent tube flame speeds, studies should continue on the mechanisms involved so that methods of duplicating the proper conditions in the laboratory can be devised. When the flame speeds can be reproduced, they may be reduced by the addition of baffles or other techniques with the objective of insuring that, in combination with available explosion prevention techniques, the fuel vent system is protected against all known ignition hazards due to lightning within the state of the art.

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# FLAME PROPAGATION THROUGH AIRCRAFT VENT SYSTEMS

G. J. Grabowski

Fenwal, Incorporated

Under contract from the Department of Transportation, Federal Aviation Agency, Fenwal Incorporated performed a test program to determine the flame propagation characteristics through aircraft vent systems and to establish criteria for vent design which would reduce or eliminate the possibility of such propagation. This was to be accomplished by tests performed on a full scale mockup of a typical turbine aircraft vent system along with a theoretical analysis based upon current knowledge of combustion propagation. Two basic assumptions were made to perform this program, namely, that an ignition source is available at any point in the vent system and that a flammable mixure is present throughout the entire volume. This paper will describe the tests and summarize the results and conclusions.

Since the majority of research and development effort on fires and explosions in aircraft has centered around the establishment of flammability and ignitability envelopes for fuels under flight conditions, it is necessary to discuss briefly the general characteristics of flame propagation in ducts to ensure understanding. The problem is complex and still not fully understood by the combustion experts, but flame propagation characteristics can be predicted for simple duct configurations.

The two velocity characteristics of a flame are burning velocity and flame speed. Burning velocity is the velocity of the unburned gas with respect to the flame; flame speed in the velocity of the flame relative to the structure. The burning velocity is a basic characteristic of the combustible, while the flame speed is dependent upon the burning velocity plus the effect of the duct physical configuration. The latter is, therefore, the larger of the two values and is the figure that concerns this investigation.

If ignition of a flammable mixture in a tube occurs at an open end, the flame speeds will be considerably less than if it takes place at a closed end because the burned gas can expand freely out the open end but is confined at the closed end. This effect is illustrated in Figure 1 for a propane-air mixture ignited in a 12-inch-diameter duct. If, however, a flame is ignited at an open end and allowed to propagate down a long duct, sufficient acceleration may occur to cause very high flame speed and possible detonation. This is shown in Figure 2 where a propane-air flame approached a speed of 1000 fps in a 100-foot-long duct. It is important to note that this phenomenon is achieved only in mixtures having relatively high burning velocities.

## TEST SETUP

The test setup for this program consisted of a full-scale mockup of a turbine aircraft vent system, as shown in Figure 3. An outer wing panel was used to obtain the vent outlet and surge tank configuration to which steel ducts and pipe were attached, duplicating the aircraft tank vent ducts in cross section and length. Three tanks of 500 gallon capacity each were attached to simulate the vapor space in the main tanks under full condition. These tanks were equipped with rupture discs to relieve the excess pressure created by explosions. Instrumentation in the form of ionization probes and pressure transducers was mounted to record the flame travel.

The combustible used in the program was primarily JP-4 since it was felt that the hazard could not be analyzed correctly with any gas such as propane. The difference in burning velocity between JP-4 and other fuel is sufficient that results and conclusions could be misleading. The fuel-air mixtures were obtained by heating the JP-4 to obtain a sufficient vapor pressure and adding air by the partial pressure method to achieve the correct ratio. The combustible mixtures were introduced into the vent system at the inboard end and the entire volume was purged until a reasonable homogeneous mixture existed. When a flow condition was necessary, the mixture was allowed to flow at time of ignition by opening the valve from the makeup tank.

## RESULTS

Since it is not possible to discuss each specific test in this paper, the results will be presented in a general form. It was found that a flame ignited at the vent

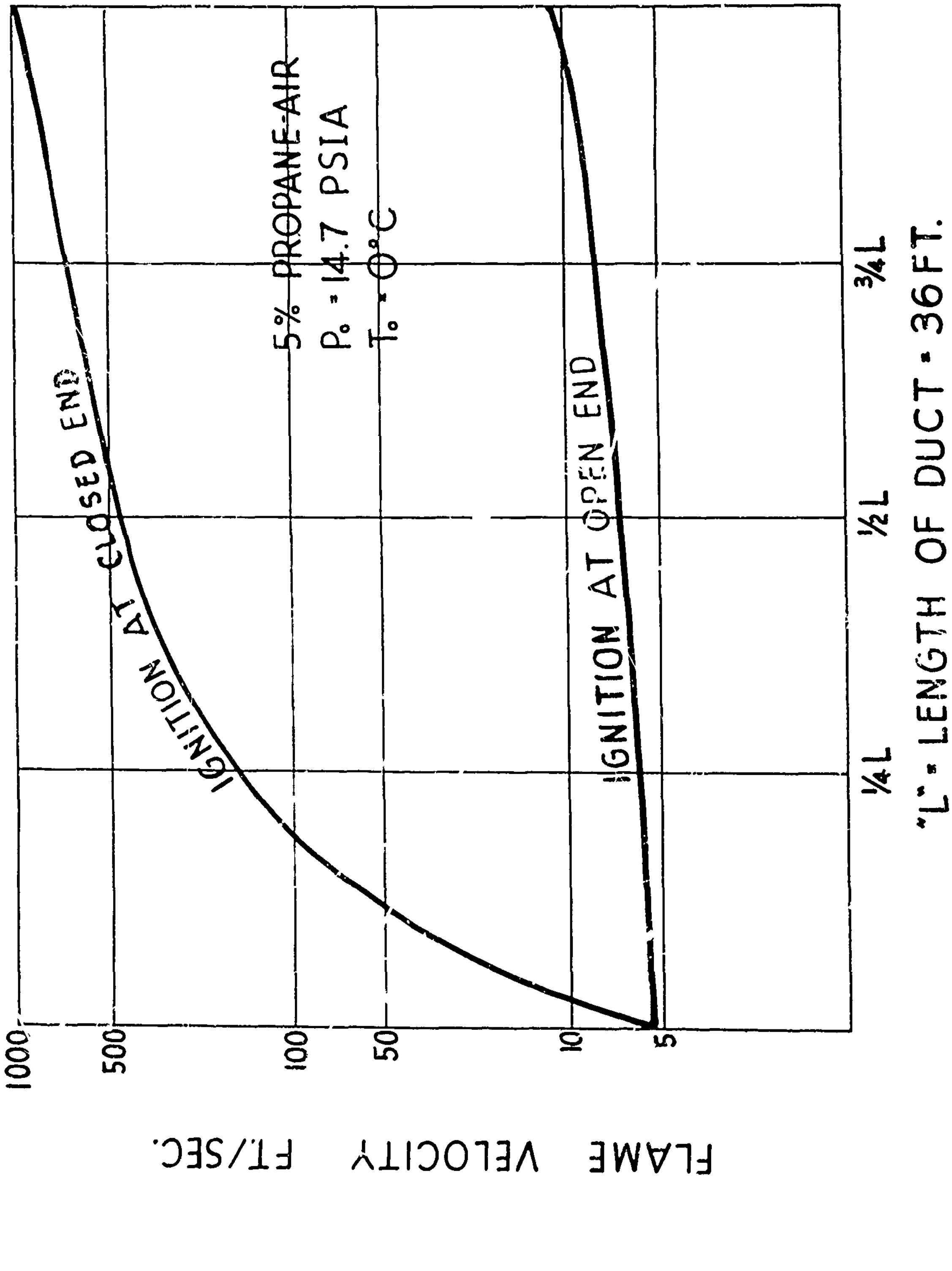



Figure 1. Flame Propagation Rates

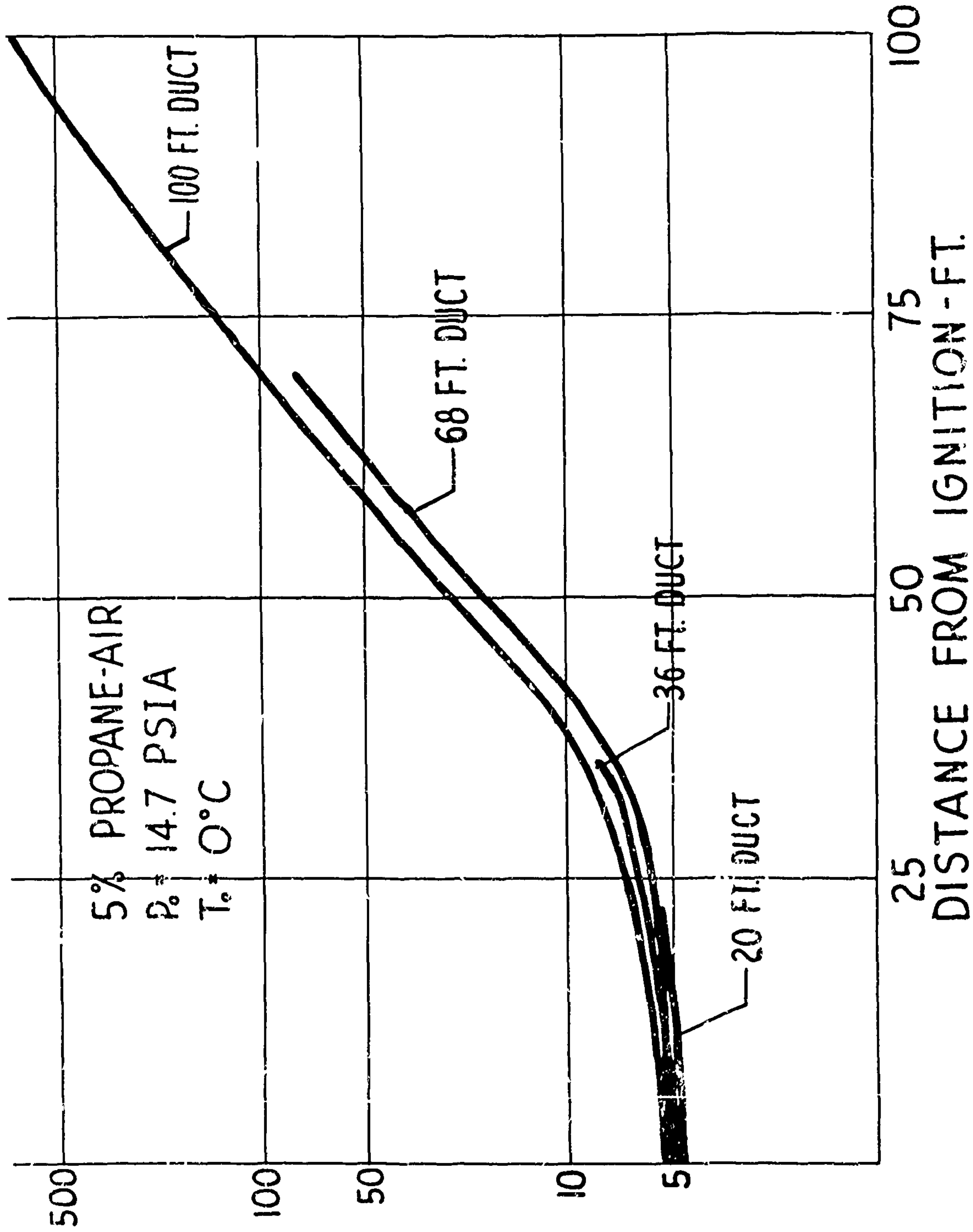



Figure 2. Flame Propagation Rates

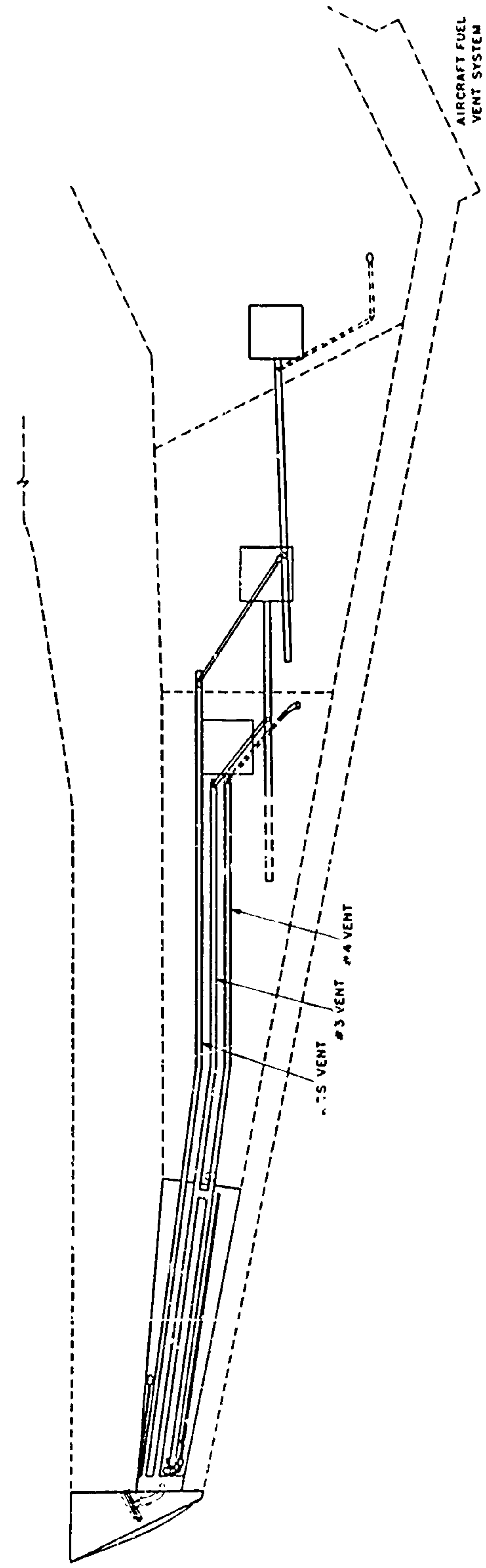



Figure 3. Test Setup

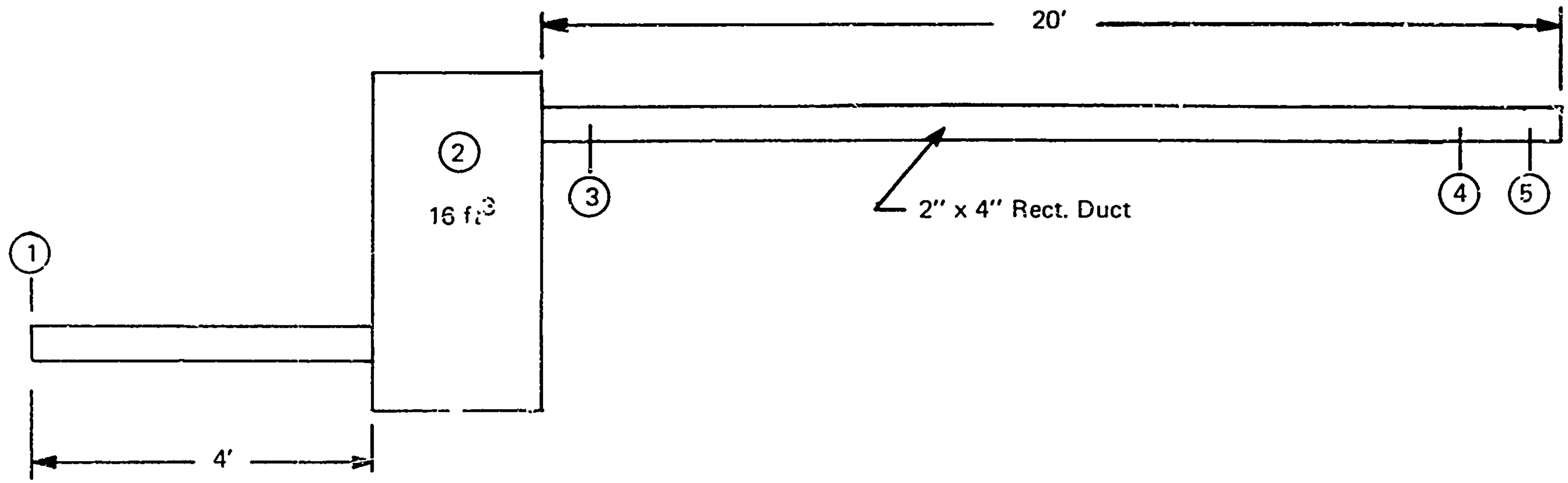
outlet would propagate through the vent system into the center main tank in a minimum of 11 seconds. The speed would vary as the result of a change in cross section or direction of the duct, and an explosion would occur within the simulated tank. The velocity ranged from the expected value of 15 to 20 fps up to 1000 fps at certain points in the vent system.

It became evident that the vent system configuration presented a very complicated model for analysis of flame propagation. The presence of very high speeds at isolated points indicated that factors were present which reinforced the flame. Such factors could take many forms, but wave reflections appeared to be the most likely. It was decided to perform tests in a simple structure to provide clues to the high speeds.

The first tests were performed with a stoichiometric JP-4 mixture in a 12-inch-diameter duct 100 feet long, with ignition in both the open and closed ends. In this structure, the maximum flame speed with ignition at the open end was 17.8 fps, while the maximum speed with ignition at the closed end was 56.5 fps. Various forms of ignition sources, including explosive charges, were used to obtain maximum flame speeds, but under no condition were speeds greater than those mentioned above; therefore, it became evident that it was not possible to obtain a detonation with this fuel unless reinforcement was obtained by the physical configuration.

To obtain further information, a simple test setup was made, as shown in Figure 4, simulating the vent outlet duct, surge tank, and tank vent. A number of tests were performed varying the location of ignition; some of the results are reflected in Figure 1. These values show the effect of the surge tank, specifically, which provides a driving force for the flame when ignition occurs within the surge tank or at the vent outlet. The mechanism involved is similar to that with the drive tube, and this undoubtedly was a major contributing factor to the high localized flame speeds in the test setup.

Other factors were evaluated and indicated that both reflection and turbulence were causing the extremely high flame speeds. The causes of some of these factors were determined, but the scope of the work did not permit an extensive investigation into this aspect.




FLAME SPEEDS – FT/SEC

| Temp. | Ignition | 1 | 2 | 3 | 4 | 5 | Combustible |
|---|---|---|---|---|---|---|---|
| 50°F | 1 | | | 6.8 | 557 | 1000 | Jp-4 |
| 70°F | 1 | | | 12.9 | 1076 | 1000 | Propane |
| 70°F | 2 | 66.2 | | 13.4 | 750 | 1000 | Propane |
| 32°F | 5 | 185 | | 8[illegible] | 4.3 | | Propane |

Figure 4. Test Setup

In general, the results of the program were as follows:

1. A flame will propagate through a typical vent system at various speeds, dependent upon the physical characteristics of the system.

2. High local flame speeds may occur due to reflections and/or turbulence, which would develop pressures that exceed the structural capability of the vent ducts.

3. Vent systems to reduce flame speeds can be designed by removing reflection and turbulence-producing features. Caution must be taken, however, to ensure that achieving these points does not affect the basic performance of the vent system.

4. The lack of complete knowledge and understanding of propagation characteristics precludes the use of an analytical approach to investigate such characteristics in a typical vent system.

AFAL-TR-68-290
PART II

# AIRCRAFT FUEL VENT LINE SENSITIVITY TO LIGHTNING EFFECTS

M. Markels, Jr., and J. M. Spurlock
Atlantic Research Corporation

J. D. Robb and J. R. Stahmann
Lightning and Transients Research Institute

On a program sponsored by the Federal Aviation Agency (Contract Nos. FA 64WA-4955, and FA 64WA-4960), scientists and engineers of the Atlantic Research Corporation* and Lightning and Transients Research Institute (LTRI) worked together to investigate the hazard to the fuel vent systems of turbojet aircraft created by lightning strikes. The facilities of both Atlantic Research and LTRI were used to accomplish the program plan, the scope of which included experimentation and analysis to: determine the existence of possible ignition sources in a typical aircraft wing tank section; characterize flame propagation in the fuel vent system arising from lightning ignition; establish the extent of the associated hazard and the effectiveness of remedial methods such as flame arresters and explosion suppression techniques; and identify as well as demonstrate some feasible ignition-source reduction measures. Thus, the overall objective of the bipartite program was the development of technology leading to methods of overcoming aircraft lightning strike hazards and the investigation of potential hazards that could be introduced by these remedial methods. Highlights of the work accomplished on the overall program, together with the more important results and conclusions, are summarized in this paper. Complete details of this joint effort comprise two FAA reports (References 1 and 2).

## BACKGROUND

Several excellent research programs have been conducted by government agencies and industry in the past to evolve methods of protecting aircraft from lightning hazards. The results of such programs provided background for the

*A Division of the Susquehanna Corporation investigation described in this paper. They were directed mainly toward the development of specific protective devices or methods for specific aircraft and aircraft components. Little generalized understanding of the hazardous mechanisms associated with lightning strikes accrued to guide the development of technology which, in turn, could lead to new methods for still further reducing lightning hazards. Thus, the major objective of the work described in this paper was the accrual of such understanding with respect to hazardous effects which might be produced in turbojet aircraft fuel vent systems by lightning. The technological background upon which the rationale of this program was based included information on lightning and lightning simulation, the ignition and combustion characteristics of aircraft fuel vent systems, and the flight environment.

Numerous sources of information are available on the mechanisms, characteristics, and effects of natural lightning (References 3-6). The more important observed properties of natural lightning discharges include: a current of $5 \times 10^4$ to $5 \times 10^5$ amperes; an initial rate of current rise from $10^4$ to $2 \times 10^5$ amperes/$\mu$sec; a charge transfer from 50 to 600 coulombs; and a voltage from $10^8$ to $5 \times 10^8$ volts. For experiments such as those described in this paper, it was necessary to effectively simulate these properties of natural lightning. The facilities of LTRI, which were designed to duplicate the measured and observed effects of lightning discharges, were used for this simulation and are discussed later in this paper. Lightning discharges produce various effects which depend on the material and configuration of the object struck, as well as the characteristics of the discharge. In the experiments of this investigation the facilities were arranged to reproduce the particular characteristics of lightning that were estimated to be significant in studying ignition and flame propagation in aircraft fuel vent lines and through flame arresters.

A large body of good information also exists on the ignition, combustion, and detonation of fuels (References 7-11). The prerequisites for ignition are a flammable mixture and a concentrated energy source. It has been shown (References 8, 10) that ignition can occur in homogeneous combustible mixtures from a source of energy of less than 1 millijoule, so flammable mixtures can be spark-ignited by commonly occurring electrostatic discharges. For example, a person walking across a rug on a dry day could conceivably initiate a discharge of energy equal to as much as 15 millijoules. Thus, it is obvious that lightning discharges can be a potent ignition source.

Fuel for commercial turbojet transports usually is kerosene (JP-1) or JP-4, a kerosene-type fuel which contains a greater concentration of more volatile components than does kerosene. The presence of these fuels in aircraft provides a flammable mixture in the fuel tanks, vent lines, etc., over a rather extensive range of altitudes and temperatures (Reference 12). In addition to the fuel vapor-air mixture, a flammable heterogeneous fuel vapor/fuel mist/air mixture can be present over a wide range of temperatures below and above the flash point (Reference 10). The vent outlet will contain fuel-air mixtures during ascent, during rapid decreases in atmospheric pressures, and in level flight just after such pressure changes. The large effluent velocity through the vent lines during rapid ascent can certainly purge the vent line with fuel-air mixtures and, in addition, provide copious quantities of the mixtures at or near the vent outlet.

In a typical turbojet transport, fuel is contained in several tanks, such as a bladder-type center tank in the fuselage plus main and reserve wet-wing-type tanks in each wing. Each wing tank is composed of many chambers since the tanks are sectioned by open ribs spaced approximately every 2 feet. The tanks are interconnected so that all engines can operate from one tank, or all tanks can supply one or more engines. In addition, each tank is vented directly or indirectly through channels (vent lines) into surge tanks, one of which is located at each wing tip. This is shown schematically in Figure 1 (Reference 13). Float valves are used in the vent lines to minimize liquid fuel transfer during banking maneuvers. The vent lines enter the surge tank from the inboard side, and a drain hole allows liquid fuel carried into the surge tank to return to the reserve tank. A duct vents the surge tank to the atmosphere, and typically is comprised of the elbow with its open end up in the surge tank and the other end flanged to an opening in the outboard rib, and an S-shaped vent tube which is flanged into a rectangular box-like vent outlet. The vent outlet is recessed into the underside of the wing, and its opening is located such that air flows across and perpendicular to its longer dimension. In the investigation described in this paper, all

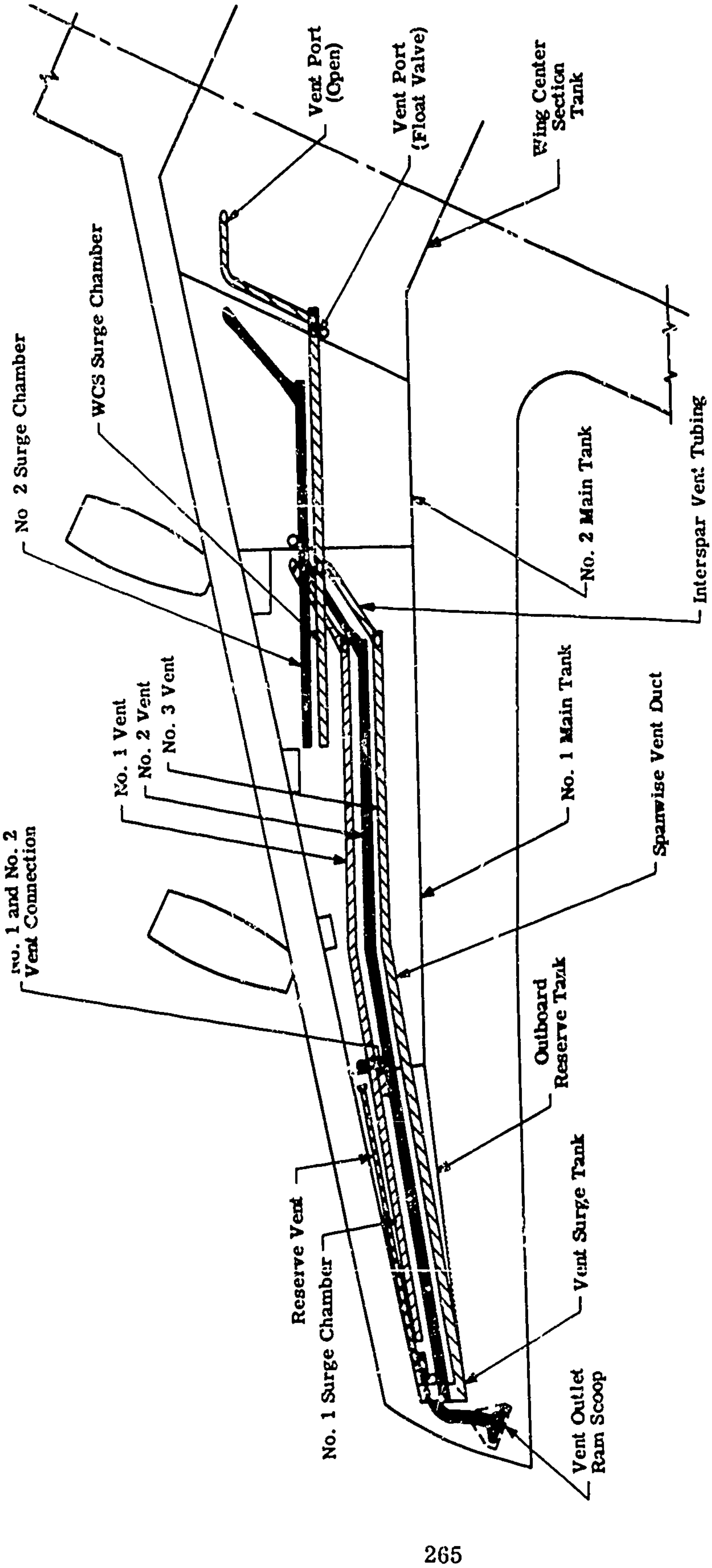



Figure 1. Schematic Diagram of Typical Turbojet Transport Fuel Vent System (Reference 13)

vent components that were used in simulation experiments were actual production aircraft parts, except for the surge tanks which were simulated for a large portion of the testing.

It follows from the foregoing discussion that a fire hazard from lightning strikes exists near an aircraft fuel vent outlet. A flammable mixture in the fuel tanks can provide flammable effluent from the fuel vents and lightning near thunderstorms can provide the ignition source. The fact that lightning does strike aircraft is well documented, and the frequency of strikes sufficiently severe to show marks on the aircraft skin is surprisingly high. Certain types of lightning strikes on aircraft in flight can occur for nearly a second, providing strike points at numerous locations across a wing or fuselage.

In the investigation described in this paper, combustion and lightning susceptibility experiments and analyses were conducted to determine the extent and severity of this type of hazard, and the potential effectiveness of some ostensibly remedial approaches.

## COMBUSTION EXPERIMENTS

These experiments were designed to elucidate flame propagation characteristics through a typical turbojet aircraft fuel vent system as a result of ignition by simulated lightning discharges and laboratory ignition equipment. They also include testing of the effectiveness of flame arresters and explosion suppression techniques using the same basic types of experimental systems. Certain other experiments were performed to explore previously reported blast pressure effects and plasma generation as the result of lightning discharge near a vent outlet.

Two identical assemblies of instrumented apparatus were constructed using actual aircraft vent system components. One assembly was used with the LTRI high-energy discharge facilities to establish the range of flame speeds of interest and the effectiveness of selected flame arresters. The other assembly was used in the Atlantic Research combustion laboratory, duplicating conditions generated by the simulated lightning, for more detailed study of flame propagation in the system.

In these experiments, it was assumed that a combustion mixture could exist in the vent system during flight operations and that lightning, acting as an ignition source at the vent, could cause a flame to propagate through the vent system into the surge tank, as suggested by results of previous studies (Reference 7). Therefore, consideration of actual free-stream effects at the vent outlet were beyond the scope of this program.

No attempt was made to simulate inflow or outflow through the experimental vent system as would often occur in an actual flight situation under the influence of flight trajectory, fuel tank void volume, and atmospheric conditions. Instead, the combustible mixture was allowed to remain stagnant in the test system. It was reasoned that this represents the most hazardous state in which flame propagation is possible (Reference 2). Outflow would impede the progress of a flame propagating toward the surge tank, and inflow sould tend to thin the mixture beyond the limits of combustibility.

Propane was chosen as the fuel in these experiments because of its suitability as a simulant for aircraft fuel vapors and the ease with which it can be obtained and manipulated. Jet aircraft fuels are usually paraffinic hydrocarbons, and the nature of combustion processes in their vapors is such that their pertinent combustion properties are nearly independent of their molecular weight (Reference 8). Therefore, it is possible to employ any of the higher homologs of the paraffin series rather interchangeably in experiments such as these.

## 1. CHARACTERIZATION OF LIGHTNING EFFECTS

Initial experiments involved the measurement of flame speed and pressures in the experimental vent system to establish the mechanism of flame propagation. The experimental vent system is shown schematically in Figure 2, including locations of the pressure transducers (points P) and thermocouples (points T) for all experiments. The drive tube and standoff sections were included only on the assembly used at Atlantic Research facilities. Three types of ignition sources were used. Two were simulated lightning strikes provided by LTRI facilities, one of which furnished electrical discharges with high currents (up to 250,000 amps) and the other with high voltages (up to 10 megavolts), both characteristic of severe lightning strokes. These facilities simulated lightning characteristics

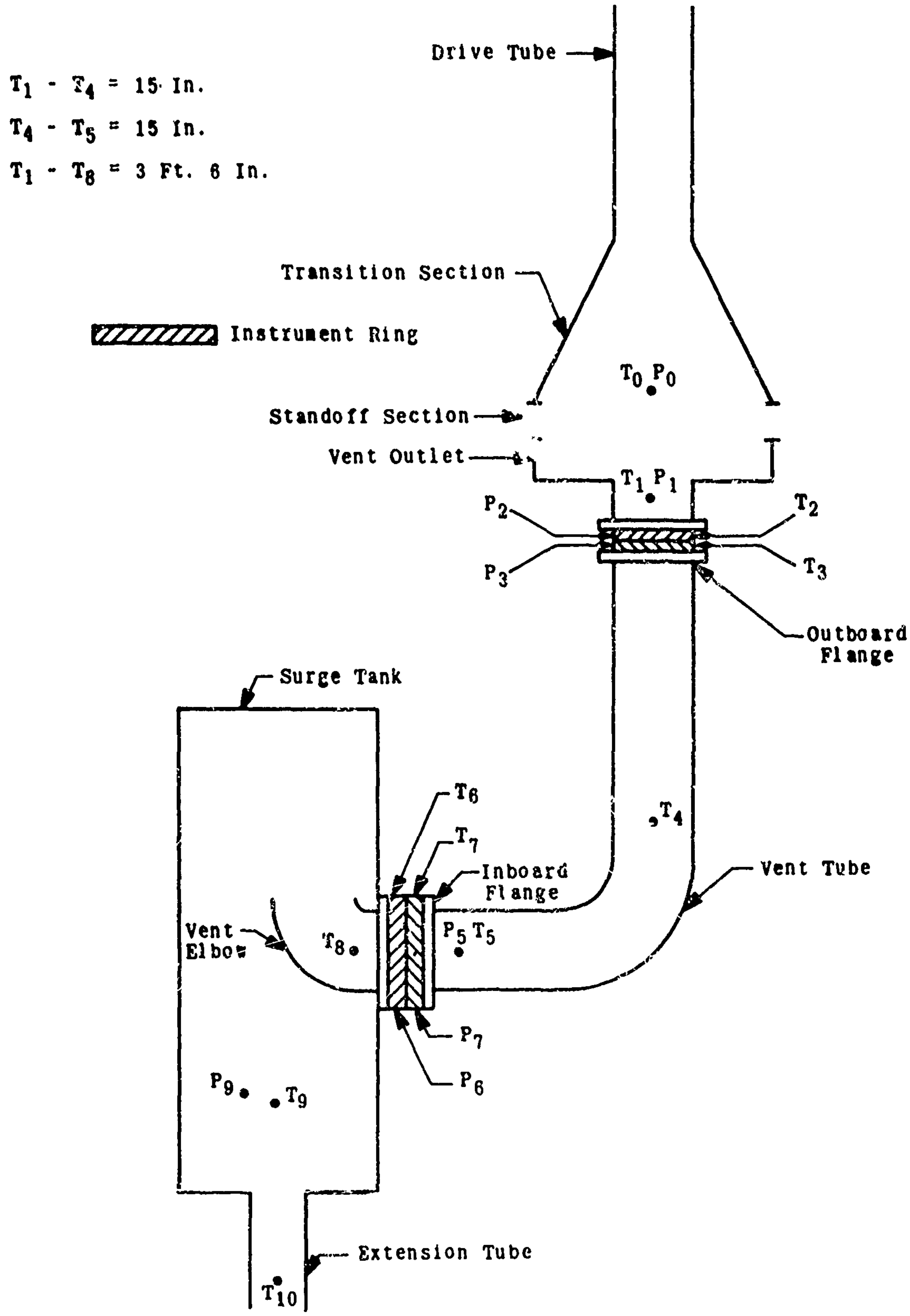



Figure 2. Schematic of Test Apparatus, Showing Instrument Locations

and effects to the best extent available within the state of the art. The third ignition source was specially designed and built into the Atlantic Research experimental assembly to produce, by simple chemical means, the fluid mechanical (shock and flow) effects of lightning ignition. Flame speeds were estimated from time-of-arrival responses of the chromel-alumel thermocouples in the system, whose experimentally determined time-response capabilities were faster than 8 msec. Calibrated and temperature-compensated pressure transducers having an operating range of 0-25 psig were used for all pressure measurements. With appropriate signal conditioning they were able to faithfully follow a pressure wave whose rise time was greater than 0.15 msec. All temperature and pressure data were recorded by a recording oscillograph.

For the experimental system used with the LTRI facilities, the fuel-air mixture was supplied from bottles of liquid propane and compressed air. The pressure of each stream was reduced, then the flows were metered, mixed, and supplied to the vent system through the vent elbow. After the system had been adequately purged with the mixture, the vent outlet was sealed, with the discharge probe from the lightning simulation facility protruding through the seal into the combustible mixture. The surge tank elbow was then blocked with an inflated balloon until just prior to ignition, when the balloon was deflated. This allowed simulation of the actual vent and surge tank configuration, but with a limited total volume of combustible mixture.

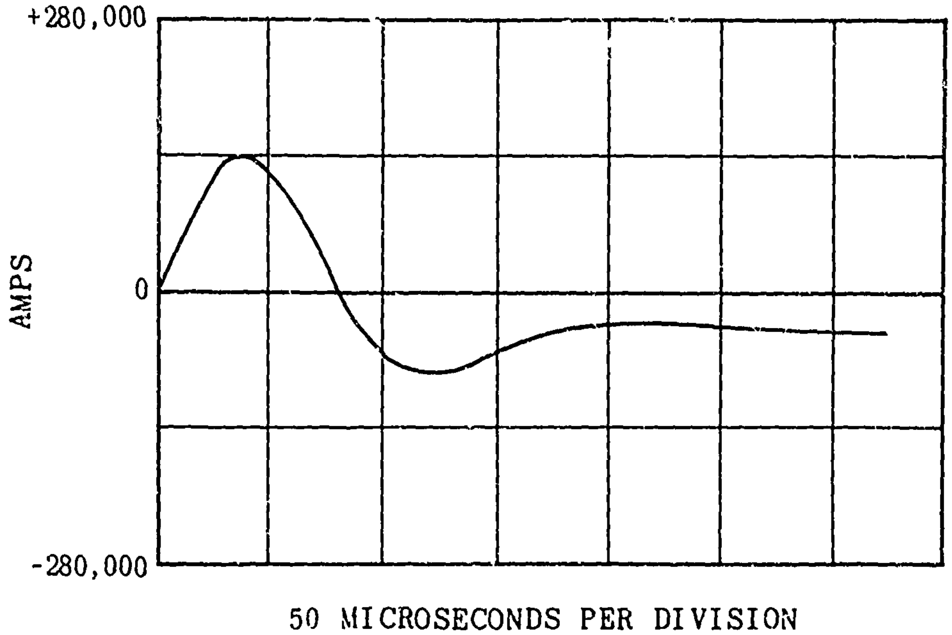
For experiments with the LTRI High-Current Facility, the vent tube, vent outlet, and mock surge tank were installed in an actual turbojet aircraft wing-tip section. The entire experimental system was electrically isolated from ground to protect the instrumentation from damage during the high current discharge. The simulated lightning discharge in this series of tests was obtained by charging a bank of capacitors whose total capacitance was 366 mf, then discharging through the discharge probe to a grounding strap placed at a selected point of discharge near the vent outlet. The discharge probe, which protruded into the combustible mixture, was positioned near this grounding strap within a distance smaller than the breakdown distance in air at the prescribed voltage, and was connected to the bank of capacitors by a mechanical switch which triggered the discharge. In all cases, the discharge occurred within the fuel-air mixture.

In some tests, the discharge probe and ground strap were connected by a small piece of aluminized tape which exploded during discharge and provided higher pressures at the vent outlet than did direct discharges of equal voltage. This was estimated to be a closer representation of the pressure wave emanating from a lightning strike.

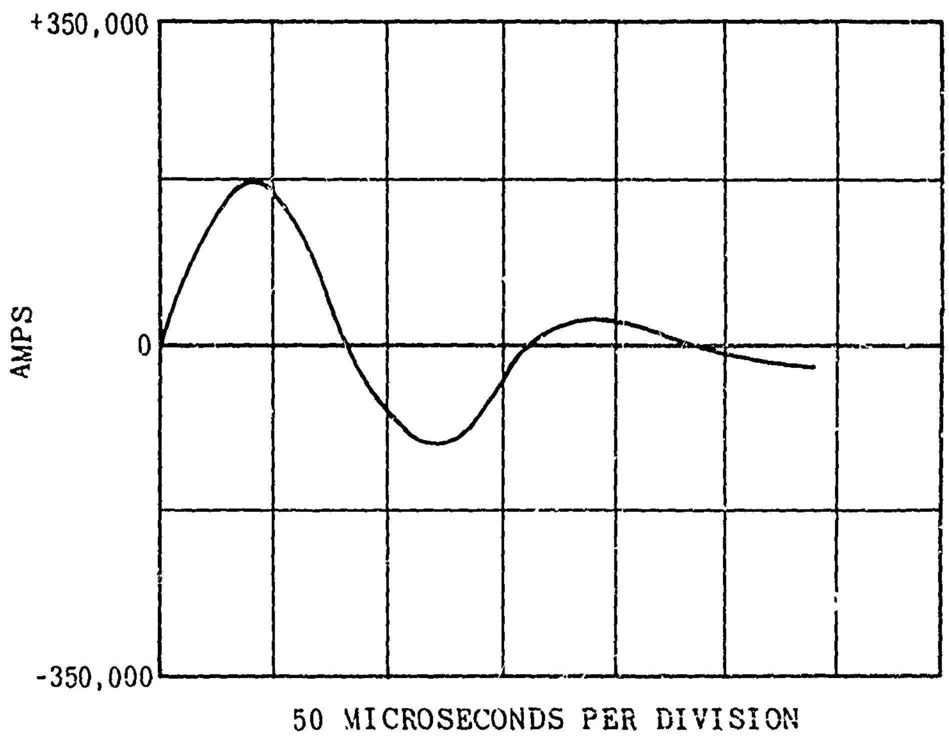
The discharge voltage was varied from 5 to 25 kv, giving peak currents up to 250,000 amperes and charge transfers to 9 coulombs. Figure 3 shows typical current wave forms of a 15 kv direct discharge and a 15 kv discharge through a strip of aluminized tape 4 inches long. The position of discharge was varied, and strikes directed to the edge of the vent outlet were considered to represent the most severe ignition source and driving force, whereas those directed at the raised portion of the vent scoop were considered to be less severe because of a larger separation from the vent outlet.

Although the fuel-air mixture ratio was varied from 70 to 150 per cent of stoichiometric (based on propane weight), most tests were conducted at 115 per cent of stoichiometric, since such a slightly fuel-rich mixture exhibits the minimum quenching distance in tubes. Results of these tests for the high-current discharge case are summarized in Table I in the form of measured values of flame speed and pressure under these simulated conditions. These values suggest the combustion response that might be expected from lightning ignition in actual turbo aircraft vent systems having similar configurations.

Experiments with the LTRI High-Voltage Facility used essentially the same apparatus and discharge arrangement as that described for the high-current discharge experiments, with the exception that the vent tube, surge tank, and air-scoop panel were installed in an aluminum box instead of the wing-tip section. The box served the same purpose as the wing tip (i.e., as a support structure and electrostatic shield) and, in addition, afforded some mounting conveniences. The discharge voltage was nominally $1 \times 10^6$ volts, with a current waveform as shown in Figure 4. The high-voltage system has an increased current rate of rise in comparison with the high-current facility. With the higher voltage, longer discharge paths could be used; however, the position of the strike could not be predetermined as in the lower voltage experiments. Therefore, photographs of




Four-Inch Aluminized Tape Discharge




Direct Discharge

Figure 3. Current Waveforms of LTRI High-Current Facility, 15 kv Discharge

TABLE I

SUMMARY OF RESULTS FROM HIGH-CURRENT EXPERIMENTS

| Discharge Point[a] | Mixture Fraction of Stoichiometric | Discharge Voltage (kv) | Pressure (psig)[b] | | | Flame Speed (ft/sec) | | |
|---|---|---|---|---|---|---|---|---|
| | | | $P_1$ | $P_5$ | $P_9$ | $T_1-T_4$ | $T_4-T_5$ | Average ($T_1-T_5$) |
| A | 1.15 | 10 (tape) | 6.1 | 6.2 | 1.0 | 40 | 73 | 52 |
| A | 1.15 | 10 (tape) | 6.9 | 9.3 | 2.0 | 96 | 125 | 108 |
| B | 1.15 | 5 (tape) | 3.7 | 3.8 | 1.2 | 33 | 26 | 29 |
| B | 1.15 | 10 (tape) | 3.7 | 6.2 | 1.2 | 37 | 42 | 40 |
| B | 1.15 | 15 (tape) | 5.1 | 9.4 | 2.3 | 54 | 83 | 66 |
| B | 1.15 | 20 (tape) | 5.0 | 10.8 | 2.4 | 89 | 15[illegible] | 114 |
| B | 1.15 | 10 (direct) | 3.0 | 2.7 | 1.0 | 16 | 18 | 17 |
| C | 1.15 | 20 (tape) | 5.0 | 8.1 | 2.6 | 37 | 50 | 43 |
| C | 1.15 | 25 (tape) | 10.6 | 14.6 | 2.6 | 415 | 83 | 139 |
| D | 1.15 | 25 (tape) | 3.7 | 15.7 | 3.1 | ?[c] | ?[c] | ?[c] |
| E | 1.15 | 15 (tape) | 3.1 | 3.7 | 1.0 | 13 | 31 | 18 |
| E | 1.15 | 25 (tape) | 1.8 | 8.1 | 1.8 | 42 | 62 | 50 |
| F | 1.14 | 15 (tape) | 6.3 | 2.6 | 1.1 | 10 | 47 | 16 |
| G | 1.15 | 15 (direct) | 5.0 | 3.4 | 0.8 | 18 | 19 | 18 |
| G | 1.00 | 15 (direct) | 5.0 | 5.0 | 1.0 | 27 | 37 | 32 |
| G | 0.90 | 15 (direct) | 3.1 | 3.8 | 0.9 | 13 | 157 | 24 |
| G | 0.80 | 15 (direct) | 5.0 | 5.0 | 1.1 | 14 | 33 | 20 |
| G | 1.50 | 15 (direct) | 4.9 | 5.2 | 1.2 | 31 | 18 | 23 |

[a] A - vent inlet; B - edge of vent inlet; C, D, E, F, - around recess of vent scoop; G - vent scoop near inlet.

[b] See Figure 2 for location of instrumentation.

[c] Instrumentation was broken.

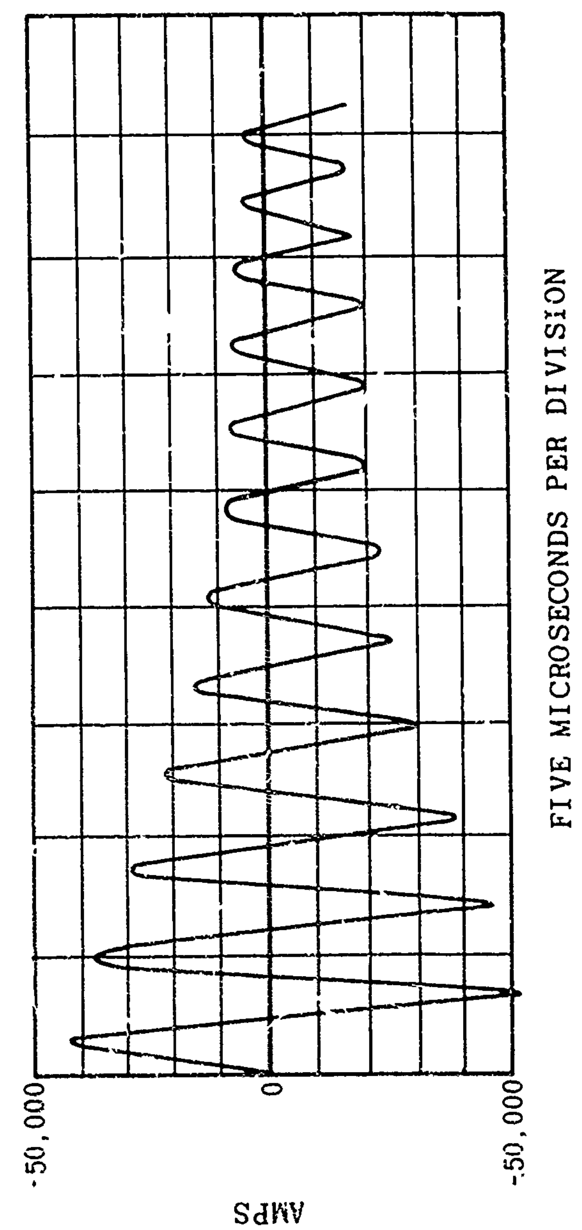



Figure 4. Current Waveform From LTRI High-Voltage Facility, $1 \times 10^6$ Volt Discharge

the discharge were recorded with a camera synchronized with the triggering mechanism of the discharge generator.

To ensure that the lightning strikes would occur in the vicinity of the vent outlet, a discharge gap of no more than 12 inches was used. The discharge probe began to discharge when the breakdown gradient in air was exceeded; and since the breakdown voltage of the probe was approximately $1 \times 10^6$ volts per meter, peak probe voltages were up to about 325,000 volts.

The fuel-air mixture ratio was 115 per cent of stoichiometric (slightly fuel-rich). Results of these high-voltage experiments are summarized in Table II. The discharge was nominally $1 \times 10^6$ volts, with peak probe voltages ranging from 128 kv for the 5-inch arc length to 325 kv for the 12-inch discharges.

The flame propagation experiments that were conducted in the Atlantic Research combustion laboratory served principally to guide the design of an assembly in which flame arresters could be tested. Thus, the assembly was designed to provide flame propagation mechanisms similar to those observed in experiments with simulated lightning strikes using LTRI facilities. This type of simplified test system obviates the necessity of complex test procedures when evaluating the performance of flame arresters in aircraft fuel vent systems. Data obtained during the experiments with LTRI facilities were used as the basis for improving the Atlantic Research test assembly.

A sketch of the Atlantic Research experimental assembly is shown in Figure 5. A 3-inch diameter pipe (the drive tube) was attached to an actual turbojet aircraft vent tube assembly by way of a matching transition section and a standoff section. The standoff was added as a modification after it was determined that the unmodified assembly did not adequately simulate the combustion characteristics that were observed during the high-energy electric discharge experiments. In the final version of the assembly, various modes of flame propagation could be produced, which provided a method for simulating a wide range of combustion effects. For example, flame speeds and pressures could be controlled by changing the drive-tube length, by changing the point of ignition, or by changing the diameter of the drive tube orifice. In some experiments,

TABLE II

SUMMARY OF RESULTS FROM HIGH-VOLTAGE EXPERIMENTS

| Nominal Discharge Voltage (volts) | Arc Length (in) | Pressure (psig) | | Flame Speed (ft/sec) | | |
|---|---|---|---|---|---|---|
| | | $P_1$ | $P_5$[b] | $T_1$-$T_4$ | $T_4$-$T_5$ | Average |
| 1,000,000 | 5 (center) | ~ 1/2 | ~ 1/4 | 5.2 | 6.8 | 5.9 |
| 1,000,000[a] | 5 (center) | ~ 1/2 | ~ 1/4 | 6.0 | 7.8 | 6.7 |
| 800,000 | 5 (center) | ~ 1/2 | ~ 1/4 | 11.6 | 8.6 | 9.3 |
| 1,000,000 | 5 (side) | ~ 1/4 | ~ 1/8 | 14.0 | 10.1 | 11.8 |
| 1,000,000 | 12 (center) | ~ 1/4 | ~ 1/8 | 15.4 | 21.6 | 18.0 |
| 1,000,000 | 7 (center) | ~ 1/4 | ~ 1/8 | 13.5 | 15.1 | 14.3 |
| 1,000,000 | 12 (center | ~ 1/2 | ~ 1/4 | 9.4 | 57.0 | 16.3 |

NOTE: Discharge probe was centered above vent inlet opening unless otherwise noted.

[a]Surge tank filled with propane-air mixture

[b]See Figure 2 for location of instrumentation.

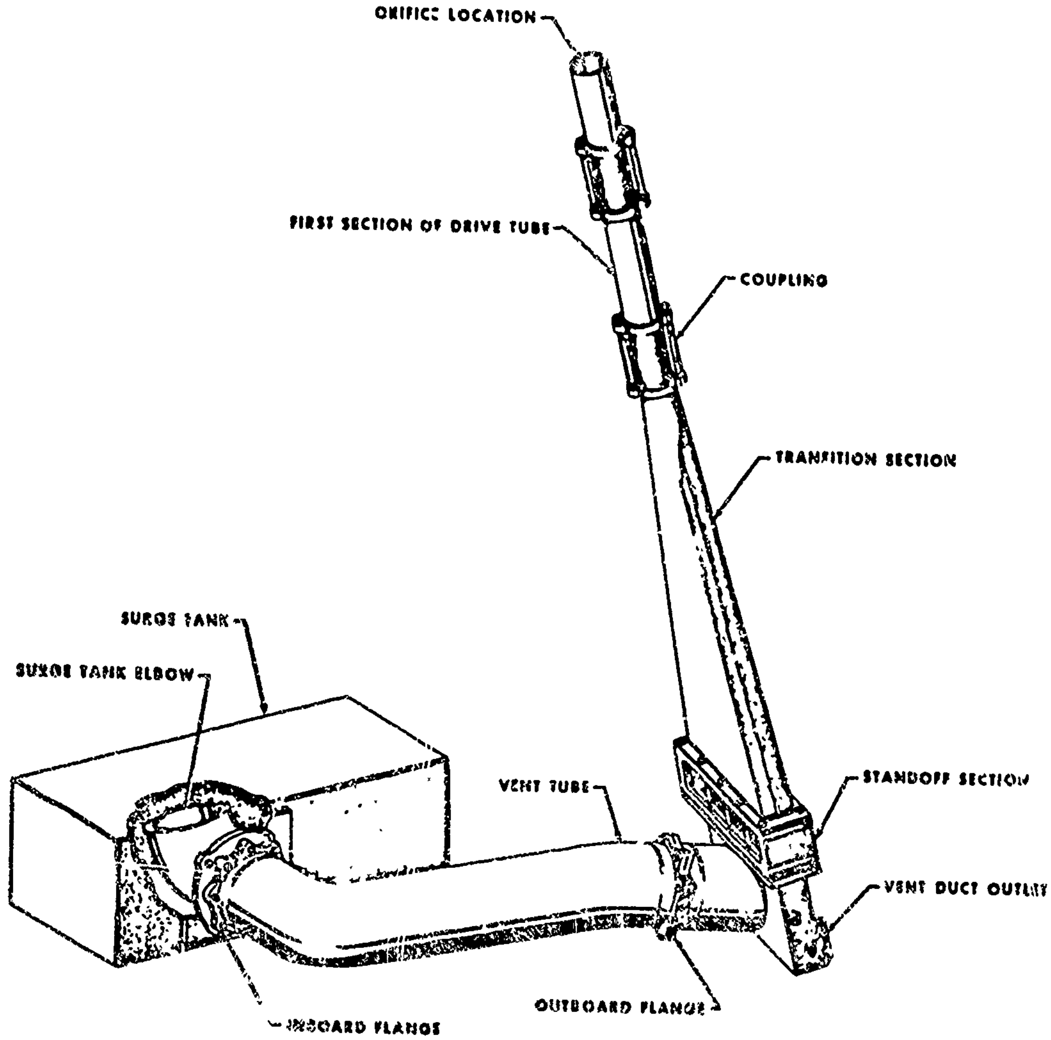



Figure 5. Sketch of Atlantic Research Experimental Assembly

significantly increased flame speeds were obtained by sealing off the end of the drive tube and incorporating two orifices in the drive tube. Some of the later tests with the drive tube produced combustion data that were remarkably similar to the data obtained during the discharge experiments with LTRI facilities.

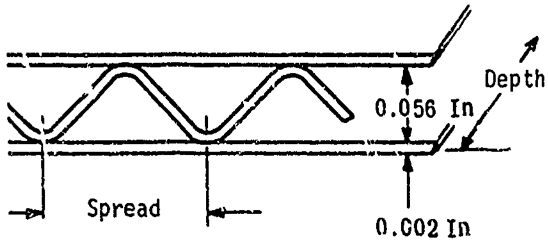
Ignition of the combustible mixture in the Atlantic Research experimental assembly was initiated by an aviation spark plug located at any one of several optional positions in the transition section and drive-tube sections. Open ports were sealed with aluminum foil during the filling operation and the seals were punctured with remotely controlled plungers just prior to ignition so they would not restrict the flow. The metered fuel-air mixture was fed into the system at the drive-tube end, and flames from burner tubes, connected through valves to either end of the system, were matched in height and color to ensure that the system was homogeneously filled. For safety purposes, the assembly was located in a combustion test cell, with all instrumentation and controls terminating in an adjacent, protected control room. In addition, the surge tank was protected by a blowout panel 8 inches in diameter. Instrumentation was identical to that used in experiments at LTRI.

This drive-tube assembly reasonably simulated the combustion mechanisms observed in the experiments with LTRI facilities, and was judged to be a convenient as well as an adequate device for testing flame arresters, etc., in a vent system.

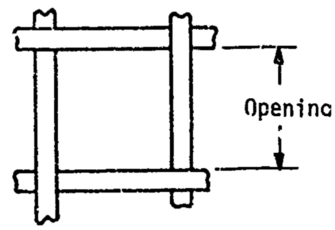
## 2. FLAME ARRESTER EXPERIMENTS

Flame arresters of several popular types were tested in the LTRI facilities and the Atlantic Research drive-tube assembly to determine their effectiveness, in general, in stopping flame propagation having the characteristics produced by lightning-discharge ignition. The test arresters were fabricated in several different ways using materials such as corrugated aluminum, corrugated stainless steel, a ceramic material, and various copper screens, with internal cell elements having the configurations shown in Figure 6. Each arrester was fitted to retaining rings and ring flanges ("instrument rings") on both faces, such that for a test it could be mounted into the vent system at the existing vent-tube flanges (see locations of inboard and outboard flanges in Figures 2 and 5).

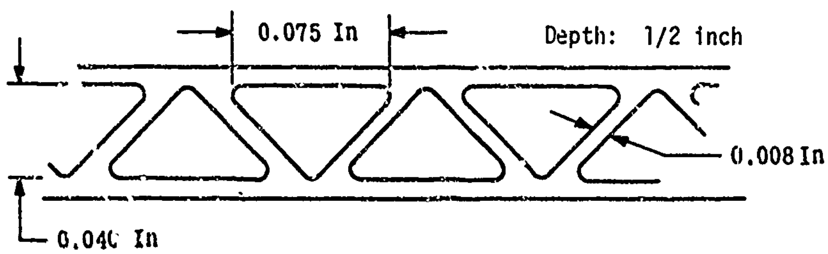
a. Cell configuration for corrugated type. Dimensions represented by two sets of numbers, depth x spread, in inches.




b. Cell configuration of screen. Single number identifies opening size, in inches.




c. Cell configuration for ceramic type. Dimensions constant and as indicated below.




d. Cell configuration for accordion type. Dimensions constant and as indicated below.

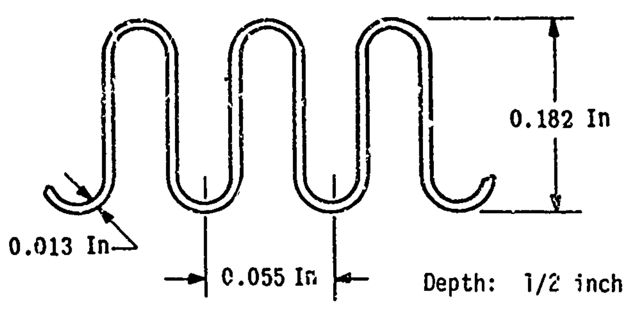



Figure 6. Dimensional Characteristics of Flame-Arrester Test Samples

The "instrument rings" were fitted with ports for a thermocouple and pressure transducer. Thus, it was possible to determine whether the flame approached and penetrated the arrester, or was stopped, by observing the responses of the upstream and downstream thermocouples. The pressure transducers measured any significant pressure drop through the arrester.

Results of flame-arrester experiments using the vent-line assembly attached to the LTRI High-Current Facility are summarized in Table III. All tests were conducted using a 115 per cent stoichiometric mixture, and the discharge was either direct or through a strip of aluminized tape, as in the combustion characterization experiments with this same facility. With the arrester positioned at the outboard flange, the tabulated flame speed was obtained only when flame penetrated the arrester. With the arrester positioned at the inboard flange, the tabulated flame speed is that for the flame as it approached the arrester.

Using the same apparatus described earlier for the high-voltage experiments, flame-arresters were tested with high-voltage discharges; results are summarized in Table IV. Arresters were tested at both the inboard and the outboard flanges of the vent tube. The fuel-air mixture and other conditions were the same as in the combustion characterization experiments with the high-voltage system.

Several flame arresters were tested in the Atlantic Research assembly. These experiments permitted a good control of flame speeds over a wide range, and provided a correlation with the results of the flame-arrester experiments performed at LTRI. Flame speeds ranging from 40 to over 400 ft/sec were attained in the drive-tube assembly, thus providing an evaluation of flame-arrester effectiveness, in the vent system of interest, over a broad range of flame propagation conditions. The important results of these experiments are summarized in Table V.

## 3. OTHER FLAME PROPAGATION EXPERIMENTS

Certain other aspects of the flame propagation phenomena associated with lightning ignition in aircraft fuel vent systems were also of interest. These included the potential effectiveness of industrial-type explosion suppression systems, and the possibility of plasma-like discharges and high-pressure

TABLE III

RESULTS OF FLAME ARRESTER EXPERIMENTS WITH HIGH-CURRENT FACILITY OF LTRI

| Arrester[a] | Arrester Position | Discharge Voltage (kv) | Pressure (psig) $P_1$ | Pressure (psig) $P_5$[b] | Flame Speed[c] (ft/sec) | Arrester Penetration |
|---|---|---|---|---|---|---|
| 1/2" x 0.050" | Outboard Flange | 5 (tape) | 2.9 | 3.1 | 41 | Yes |
| | | 10 (tape) | 6.3 | 6.7 | 125 | Yes |
| | | 15 (tape) | 8.3 | 8.8 | 150 | Yes |
| | | 20 (tape) | 3.8 | 11.3 | 125 | Yes |
| 1" x 0.050" | Outboard Flange | 5 (tape) | 2.4 | 2.5 | 44 | Yes |
| | | 5 (tape) | 2.9 | 3.1 | 49 | Yes |
| | | 10 (direct) | 0.5 | 1.0 | - | No |
| | | 15 (direct) | 0.6 | 1.1 | 10 | Yes |
| | | 20 (direct) | 2.9 | 3.1 | 48 | Yes |
| 1/2" x 0.050" | Inboard Flange | 5 (tape) | 2.5 | 4.5 | 16 | No |
| | | 15 (tape) | 8.8 | 10.6 | 80 | No |
| | | 20 (tape) | 2.5 | 13.8 | 75 | No |
| 1" x 0.050" | Inboard Flange | 5 (tape) | 1.6 | 3.1 | 19 | No |
| | | 20 (tape) | 15.8 | 8.7 | 120 | No |
| | | 25 (tape) | 16.3 | 23.7 | TC's Broken | |
| 0.055" Screen | Inboard Flange | 5 (tape) | 2.1 | 4.2 | 20 | Yes |
| Accordian | Inboard Flange | 5 (tape) | 0.7 | 3.5 | 21 | No |
| | | 10 (tape) | 3.5 | 11.0 | 60 | Yes |

[a] See dimensional explanations in Figure 6.

[b] See Figure 2 for instrumentation location.

[c] Measured between $T_1$ and $T_5$.

TABLE IV

RESULTS OF FLAME ARRESTER EXPERIMENTS WITH HIGH-VOLTAGE FACILITY OF LTRI

| Arrester[a] | Arrester Position | Discharge | | Pressure (psig)[c] | | Flame Speed (ft/sec) | | | Arrester Penetration |
|---|---|---|---|---|---|---|---|---|---|
| | | Voltage | Length[b] | $P_1$ | $P_5$ | $(T_1-T_4)$ | $(T_4-T_5)$ | Average $(T_1-T_5)$ | |
| 1" x 0.050" | Outboard Flange | 1,000,000 | 5" | 0 | 0 | - | - | - | No |
| 1" x 0.050" | Outboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | - | - | - | No |
| 1/2" x 0.050" | Outboard Flange | 1,000,000 | 5" | 1/4 | 1/8 | - | - | - | No |
| 1/2" x 0.050" | Outboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | - | - | - | No |
| Ceramic | Outboard Flange | 1,000,000 | 5" | 1/4 | 1/8 | - | - | - | No |
| Ceramic | Outboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | - | - | - | No |
| 0.067" Screen | Outboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | 10 | 15 | 12 | Yes |
| 0.067" Screen | Outboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | 14 | 33 | 19 | Yes |
| Ceramic | Inboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | 6.6 | 8.3 | 7.3 | No |

TABLE IV (CONTD)

| Arrester[a] | Arrester Position | Discharge Voltage | Discharge Length[b] | Pressure (psig)[c] $P_1$ | Pressure (psig)[c] $P_5$ | Flame Speed (ft/sec) $(T_1-T_4)$ | Flame Speed (ft/sec) $(T_4-T_5)$ | Flame Speed (ft/sec) Average $(T_1-T_5)$ | Arrester Penetration |
|---|---|---|---|---|---|---|---|---|---|
| 1/2" x 0.050" | Inboard Flange | [illegible],000,000 | 12" | 1/4 | 1/8 | 8.4 | 6.9 | 7.6 | No |
| 1" x 0.050" | Inboard Flange | 1,000,000 | 12" | 1/4 | 1/8 | 18.3 | 15.2 | 16.7 | No |

[a]See dimensional explanations in Figure 6.

[b]The discharge probe was located above the center of the vent inlet in all tests. Mixture ratio was 1.15 stoichiometric in all tests.

[c]See Figure 2 for location of instrumentation.

TABLE V

RESULTS OF FLAME ARRESTER EXPERIMENTS WITH DRIVE-TUBE ASSEMBLY

| Arrester[a] | Location | Arrester Penetration | Average Flame Speed (ft/sec)[b] |
|---|---|---|---|
| 1/2" x 0.050" | Outboard Flange | Yes | 166[c] |
| 1/2" x 0.050" | Outboard Flange | Yes (damaged arrester) | 190[c] |
| 1" x 0.050" | Outboard Flange | No | - |
| 1/2" x 0.050" | Inboard Flange | No | 156 |
| 1/2" x 0.050" | Inboard Flange | Yes (damaged arrester) | > 400 |
| 1" x 0.050" | Inboard Flange | No | 40 |
| 1" x 0.050" | Inboard Flange | No | 100 |
| 1" x 0.050" | Inboard Flange | No | 114 |
| 1" x 0.050" | Inboard Flange | No | > 400 |
| 0.050" Screen | Inboard Flange | Yes | 166 |
| Ceramic | Inboard Flange | Yes (shattered arrester) | 139 |

NOTE: All tests conducted with 1.15 stoichiometric mixture. Surge tank filled for all tests.

[a]See dimensional explanations in Figure 6.
[b]Average flame speed measured between $T_3$ and $T_7$ (See Figure 2).
[c]Indicated flame speed is that measured after penetration.

blast effects being generated, as suggested by previous investigators (Reference 7). Therefore, experiments were performed to study these problems in a preliminary manner.

Experiments to demonstrate the effectiveness of explosion suppression techniques were performed in conjunction with manufacturers of these devices. The components were of industrial type, and were not designed specifically for aircraft, but they served as convenient demonstrators of the factors involved. An explosion suppression system, in general, involves a rapid-response sensor to detect the presence of flame, and a unit to receive a signal from the detector, actuate a detonator, and rapidly release a quantity of liquid suppressing agent as a mist to inert the unburned vapors around the expanding flame front, all in time to arrest an explosion (a few milliseconds). Each example system was mounted in the Atlantic Research drive-tube assembly, using modifications as required to obtain the desired information. Typically, the effective responsiveness of each example system was observed for flame speeds up to 500 ft/sec. The results of these experiments are discussed later in this paper.

To clarify the role of plasma effects in the vent system following lightning discharges, additional experiments were conducted at the LTRI facilities with the vent-system assembly described previously for the discharge experiments. The objective was to determine the existence of plasma, and, if the plasma existed, to study its velocity and effective length of propagation into the vent system. Special plasma detectors were installed in the vent outlet and vent tube. These consisted of spark-plug probes for which it was established that plasma ions would produce a current flow across the probe's gap, through a series resistor to an oscilloscope and producing detectable signals on the oscilloscope as the ions passed between the probes. The number of probes along the path which indicated a signal would be a measure of the effective distance of penetration. However, no plasma was detected at any point in the tube during several tests with simulated lightning strikes. Time was not sufficient to permit a detailed study of this less consequential phenomenon, so no attempt was made to improve instrumentation sensitivities, identify any artifacts, etc.

Blast measurements also were made using apparatus located at the LTRI facilities and discharges from the High-Current Facility. Tape discharges were initiated both parallel and perpendicular to the vent opening at voltages from 5 to 20 kv. An Atlantic Research Corporation LC-13 blast probe was mounted in the center of the vent duct at the outboard flange, with adequate isolation from ground, vibration protection, and calibration. There was no evidence of any large-magnitude pressure pulses in the vicinity of the probe. If any were present, their duration had to be less than 1$\mu$sec to escape detection by the blast-probe instrumentation.

## LIGHTNING SUSCEPTIBILITY EXPERIMENTS

The vent-outlet lightning susceptibility experiments conducted by LTRI were of three types. (a) high energy direct discharges; (b) high voltage direct discharges; and (c) streamering tests. Most of the high energy tests were accomplished in connection with the combustion test program already discussed. In this work, a transparent plastic skin was placed over the vent outlet confining the fuel-air mixture inside the vent enclosure. The discharges were fired inside the plastic to ignite the fuel mixture. Ignition was obtained in every case. A few brief separate tests were conducted to verify the ignition of an effluxing air-fuel mixture by high energy electrical discharge. The high energy discharges were found not to ignite the fuel vapor on every discharge. Apparently the blast from the extremely high energy of the arc extinguished the flame in some cases.

A series of high voltage discharge tests were made on a KC-135 wingtip to determine the possibility of direct strikes to the vent outlet. As the point at which an artificial lightning discharge strikes a test object is determined to some extent by the source potential, the question arose as to the best potential to be used for tests of the vent system. The maximum voltage which can be developed on a given test configuration is a function of the gap length. The voltage rises across the test gap until sparkover occurs, limiting the maximum potential to this sparkover voltage. The sparkover potential varies with the potential rate of rise, but a rough figure of about one million volts per meter may be used for estimation.

Potentials as high as 10 million volts were available for these tests, but lower voltages and closer spacings had been found to give the maximum probability of strikes into the vent region. With potentials of 10 million volts and the required 30 foot spacing, discharges may strike anywhere within a large radius about the test object and are therefore very unlikely to hit the shielded configuration of the test vent outlet. With a spacing of about five feet to represent a closely approaching discharge channel streamer from the most critical angles, discharges contacted the outer edge of the vent scoop (defined as the indented area surrounding the vent outlet) but did not strike the vent outlet itself.

Streamering studies of the vent area were made with potentials of about 10 million volts. An observer-cage of aluminum was mounted on top of the high voltage generator for nearby photographic recording of the wingtip streamering. A photograph of the streamering of the wingtip taken from the observer cage 2 meters away, is shown in Figure 7 for a 10 million volt open circuit firing. A separate photograph at the same angle of the wingtip installation is shown in the upper photograph to facilitate orientation as to the vent location in the dark area of the photograph showing the streamering. The test disclosed severe streamering from the wingtip but none from the vent scoop or vent outlet; thus, none of the tests completed in this project have shown any streamering from the recessed vent outlet. It can be commented on the basis of these tests that, while many other types of vent openings would be subject to streamering, the type tested, the KC-135 recessed internal vent edge, is probably immune from cross-field streamering. Some uncertainty remains on all types of vents in relation to possible effects from closely approaching stepped-leader type streamering from a direct lightning discharge. Continuation of research to test voltages of about 20 million including wind sweeping effects may permit duplication of streamers exhibiting the stepped-leader process of natural lightning environment, and would more completely simulate the processes by which lightning discharges in the vent area may arise.

A final series of tests was made to evaluate the possibilities and limitations of lightning diverters for the protection of fuel vent areas. A special lightning rod utilizing a graded resistance diverting principle was tried. This type of rod has been used on aircraft for several years to divert lightning discharges away from vulnerable areas to the base of diverter rod, which has sufficient cross sectional area to carry the stroke currents without damaging the skin or airframe. The graded resistance diverter rod differs from a conventional lightning

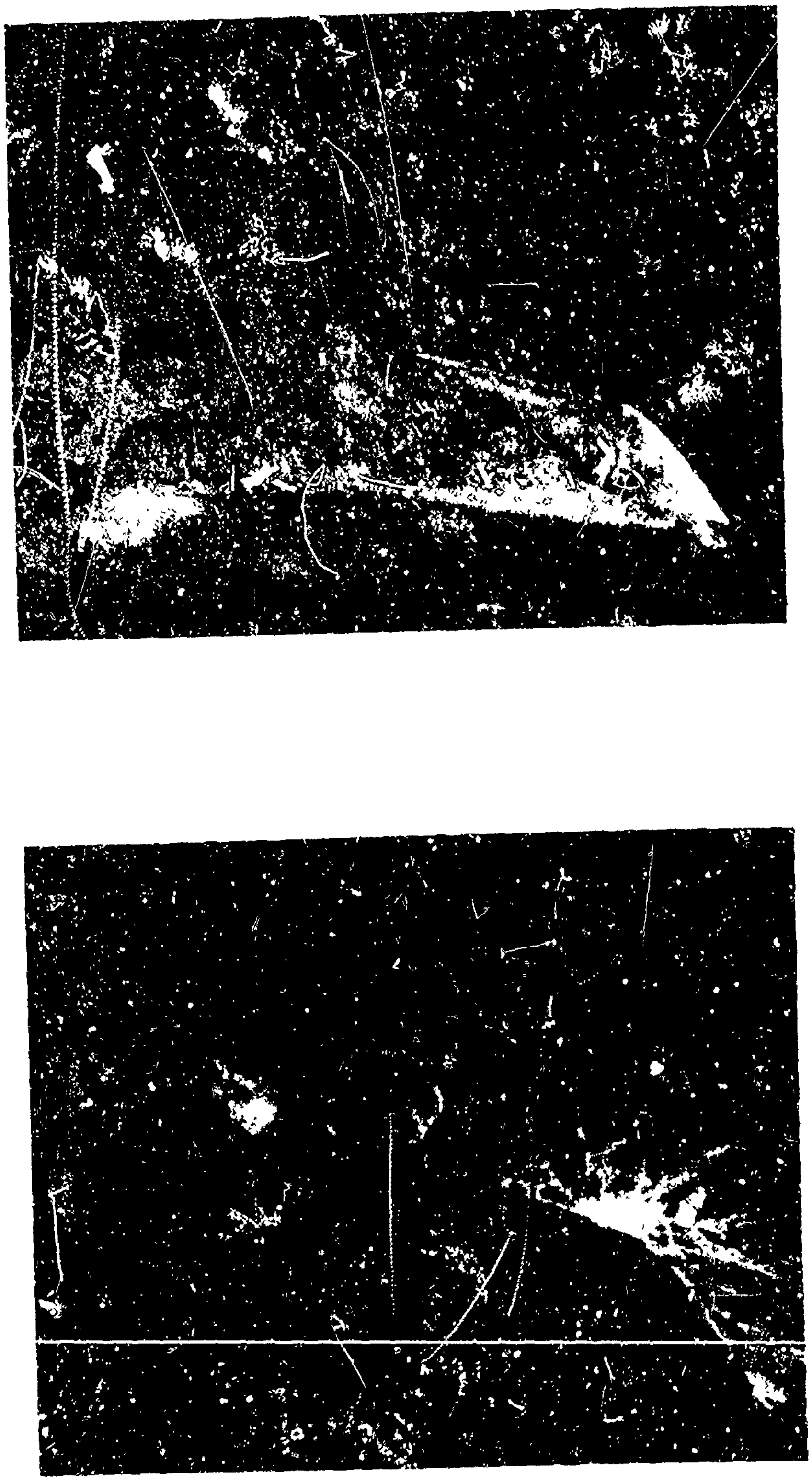


Figure 7. Streamering on Wingtip Produced by 10-Million Volt Discharge. (Viewed from test tank (below), with view of tip and vent area (above) for orientation.)

rod in that it utilizes high resistance coatings to perform its diverting function without introducing radio interference. Diverter rods have been successful in reducing maintenance in areas where much lightning damage occurs, such as in Air Force-Navy Airborne Early Warning operations. These aircraft operate in spite of weather, and therefore receive considerable lightning damage.

The use of graded-resistance lightning diverter rods for aircraft fuel vents has been approached with considerable caution because of the lack of basic information on lightning hazards mechanisms at fuel vents. As lightning rods provide a local diverting action the question arises — Is it better to divert a discharge away from a fuel vent which might otherwise be struck and in so doing attract a discharge into the fuel vent area which might not otherwise be struck? The basic information which is lacking is the distance from a fuel vent at which lightning may safely be permitted to strike. If this were known, lightning diverters could be located in questionable areas just outside the critical zone. At present, lightning diverters are used near fuel vents only when direct strikes are almost certain to occur to the vent outlet. In this case, the use of diverters to get the high energy stroke plasma away from the vent outlet is obviously desirable.

High voltage discharges were fired to the KC-135 wingtip, as illustrated in Figure 8, to demonstrate the operation of the diverter rod in bringing discharges to the rear of the wingtip. This seems to be the best point for lightning to strike, as it is farthest from the vent outlet. Model studies using high voltage discharges to determine the area of protection provided by a single diverter rod located at the trailing edge of the wingtip have indicated that discharges approaching the midspan area near the vent would be diverted to the rear, but not discharges approaching the front unless the diverter rod were unusually long and positioned at an extreme angle out from the wingtip. A diverter at the rear of the wingtip positioned 30° out from the line of flight would divert from 50 to 90% of strikes, depending on the diverter length.

## ANALYSIS OF RESULTS

The results of these experimental investigations were evaluated and correlated analytically. Particular consideration was given to establishing

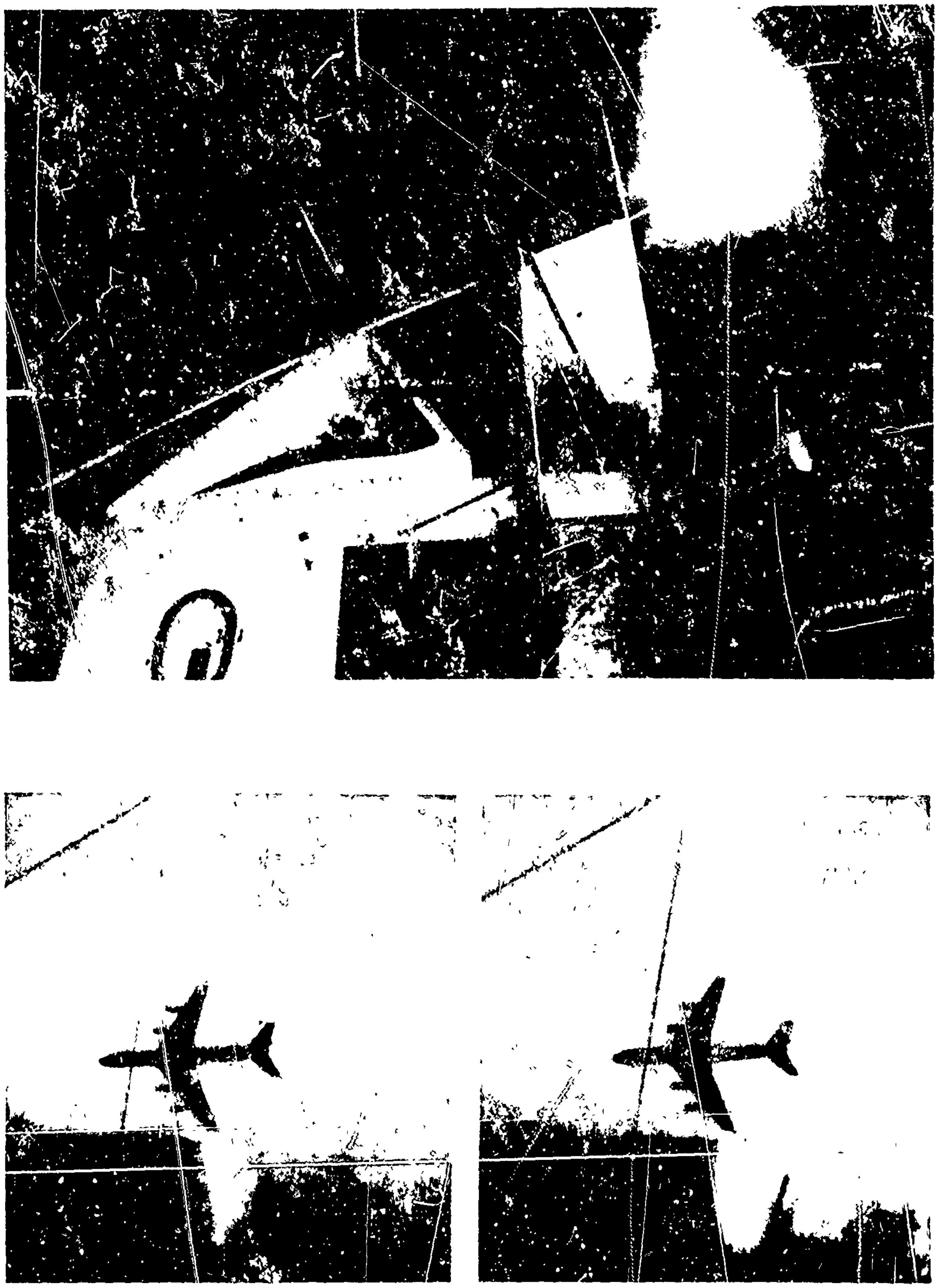


Figure 8. Two-Million-Volt Discharge to Diverter Rod Mounted on Trailing Edge of Actual Wingtip (above); Model Studies (below) Show Patterns for Discharge Strike Forward (left) and Trailing Edge (right) of Wingtip

the significance of the combustion phenomena observed, the inferable lightning hazard, and the apparent effectiveness of protective devices.

## 1. COMBUSTION PHENOMENA

Results of the flame propagation experiments established that simulated lightning strikes to the vent outlet can initiate flames which will propagate through aircraft vent systems similar in design to the test system, and into the fuel tanks when a flammable mixture is present. Flames initiated by the simulated lightning propagated through the vent duct with speeds up to 150 ft/sec and pressures up to 15 psig. The flame speeds and pressures appear to depend upon the total energy released in the vicinity of the vent outlet, and are not strongly affected by the current rate of rise. The high-current experiments produced the stronger combustion effects, showing average flame speeds higher than the laminar burning velocity of propane-air mixtures (1.4 ft/sec) by factors up to 100. Varying the mixture ratio from 0.80 to 1.50 stoichiometric had little effect on pressures or flame speeds.

Observed cyclic variation of the pressure data for the high-current experiments resembled the characteristics of an established standing wave which subsequently becomes damped. Theoretical considerations suggested that the vent system would respond to acoustic oscillations in a manner similar to an organ pipe. In an organ pipe, resonance generates a standing wave with a pressure node at the closed end and an antinode at the open end. During an acoustic cycle, air flows into and out of the open end, but the air at the closed end remains essentially at rest. At the open end, the cyclic variations of the particle velocity attain their maximum amplitude while the pressure remains constant and equal to the ambient. At the closed end, the particle velocity remains zero and the pressure oscillates at maximum amplitude for the system. Experimental pressure traces showed that after the initial lightning-generated pulse, the pressure oscillations near the vent-duct outlet vanished rapidly, while at the other end (near the surge-tank elbow), the oscillations decayed rather slowly. The recorded frequency agreed substantially with the calculated acoustic frequency for the system. It appeared that in the development of acoustic resonance the surge-tank elbow acted as a closed end, and the vent duct outlet as an open end. Thus, a fluctuating flow field is generated near the duct outlet where the lightning strikes and the flame enters. The lightning stroke provides the ignition source and the initial acoustic energy. The resulting flow field has the effect of accelerating the propagation of the entering flame under the influence of the flow turbulence. Not only does this mechanism account for the very high flame speeds that were observed, it could also mean that the inflow phase of the acoustic particle velocity may defeat a flame arrester by transporting ignited gas through the arrester channels.

## 2. EFFECTIVENESS OF PROTECTIVE DEVICES

The experiments conducted with typical flame arrester configurations under simulated lightning discharges in the vent system produced some significant information. Flame arresters of certain types will prevent flames initiated by simulated lightning discharges from propagating through the duct into the surge tank and fuel tanks. The flame arrester openings should have characteristic cell dimensions of the order of 0.05 inch. Stainless steel honeycomb or rolled corrugated strip with cell depths of 1 inch proved quite satisfactory, but screens and wire gauzes were ineffective in stopping high-velocity flame fronts generated by certain types of simulated lightning. The acoustical theory qualitatively explains the mechanism of flame propagation into the vent system and the reason for the defeat of flame arresters under certain conditions. This theory suggests that a careful acoustic analysis of the system should be an important consideration in locating flame arresters. Arresters mounted in the vent duct near the vent outlet were ineffective due to the inward mass flow developed in the duct by the simulated lightning discharge; however, those located near the surge tank were effective.

An analysis was also performed to study the potential hazards that might be introduced by the use of flame arresters in vent ducts. The results of this analysis, although not completely conclusive, nevertheless offer some useful guidelines for design considerations. For example, the pressure drop introduced by the 0.5-inch and 1-inch deep flame arresters with 0.050-inch openings is a small fraction of the total pressure drop through the vent system, and would require a small decrease in allowable fueling rate to maintain the design pressure drop through the vent system. However, the analysis showed that icing could block the flame arrester and the vent duct after repeated ascents and descents under the most severe atmospheric icing condition. This potential problem requires further investigation.

Results of the rather small sampling of experiments with standard industrial-type explosion suppression units indicated that flames which traversed the vent tube with average flame speeds of 140 ft/sec or less were prevented from propagating through the surge tank, but not flames having speeds of 230 ft/sec or greater. This suggests that these explosion suppression systems might prove effective in preventing the propagation of flames having speeds comparable to the maximum flame speeds measured during simulated lightning experiments, through the surge tank into the reserve tank. The margin of safety for the particular systems tested is limited, however, since the system may not be effective for flames with somewhat greater speeds.

The analytical studies conducted during this investigation also considered other methods for preventing flame propagation through the vent system after a lightning strike. The use of mechanical valves in the vent tube to isolate flow paths did not appear to be an effective or practical technique. Similarly, bladders to contain bulk fuel and separate it from air appeared to be a poor choice since the fuel may permeate the bladder and permit a flammable mixture to exist between the bladder and the wing skin. It appeared that inerting systems might be effective in preventing flammable mixtures from occurring, but very large quantities of inerting gas may be required to ensure nonflammability of the mixture under all conditions. Another method considered for preventing flame propagation through vent tubes is to increase the velocity of effluent vent gases to a point above the critical flow for flame flashback. This could be accomplished by addition of free-stream air or engine bleed air to the vent system. Analysis indicated that a continuous net outflow rate of 5 ft/sec is required to prevent flashback in a 5-inch diameter vent tube, either through the core of the gas flow or along the walls. Since flames may propagate into the duct at speeds of the order of 150 ft/sec, a minimum total outflow rate of the same order may be required to provide the desired outflow conditions at all times. New aircraft designs might consider the use of a scoop to bring in air, and mix it with the fuel tank effluent to provide the required high exit velocities to the mixture.

CONCLUSIONS

If lightning can initiate combustion near the vent outlet, then under certain conditions, the flame could propagate through the vent system into the fuel tanks. Acoustical effects, arising from resonant conditions produced by the lightning strike, lead to flow fluctuations and relatively high flame speeds through the vent lines. In this respect, the surge-tank area and components are the most critical elements of the duct configuration that permit flame arresters (which appear at this time to be the most promising passive devices for protection against flame propagation) to perform successfully at the inboard position. Careful acoustic design of a vent system, it appears, could provide locations for flame arresters that would enhance their capabilities. Explosion suppression systems, increased vent velocities, and possibly inerting, also show promise and should continue to be considered.

Recessed vents such as are used on the KC-135 aircraft resist direct lightning strikes and streamering, but stepped-leader strikes and wind sweeping may enhance the probability of strikes to the vent. Therefore, a graded resistance lightning diverter may have application in protecting the vent area. More information on the extent of the area to be protected and the mechanics of the phenomena of lightning strikes to aircraft is required before this approach can be used with confidence.

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AFAL-TR-68-290
PART II

# A METHOD FOR ESTABLISHING LIGHTNING-RESISTANCE/ SKIN-THICKNESS REQUIREMENTS FOR AIRCRAFT

R. O. Brick

The Boeing Company

Lightning is an electrical phenomenon that consists of a discharge of static electricity between two charge centers or between a charge center and ground. The intent of this paper is to examine the contact of a lightning channel with an airplane in flight. It will be necessary to discuss some aspects of the lightning-discharge channel to arrive at a better understanding of how the contact mechanism relates to requirements on aircraft-skin thickness for critical areas. The paper will not go into detail on the mechanism of charge buildup; however, the discharge mechanism will be discussed.

## THE LIGHTNING DISCHARGE

Lightning is usually generated in the cumulonimbus, or thunderhead, cloud formation. The charge within the cloud increases until the electric field intensity between charged centers is sufficient to produce a faintly luminous low-energy step leader. The thread-like leader advances in hesitating steps to another charged center within the same cloud structure, another cloud, or the ground. The advancing step leader can either have no or many side branches, depending upon variations in the air. When the advancing leader contacts another leader advancing from the ground or another cloud, the charge centers are connected together electrically as if by a fine wire and a bright visible flash occurs, accompanied by a high electrical charge transfer. The sudden change in electrical field may cause other electrical charges nearby to discharge into the same path and sustain the discharge for a considerable time (occasionally more than 1 second). A visible flickering effect may sometimes be noted. The discharge of other electrical charges into the same path is sometimes called a restrike, although it occurs in the same flash that is seen visually. The time between restrikes is usually in the order of milliseconds. A current-versus-time waveform for a typical lightning strike is shown in Figure 1 (Reference 1).

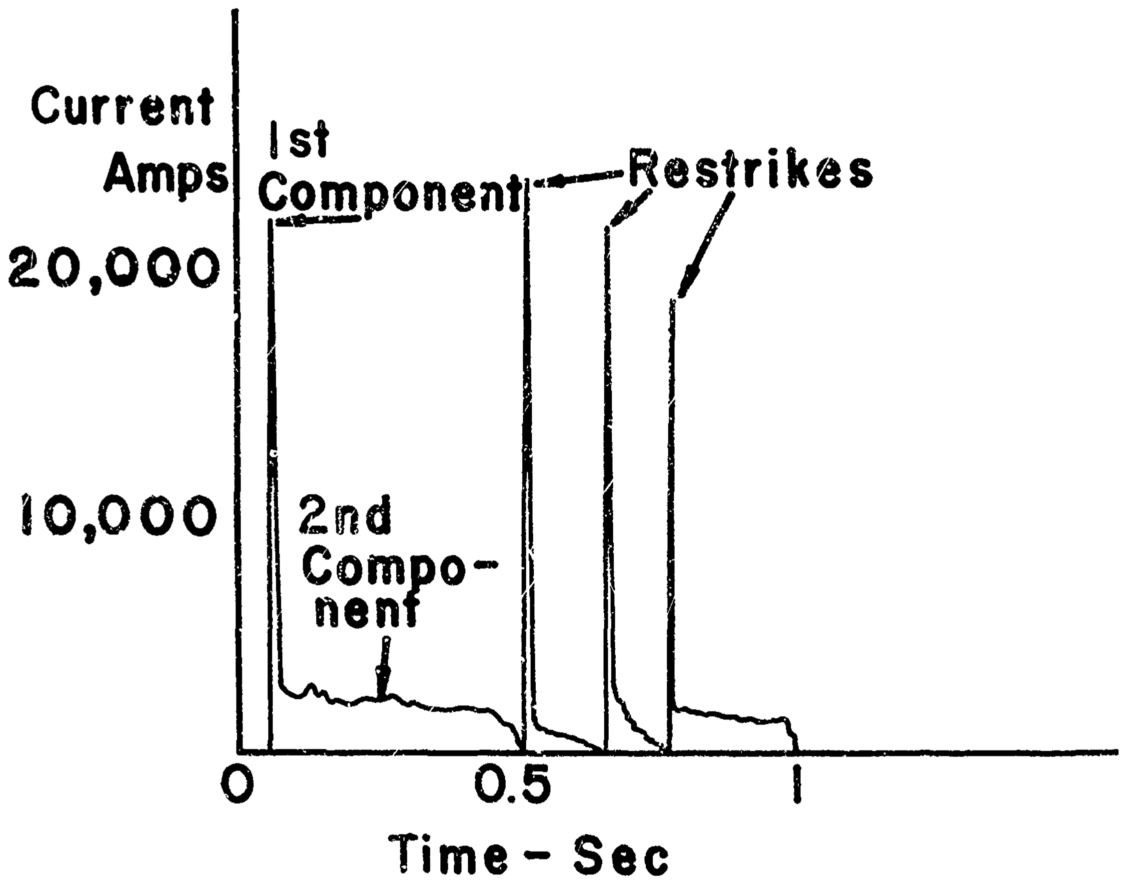


Figure 1. Typical Lightning Strike Current Vs. Time Waveform

The maximum initial current and restrike current may exceed 50,000 amperes (218,000 amperes is the maximum current ever recorded); however, they represent only a small portion of the total charge transfer because of their short (microseconds) duration (Reference 1). The major portion of the charge is transferred after the high current surges. The total charge transfer in coulombs is equal to the current (rate of charge flow) multiplied by the time. The initial high current surge may be only a few coulombs, whereas the longer duration, secondary, low current surge may be several hundred coulombs. It is the high-coulomb transfer that causes major damage to metal aircraft skins, especially if the contact point remains in a localized area.

## LIGHTNING STRIKES TO AIRCRAFT

An aircraft in flight may be contacted by a step leader that is advancing toward another charge center or ground. The step leader contacts the aircraft at one extremity and leaves from another extremity. The ionized lightning channel is relatively stationary with respect to the movement of the aircraft; depending upon the orientation of the channel with respect to the aircraft's flight-path, the channel may or may not be swept over the surface of the aircraft.

Figure 2 shows several possibilities for lightning-channel contact. Air in the portion of the lightning-discharge channel is at a very high temperature. An aircraft provides a good conductor in the path of the lightning-discharge channel. Current density per unit of volume is low on the aircraft and thus minimum heating occurs, except at the entry and exit points where the high-temperature channel, which may be only several millimeters in diameter, contacts the aircraft. If the high-temperature channel remains at one point, severe melting, burning, and puncture of the surface can occur. Any movement of the channel over the surface of the aircraft will reduce the damage because the energy is distributed over a larger area.

### 1. DAMAGE TO METAL SKIN

Metal skin surfaces of an aircraft receive varying amounts of damage from lightning. Damage of the extremities is the most severe because lightning usually makes initial contact with these surfaces, due to the high electrical

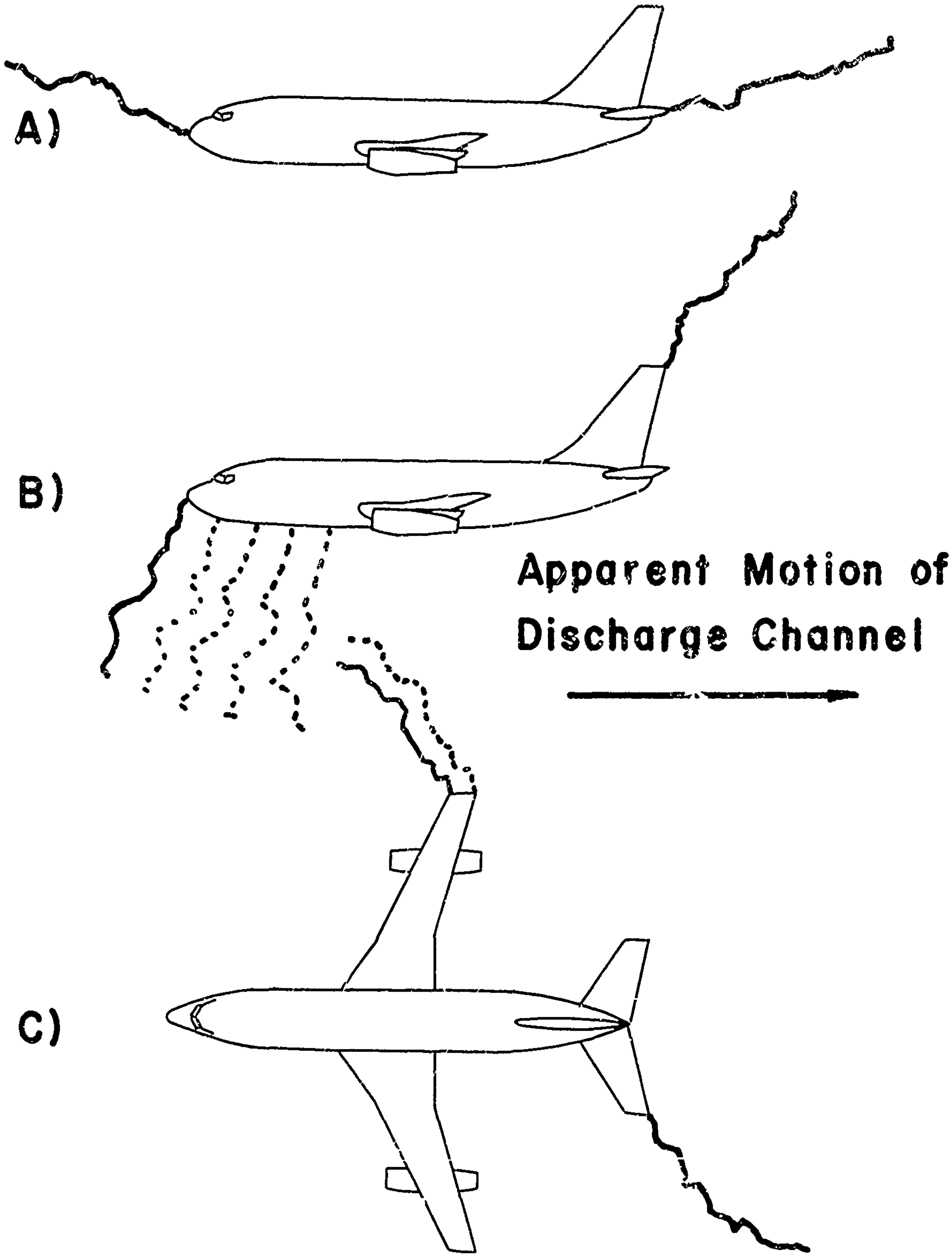


Figure 2. Lightning Discharge Attachment to Airplane

(a) Lightning stroke in flight path does not sweep aft but tends to remain at both attachment points.
(b) Stroke in vertical plane sweeps aft.
(c) Stroke in horizontal plane sweeps aft.

field gradients, and can remain attached for the duration of the strike. These areas are called direct-stroke zones and are defined by FAA (Reference 2) as Zone 1 areas. Damage to a Zone 1 area (wingtip) is shown in Figure 3.

Metal skin surfaces inboard on the airplane do not receive extensive damage due to the movement of the channel attachment point and the relatively low field gradients. The channel may try to attach to a local area, as evidenced by spot weld type melting of the skin surface, but airflow will move or sweep the channel aft from the initial extremity contact point to a trailing edge point until the energy transfer is completed. Areas of the airplane that are subjected to sweeping action of the lightning channel contact are called swept-stroke zones and are defined as Zone 2 areas. Figure 4 shows typical aircraft damage experienced in Zone 2.

2. DAMAGE TO NONCONDUCTING SKIN SURFACES

Nonconducting skin surfaces can be punctured by the lowest energy direct or swept lightning strike. If the surface is in a critical area, it may be protected by metal gridwork or with a metal skin bonded to the outer surface. If this is done properly, the lightning/fuel ignition threshold resistance can be increased significantly.

LIGHTNING-RESISTANCE/SKIN-THICKNESS REQUIREMENTS

When lightning contacts skin surfaces covering fuel or fuel vapor areas, skin thickness plays a very important part in the prevention of ignition due to skin puncture or interior surface heating. Other ignition sources exist, such as joint sparking and induced voltages into the aircraft's wiring; therefore, skin thickness is only one of many lightning protection considerations. However, skin thickness requirements are important because severe weight penalties result if skin thickness is increased arbitrarily to meet unrealistic lightning protection criteria.

Skin thicknesses to safeguard against ignition of fuel or fuel vapor are presently dictated by the most severe lightning damage, Zone 1. The skin thickness recommended for Zone 1 areas by the FAA (Reference 1) is 0.08 inch (aluminum or equivalent). However, fuel tanks are usually located in minimum

Figure 3. Lightning Damage to Zone 1 Area (Wingtip)

Figure 4. Lightning Damage to Zone 2 Area (Engine Pod)

damage swept-stroke Zone 2 areas, where no lightning damage occurs. The extremely conservative approach for fuel tank skin thicknesses was tolerated in the past because the gage aluminum to meet structural requirements has always exceeded that required to resist skin puncture by swept lightning. Tests have shown that for aluminum skins, puncture by the lightning channel is required before ignition of fuel will occur (Reference 3).

With advances in structures technology, more realistic skin-thickness criteria should be established for enclosing fuel and fuel-vapor areas in the swept-stroke zones. It would also be desirable to identify other types of skin for use in Zone 1 areas that would be equivalent to 0.08-inch aluminum.

The basic problem in establishing a better skin thickness design criterion for fuel and fuel vapor areas, where integral tank techniques are used, is to determine the maximum amount of lightning energy incident upon the Zone 1 and Zone 2 areas. If fuel or fuel vapor is located in a Zone 1 area, the maximum amount of energy can be determined on a skin thickness equivalent basis. Since a skin thickness equivalent to 0.08-inch aluminum is required for Zone 1, equivalent thicknesses can be determined by simulated lightning tests. Recent studies have shown that the test condition must be carefully chosen to ensure that the criteria are valid. If fuel or fuel vapor is located in Zone 2 areas, the problem is more difficult because there is no aluminum skin thickness equivalent established for Zone 2, although, it is known from aircraft damage records that less than 0.08-inch aluminum would be needed.

A study program was initiated by The Boeing Company to aid in providing an experimental and analytical basis from which skin-thickness criteria could be established. The objectives of the study were to:

(1) Establish the lightning-energy-puncture threshold for sheet aluminum with wide variations in current and time.

(2) Investigate hot spot ignition thresholds for a typical poor electrical conductor. Titanium was selected for these tests and analyses because it is the basic skin material being considered for the Supersonic Transport.

(3) Simulate the lightning channel attachment in swept-stroke zones to learn more about the attachment mechanism and to provide a basis from which test techniques could be established for detail design studies and design-verification tests.

(4) Review skin damage caused by lightning in direct- and swept-stroke zones to provide a natural-lightning damage record to which artificial-lightning test results could be compared.

The method for establishing skin thickness requirements was derived from the results of the study programs. It was assumed at the beginning of the program that if one study area was unsuccessful, enough information could be obtained from the other studies to develop skin-thickness requirements; however, this was not the case as all areas of study provided vital information which was required to develop skin thickness criteria. The areas of study are briefly discussed below.

## 1. ALUMINUM-SKIN-PUNCTURE THRESHOLD

Tests for obtaining a puncture-ignition threshold for aluminum were based upon an energy-flow rate corresponding to hundreds of amperes of current in a 2- to 5-second time period; however, the coulomb value to penetrate aluminum from one test to another varied significantly. This was previously attributed to the wandering nature of the arc or the inaccuracies in measuring the coulomb values.

Tests were conducted at Boeing to determine the cause of some of the above variations (Reference 4). The test setup is shown in Figure 5. A photo cell was used to detect the spark showers on the back side of the panel to determine the exact time of puncture as displayed on a dual-trace oscilloscope, which also recorded the current value. The arc was stabilized by using a tungsten probe located directly above the panel; this simulated a condition much more severe than would be encountered in swept-stroke zones, and was more representative of the direct-stroke areas where the curvature of the extremities tends to make the arc channel remain at one spot. The most significant result from these tests was the variation in the electrical charge transfer (coulombs) required to puncture the aluminum, as shown in Figure 6.

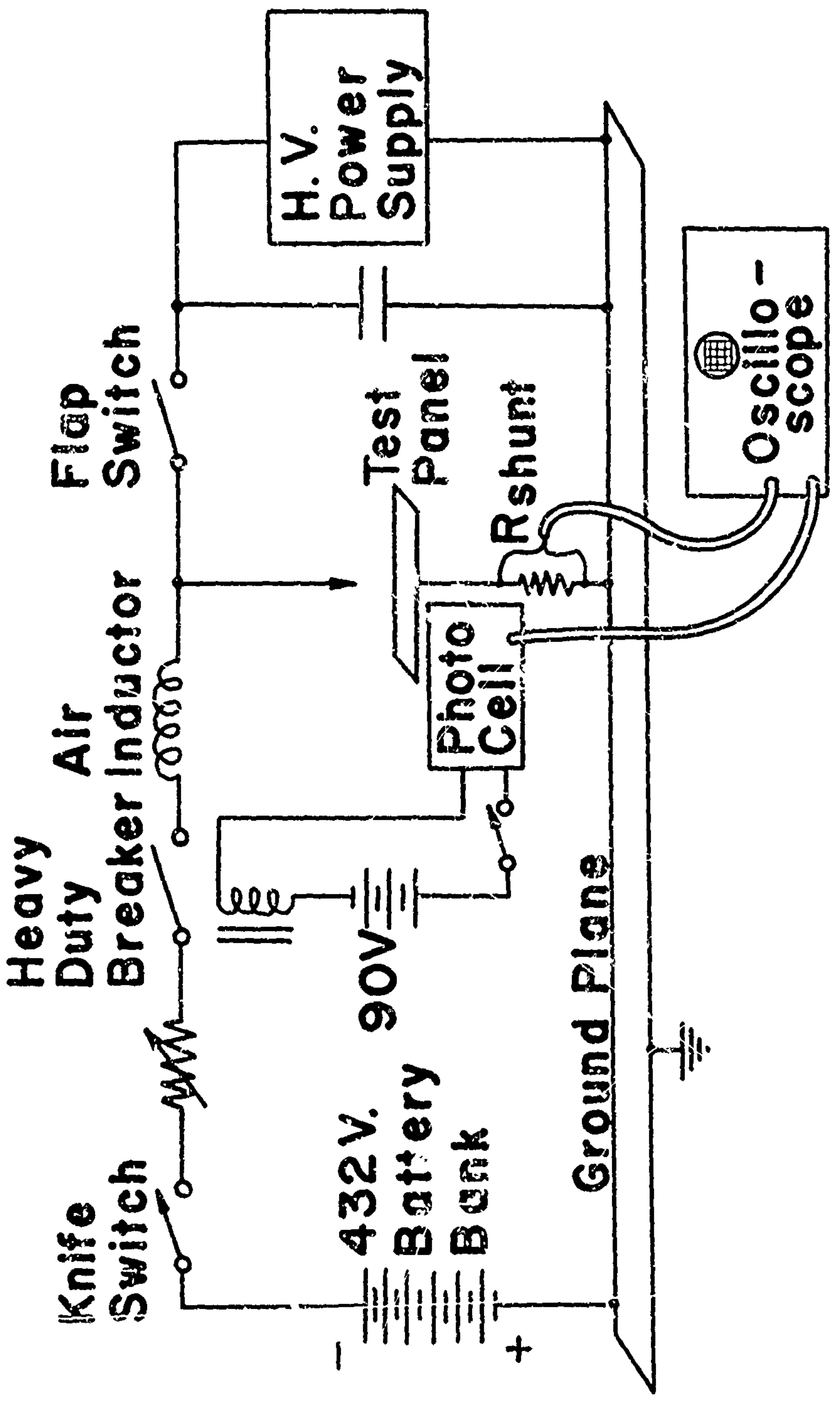


Figure 5. Aluminum Skin Puncture Test Setup

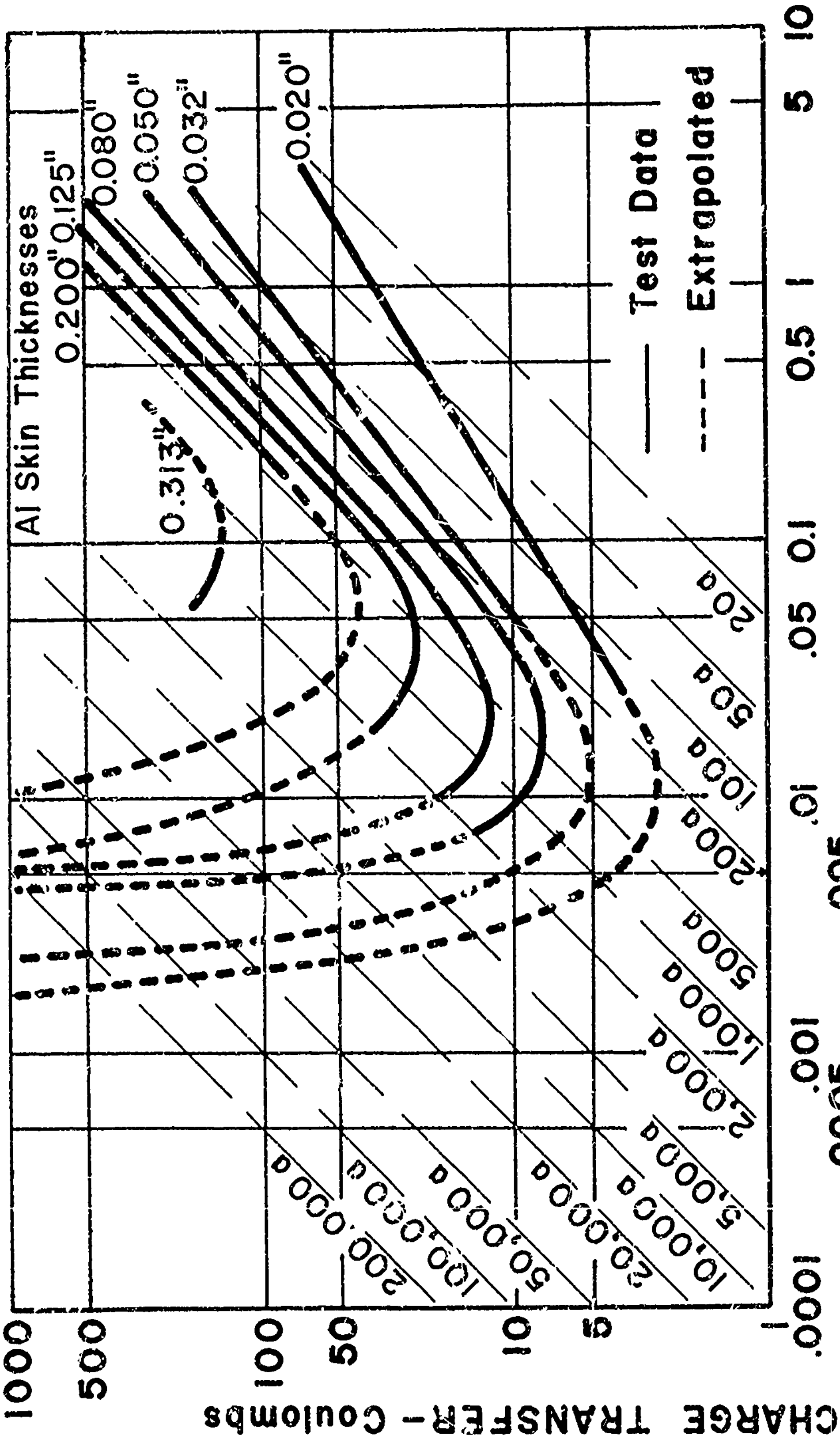


Figure 6. Aluminum Skin Puncture Threshold

All of the graphs are plotted on log-log scales; time in seconds is plotted on the abscissa, and the total charge in coulombs on the ordinate. Constant-current lines are shown as straight lines with a positive slope.

The tests showed that a constant-coulomb value could not be used for test criteria, and that the test conditions for Zone 1 would be significantly different from those for Zone 2 since the energy required to puncture the aluminum was related to the rate of charge transfer (current value). This accounted for some of the variations in the coulomb value required to penetrate a given thickness of aluminum.

## 2. TITANIUM HOT-SPOT THRESHOLD

Fuel vapor enclosed in titanium tanks can be ignited by simulated lightning without puncturing the skin surface (Reference 3). Ignition is due to the temperature rise of the inner skin surface, and occurs when the hot-spot reaches a threshold temperature of approximately 2400°F.

The hot-spot thresholds for titanium are shown in Figure 7 (Reference 5). The tests were conducted in a manner similar to those for aluminum-puncture except a temperature-sensitive paint was used to determine the threshold conditions, and the charge transfer was started with a fine wire. The test setup is shown in Figure 8.

The surprising thing about the titanium tests was that the hot-spot threshold corresponded to a constant coulomb value for each thickness of titanium and was independent of current value except at very short time durations.

Since the slope of the curves for aluminum and titanium are completely different, a simple titanium-to-aluminum thickness-equivalent to prevent lightning ignition cannot be established. It is therefore necessary to examine in greater detail the attachment mechanism of the lightning channel in swept-stroke zones (where fuel tanks may be located).

## 3. SWEPT-STROKE ATTACHMENT MECHANISM

The Boeing Company retained Lightning and Transients Research Institute to investigate techniques in simulating swept-stroke phenomena (Reference 6).

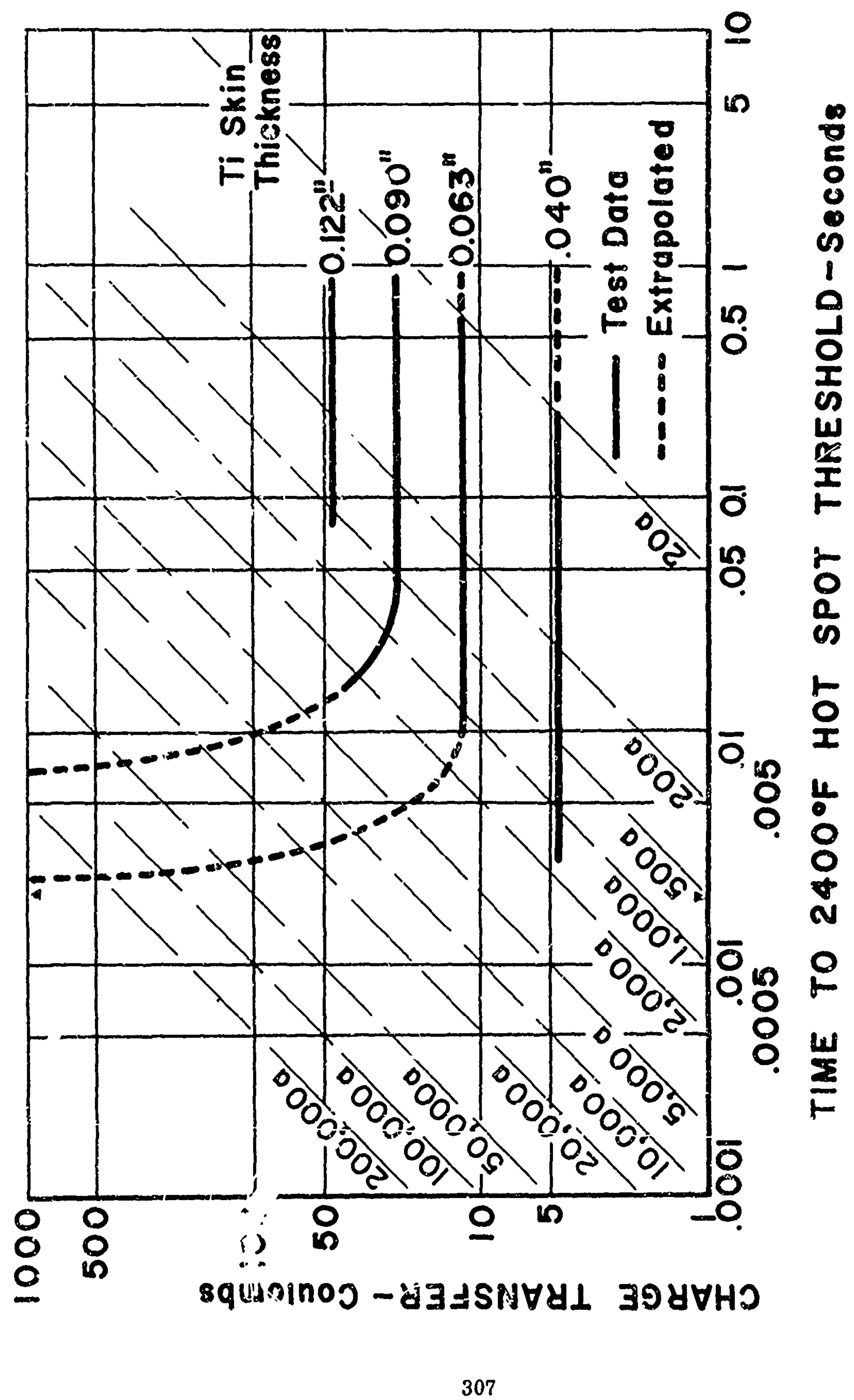


Figure 7. Titanium Hot Spot Threshold

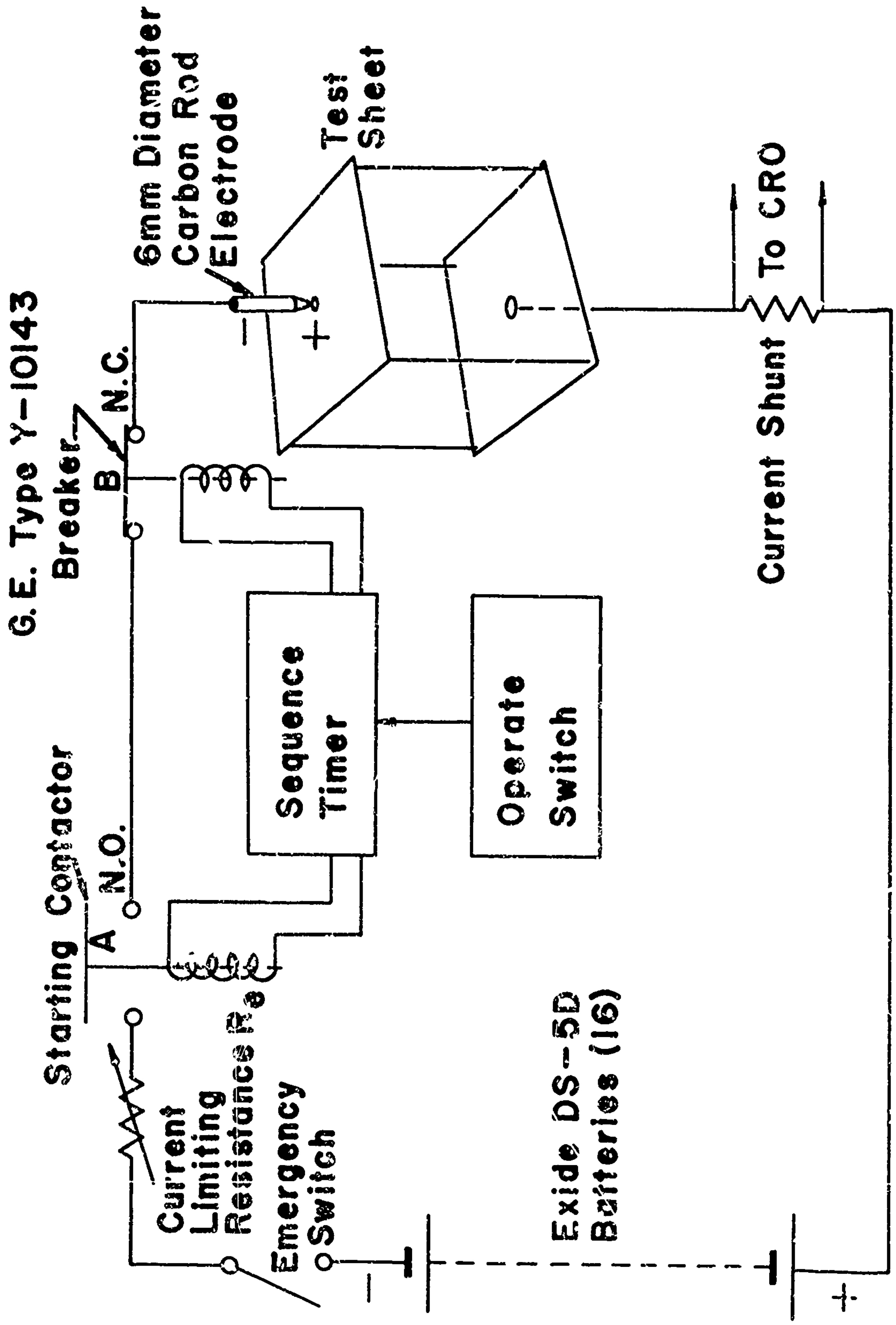


Figure 8. Titanium Hot Spot Test Setup

Two techniques were investigated to simulate the relative motion between the aircraft-skin surface and the lightning channel, which remains relatively stationary with respect to the air. The first method was to discharge simulated lightning to a rotating disk to simulate the aircraft surface. The second method was to use a blower to move air over a stationary skin panel. A simulated lightning discharge was initiated at the upstream end of the panel and was allowed to track downstream with the moving airflow. The second method was used to determine the time that the channel would remain at any one spot on the panel. Fastax movies, taken from the side as shown in the test setup in Figure 9, were used to determine the time for each attachment.

The channel-attachment movement is shown in Figure 10 with a sequence of selected Fastax photos. The minimum time between photos is approximately 0.8 millisecond. The time between frames in the original film was approximately 0.4 millisecond. By analyzing the film data, the channel-attachment time was determine. The materials investigated were anodized and alclad sheet aluminum, and type 6-4 sheet titanium.

The swept-stroke experiments revealed that the lightning attachment point moved in a stepping manner. Airflow was the main factor in moving the attachment point; the channel bent over close to the skin in order to contact and reattach to the skin. The most significant result of the tests was that the channel reattached during the high-current flow and the current did not approach zero before reattachment occurred.

The material-surface condition was an important factor in channel reattachment and must be considered in the selection of skin materials for fuel and fuel vapor areas. Of the three materials studied, the lightning channel attachment point dwelled longest on anodized aluminum, approximately 0.005 seconds with an air velocity of approximately 155 mph. The long dwell time was probably due to the insulation qualities of the anodized surface which tended to prevent reattachment until the voltage drop on the channel close to the skin surface exceeded the dielectric voltage breakdown strength of the anodized coating. The channel then punctured the coating and reattached to the skin.

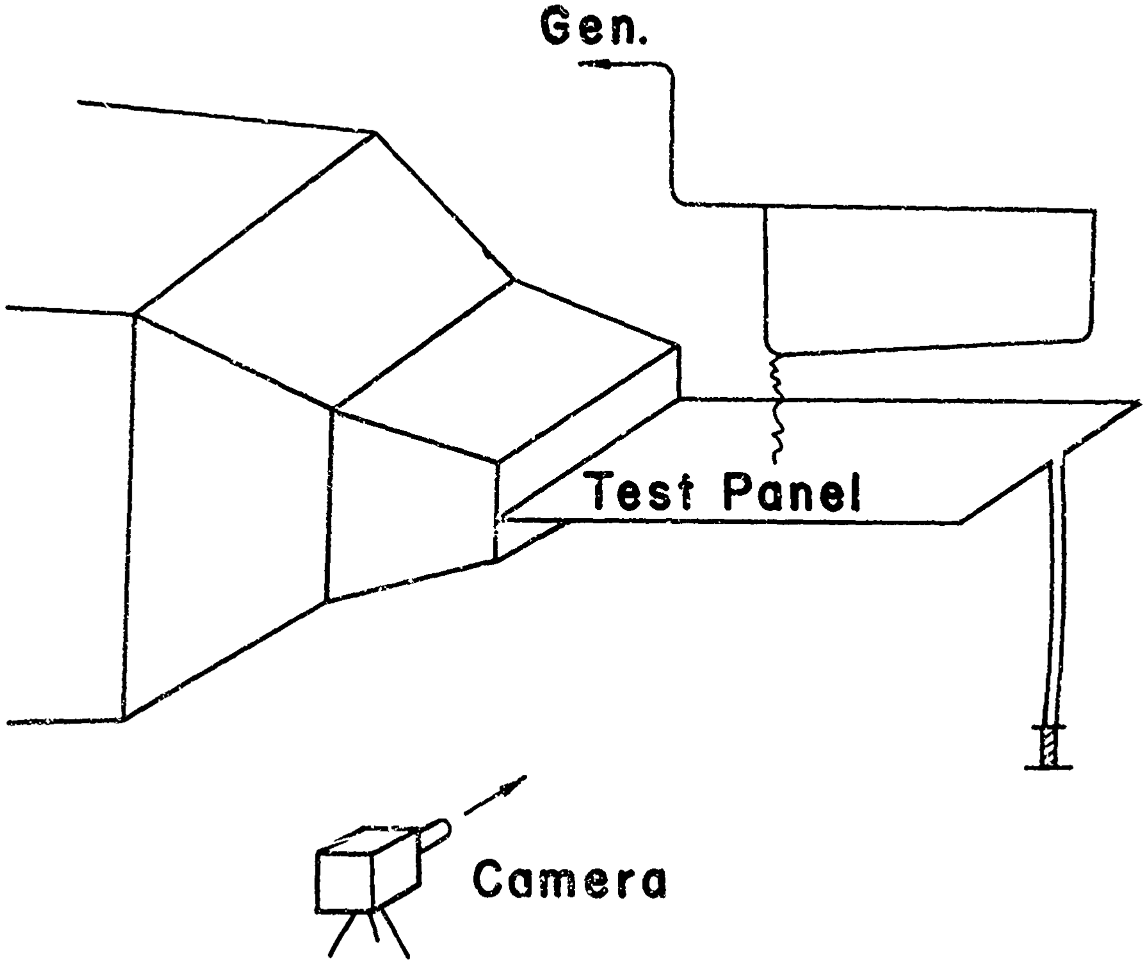


Figure 9. Swept Stroke Simulation Test Setup


Figure 16. Swept-Stroke Channel Movement

(a) Direction of Air Flow
(b) Attachment of Lightning Channel to Test Panel

Reattachment also varied with the electrical polarity of the discharge. The dwell times were usually longer when the skin surface was negative with respect to the probe.

4. NATURAL AND ARTIFICIAL LIGHTNING DAMAGE COMPARISON

The artificial lightning damage compared very well to natural lightning damage to airplanes. Aritificial lightning dendritic patterns and pitting patterns were similar to patterns produced on aircraft by natural lightning in swept-stroke zones. The data indicate that the sweeping motion of the lighining channel attachment point can be simulated in the laboratory and the test techniques can be a useful design tool.

5. THE METHOD FOR ZONE 1 SKIN THICKNESS REQUIREMENTS

For Zone 1, the skin thickness must be equivalent to 0.08 inch aluminum; the method to establish lightning-resistance/skin-thickness requirements is as follows:

(1) Determine the ignition mechanism for the material, e.g., skin puncture, hot-spot, etc.

(2) Establish the ignition threshold for a wide range of current-time and material thicknesses being considered for fuel or fuel vapor areas.

(3) Compare the ignition threshold curve to the 0.08 aluminum curve, (Figure 11). The coulomb value must be equal to or exceed the values for aluminum for the maximum time duration of natural lightning, i.e., the curve must lie above the 0.08 aluminum curve. The maximum strike duration is approximately 1.5 seconds (Reference 1) and it is proposed that a maximum time of 2 seconds be used for test purposes.

If single sheet titanium were used to enclose fuel in a Zone 1 area, the material would have to be approximately 0.4-inch thick (see Figure 11) and probably would not be considered because of the excessive weight penalties; however, lightweight composite sandwich skin panels that use electrical/thermal insulating materials between the skins can be used to provide the required resistance to lightning for Zone 1 fuel tank areas. Composite skin panels are currently being tested. Preliminary testing for sandwich skins has shown that panels using a

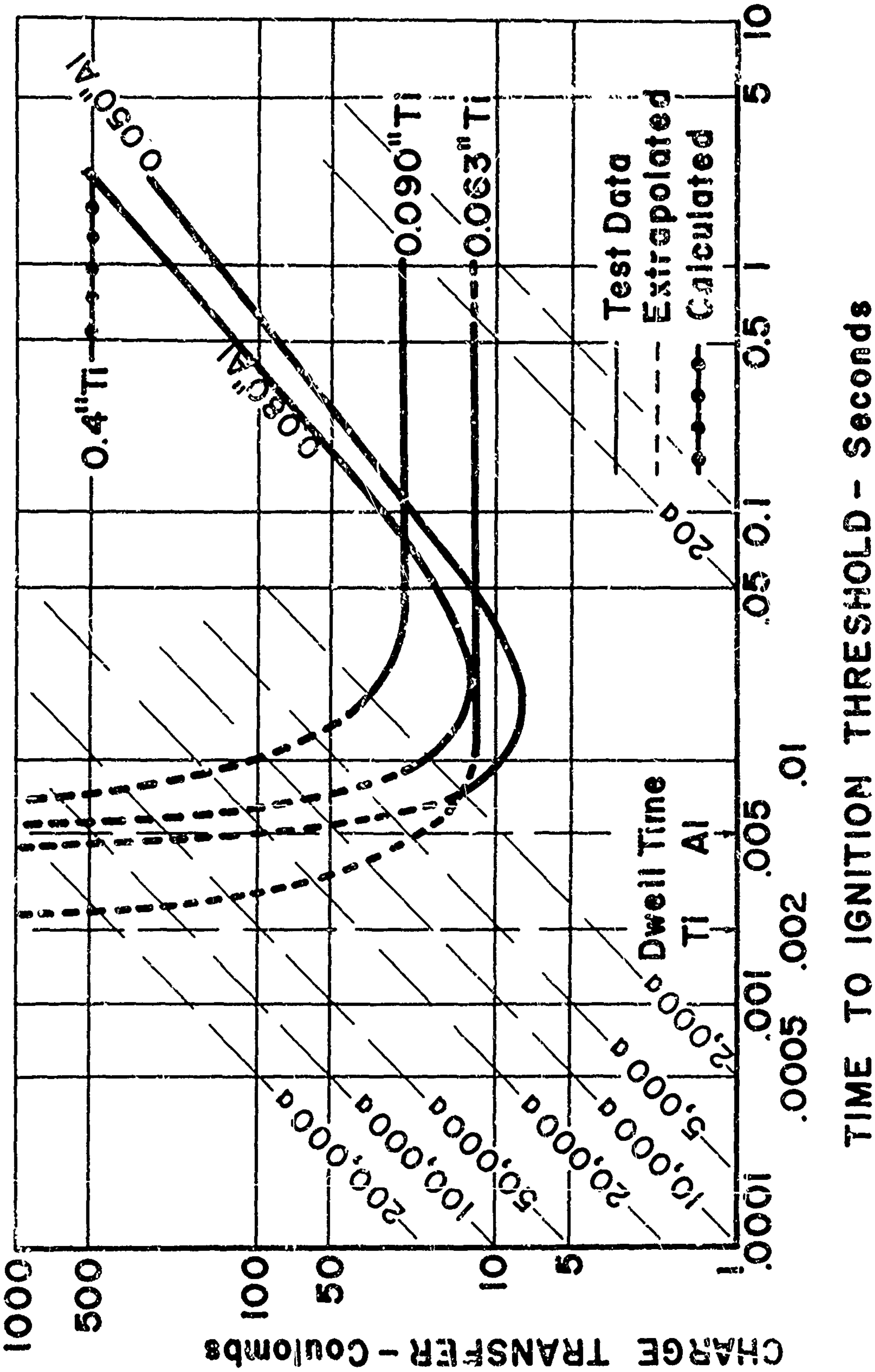


Figure 11. Comparison of Aluminum and Titanium Ignition Thresholds and Dwell Time

metal skin and a plastic core, or an insulating surface between the metal skin and metal core, will have excellent lightning resistance characteristics. Panels that use metal skins and cores welded or brazed together have only fair lightning resistant characteristics; however, they should be better than an equivalent weight single sheet titanium skin.

## 6. THE METHOD FOR ZONE 2 SKIN THICKNESS REQUIREMENTS

For Zone 2, the swept-stroke dwell time data is used to establish the lightning-resistance/skin-thickness as follows:

(1) Determine the ignition mechanism for the material, especially for short-duration high-current pulses.

(2) Establish the ignition threshold for a wide range of current-time and material thickness variations being considered for fuel and fuel vapor areas.

(3) Establish the maximum lightning channel dwell time for the material by swept-stroke test techniques.

(4) Compare the ignition threshold data to maximum dwell time (Figure 11). The safe skin thicknesses are represented by the ignition threshold curves which are to the right of the maximum dwell time line for the material.

Referring to Figure 11, and assuming that the extrapolation of the 0.063 curve is realistic and the dwell times for aluminum and titanium are valid, the minimum titanium skin thickness for swept-stroke zones that would be comparable to 0.08-inch aluminum would be 0.063-inch titanium. Additional testing will be required to verify the extrapolated data and the dwell times.

## CONCLUSIONS

A method to establish lightning-resistance/skin-thickness requirements for aircraft has been described. The same test and anlysis techniques could be applied to sandwich-type skins and nonmetal skins.

The importance of the swept-stroke studies is apparent because the dwell time data tends to define the lightning charge transfer levels that could be expected in Zone 2 areas. The maximum charge transfer to an attachment point would not be greater than the product of the dwell time and the current value.

For example, the lightning channel stayed attached to a localized area of anodized aluminum for approximately 5 milliseconds. This dwell time was so short that it would be unlikely for natural lightning to transfer enough charge to the attachment point to puncture 0.08-inch aluminum in a swept-stroke zone. Boeing service damage records do not contain a case of puncture of 0.08-inch aluminum skins. If the dwell times measured in these tests are representative of natural lightning, a skin thinner than 0.08 inch aluminum could be considered for swept-stroke zones, especially if the aluminum has no insulating coatings on the outer surface.

For the swept-stroke experiments conducted, titanium had a very short dwell time and in some cases the Fastax camera data indicated that the attachment point moved continuously along the skin surface implying dwell times of less than 1 millisecond; therefore, the energies to the individual attachment points would be considerably less than for anodized aluminum.

In concluding, some general observations and comments will be made:

1. Some of the tests indicated that the lightning channel could consist of several separate channels which would tend to spread out the energy over a larger area of the skin surface.

2. Additional swept-stroke testing must be accomplished to determine the effect of skin joints, surface coatings, and other electrical discontinuities on the dwell times.

3. The hot-spot and puncture tests results are conservative in that the channel was not allowed to move. In addition, the hot-spot threshold time did not include the ignition delay time that would occur in actual ignition tests. Delay times may be of the order of milliseconds to tens of milliseconds for lightning energies.

4. Tests should be conducted to simulate very high energy transfer in the millisecond range, to verify the extrapolated portion of the curves.

5. Skin puncture data for swept-stroke areas should be obtained for other types of aircraft.

6. An upper limit of coulomb values should be established for short duration natural lightning in swept-stroke zones.

ACKNOWLEDGEMENTS

The author is indebted to a number of people at The Boeing Company and at Lightning and Transients Research Institute (LTRI) for their efforts in the study program.

In particular, I wish to acknowledge the swept-stroke test work accomplished by J. D. Robb and J. R. Stahmann of LTRI and the laboratory and analytical work of Dr. M. J. Kofoid of the Boeing Scientific Research Laboratory.

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# SURGE TANK PROTECTION SYSTEM

Frederick W. Newman

Fenwal Incorporated

The Fenwal Surge Tank Protection System (STP) consists of a broad band radiation-type sensor located near the aircraft vent outlet, suppressors containing the inerting agent (the number depends upon the size of the area to be protected) and an indicator to tell when the system has been actuated. The fourth item, the system test unit to activate the test circuit within the detector, is usually provided by the user. The system as installed on the Boeing 707 is shown in Figure 1. The detector is located in the vent tube and two suppressors are mounted on the outboard rib of the surge tank. The indicator is mounted on the bottom surface of the wing tip and can be viewed from the ground on walk-around inspections to determine whether or not the system is actuated. In this installation, the detector surveys the vent tube through an opening in the wall to sense flame traveling toward the surge tank.

The 727 installation is shown in Figure 2, with the detector mounted on the vent tube and the indicator mounted on the underside of the wing surface. Again, two suppressors are used to protect the surge tank. For this installation, it was necessary to mount the reservoirs vertically from the upper stringers.

In both the 707 and 727 installations, the detector, reservoirs, and indicator are identical. Slight modifications were necessary in the initiator that actuates the suppressor unit for the 727 because of the in-tank mounting of the reservoirs; it was necessary to equip the initiators with long pigtails which were brought through fittings to the forward side of the surge tank.

Very briefly, the performance required of the broad band detector is to sense any flame traveling toward the surge tank and initiate the signal to the initiators within the suppressor units within 100 microseconds. These suppressor units must then rupture to discharge the inerting agent into the area to be protected in not more than four milliseconds; actual performance is closer to two milliseconds. The suppressor, a grooved tubular stainless steel container,

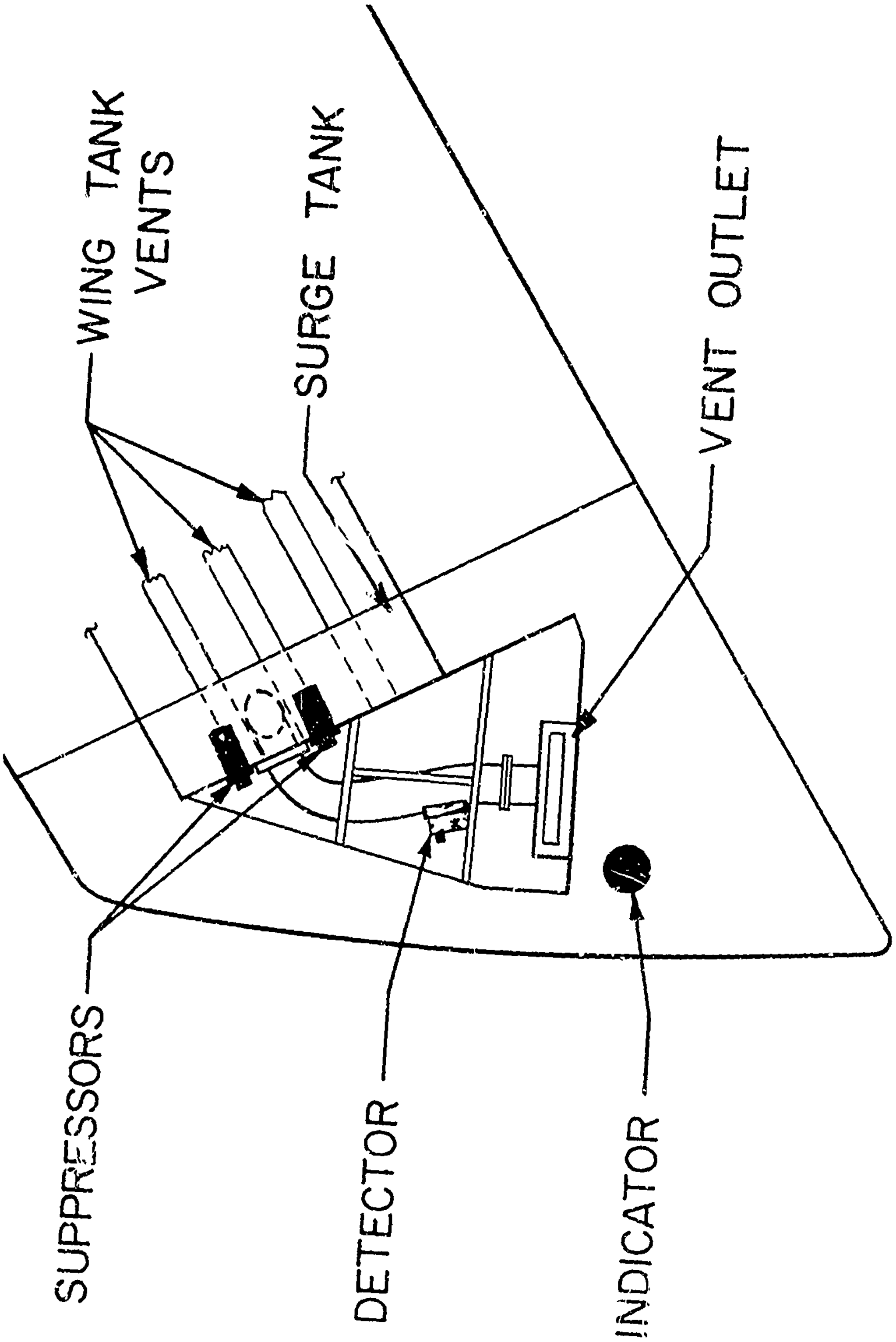


Figure 1. STP System for 707 Aircraft

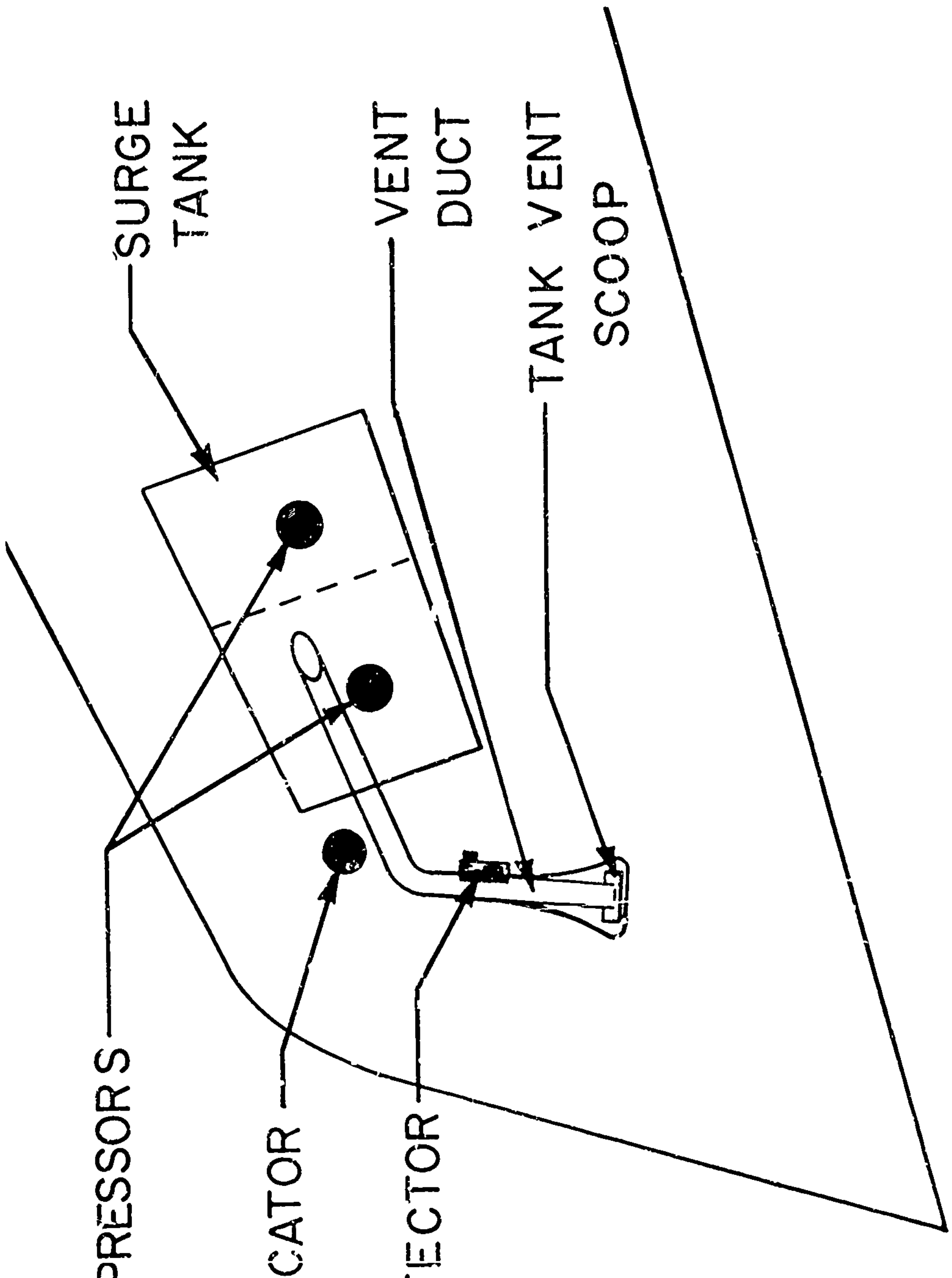


Figure 2. STP System for 727 Aircraft

has a capacity of 250 cc of Freon 2402. It also contains an initiator (in a well so that it is not in contact with the agent) which transmits a force to the grooved shell and causes a controlled discharge of the Freon 2402 inerting agent. Because of the grooves, a peeling action occurs upon initiation, and allows the suppressant to be expelled at speeds greater than 240 fps.

While the 747 installation, shown in Figure 3, employs exactly the same principles for detection and inerting as the 707 and 727 installations, certain modifications were required because of the extreme size of the surge tank on this aircraft. The most important change to the detector was to increase its capacity so as to operate four suppressors in series, rather than two as in the 707 installation. As you can see in Figure 3, three suppressor units are located within the surge tank, while a fourth is mounted essentially at the inboard end of the vent pipe. The length of the vent pipe is this installation makes it possible to arrest any flame within the pipe. The three suppressors within the surge tank are intended to inert this area to provide the required long-term inerting capability.

The suppressors were also modified, the grooves in the side of the tube eliminated, and a petaling disc added to the end of the suppressor. A screen has also been added over the end of the petaling disc which serves a two-fold purpose. It serves as a spreader to control the distribution pattern of the agent upon discharge, and also acts as a trap for the small metallic particles which are generated upon actuation of the initiator. The units have a flange for mounting and are accessible for mounting without requiring entrance to the inside of the surge tank.

In December 1967, the Fenwal flame suppression system installed on TWA 707 and 727 aircraft had a failure rate of 0.8 per 10,000 hours, or an MTBF of 12,500 hours, as reported at the conference of Fire Safety Measures for Aircraft Fuel Systems. As of May 1968, that figure had decreased to 0.43 per 10,000, or an MTBF of 23,000 hours. As of July 23, 1968, the system was installed on 98 TWA aircraft (86 707's and 12 727's), and the MTBF has increased to more than 26,000 hours. The figures quoted are for the detector only. The MTBF for the initiator is in excess of 800,000 hours and for the indicator in excess of 400,000 hours.

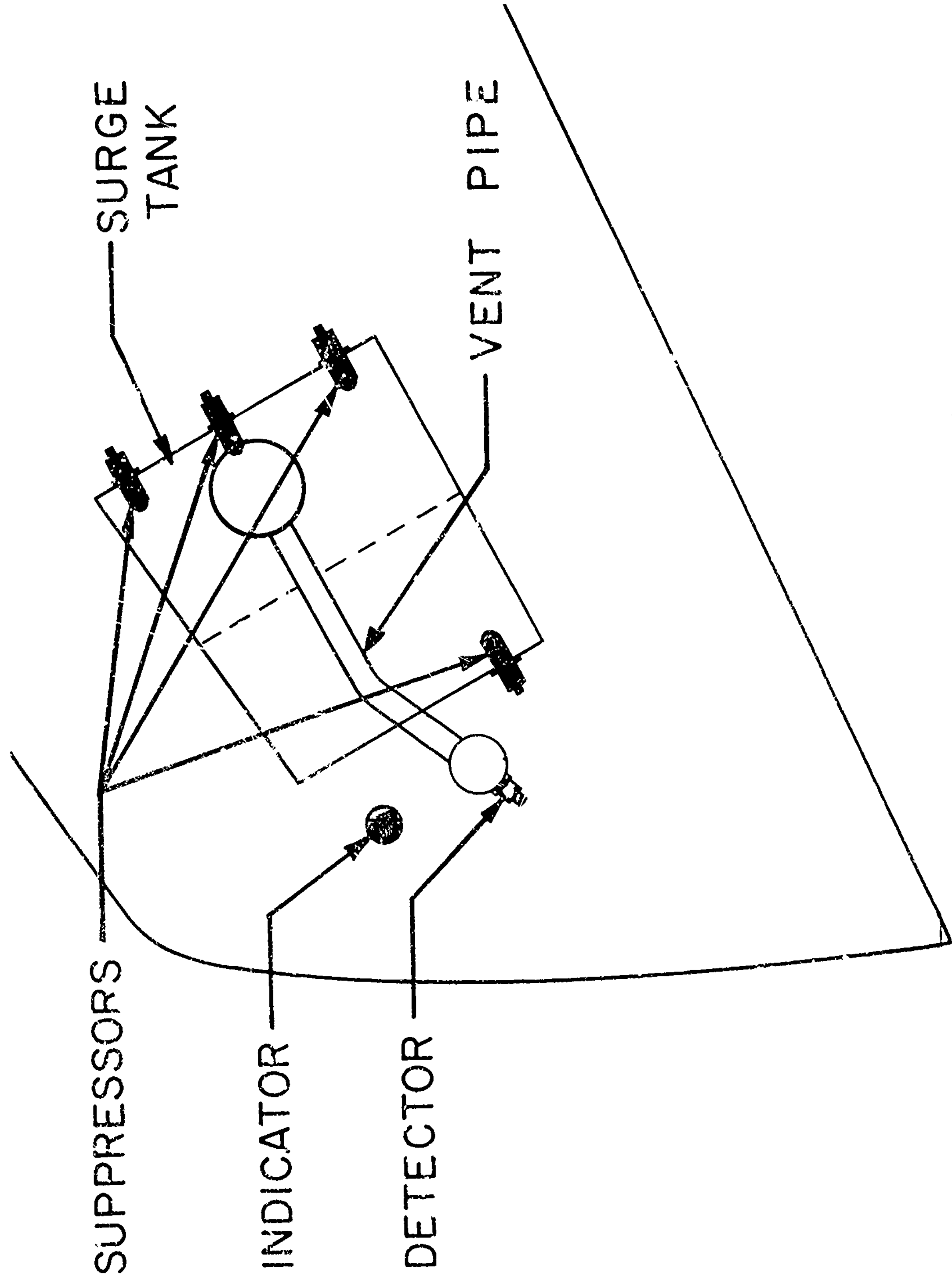


Figure 3. STP System for 747 Aircraft

The figures quoted to you today indicate a steady improvement in MTBF since the first installation in March 1967. This has not been achieved without intensive effort in the analysis of performance data, problems reported, analyzed, and solutions found.

The very first problem encountered was a susceptibility of the system to false fire if the aircraft was struck by lightning. An intensive investigation was conducted to determine the magnitude of this susceptibility and an evaluation of the fix. The 707 detector, manufactured with a stainless steel hermetically sealed case, was not sufficiently shielded against lightning strikes. It was determined that an aluminum overbox would provide adequate protection for lightning strikes up to 200 kiloamperes.

The second problem was also a random occurrence of false system actuation in some of the TWA fleet. This was finally traced to a very high frequency transient generated by the retractable landing light. This light is supplied with 28 VDC for the motor drive and 400 hz 100 VAC with an autotransformer to supply 400 hz 28 VAC to the lamp filament. Transients were measured with amplitudes of 1600 volt peak to peak, at a frequency of 10 mhz.

Corrective action was the addition of two filter capacitors and an internal circuit to case ground. All detectors have been modified at no cost to the user. Aircraft fitted with modified detectors have not had a false system actuation.

Fenwal also made a complete performance analysis of the detector. Based upon the results of this investigation, additional in-house testing was added to our production line and a component change was made in a limited number of early production units during the modification program.

Anticipating new requirements for additional protection on newer aircraft, Fenwal has continuing investigations in the areas of flame sensing and inerting and agent containment and discharge. Much of the work in this area is empirical in nature with most of the effort being company funded. We do feel the effort is necessary if Fenwal is to maintain its position in developing techniques and equipment for protection against fire and explosion hazards in military and commercial aircraft.

# NITROGEN INERTING OF AIRCRAFT FUEL TANKS

K. R. Bragg

C. C. Kimmel

P. H. Jones

Parker Hannifin Corporation

Aircraft fuel tank vapor and vent spaces are normally filled with a mixture of air and fuel vapor. Under many conditions of flight, the air/fuel vapor combination yields a combustible mixture which will explode if ignited. In the past, the isolation of ignition sources was considered adequate protection — the logic being that having an explosive mixture in the tanks was tolerable so long as it was not ignited. The record of incidents resulting in fuel tank explosion, however, dictates that more positive steps for protection must be taken. Fuel tank inerting is such a step.

The concept of inerting is to render the air/fuel vapor mixture in the vent and vapor spaces, or "ullage," nonreactive by replacing the oxygen in the mixture with the inert gas nitrogen. By creating an atmosphere void of oxygen, combustion and thus explosion simply cannot occur regardless of the presence of ignition sources.

The oxygen in the vent spaces and in the vapor space, or ullage, comes from two sources. First, oxygen is present in the air which enters the fuel tanks to fill the spaces vacated as the fuel is consumed, or to adjust for increasing pressure as the aircraft descends. Second, stored fuel absorbs air in large quantities and then releases it as the pressure decreases during ascent. An analogy is a carbonated beverage charged with carbon dioxide which is held under pressure by the cap; when the cap is removed, the pressure is released and the carbon dioxide comes out of solution and bubbles to the surface.

In several past attempts at fuel tank inerting, the inert gas was merely introduced as a blanket over the fuel; no means was provided for handling the dissolved air in the fuel. As a result, the dissolved air, when released, could contaminate the presumably inerted vapor space with an unpredictable oxygen concentration. The Parker inerting system however embraces a positive method of conditioning the fuel and the vapor space to provide a predictable oxygen concentration in the vapor that is well below the ignition or explosive level.

THE CASE FOR INERTING

In analyzing the hazard presented by oxygen in the fuel tanks, it is important to consider the potential sources of ignition and when they may occur. The following are some of the potential sources of ignition with a brief analysis of each.

1. LIGHTNING STRIKES

Aircraft are subject to being struck by lightning. Strikes can occur in the range from ground level to 40,000 feet, with the most probable strike zone being ground level to 15,000 feet. The tragic loss of a 707 near Elkton, Maryland on 8 December 1963 brought this hazard into focus and resulted in a considerable increase in the pace of the study and testing aimed at understanding and establishing the hazards of lightning to aircraft.

Briefly, it has been demonstrated that a lightning strike can ignite a fuel-air mixture either by direct ignition (such as a strike hitting at an open vent or burning a hole through the tank wall) or indirectly by inducing arcing in the fuel tanks as a result of charging the aircraft to a high potential. In the case of aircraft constructed of high-melting-temperature alloys such as titanium, a lightning strike could cause a local "hot spot" which will ignite the fuel vapors by a "glow-plug" action. If the aircraft is not inerted, other protection must be provided, such as insulation or separation of the fuel tanks from the skin. It is believed that flying through strong electrical fields of a lightning storm can also cause arcing across a vent opening or within the fuel tanks without the aircraft being struck. This is the probable cause attributed to the loss of a Constellation over Milan, Italy on 26 June 1959.

2. TAKEOFF, FLIGHT, AND LANDING MISHAPS

Mishaps such as these which result in fire can subsequently cause ignition and explosion of the fuel tanks. An example of this hazard was the loss of a 707 as a result of a takeoff mishap in Rome on 23 November 1964. In that incident, a crash-initiated fire apparently propagated down the vent lines, igniting and exploding the fuel tanks. Similarly, the explosion of the fuel tanks in the BOAC 707 at London airport on 8 April 1968 greatly magnified the difficulty of passenger escape. As was demonstrated by these incidents, a fire external to the fuselage and an explosion of the fuel tanks can be the difference between life and death. Penetration of fuel tankage by metal fragments from a disintegrating engine poses another potential source of ignition of tank vapors.

### 3. MILITARY BATTLE DAMAGE

In military applications, the problem of fuel tank explosion after severe battle damage or from small arms ground fire has a serious impact on the current aircraft inventory in Southeast Asia. Losses could be greatly reduced if properly inerted fuel tanks were employed. For example, in the event of a damaging strike by heavy ammunition, an inerting system can prevent an explosion in the tanks and buy time for the pilot to eject before the aircraft disintegrates, to get to an area where he can expect to be rescued, or to assess the damage and try to save the aircraft.

Small arms ground fire is less hazardous to aircraft if the explosion of tanks cannot occur. Inerting can save the pilot and possibly the aircraft in take-off and landing accidents where fuel tank explosion would have caused complete destruction. It can minimize the battle damage to aircraft while parked, and can prevent damage to adjacent aircraft which may occur if a noninerted fuel tank explodes.

## DEVELOPMENT BACKGROUND

In 1956 Parker was selected to develop a positive means of inerting the fuel tanks of the XB-70 air vehicle. In this case, the flight profile included speeds which raised the temperature of the fuel tank walls well above the ignition point of the fuel vapor in the tanks; therefore, it was mandatory that the oxygen concentration remain below the explosive level.

An exhaustive analysis of the conditons contributing to the problem of fuel tank protection resulted in the highly successful fuel tank inerting system in the XB-70.

This analysis revealed the importance of handling not only the initial oxygen present in the tank vapor space but also the oxygen in the air dissolved in the fuel. The XB-70 system pressurizes the tanks with nitrogen and prevents the admission of free air. Nitrogen gas is utilized on the ground during refueling to displace the dissolved oxygen in the boarding fuel, thereby achieving a predictable safe level of oxygen concentration in the tanks.

The process of treating the fuel on the ground during refueling, while acceptable in the highly experimental climate of the XB-70 development program, is not attractive from a cost and logistics standpoint for production military aircraft or commercial airlines. Accordingly, Parker continued the development of the total inerting concept, and in 1967 solved the problem of treating the onboard fuel to reduce the oxygen concentration to a predictable and safe level. Prototype components were fabricated and a small-scale system was assembled in the laboratory. A 500-gallon tank was used in a system capable of simulating the climb, cruise, and descent profiles of various aircraft. This system provided a direct means of correlating theory with practice and a method of achieving predictable oxygen concentrations was established.

Subsequently, flightworthy hardware was built and tested in full-scale systems at the Parker Full-Scale Fuel Test Facility. This facility provides a means of observing and demonstrating the complete function and performance of the Parker inerting system for any application for which it is considered.

## TECHNICAL ANALYSIS

In arriving at a solution to the problem of oxygen control in fuel, it was necessary to analyze the combustion process associated with hydrocarbon fuels. In fuel tanks of aircraft employing such fuel, there is always the hazard of a fire, which can involve either a slow or a rapid oxidation process. Two factors must be considered in the control of this oxidation process:

(1) Oxidation by ignition sources.

(2) Oxidation by autoignition sources.

For subsonic aircraft, only oxidation by ignition sources need be considered for fuel tank internal protection. For supersonic aircraft, both ignition and autoignition sources may have to be evaluated, depending upon flight speeds and mission profiles. In order for oxidation to occur with hydrocarbons, it is necessary to have three basic elements in contact with each other, as shown in Figure 1. The oxidation process is further complicated by the necessity of having the elements arranged so that each meets certain boundary restraint limits (Figure 2). Only within these limits can the oxidation process, as we know it, take place.

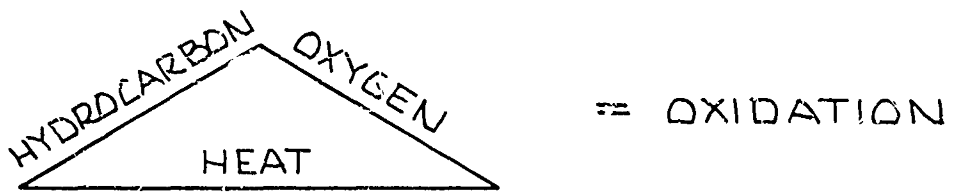


Figure 1. Oxidation Process

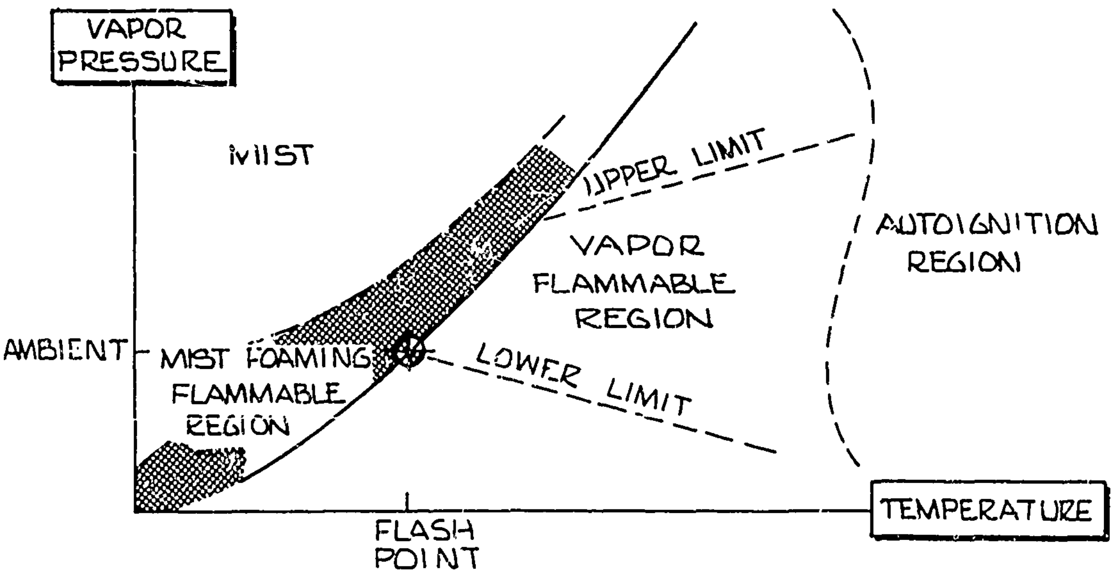


Figure 2. Flammability Characteristics of a Typical Hydrocarbon Fuel-Air Mixture

In addition to the basic boundary limits shown in Figure 2 for pure hydrocarbons, the system becomes even further complicated by the synthesis of fuel blends to meet the performance and cost goals of the airplane and engine manufacturer. This results in a varying vapor pressure and hydrocarbon molecular content, depending upon the fuel purity, distillation, and weathering. These factors can be determined generally within a narrow range for a specific fuel type, blend, and manufacturer.

The flash point is considered to be a relative indication of safeness, a probable carry-over from early mining days. In order to prevent the pioneers from killing themselves when using kerosene lanterns, a general consideration of making the fuel safe at a temperature of about 110°F was set up. Thus, if a lantern tipped over and spilled kerosene, the vapors would not ignite at this temperature. The validity of using the flash point as a method of evaluating fuel performance could, therefore, be considered.

As shown on the flammability chart (Figure 2) there are three regions where oxidation can exist:

(1) Vapor phase above the flash point within limits.

(2) Vapor phase at elevated temperatures beyond the flammability limits.

(3) Vapor/liquid region below the flash point when the mixture is foaming.

In these regions, oxidation can be initiated by fire, electric spark, or electrostatic discharge. With the basic aircraft fuels, the limits of noticeable oxidation are a function of the concentration of gases in the vapor phase mixture and the total pressure and temperature of the system. For example, with a fuel having a low Reid vapor pressure (high flash point, above 110°F) such as Jet "A," a typical ignition limit for equilibrium conditions is as illustrated in Figure 3.

With the addition of an inerting gas, the limits of flammability can be altered so that the rapid oxidation process (i.e., burning) cannot occur (Figure 4). For spark ignition, adequate vapor formation at ambient for combustion occurs at about 0.9 percent fuel, and corresponds to the flash point. Below the 10 percent oxygen concentration level, the system is considered safe from an oxidation process that could result in sustained burning or explosion.

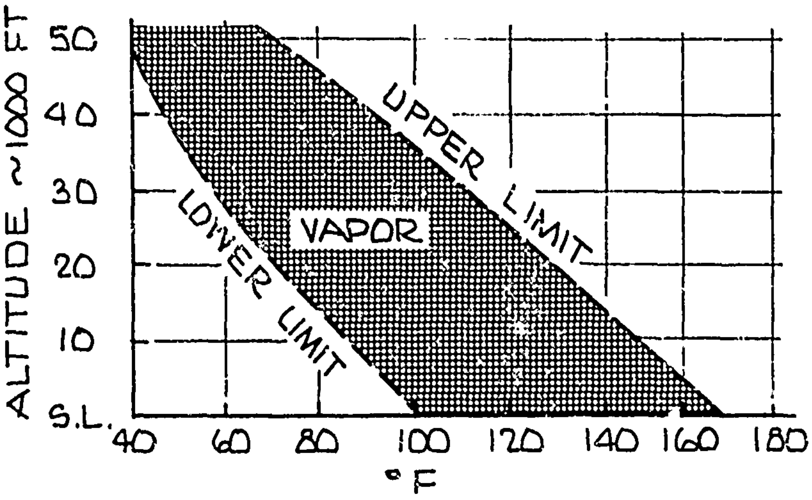


Figure 3. Typical Altitude/Temperature Flammability Limits of Jet "A"

For the foam region, it is possible to have the oxidation occur at conditions otherwise considered to be nonflammable (as defined by Figure 4). This actually extends the ignition limit curve for the oxygen concentration region above 10 percent (Figure 5).

Hydrocarbon fuels in contact with the atmosphere dissolve air. The amount of air dissolved results in the same relative ratio of oxygen to nitrogen as found in the atmosphere, except that there is a greater affinity for oxygen. This poses a severe problem in the determination of the vapor concentration of oxygen, fuel, and nitrogen. In an aircraft, two conditions which are important in considering this problem are refueling, and climb-to-altitude.

PROCESS OF AIR RELEASE

Oxygen and nitrogen gas entrained (dissolved) in the fuel tend to emerge when the pressure on the system is reduced. This is a slow process when performed under near quiescent conditions. The rate of release is suppressed by the surface tension of the bubble. These bubbles tend to oppose the surface tension of the liquid when they are very small. For release to occur, a cavity of a minimum size must be created in the fuel by mechanical means.

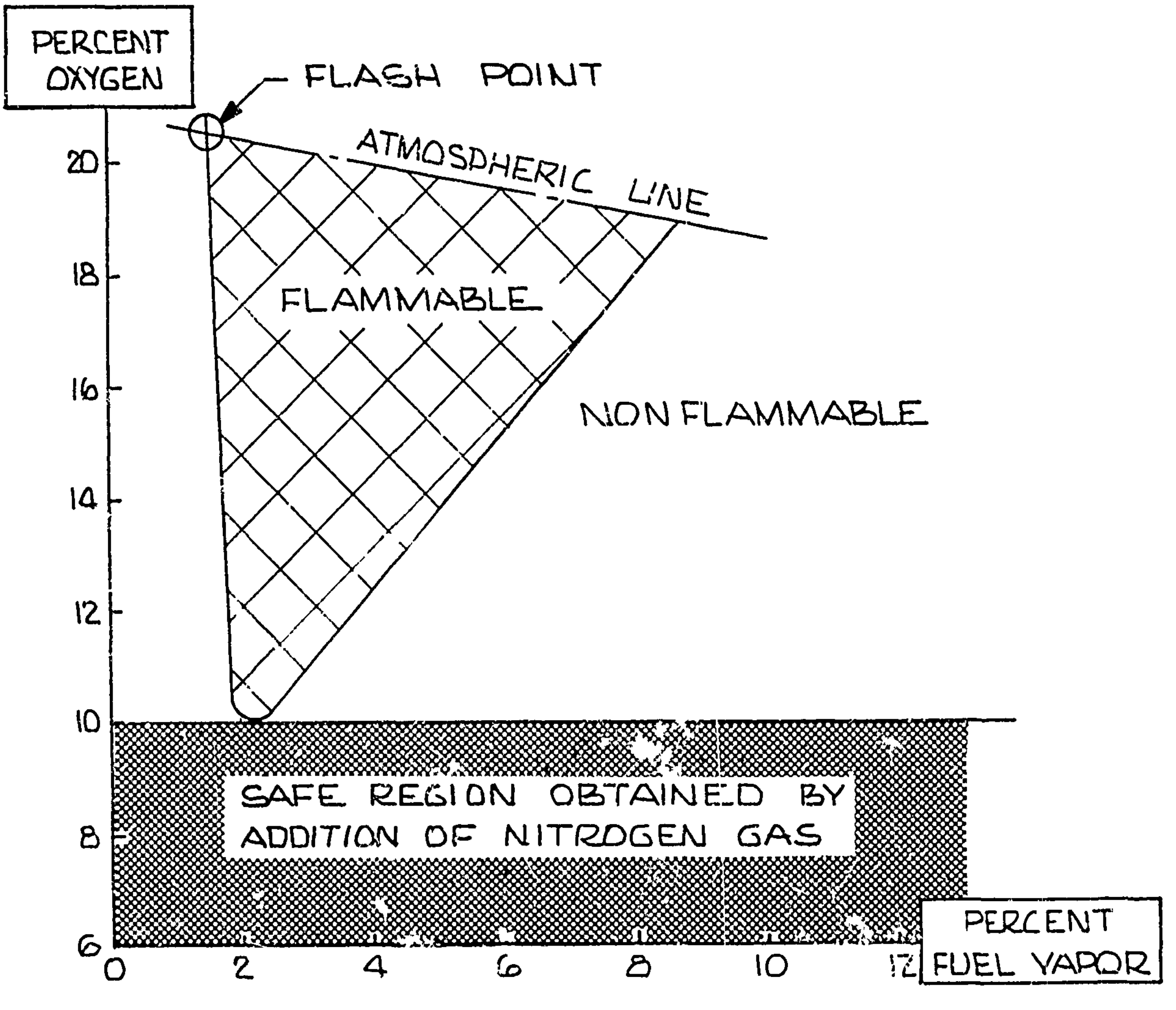


Figure 4. Flammability Limits of Typical Jet "A" Fuel-Air Mixture With Gaseous Nitrogen Added

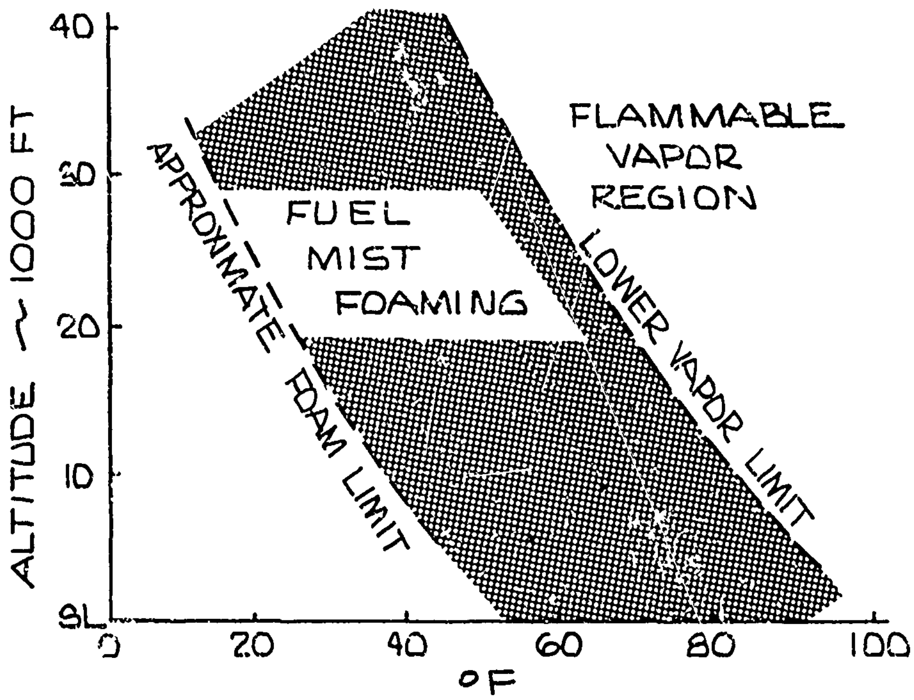


Figure 5. Flammability Region of Fuel Mist Foam for Typical Jet "A" Fuel

A refueling operation employing a standpipe that sprays fuel into the tank, induces some release of the dissolved gases. This action can be quite violent in an aircraft fuel tank. These aircraft fuels are generally distillate compounds containing nonpolar hydrocarbons, and as such are good electrical insulators. However, some organic and related compounds in small concentrations can also be present, which are capable of ionizing and thus altering the electrical behavior of the fuel. In this instance, it is possible to set up a relative motion difference between the fuel and the fuel interfaces that results in polarity shifts that can set up an electric charge. This can occur by bubbles rising to the top and breaking the surface. Normally this dissipation rate is very high; however, concentration can occur and an electrostatic charge can be released as noted in the Table I. A certain minimum charge is required in order to ignite the vapor/foam mixture. With the foam mixture, individual foam particles can contain higher oxygen concentrations than the free vapor space due to the uneven release of oxygen and nitrogen from the fuel. This is what has caused the ignition limits to shift so drastically.

TABLE I

CHARACTERISTICS OF ELECTROSTATIC CHARGE

| Condition | Description |
|---|---|
| Sparks from tank to other | Electric discharges resulting from liquid or gaseous friction. |
| Sparks in vapor | Electrical charges produced by formation of liquid particles or friction of gas to solid particles. |
| Sparks from fuel to objects | Electrical charges on solid particles in fuel produced by friction with walls or filters. |
| Fuel autoignition | All fuel/oxygen mixtures are spontaneously reactive above 400°F. |
| Component overheating | Case temperature above 450°F can cause ignition. |
| Fuel mixtures | Mixing of various fuel grades can extend the flammability region beyond that of either fuel type used by itself. |

During the aircraft climb to altitude, the effect of pumps and other agitating devices can cause the dissolved gases to be released at a rate inducing high foaming. During this process and during refueling, the release of the dissolved gases causes the vent spaces to become richer in oxygen. The oxygen concentration of a typical fuel saturated with air at sea level can exceed 40 percent at altitude, as shown in Figure 6.

To protect against fuel vapor ignition in subsonic aircraft, it is desirable to do the following:

(1) Remove the dissolved oxygen in the fuel to a safe level.

(2) Maintain the vent space oxygen concentration below 10 percent.

In addition to the refueling hazard, a possibility exists of lightning strikes causing vapor ignition. The region of highest lightning strike potential is shown in Figure 7.

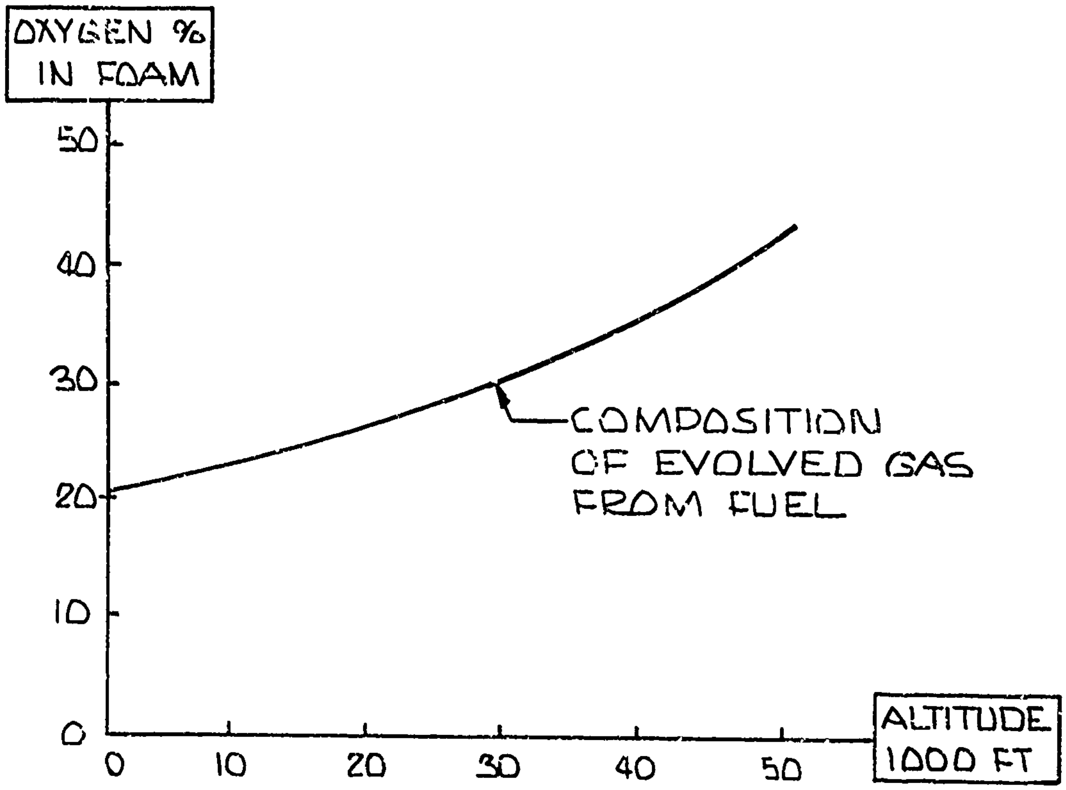


Figure 6. Oxygen Concentration in Foam Mist Evolved From Fuel During Climb to Altitude

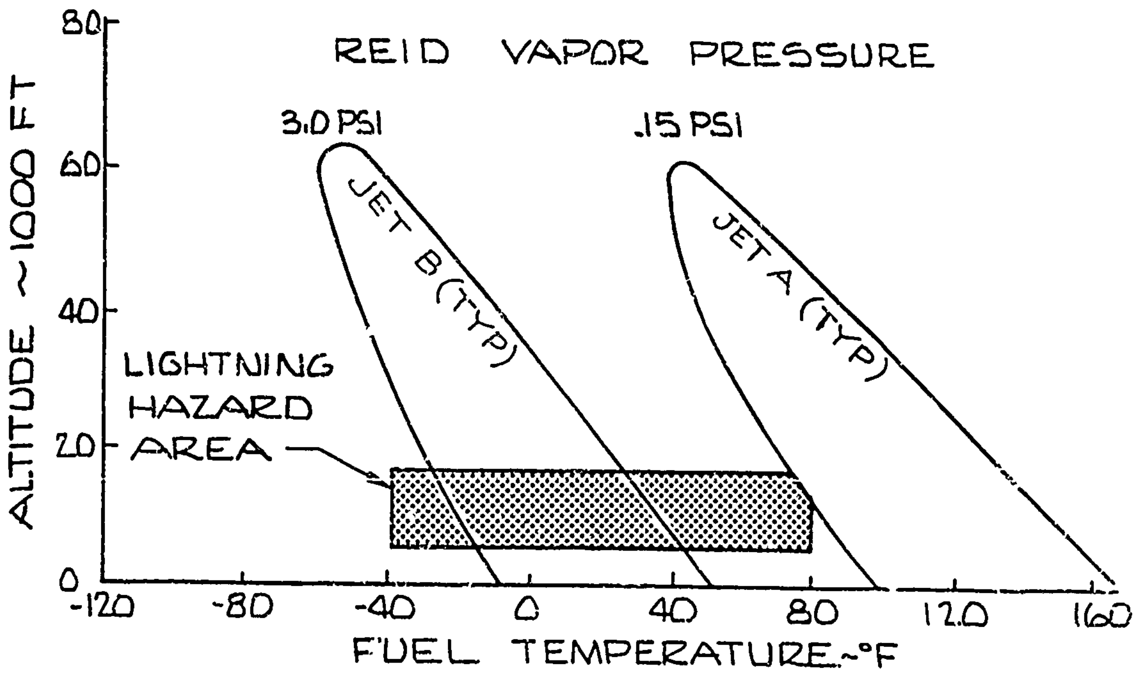


Figure 7. Altitude/Temperature Flammability Limit of Typical Jet Fuels

Protection against fuel-vapor ignition in supersonic aircraft requires the oxygen concentration to be less than 10 percent. The actual limit is dependent upon the initial conditions of pressure, temperature, and fuel-vapor mixture ratios and the allowable structural limits. In order to prevent any autoignition reaction, all the oxygen must be removed. This restriction is considered unnecessary, however, and economically unfeasible. Low order reactions involving slight pressure rises and slow oxidation would not create a situation detrimental to the aircraft structure. A typical autoignition region is shown in Figure 8. For a specific fuel distillate blend and known concentration of temperature, pressure, and oxygen, the pressures resulting from ignition can be evaluated.

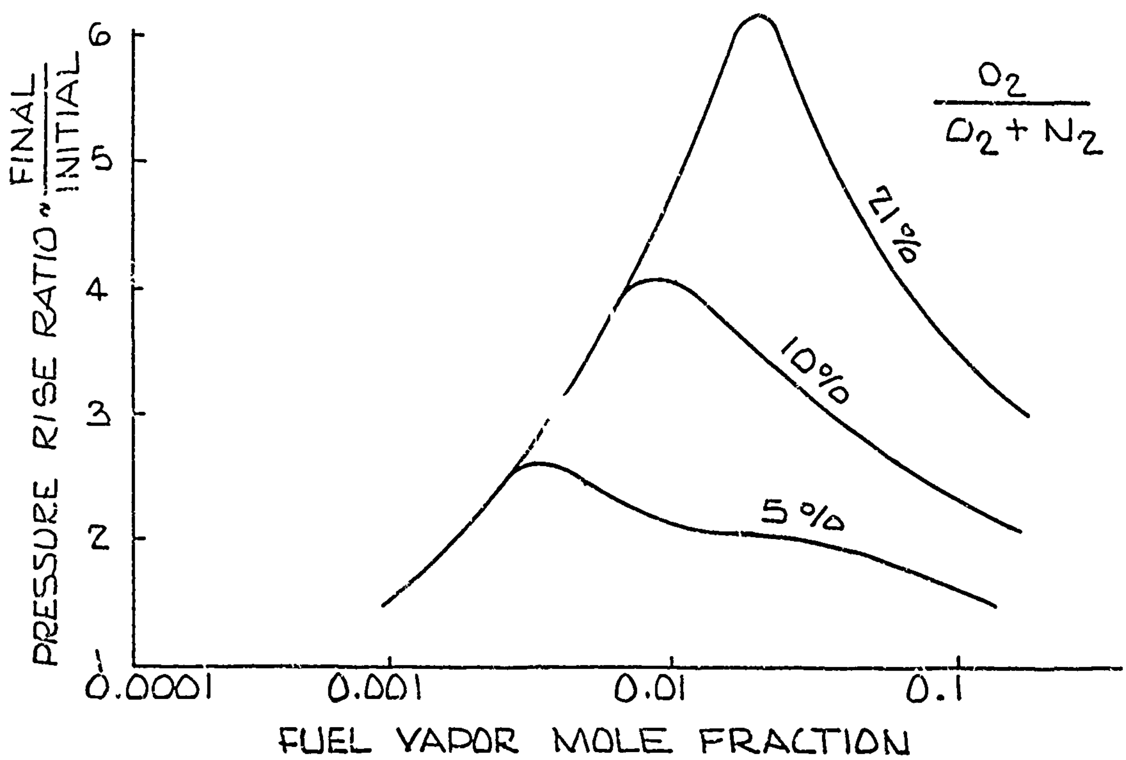


Figure 8. Reaction Pressure Rise Ratio for Typical Hydrocarbon Base Fuel Autoignition

INERTING SYSTEM DESCRIPTION

Figure 9 shows the general arrangement of elements of the system in the aircraft. The system consists of the following components:

(1) Liquid Nitrogen Dewar -- The liquid nitrogen dewar is an aluminum vacuum insulated dewar with a fill port, a vent port, a saturation and discharge port, and passive provisions for quantity gaging by means of a differential pressure gage.

(2) Control Package — The control package consists of a scrub solenoid valve, a pressurization solenoid valve, and a mechanical pressurization regulator which are packaged together for convenience of installation. These three devices control the flow of nitrogen to the fuel tanks.

(3) Scrub Sequencer — The scrub sequencer is a package consisting of a solenoid valve, a differential pressure switch, and a reference volume. Its function is to regulate the scrubbing process.

(4) Distribution Manifold — The distribution manifold is an arrangement of aluminum tubing ranging from 1-1/2-inch to 1/2-inch in diameter, connected by conventional fittings. It supplies nitrogen from the control package to each fuel tank.

(5) Scrub Manifold — A scrub manifold is installed in each fuel tank. Each provides a means of mixing nitrogen from the distribution manifold with fuel tapped from the boost pumps in the tank or, in the case of the reserve tanks, in the scrubbing pump. The mixture is discharged into each compartment of the fuel tank through orifices to control the flow rates of nitrogen and fuel. A spring-loaded check valve and the tubing routing prevent the backflow of fuel into the nitrogen distribution manifold.

(6) Fog Nozzle and Relief Valve — A fog nozzle and relief valve is installed in each of the four main tanks and the two centerwing tanks in parallel with the scrub manifold. The fog nozzles break up liquid nitrogen into a fog. Thus, the liquid is suspended in the vapor space until it vaporizes and cannot collect as a puddle and create thermal shock problems. This eliminates the need for a heat exchanger to vaporize the liquid nitrogen.

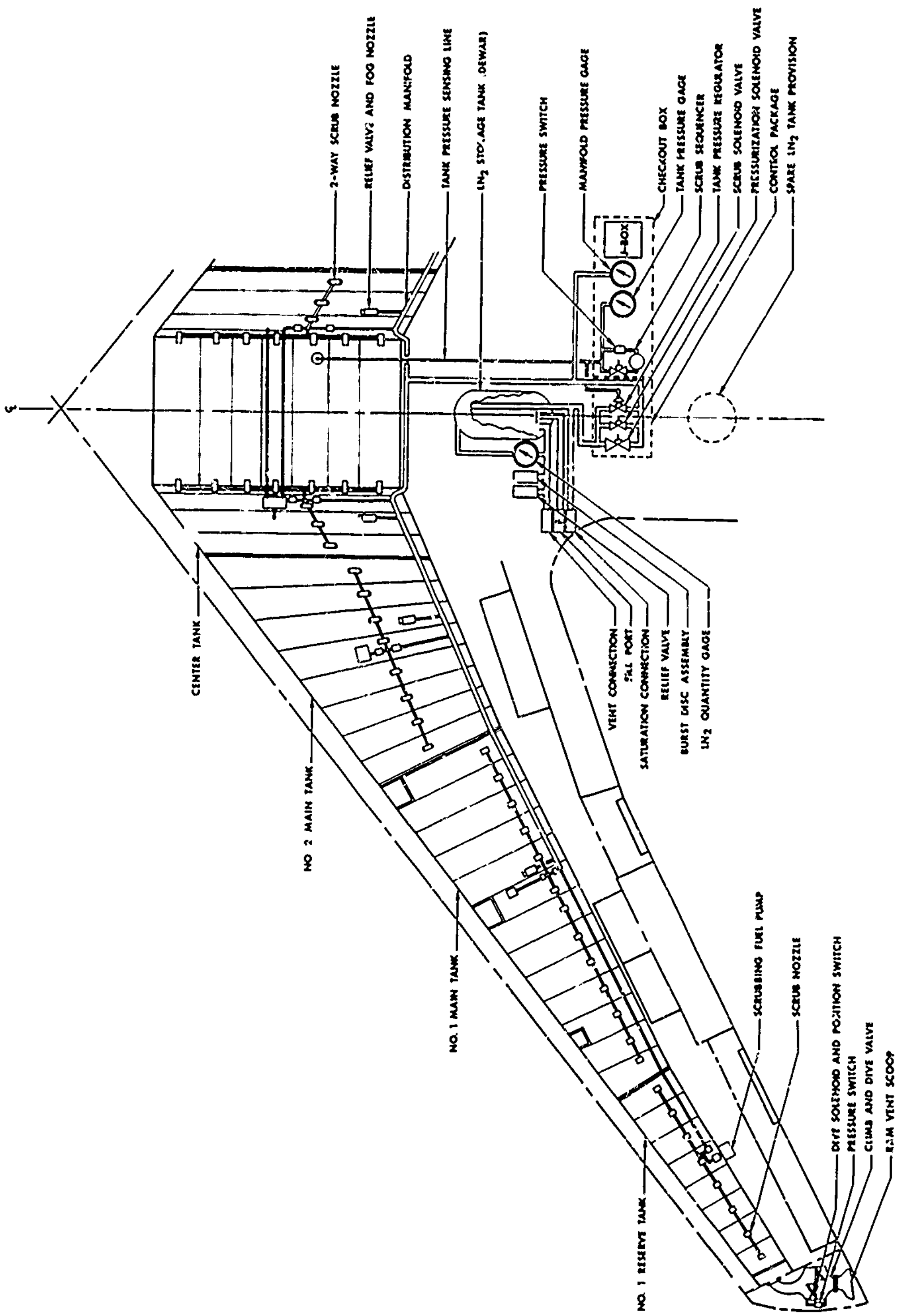


Figure 9. Typical Transport Installation

(7) Vent Valves — The vent valves are installed in each wing tip at the vent outlet. They consist of swing flapper dive and climb valves and a differential pressure switch. The flappers are spring-loaded to operate as relief valves. In their design, extreme care has been taken to ensure that the valves will not stick or ice shut during operation.

(8) Fuel Bleed Block Valve — A fuel bleed block valve is installed in each scrubbing fuel bleed line. In each tank presently containing boost or boost override pumps, the fuel for scrubbing will be bled from the pair of pumps located in the tank. Should one of a pair of pumps fail, the block valve would automatically shut off the bleed of scrubbing fuel in that tank. The flow of scrubbing fuel would continue normally in all unaffected tanks.

## INERTING SYSTEM OPERATION

The basic system operation is concerned with the delivery of nitrogen gas into the fuel tanks to perform two functions: (a) dilute the ullage oxygen concentration to a level that prevents ignition possibility and (b) prevent admission of air into the fuel tanks. Detail operations associated with these functions are discussed as follows.

### 1. NITROGEN SERVICING

Nitrogen for inerting the fuel tanks is stored aboard the aircraft in a dewar assembly. Liquid nitrogen from a ground cart is saturated to about 100 psia in the dewar during servicing. The ground cart has three connections, fill, vent, and saturation. Liquid nitrogen and gaseous nitrogen are introduced into the dewar through a specially designed fill manifold. This manifold simultaneously mixes the gaseous nitrogen with liquid nitrogen so as to permit saturation. The ground cart automatically controls this pressurization and servicing function. No flight functional equipment is used during the servicing.

Nitrogen is loaded in this saturated condition so as to eliminate any requirement for pressure building equipment. Now the nitrogen can be discharged from the dewar at any time upon demand. The final temperature of the nitrogen after loading is about -280°F.

2. FUEL SERVICING

Assuming that the fuel tanks were inerted from the previous flight, the initial oxygen concentration should be less than 1 percent prior to refueling.

Normal design practices recommended that the refueling level control valve outlet be as low as possible to minimize static electrical discharges. With an inerted tank this recommendation is not mandatory. It has been shown by test that some of the dissolved oxygen in the fuel will come out into the vent space during refueling. The amount released is dependent primarily upon the exit location of the refueling line and the agitation imparted to the fuel as it exits. For outlets submerged before the tank is one-fifth full, the resulting oxygen concentration will generally be under 2 percent. Locating the exit to a place where the tank can be three-fourths full before being submerged can result in an oxygen concentration of about 7 percent after servicing. These values have been verified by full-scale test of a 2200-gallon fuel tank. Thus, the aircraft is inherently safe during refueling with no function required from the inerting system other than maintaining an inert vent space atmosphere at the start of fueling.

3. AIRCRAFT STANDBY

During standby, a low pressure regulator bleeds sufficient nitrogen into the tanks to maintain them at a pressure level of 1/8 psi above ambient pressure. Pressure levels over this nominal value will stop nitrogen flow. In addition to this normal function, the low pressure regulator will provide nitrogen flow into the tanks should a rupture or fuel leakage occur.

4. DURING CLIMB-TO-ALTITUDE

The flight period during which the aircraft climbs to cruise altitude is the most significant time for inerting protection. During this short time span, the release of oxygen dissolved in the fuel must be controlled. Fuel that is serviced into the aircraft can be saturated with air. The amount of oxygen gas dissolved in the fuel is about 32 percent by volume, compared with 21 percent by volume in the free vent space above the fuel surface in the ground storage tanks. As explained above, it is possible to load fuel onboard with a minimal loss of this dissolved oxygen. The oxygen and nitrogen dissolved in the fuel tend to remain in solution in a supersaturated state as the pressure in the vent space is reduced. These gases can be induced to come out by applying violent agitation or vibration or by reducing the relative partial pressures of these dissolved gases by the Parker "scrubbing" process. With either agitation or vibration, no easy way exists to control the rate of gas release or to limit the total oxygen content. The Parker "scrubbing" process prevents supersaturation by injection billions of small nitrogen bubbles into the fuel. These bubbles diffuse throughout the fuel and provide a large gas-to-liquid interface. The dissolved gases sense a local pressure unbalance and flow into the nitrogen bubbles in an attempt to equalize their partial pressures. The bubble rise rate is so slow that there is sufficient time for the diffusion to take place. The oxygen is diluted with nitrogen and fuel vapor as the bubble breaks the surface, so there is no tendency for stratification and therefore no need to flush the vent space with nitrogen.

## 5. AIRCRAFT CRUISE

After attaining cruise altitude, the reference volume will contain the initial cruise altitude pressure. Since the ullage pressure differential change due to the altitude climb has stopped, the scrub solenoid will close when the timer runs out (about 2 minutes). As fuel is consumed by the engines, the vent space pressure decreases. When this pressure decreases to about 1/4 psi above ram pressure, the pressurization solenoid will open and nitrogen will be discharged into the vent system to pressurize the tanks. The flow will stop when the tank pressure reaches about 0.4 psi above ram pressure.

## 6. AIRCRAFT DESCENT FROM ALTITUDE

As the aircraft descends from altitude, the increase in external ambient pressure is detected by the same switch used to maintain the tank pressure at 1/4 psi above ram pressure for cruise conditions. For normal descents from altitude, the rate of pressurization is faster than the aircraft rate of descent, causing the pressurization solenoid to cycle. The flow capacity of the inerting pressurization system is sized to match the fastest descent rate and ullage volume combination possible in the aircraft. For the duration of descent, the pressurization solenoid would remain open. Normal descent rates would require a pressurization flow rate of about 7 lb/min.

BENEFITS OF INERTING

The advantages of the Parker total inerting system are obvious from the standpoint of preventing destructive combustion, fire, or explosion in the fuel tanks, regardless of the ignition source. During fueling, takeoff, climb, cruise, descent, landing, and stand-by, with electric power on or off, and with the aircraft attended or unattended, protection is insured. When a crash ruptures the fuel tanks, active protection against propagation of fire to the tank interior is offered for several minutes.

The inerting system permits the use of low-vapor pressure fuel without the increased hazard of explosion associated with these fuels. Also, by preventing the admission of air into the tankage through the vent system, the condensation of water in the fuel tanks is totally eliminated. Tests have shown that the "scrubbing" process for removal of dissolved oxygen also removes all dissolved water. Unless gross quantities of free water are pumped aboard during fueling and then not drained, the inerting system will keep the fuel system completely free of water. This will totally eliminate the problems of fuel system icing and microbiological growth, thereby resulting in significant maintenance savings.

All in all, the Parker inerting system now available for military and commercial aircraft is considered a major step forward in flight safety.

AFAL-TR-68-290
PART II

# SPARK IGNITION

## (Ignition of Flammable Mixtures as a Consequence of Gaseous Electronic Discharge)

Elton L. Litchfield

Bureau of Mines

## INTRODUCTION

The prerequisites of a spark ignition are (1) a flammable mixture, and (2) the discharge of sufficient energy, under the appropriate conditions, to ignite the specific mixture. Within the flammable range of a composition, flame can result only if sufficient energy is injected into an adequately small gas volume and if certain other conditions are satisfied. Essentially, the "other conditions" are those that permit the injected energy to produce ignition rather than be removed from the developing flame kernel by the injection apparatus. The minimum sufficient energy to produce ignition depends upon the types of fuel and oxidizer, their relative proportions, the ambient pressure, and, probably the ambient temperature. Present knowledge of this minimum sufficient energy rests upon the minimum ignition energy concept and laboratory data establishing the values of the significant parameters.

The minimum ignition energy concept and the more important attributes of the laboratory data are summarized in the first portion of this paper. The second portion considers the conditions which must be imposed to ensure that the minimum ignition energy cannot be derived from a triboelectrified fluid. Since the latter portion is expository, liberties have been taken with geometrical form factors, transfers from resistivity to resistance, and similar items.

## THE MINIMUM IGNITION ENERGY CONCEPT

The minimum energy concept of spark ignition originated with Lewis and von Elbe (Reference 1). In essence, the ignition is considered to result from localized and instantaneous energy injection into a small volume of combustible gas mixture. The injected energy is utilized as heat to produce a high-temperature gas kernel. Thermal ignition occurs within this kernel and gives a flame which propagates throughout the combustible mixture. The Bureau of Mines has expended considerable effort in evaluating the parameters and circumstances contributing to ignition with minimum spark discharge energies. Data from diverse sources have been shown to correlate well when compared within the framework of the minimum energy concept (Reference 2).

A number of "rules of thumb" have been developed from the experimental data. For a given oxidizer, the energy required to ignite the most sensitive mixtures depends primarily upon the type of molecular bonding in the fuel molecule. For example, all saturated straight-chain hydrocarbons (alkanes) have essentially the same ignition energies for their most sensitive mixtures. (Ignition energies of the most sensitive mixture of an alkene with air are less than the corresponding energy for an alkane with air, etc.) Such an effect precludes any alleviation of the ultimate hazard from straight-chain hydrocarbon fuels by the blending of refinery fractions. The most common or frequent flammable mixture may be changed by such blending since it changes relative vapor pressures. Although this most frequent mixture may be appreciably less sensitive than the most easily ignitable mixture, the energy required to ignite the most sensitive mixture cannot be modified by such blending.

The concept of an "ignition quenching distance," which represents the smallest electrode separation at which the experimental apparatus does not extract significant heat from the developing flame kernel, suggests that there is a minimum potential difference for minimum energy ignition. At electrode separations less than the ignition quenching distance, the energy requirements for ignition are extremely sensitive to electrode geometry. The characteristic is demonstrated in Figure 1, which shows the effects of various electrode geometries and separations upon the ignition energy requirement. It may be seen that the glass flanged electrode (Reference 3), when used within the ignition quenching distance, is quite effective for increasing ignition energy requirements. Although little effort has been directed at the use of this effect as a safety measure, it is a simple and effective one where applicable. The concept of propagation-quenching distance is, of course, fundamental to the design of flame arrestors.

The effect of relative proportions of fuel and oxidizer upon the energy required for spark ignition is demonstrated in Figure 2. Ignition energy is figured as $CV^2/2$, where C is the energy storage capacitance in microfarads, V is the discharge voltage in kilovolts, and the energy is in joules. It will be noted that there is a "most sensitive mixture" or "most easily ignitable mixture," and that ignition energy varies slowly around that composition and then increases rapidly as either flammable limit is approached. Note particularly that the ordinate in Figure 2 is plotted on a semi-logarithmic scale so that the ignition energies have varied by orders of magnitude over the range of the determinations. The composition of the most sensitive mixture varies with the relative molecular weights of fuel and oxidant (where data are available). For equal molecular weights, the stoichiometric mixture is the most easily ignited; if the fuel is lighter than the oxidant, the most easily ignited mixture is one which is fuel lean; if the fuel is heavier than the oxidant, the most easily ignitable mixture is one which is fuel rich.

If the fuel is hydrogen, the minimum ignition energy for the most sensitive mixture with air is about ten times as large as that for the most sensitive mixture with oxygen. For fuels containing carbon and hydrogen, the corresponding ratio of ignition energies is approximately 100. Table I lists minimum ignition energies and ignition quenching distances for representative fuels with oxygen or with air. It may be noted that hydrogen and the saturated hydrocarbons have essentially equal ignition energies in oxygen atmospheres.

The effect of pressure upon ignition energies and ignition-quenching distances is such that ignition energies vary approximately as the inverse square of the pressure ($H \sim p^{-2}$), and ignition quenching distances vary as the inverse first power ($L \sim p^{-1}$). Since the inverse relationship of the quenching distance and Paschen's law for spark breakdown imply a constant breakdown voltage for minimum energy ignition, it follows that the required energy storage capacitance varies inversely as the square of the pressure.

To this point we have considered only the possibility of ignition resulting from a clearly defined gaseous discharge occurring between two metallic electrodes. It has been implicitly assumed that the sparks occur as single isolated sparks. Four other circumstances deserve note.

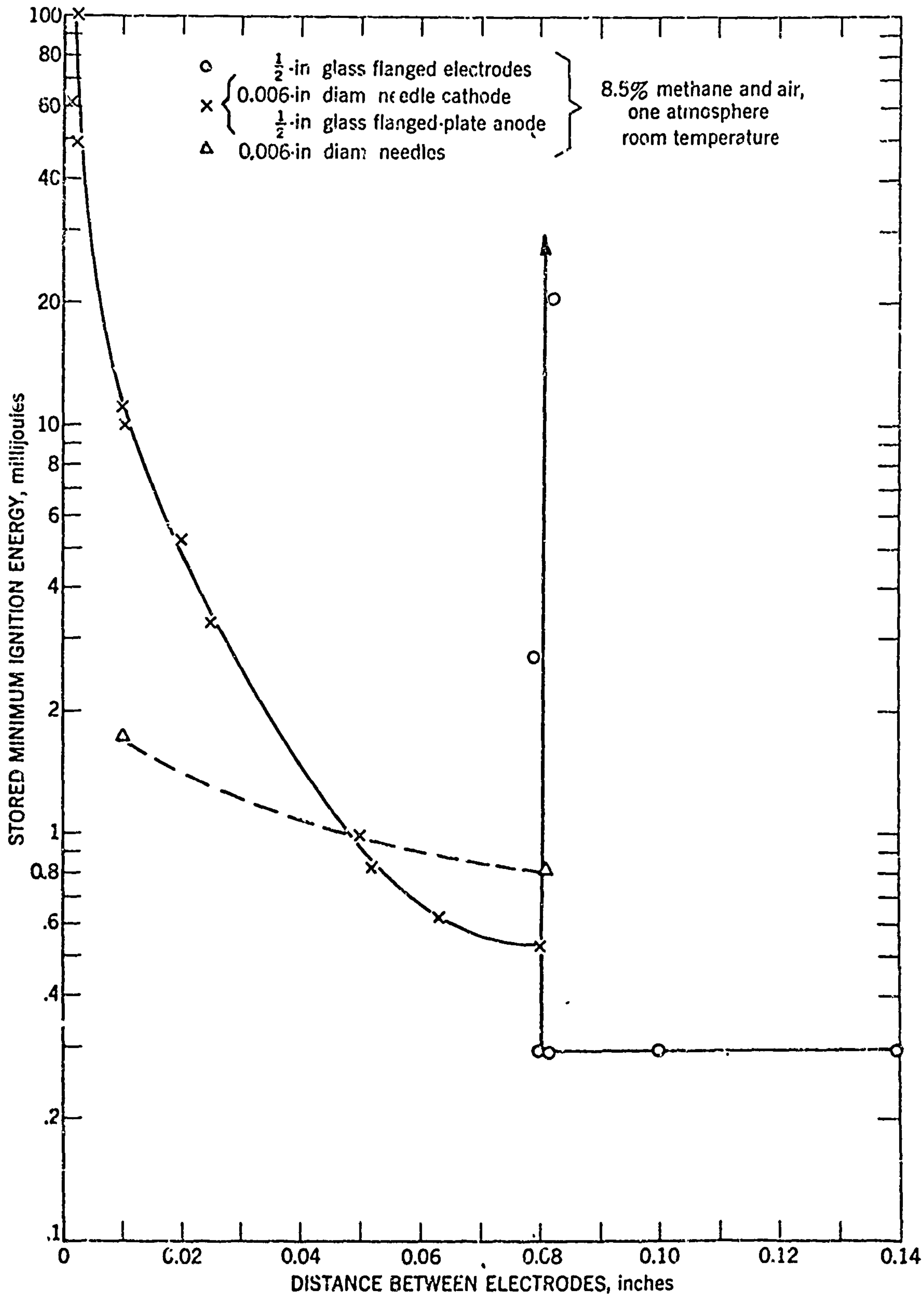


Figure 1. The Effects of Electrode Geometries and Separations Upon Minimum Ignition Energy Requirements

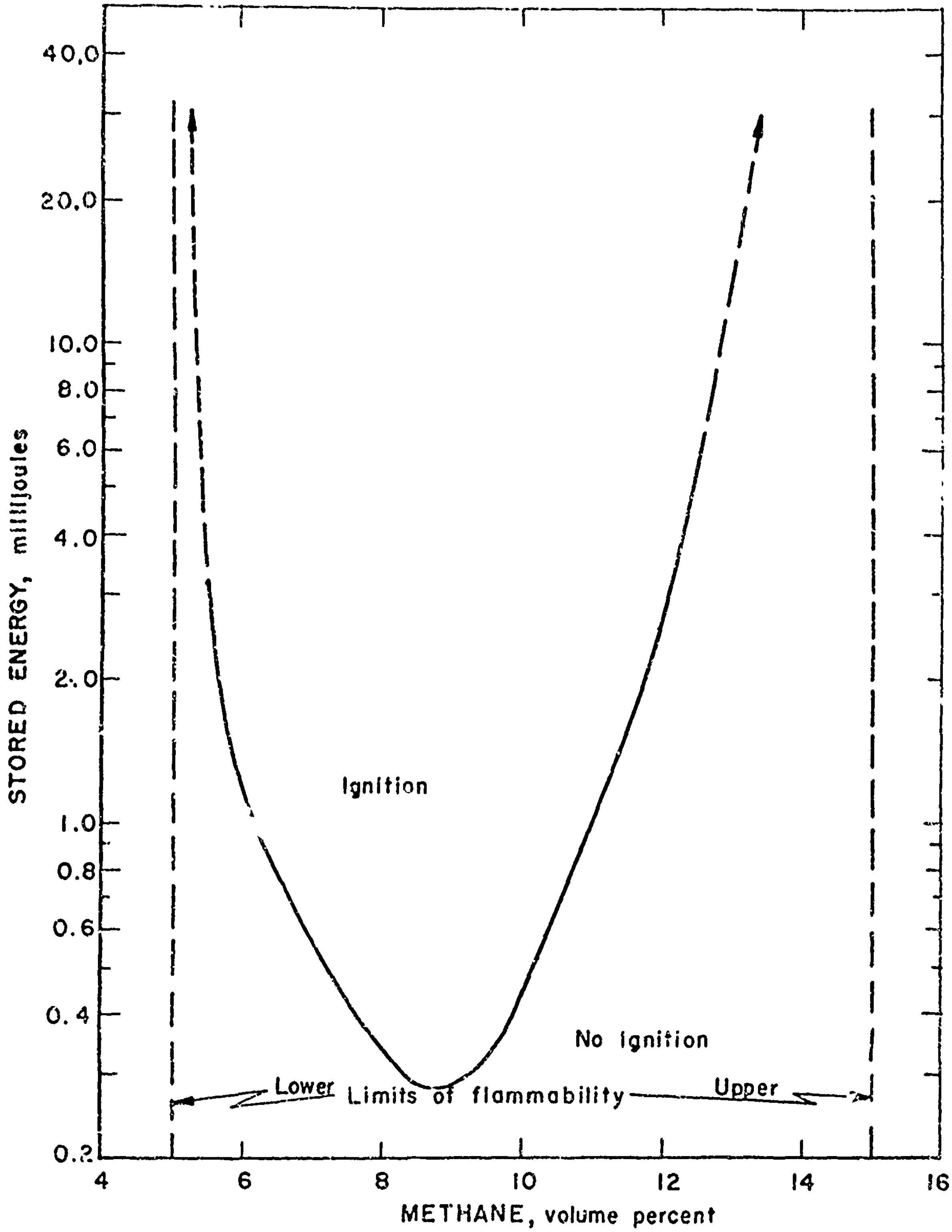


Figure 2. Minimum Ignition Energy Vs. Composition for Methane-Air

TABLE I

MINIMUM IGNITION ENERGY AND IGNITION QUENCHING DISTANCES FOR REPRESENTATIVE FUELS WITH VARIOUS OXIDIZERS

| | Quenching distance, cm | Experimental ignition energy, mj |
|---|---|---|
| Acetylene-oxygen | 0.018 | 0.0002 - 0.0004 |
| Acetylene-air | .064 | .017 - .018 |
| Hydrogen-oxygen | .025 | .0012 - .0014 |
| Hydrogen-air | .064 | .017 - .018 |
| Ethylene-oxygen | .023 | .0009 - .001 |
| Ethylene-air | .122 | .07 - .08 |
| Methane-oxygen | .030 | 0.0027 |
| Methane-air | .203 | .3 |
| Nitric oxide-hydrogen | ~ .635 | 8.7 |
| Methane-nitric oxide | ~ .635 | [illegible].7 |

(1) If the sparks have repetition rates of several per second or more, it is possible to get ignition with an energy per spark which is less than the minimum ignition energy for a single spark. This result is a consequence of finite thermal relaxation times. It is important primarily at low pressures where the long ignition quenching distances may give thermal relaxation times of appreciable fractions of seconds.

(2) There are literature references to "ignition by corona currents." Typically, the ignition currents are several milliamperes, but accompanying data do not establish the gaseous discharge state. The author has been unable to obtain "corona" currents of such magnitude in gaseous mixtures without disruptive discharges which should be called sparks (Reference 4). Leonard and Carhart (Reference 5) have made a valuable study of the nature of discharges from fuel surfaces and demonstrated failure to produce ignition under their experimental conditions.

(3) If a short-duration, high-voltage pulse is applied between a metallic electrode and dielectric surfaces, it is possible to obtain ignition of combustible

gases without the occurrence of a bright discharge channel. The energetics of these circumstances have not been extensively investigated; in the laboratory demonstrations to date, both voltages and energies have appreciably exceeded those involved in normal minimum energy ignitions (Reference 4).

(4) Within the last few years, ignition sparks have been drawn from the surfaces of charged plastics of the polymethyl-methacrylate type (References 6, 7). The surface of the plastic was charged, an electrode was advanced toward the plastic surface, and the ensuing discharge passed through a combustible mixture. The system with the charged plastic has considerable similarity to that with a charged, high resistivity liquid, which is the subject of the second portion of this paper.

## TRIBOELECTRICICATION OF LIQUIDS WITH REGARD TO THE IGNITION OF FLAMMABLE VAPOR-AIR MIXTURES

Varying amounts of charge transfer occur as liquids are pumped through pipes. A parameter, known as the charge relaxation time, is defined as $K\rho\epsilon$, where K is the dielectric constant, $\rho$ is the resistivity of the dielectric, and $\epsilon$ is the conversion factor appropriate to the set of units (Reference 8). If the charge relaxation time is sufficiently long, significant amounts of charge may be pumped into a receiving vessel or tank, which may have a flammable vapor-air mixture above the liquid. As hydrocarbon fuels are refined to higher and higher resistivities, the charge retention in the liquid becomes more apparent and the question naturally arises as to whether such electrification can cause inadvertent ignition. The liquids with which we are concerned here are fuels, not monopropellants, so that the possibility of ignition is entirely restricted to the flammable mixtures existing in the ullage. To examine the question of the possibility of ignition, we assume the worst case — namely, that the ullage is uniformly filled with the most easily ignitable mixture of alkane-air composition. The following exposition demonstrates that lack of ignition can be achieved only by maintaining the electric field strength below the breakdown strength of air and by satisfying certain additional requirements.

The normal spark discharge or spark transition is one where the gas goes along a nonreversible path from its normal, essentially nonconducting state to a conducting one. There are two terminal states which may be relatively stable end points of the transition (References 9, 10). One of these is the glow discharge state, which is characterized by currents that are typically tens of milliamperes and a few tens or hundreds of voltages across the discharge channel; the current carriers are formed within the gas discharge column. The other stable end state is the arc state, normally occurring between metallic electrodes, which is characterized by ampere currents, voltages of a few volts or a few tens of volts across the discharge column, and the production of charge carriers at the electrodes rather than in the discharge column. The most efficient use of electrical energy to heat gas molecules probably occurs in the regime between the initially nonconducting state and the glow discharge state; this portion of the transition is believed to be the significant part of the minimum energy spark ignition.

It is well known that gaseous electrical discharges are characterized by "negative resistance characteristics" (Reference 10), which is to say that the tendency within the discharge channel is for the current to increase as the voltage across the discharge channel decreases. Nevertheless, such discharges can be controlled by external series resistance. The control condition is simply that the current supply be at sufficient potential to allow a series resistor large enough that the positive control characteristic of the series resistor overcomes the negative dynamic characteristic of the gas discharge.

Since the steady gaseous discharges can be controlled or prevented by external series resistance, it would seem that transient discharges should be preventable by the same technique. Two factors, the capacitance and the required safe level of resistance, prevent practical use of such a technique. The conceptual capacitor is usually described as having a metallic charge collection surface such that there is no difference in potential over the plates during current flow. However, the electrostatic definition of capacitance (Reference 11) involves only geometry and dielectric constants. (More complex definitions can be employed, as described in Reference 8, but produce equivalent results in this discussion.) Assume that a small portion of the surface of a $10^{14}$ ohm/cm liquid supports a thin layer or object of dissimilar material. (The capacitance of the object or layer does not depend upon its resistivity.) If the effective surface resistivity of the body or layer is $10^{6}$ ohms/square, the potential drop over the "condenser plate" for a 1-microampere current will be about 1 volt; for a 1-milliampere current, it would be about 1 kv.

Existing data do not suffice to determine a current level at which the spark transition becomes irreversible and/or hazardous; there is justification, both from Bureau of Mines laboratories (Reference 4) and from the literature (Reference 10), for considering that this current level is between 1 microampere and several milliamperes. Although dark currents and/or corona currents of 1 microampere have not been reported to produce ignition, currents of several milliamperes have been reported to produce ignitions and should be representative of glow discharges.

Suppose there is a metal tank with a distance of 2 feet from the center of the liquid to the metal wall. The dielectric breakdown strength of air is ordinarily taken as 30,000 volts/cm, so that a potential difference of about 1.8 megavolts could exist before breakdown occurs. Considering only the liquid surface, it is seen that if the surface resistivity is about $2 \times 10^{12}$ ohms/square, a 1-microampere current cannot flow. However, if a patch of a few square centimeters exists where resistivity is only $10^{6}$ ohms/square, milliampere transient currents could be obtained while reducing the 1.8 megavolts total potential difference by only a few tens of kilovolts. Thus, if it is possible to guarantee the surface resistivity of the liquid, research on the critical heating and transition current flows would suffice to specify a surface resistivity which is high enough to prevent hazard, at least to the point where internal dielectric rupture of the liquid would become possible.

It is probably impossible to ensure that surface patches of resistivity of $10^{6}$ ohms/square or less do not appear on the surface of liquid within the storage tank during pumping operations. In the absence of an assured high surface resistivity, the only way that electrostatic spark discharge can be assured is to hold the electric field strength at all points below the dielectric breakdown strength, thus ensuring that a spark cannot form.

Although a considerable number of effects--innate energy of the electric field, the energy of the electrical double layer, electric charge in the ullage, the modification of various parameters by the electric field, etc.--have not been included explicitly in this discussion, the effects of all are included within the safety prescription given. Individually, it seems that although most of these separate effects can contribute to hazard, they are only secondarily relative to the surface conductivity and capacitance effect. Such a statement relative to the charge in the ullage is dependent upon assumptions about droplet size and other parameters. Abbas, Azad, and Latham (Reference 12) have reported upon the complexities of the problem, and the discharges of Leonard and Carhart (Reference 5) may well reflect the influence of charge in the vapor space.

There is some question as to whether the electrostatic-potential distribution within the storage tank can be computed for arbitrary pumping conditions. Good approximations to the potential distribution can certainly be computed if the tank dimensions and the fluid-motion gradients are such that little relative fluid motion occurs during two or three charge relaxation times.

Three further items deserve specific note, as follows:

(1) Any type of object floating on the surface of the liquid having low to intermediate resistivity would be the counterpart of the film patch capacitor. Since this capacitor may be mobile and may have a charging-time constant of seconds or minutes, a hazard can exist even though the liquid-potential gradients are such that breakdown would not otherwise occur.

(2) An object inserted into a tank at a rate which is rapid as compared to the dielectric relaxation time of the liquid has the effect of causing rapid variations in potential and may cause discharge from the "film patch condensers" on the liquid surface. Therefore, such items as gaging poles should be avoided, if possible.

(3) Plastic liners and plastic piping should be considered in terms of the specific resistivity of the plastic, the effective resistance of the plastic items, and the magnitude of the triboelectric current. If the electrical resistivity of the plastic is several orders of magnitude smaller than the resistivity of the fuel, then the plastic effectively short-circuits the liquid. Under such circumstances, plastic liners would not increase the hazard in the interior of the tank, per se; however, potential differences along a plastic pipe would depend upon the triboelectric current and the effective resistance of the pipe. "Charged Man" hazard must be considered to be increased in any instance where one may use "nonconductive" plastic piping (specific resistance greater than about $10^3$ ohm/cm$^2$).

SUMMARY

Mixtures of saturated straight-chain hydrocarbon (alkane) fuels with air may be ignited by the discharge of about 0.25 mj energy at 1 atmosphere pressure. It does not appear possible to ensure that such energy cannot be discharged from a triboelectrified fuel surface if dielectric breakdown occurs. The only method of ensuring complete safety, therefore, is to ensure that electric field strengths within the tank ullage are not permitted to equal or exceed the dielectric breakdown strength of the fuel-air mixture. Since the severity of the ignition hazard is dependent upon ambient pressure and the specific flammable mixture present, complicated pressure dependence may result.

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AFAL-TR-68-290
PART II

# INVESTIGATION OF TURBINE FUEL FLAMMABILITY WITHIN AIRCRAFT FUEL TANKS

L. Maggitti and L. J. Nestor

Naval Air Propulsion Test Center

Under the sponsorship of the FAA, the Aeronautical Engine Department of the Naval Air Propulsion Test Center undertook this study to establish some of the effects that dynamic aircraft environments might have on fuel tank flammability.

The flammability of the vapor space in a fuel tank varies according to the relative concentrations of evaporated fuel to air. Reducing the fuel/air ratio below a definite limiting value produces a mixture that is too lean in fuel to burn. Likewise, another limiting value exists where the fuel/air ratio is too rich in fuel to burn. The particular fuel/air ratio which exists is determined by temperature and pressure. The temperature determines the vapor pressure of the fuel, and the altitude determines the quantity of air. Therefore, by a suitable combination of temperature and altitude, the fuel tank ullage can be made either flammable or nonflammable. The flammability of a fuel tank is designated by a flammability envelope, plotted in terms of altitude and temperature. The borders of such an envelope are the lean and rich limits. For turbine fuels, these flammability limits, when expressed on a weight basis, are nominally considered to be 0.035 for the lean limit and 0.26-0.28 for the rich limit (Reference 1).

These nominal fuel/air ratios are too crude to permit defining the flammability envelope of a specific turbine engine fuel with any degree of accuracy or confidence because fuels are rather complex blends of hydrocarbons. A wide-cut turbine engine fuel, such as JP-4 or Jet B, may contain as many as 5000-10,000 different hydrocarbon compounds (Reference 1). Each compound exhibits its own characteristic vapor pressure and limiting flammable fuel/air ratio. Relatively small differences in the composition and volatility between several batches of the same type fuel are reflected in differences between their respective flammability envelopes.

In addition to the effects of pressure, temperature, and compositional variations in jet fuels, the fuel/air ratio of the tank ullage can be altered by mechanically dispersing the fuel. Mechanical dispersion in the form of spray can occur as a result of sloshing and vibration by atmospheric turbulence and other sources of agitation. The nature and the extent of the spray formation varies in accordance with the magnitude of the disturbance. In the final analysis, the fuel/air ratio of the tank ullage, regardless of the physical state of the fuel, is altered because of the existence of spray. This paper, based on some of the work done in the FAA sponsored investigation, discusses:

(1) The apparatus and method which were designed primarily for overcoming the experimental difficulties inherent in defining the flammability characteristics of multicomponent liquids such as the turbine fuels.

(2) The equilibrium flammability envelopes of a representative wide-cut turbine engine fuel, an aviation kerosene, and mixtures of the two.

(3) The deviations from the equilibrium flammability envelopes that are introduced by the presence of sprayed fuel produced by simulated aircraft vibration.

## DYNAMIC COMBUSTION APPARATUS

A dynamic combustion apparatus was designed to determine the flammability characteristics of turbine fuels at both static (equilibrium) and dynamic conditions. Vibration was selected as the most representative form of aircraft dynamics, and was incorporated solely for producing sprayed fuel. The principle of design incorporated in the apparatus was unique in that fuel was maintained in the combustion vessel at the time of ignition. Through instrumentation of the apparatus, the resultant temperature and pressure rises occurring at combustion were recorded. These flammability parameters were then related to fuel type, ambient temperatures, altitudes, and fuel spray.

The combustion apparatus used in these investigations is shown schematically in Figure 1. The combustion bomb was constructed from a surplus 2-1/2 gal hydraulic accumulator, having the approximate dimensions of 9 in. in diameter and 15 in. in height. It was supported within a temperature controlled chamber by an extended mounting to an electromagnetic shaker. Instrumentation

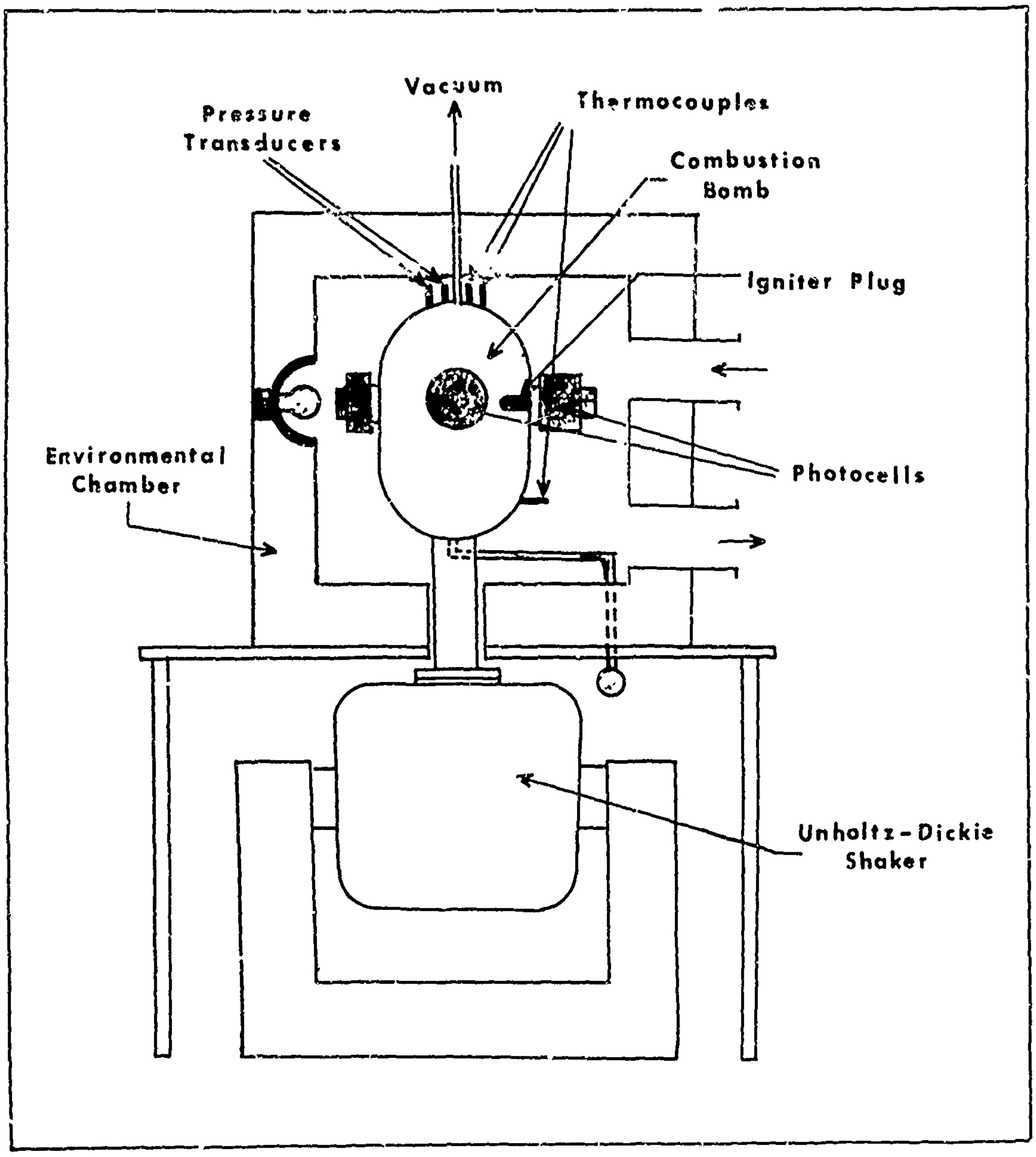


Figure 1. Dynamic Combustion Apparatus

of the combustion bomb included photocells mounted in viewing ports for scanning the section where combustion and ignition were centralized. The existence of spray and its position relative to the point of ignition could be monitored by these photocel through variations in scattered and transmitted light. Two thermocouple readouts enabled the measurement of the steady state temperatures of the liquid fuel and the tank ullage. A third thermocouple, monitored on one channel of a dual trace oscilloscope, recorded the temperature rise within the tank ullage produced by ignition. Two pressure transducers completed the instrumentation of the combustion bomb. One pressure transducer monitored steady state pressure conditions; the second transducer recorded the pressure transients occurring at ignition on the second channel of the oscilloscope. Thus, the pressure and temperature rises occurring at ignition were recorded simultaneously. Ignition of the vapor space was initiated by a J-57 igniter plug. The igniter plug was energized by a capacitor discharge electronic ignition system such as is used to start turbojet and turboprop aircraft engines. The ignition energies of the sparks were in the range of 16-24 joules per spark.

## EXPERIMENTAL PROCEDURE

The procedures employed involved the introduction of a constant volume of the test fuel into the combustion bomb, which had previously been evacuated to a vacuum of 29-29.5 in. of mercury. The volume of fuel tested was standardized to produce a system ullage of 87.5%. Upon stabilization of fuel and bomb temperature, dry air was introduced into the chamber, increasing the pressure until the predetermined altitude equivalent had been reached. To ensure the existence of equilibrium conditions, extreme vibration of the combustion bomb was induced which, in turn, caused violent agitation of the fuel. This scrubbing of the air by the agitated fuel ensured complete saturation of the air with evaporated fuel. Saturation is, in essence, equilibrium. Techniques were developed and ullage monitored optically to preclude the existence of dispersed fuel droplets in the vapor space upon conclusion of the scrubbing procedure. Ignition of the tank ullage was produced by sparking of the igniter plug at the various combinations of altitude, temperature, and dynamics. The resultant temperature and pressure phenomena were recorded on an oscilloscope.

Combustion studies were conducted within the framework of the following conditions:

(1) Static combustion — characterized by equilibrium vapor space conditions.

(2) Dynamic combustion — characterized by the presence of sprayed fuel within an equilibrium vapor space with:

(a) The point of ignition external to the spray envelope.

(b) The point of ignition internal to the spray envelope.

The varying degrees of agitation of the fuel to control the relative position of the igniter plug within or outside the spray were provided by suitable selections of vibration frequencies and amplitudes. The relative position of the igniter plug to the spray envelope was established from prior optical calibration of the combustion bomb. For condition 2a, where spray was produced but was not near the spark, the shaker conditions employed were a vibration frequency of 10 cps and an amplitude of ±1/16 in. These conditions produced a spray pattern with a maximum height of approximately 3 in. above the liquid fuel level. Since the fuel level was approximately 5 in. below the igniter plug, the spark was about 2 in. above the maximum height of the spray. Where the spray pattern completely enveloped the igniter plug, condition 2b, the vibration frequency of 15 cps and an amplitude of ±1/8 in. was employed. This vibration setting resulted in violent agitation of the fuel, to the degree that the entire vapor space of the bomb was completely filled with sprayed fuel droplets.

## RESULTS

### 1. TYPES OF TURBINE ENGINE FUELS

Turbine fuels basically belong in two categories, wide-cut turbine engine fuel and aviation kerosene. The wide-cut turbine engine fuel can be considered to consist of both a gasoline and a kerosene fraction (Reference 2). It has a relatively broad distillation range of approximately 175-500°F and a Reid vapor pressure of about 2-3 psig. The second fuel type, the aviation kerosene, is characterized by a narrower distillation range of approximately 350-500°F and has a negligible Reid vapor pressure. Therefore, the vapor pressures are not controlled by specification requirements. These two fuel types are purchased for operational use according to military or commercial specifications. In this report, the two fuel types are represented by Jet A-1 and Jet B. Jet A-1 is one of two commercial designations for aviation kerosene. (The other kerosene turbine fuel is Jet A, and its military counterpart is JP-5.) Jet B is the commercial designation for the wide-cut turbine engine fuel, and its military counterpart is JP-4.

## 2. NATURE AND INTERPRETATION OF INSTRUMENTAL FLAMMABILITY DATA

Some representative oscillograms used to define flammability of Jet A-1 are shown in Figure 2. These oscillograms record the temperature and pressure rises produced by ignition at the designated tank environmental conditions. The upper left oscillogram represents ignition at a simulated altitude of 30,000 ft and temperature of 50°F; ignition failed to produce a pressure or temperature rise. At an increased environmental temperature of 62°F, and still maintaining an altitude of 30,000 ft, however, ignition was accompanied by a temperature increase of 135°F and a pressure increase of 15 psi. This is shown in the upper right oscillogram of Figure 2. The two oscillograms shown at the bottom of Figure 2 represent ignition data obtained at sea level. The bottom left oscillogram recorded ignition reactions of a low magnitude at sea level and 87°F. By raising the environmental temperature to 95°F, the ignition reaction was recorded to be one of a high magnitude.

The demarcation between the high and low reaction regions can be established by the empirical limiting values of a 4 psi rise in pressure and/or a 50°F rise in temperature at ignition. The particular significance of these values are demonstrated in Figures 3 and 4. In Figure 3, for example, it can be seen that the low reaction region is characterized by pressure transients which are less than 4 psi. The 4 psi limiting value pressure is valid at both the altitude ranges shown, that is, 20,000-30,000 ft and at sea level to 10,000 ft. Similarly, the transition temperature of 50°F temperature rise is validated by the data shown in Figure 4. It is believed that there is no theoretical significance to these limiting values. Their quantitative nature is probably related only to this particular combustion apparatus, spark energy, and instrument response. Based on

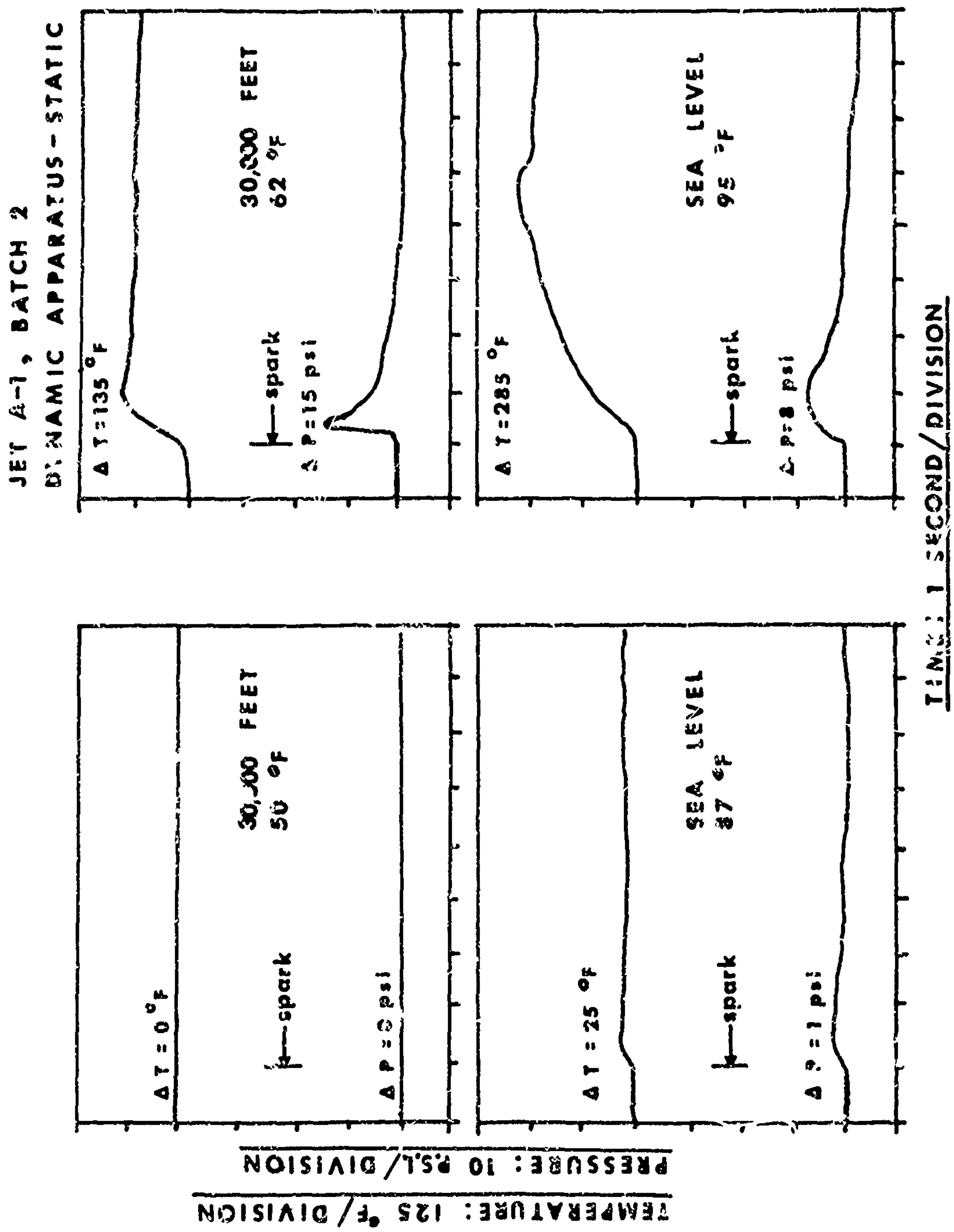


Figure 2. Representative Oscillograms Showing the Ignition Phenomenon of Jet A-1 Under Static Conditions

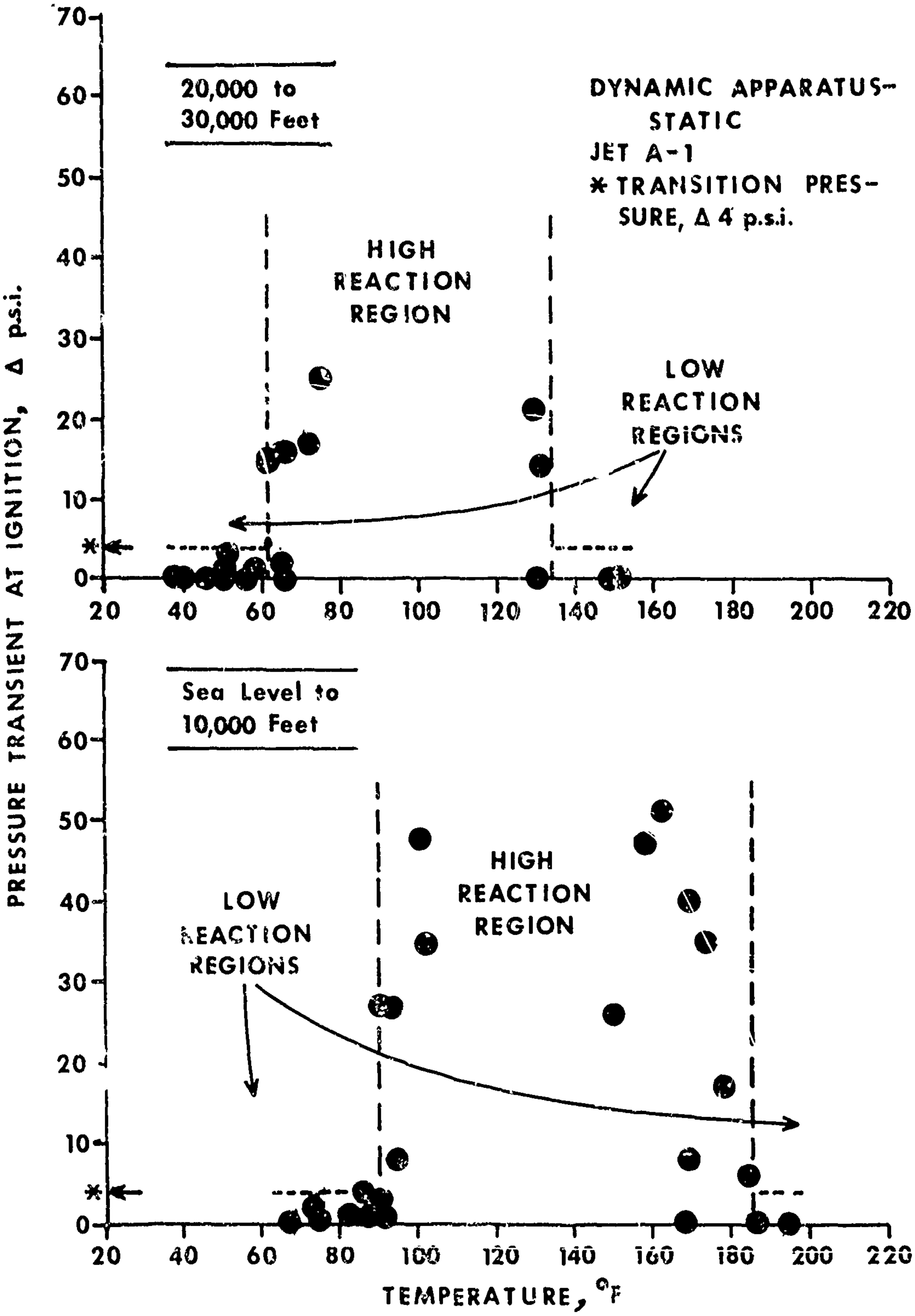


Figure 3. Transitions of Pressure Transients Between High and Low Reaction Regions

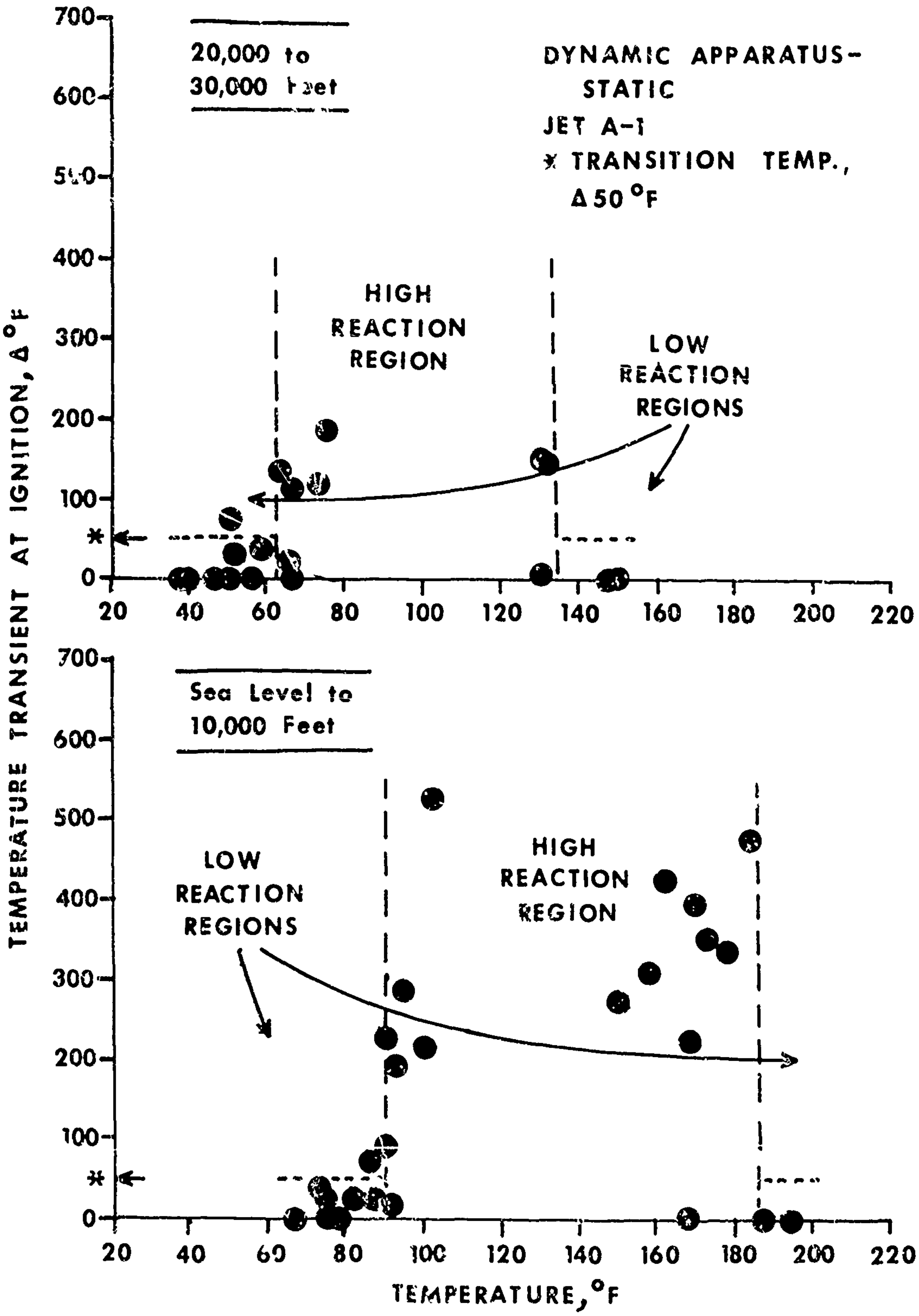


Figure 4. Transition of Temperature Transients Between High and Low Reaction Regions

these empirical distinctions, ignition reactions were classified in the following manner:

(1) No reaction — no recorded pressure rise or temperature rise.

(2) Low reaction — pressure rise of less than 4 psi and a temperature rise of less than 50°F.

(3) High reaction — pressure rise of 4 psi or greater and/or a temperature rise of 50°F or greater.

## 3. EQUILIBRIUM FLAMMABILITY CHARACTERISTICS OF JET A-1

Using the preceding classification of ignition reactions, the equilibrium flammability envelopes of the two basic fuels were developed. The flammability envelope of Jet A-1, representing the aviation kerosenes, is shown in Figure 5. The envelope designates the high reaction region, which is bordered on the left by the lean limit and on the right by the rich limit. Beyond these limits, the pressure rise was below 4 psi and the temperature rise was less than 50°F.

The equilibrium flammability envelope of the same Jet A-1 fuel was also determined using the more classical glass tube combustion method (Reference 3). The glass tube combustion method is based upon visually confirming the existence and propagation of flames produced by a high energy spark. The tube was 4 in. in diameter and 4 ft in length. This flammability envelope, based upon glass tube combustion, is also shown in Figure 5, in comparison with the envelope defined in terms of ignition reactions. The two envelopes, although not identical, were sufficiently similar to conclude that the high reaction region is roughly comparable to the flammability region, as determined by visual confirmation in a glass tube. To distinguish between the two envelopes, the limits obtained with the dynamic combustion apparatus are referred to as the transition limits. The limits obtained by visually observing flames are designated as flammability limits.

The pressure profile describing the pressure rise characteristics of Jet A-1 at equilibrium is shown in Figure 6. These pressure rises are plotted against the temperature differential between the actual test temperature and the transition limit temperature for the altitude test condition. The pressure

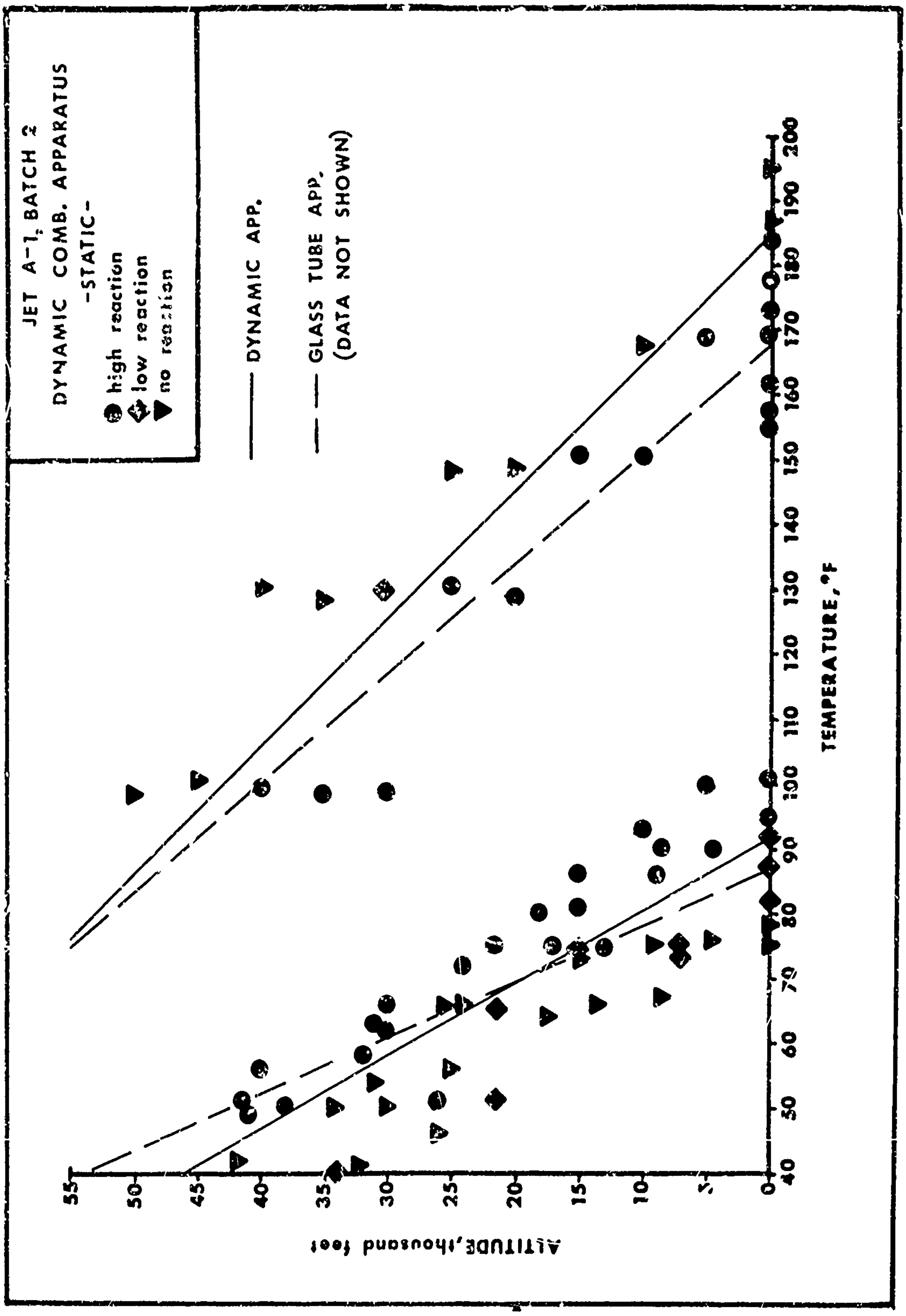


Figure 5. Equilibrium Flammability Envelope of Jet A-1

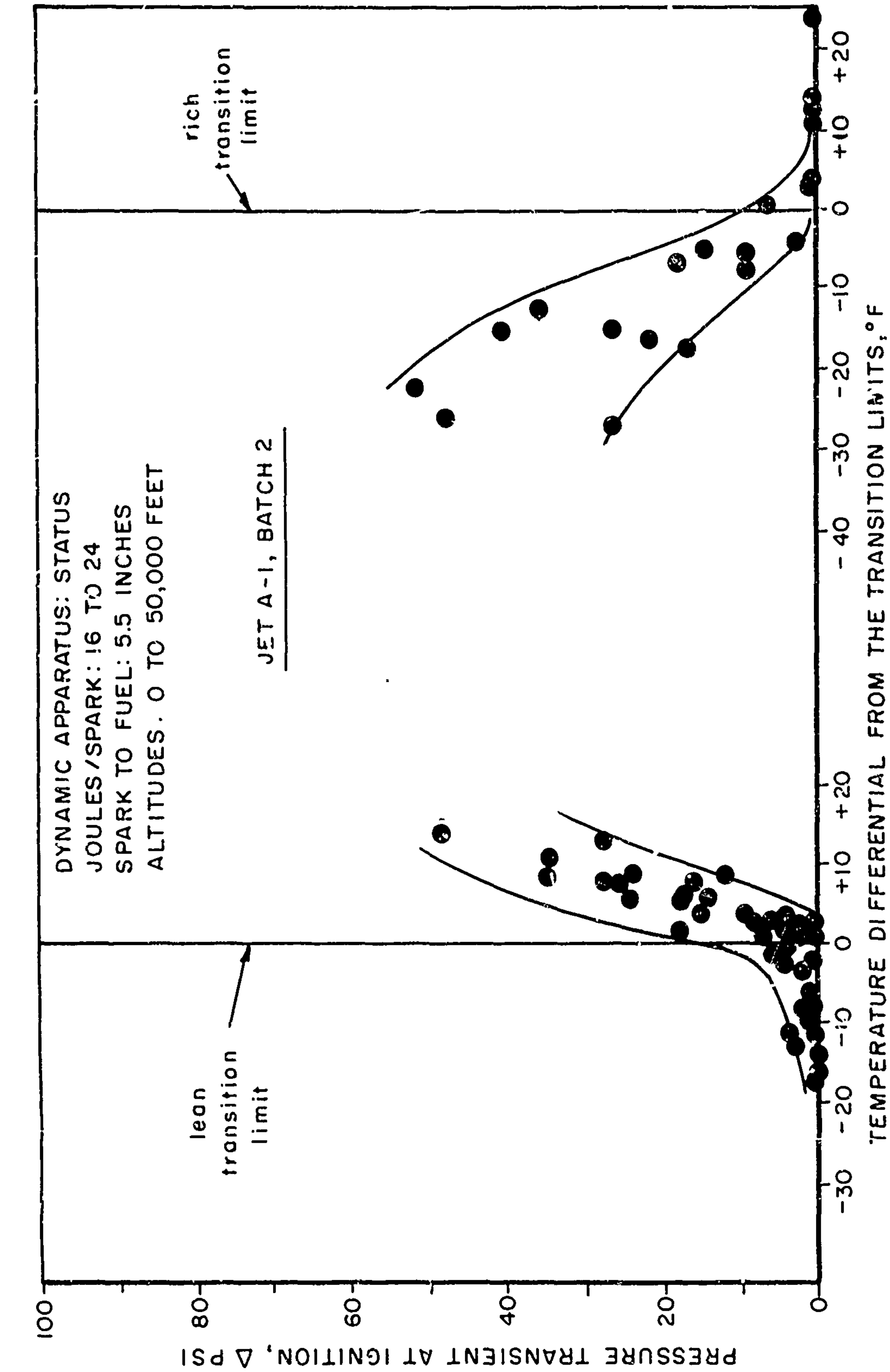


Figure 6. Pressure Profile for Jet A-1 at Equilibrium (Static) Conditions

profile of Jet A-1 demonstrates that, as the environmental temperatures are varied, representing a transition through the flammability envelope from left to right (lean to rich), the resultant ignition pressure rises achieve a maximum value somewhere within the envelope. The maximum pressure rises were not documented experimentally because there were overload restrictions of the pressure transducers, which prevented the complete investigation of the pressure profiles.

The temperature profile for the equilibrium combustion of Jet A-1 is shown in Figure 7. In principle, the temperature profile is plotted on an identical basis as the pressure profile. The appearance of the temperature profile is like that of the pressure profile, in that the maximum temperature rises are produced well within the flammability envelope.

## 4. EQUILIBRIUM FLAMMABILITY CHARACTERISTICS OF JET B

The equilibrium flammability envelope for the wide-cut turbine engine fuel, Jet B, is shown in Figure 8. The fact that Jet B is a more volatile fuel than Jet A-1 is reflected in the relative temperature ranges of their flammability envelopes. The sea level transition limits of Jet A-1 were 92°F and 184°F, which compares with the equivalent limits for Jet B of -8°F and 56°F. Although the temperature ranges of the flammability envelopes of the two fuels were different, they resemble one another qualitatively because the slopes of their respective limits are similar. This similarity also exists in the pressure and temperature profiles for the two fuels. The pressure and temperature profiles of Jet B at equilibrium are shown in Figures 9 and 10. A comparison between the pressure profiles of Jet A-1 and Jet B suggests that Jet B tends to produce pressure rises that are somewhat greater than those of Jet A-1. The temperature profiles of the two fuels did not appear to be different.

## 5. FLAMMABILITY CHARACTERISTICS OF JET A-1 UNDER DYNAMIC CONDITIONS

From experiments involving aircraft dynamics, it was concluded that, for the most part, the net effect of such motion as rocking and vibration on combustion could be studied in terms of spray within the fuel tank. Vibration was determined to be the most significant factor for producing sprayed fuel (Reference 3).

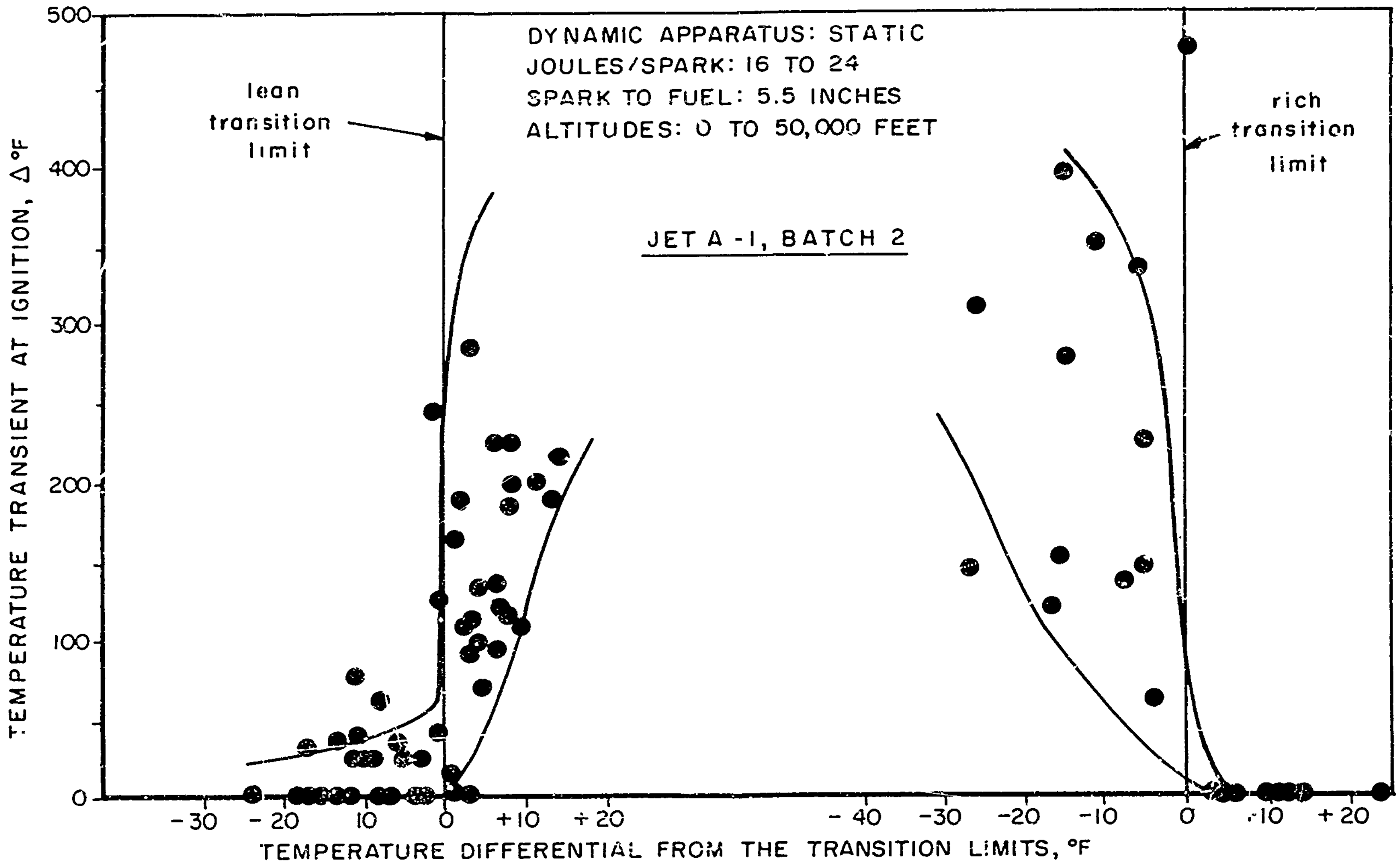


Figure 7. Temperature Profile for Jet A-1 at Equilibrium (Static) Conditions

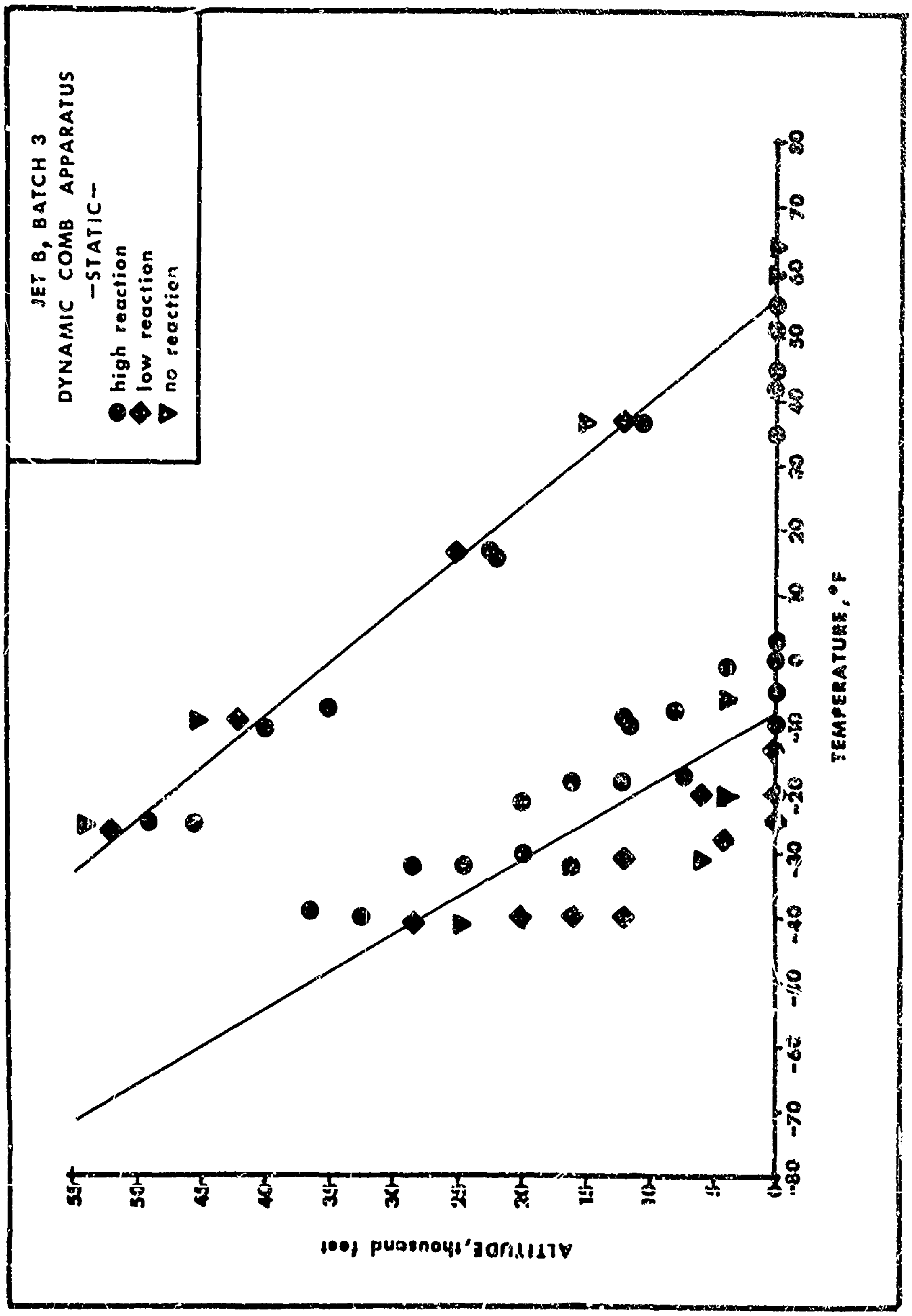


Figure 8. Equilibrium Flammability Envelope for Jet B Fuel

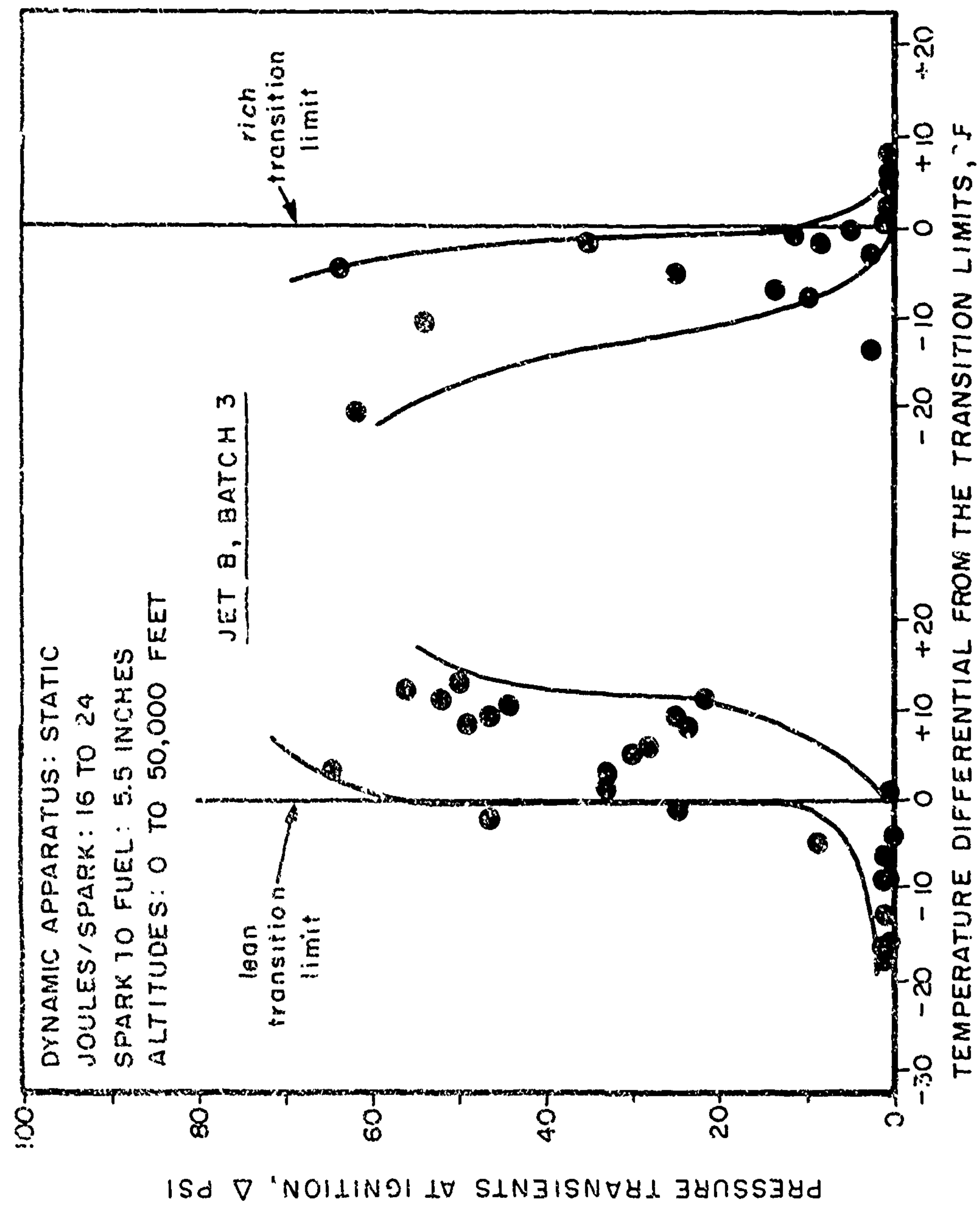


Figure 9. Pressure Profile for Jet B Fuel at Equilibrium (Static) Conditions

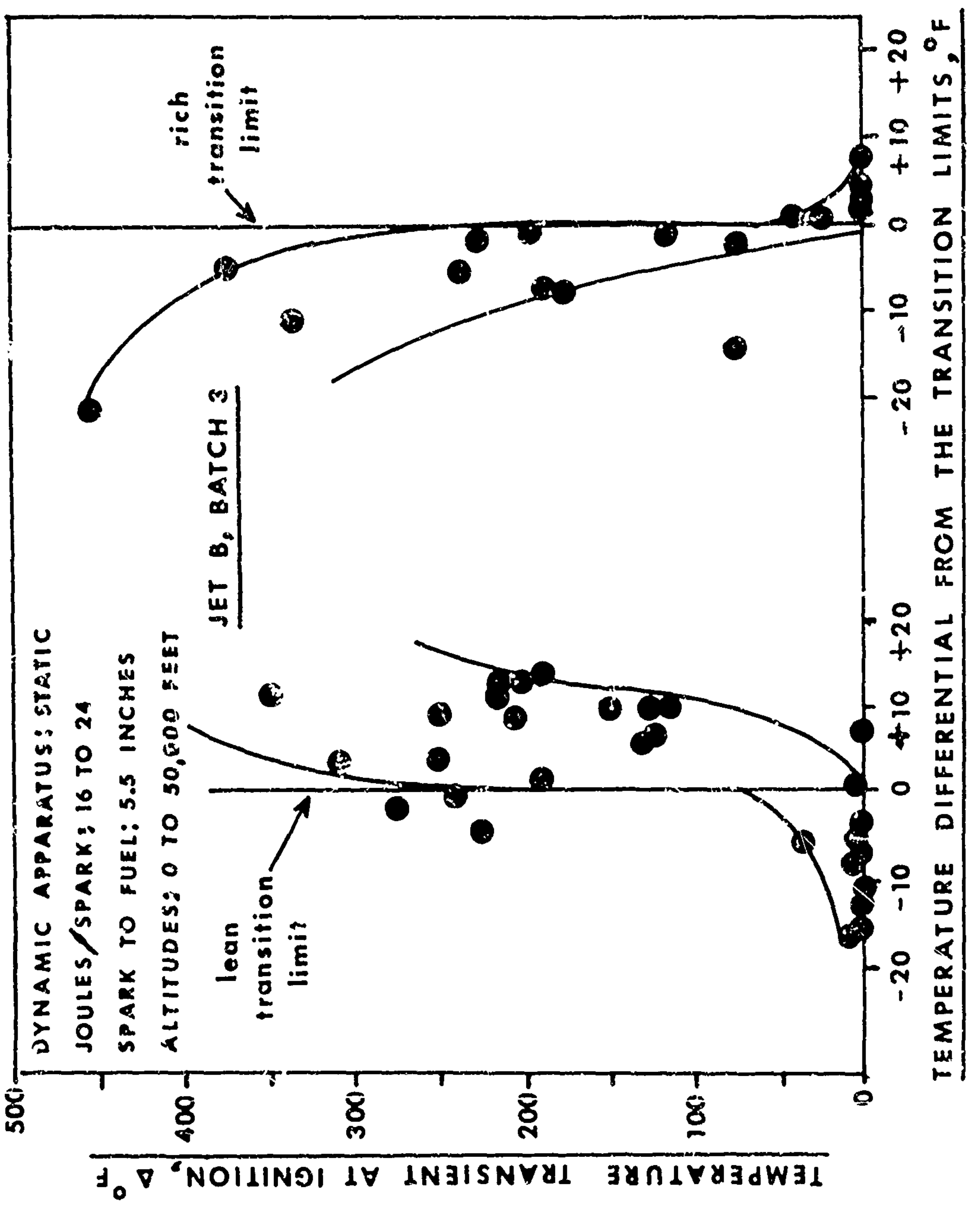


Figure 10. Temperature Profile for Jet B Fuel at Equilibrium (Static) Conditions

Therefore, when discussing sprayed fuel, the term vibration is considered to be synonymous with turbulent flight conditions or aircraft dynamics. Two conditions were studied: the first involved the ignition source external to the spray pattern, and the second the ignition source within the spray pattern of the fuel.

a. Ignition Source External to Spray Pattern

By maintaining a vibration frequency of 10 cps and vibration amplitude of ±1/16 in., fuel spray was produced, but the height of the spray pattern failed to reach the igniter plug. With this vibration, according to prior optical calibration, it was known that the maximum height of the spray pattern was approximately 2 in. from the point of ignition. Maintaining this relative position of the spray, the resultant pressure and temperature rises produced by ignition were determined for Jet A-1. The pressure and temperature rise data, produced under these circumstances where the spray did not extend beyond the point of ignition, are shown in Figures 11 and 12. These data are shown compared to the normal profiles representing ignition under static conditions. These dynamic data are generally consistent with the equilibrium profiles, thereby indicating no significant change in flammability characteristics.

b. Ignition Source Within Spray Pattern

The vibration frequency of 15 cps and amplitude of ±1/8 in. produced a spray pattern of fuel which completely enveloped the igniter plug. The resultant pressure and temperature rises produced by ignition were recorded and compared with the data established under equilibrium conditions. It was determined that when the point of ignition was within the spray pattern, the flammability envelope of Jet A-1 was altered considerably. The combustibility was affected by extending the lean limit; little or no effect was noted on the rich limit. These effects are shown in Figures 13 and 14.

The pressure profile produced when the point of ignition was within the spray pattern is shown in Figure 13. Ignition data were produced only at two altitudes, sea level and 25,000 ft. In Figure 13, these pressure rise data are compared with the pressure profile representing equilibrium flammability. It is quite obvious from their comparison that the presence of spray about the point of ignition has

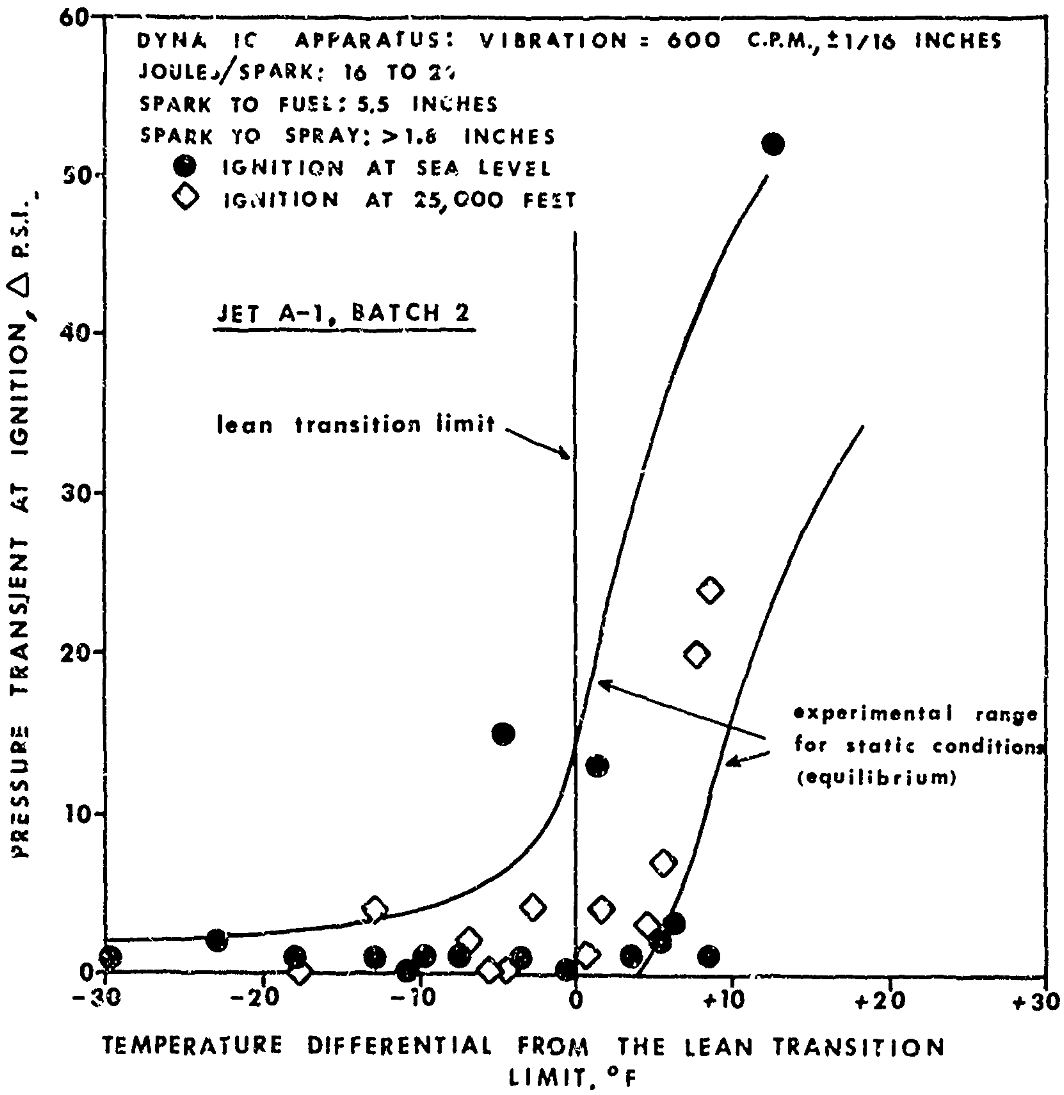


Figure 11. Dynamic Pressure Profile of Jet A-1 with Spark Outside Spray

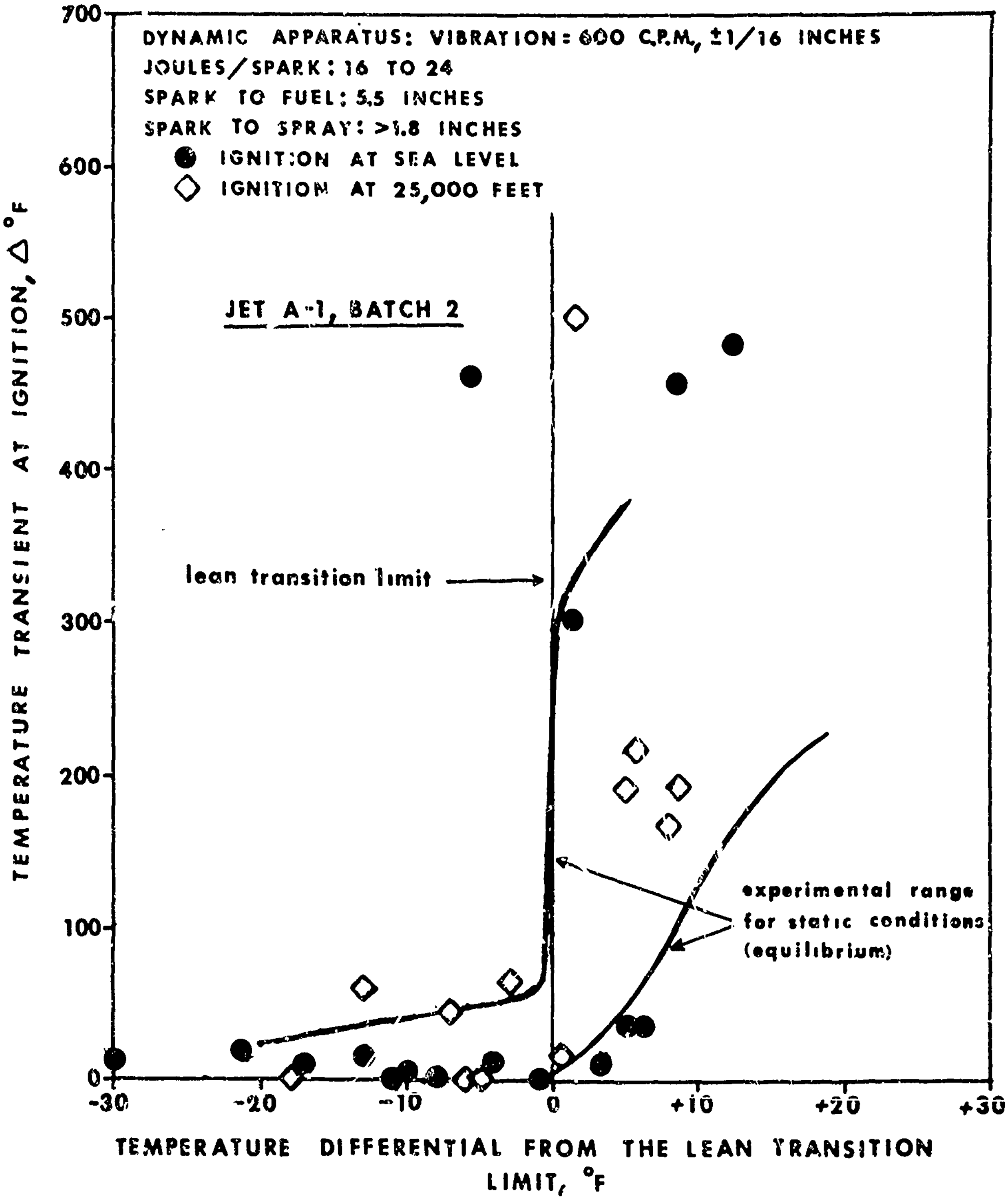


Figure 12. Dynamic Temperature Profile of Jet A-1 with Spark Outside Spray

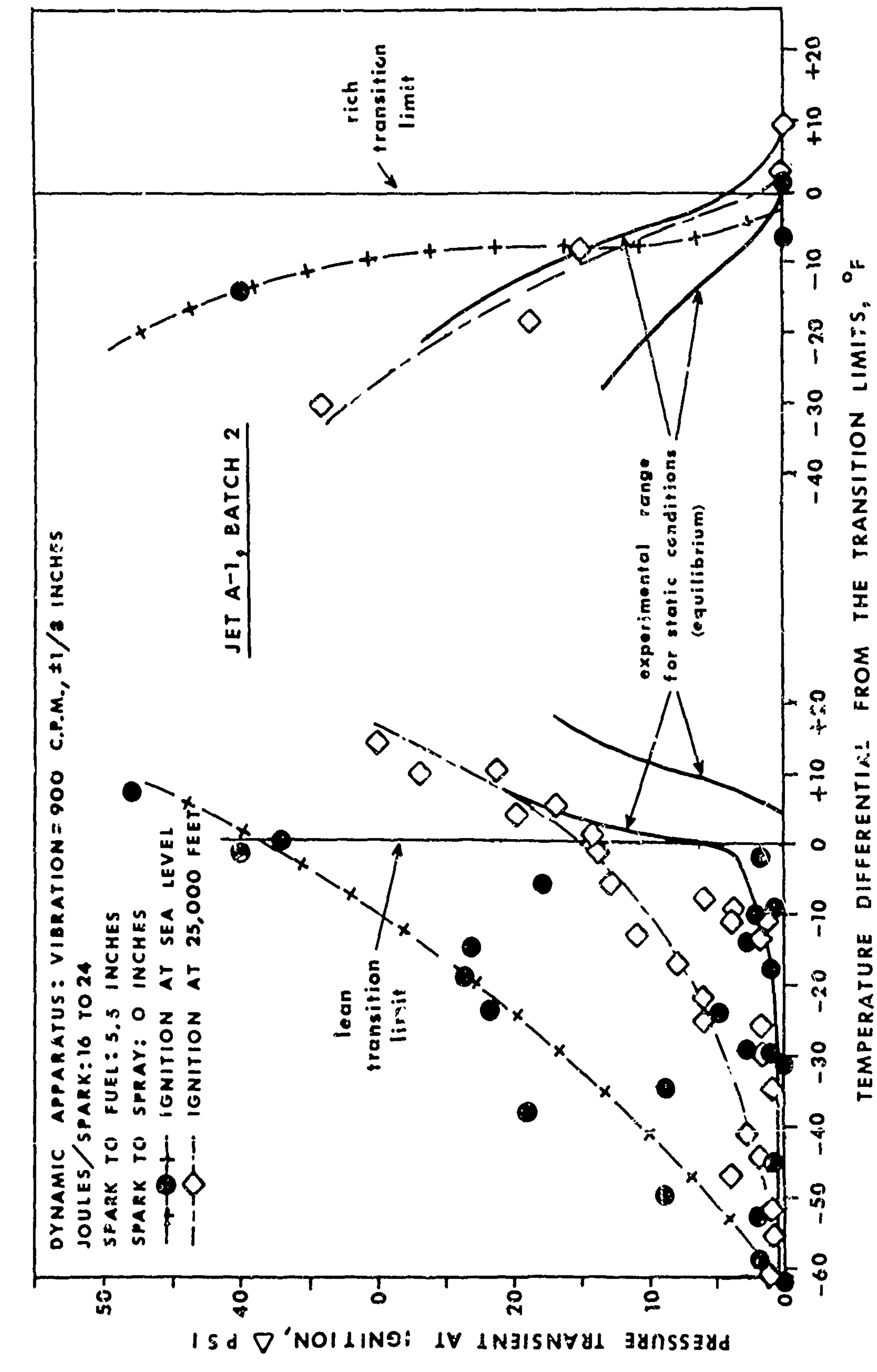


Figure 13. Dynamic Pressure Profile of Jet A-1 with Spark Within Spray

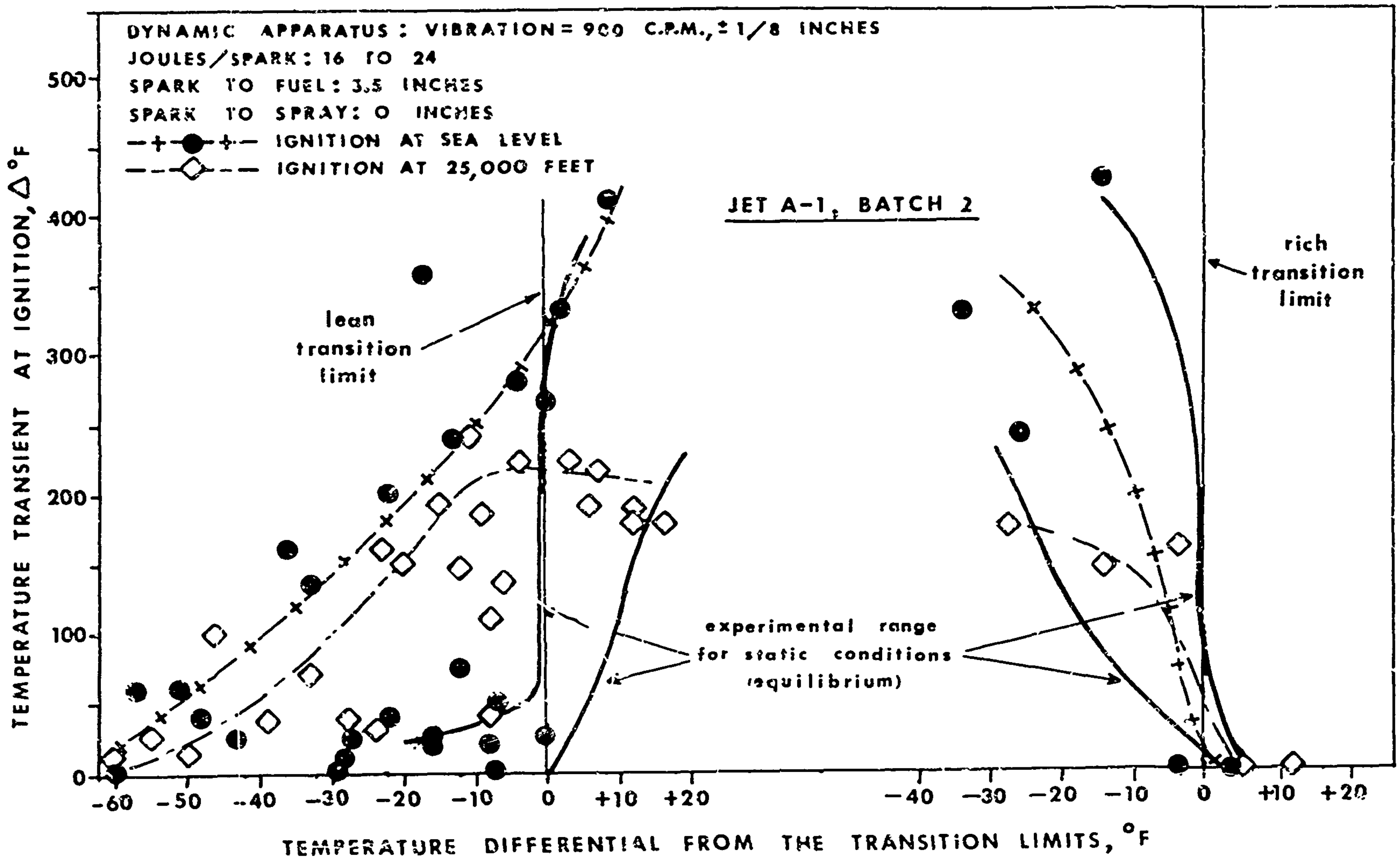


Figure 14. Dynamic Temperature Profile of Jet A-1 Fuel with Spark Within Spray

a marked effect. Using the definition that a high reaction is characterized by a minimum pressure rise of 4 psi, the lean transition limit was extended 54°F at sea level and 32°F at 25,000 ft.

From a similar analysis with respect to temperature rises, according to Figure 14, the lean limit was extended 50°F at sea level and 42°F at 25,000 ft. By maximizing the combined effects from both the pressure and temperature profiles, the lean limit was considered to be extended 54°F at sea level and 42°F at 25,000 ft. Taking these two points as defining a straight line, the lean limit of the dynamic combustibility envelope was constructed.

This dynamic flammability envelope and the equilibrium envelope are shown in Figure 15 in conjunction with the envelopes of Jet B. The net effect of ignition in the midst of spray was to extend the lean limit of the Jet A-1 into the envelope of Jet B. No effect of spray with respect to a change of the rich limits was noted.

The Coordinating Research Council reported fuel temperature ranges estimated to cover approximately 95% of all commercial operations at the cruising altitudes of 35,000-40,000 ft (Reference 4). By combining these temperature ranges with the information available from the flammability envelopes shown in Figure 14, the relative changes produced by spray in the tank become increasingly apparent. Figure 16 documents, theoretically, the extent that the two different fuels would be flammable during the course of flight under static and dynamic conditions. It can be seen that spray, if ignition were within the spray, severely increases the duration of exposure of potential flammability with respect to Jet A-1. At the highest flight temperatures shown, the Jet A-1 tank space would reach a safe condition after 1-1/2 hr. of flight if static equilibrium conditions prevailed. Under dynamic conditions, which produce spray in the vapor space, a safe region would be attained only if the fuel temperature dropped below 0°F. Because of the in-flight temperatures involved, an increase in flammability potential due to spray was not noted with respect to Jet B.

## 6. EQUILIBRIUM FLAMMABILITY CHARACTERISTICS OF MIXED FUELS

Another reason for the existence of a flammable condition in the "safe" temperature zone between wide-cut fuel and kerosene is mixing of fuels. A series

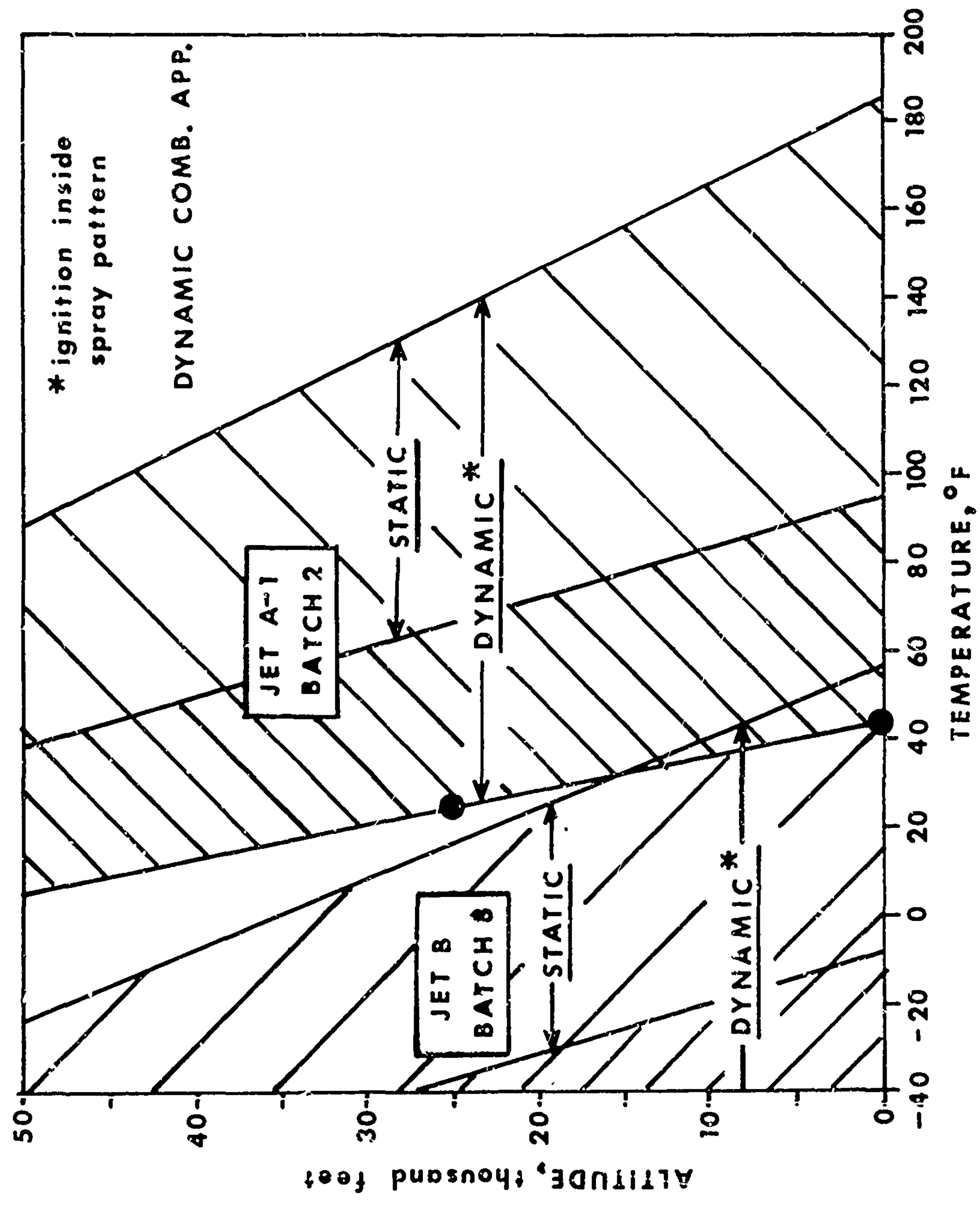


Figure 15. Flammability Envelopes of Turbine Fuels Under Static and Dynamic Conditions

(a) Transition limit temperatures at 40,000 feet

(b) Ignition inside spray pattern

(c) Temperature ranges estimated to cover approximately 95% of operations: mach 0.75 - 0.85, 35 - 40,000 ft. cruise altitude (reference 1)

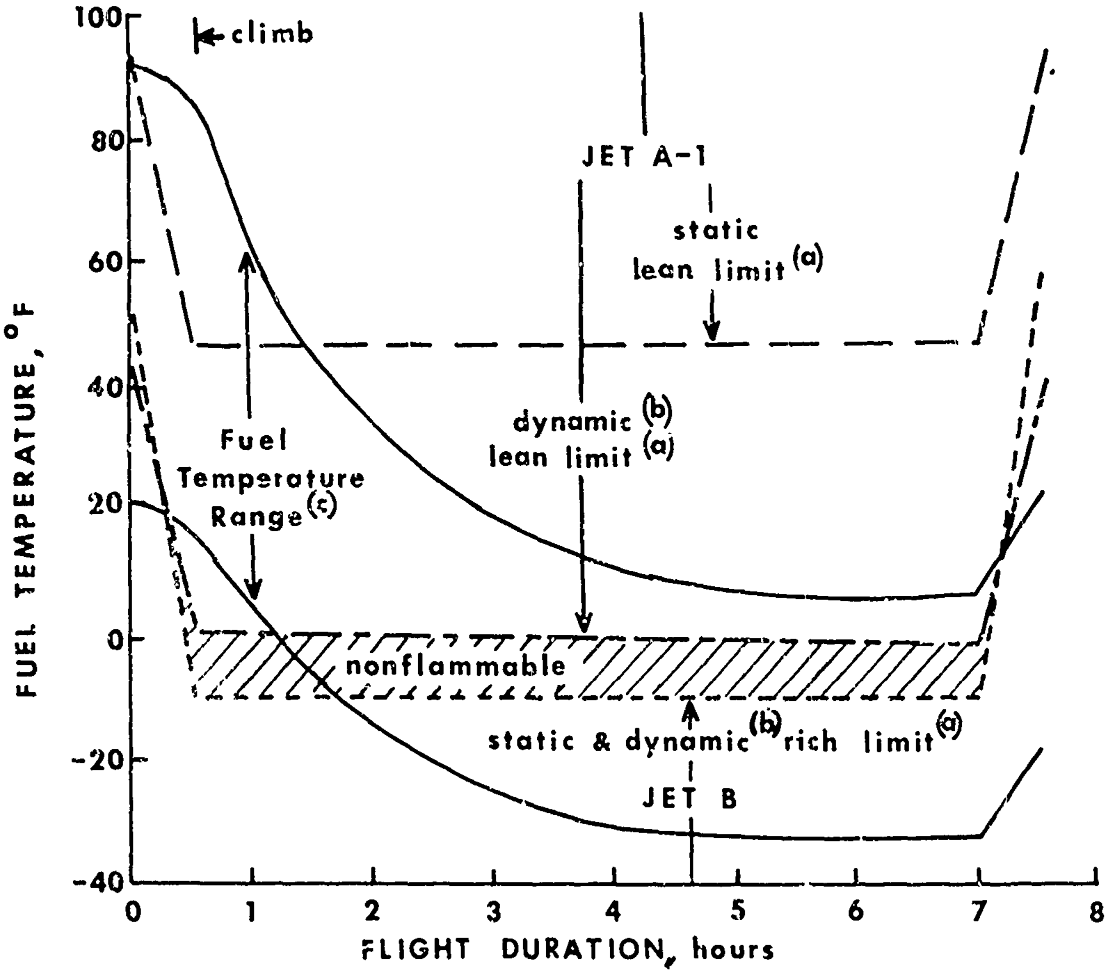


Figure 16. Estimated In-Flight Fuel Temperature Compared to Flammability Envelopes

of combustion experiments was performed under static conditions to determine the limits of flammability of mixtures of Jet B and Jet A. This test simulated the practice of "topping" off an aircraft fuel tank with a different fuel from the one already there. It was found that the rich and lean limits of Jet A will both move to lower temperatures as Jet B is added. The variation of the limits at sea level with relative concentration of Jet A and Jet B is shown in Figure 17. The relationship is nonlinear, with the higher vapor pressure material (Jet B) having the greater effect on the properties of the mixture. It was estimated that a mixture of 85% Jet A and 15% Jet B would have a flammable range straddling the nonflammable region that normally separates the two fuels at equilibrium. This result is similar to that produced by the dynamic conditions with kerosene, and the effect on the probability of a flammable condition existing during a typical flight would also be similar.

## 7. STATIC ELECTRICITY GENERATED BY DYNAMIC CONDITIONS

The high levels of turbulence and fuel sloshing during a dynamic condition, such as in a thunderstorm environment, might raise a question as to the possible ignition of a flammable mixture by discharge of accumulated static electricity. Although the subject was not formally included in this program, a tank used for preliminary studies of fuel dynamics was instrumented to measure the potential generated by sloshing and vibration. Figure 18 is a schematic of the apparatus used for this test. The tank was of transparent plastic with an aluminum frame and was insulated from the rocking and vibration table with a resistance to ground of $10^{10}$ ohms. The accumulation of static electricity in the tank wall was measured by an electrometer with an input resistance of $10^{14}$ ohms. Measurements were made at rocking rates up to 25 cycles per minute and vibration from 0 to 15 cycles per second with an amplitude of plus and minus 1/8 in.

The tank wall potential was found to increase with time until a maximum value was reached, after which it leveled off. Within the range of variables tested, vibration had the greatest effect on generation of static electricity. The potential increased with increases in both vibration and rocking frequencies. The data for the most severe condition of 25 cpm rocking and 15 cps vibration are summarized in Figure 19. The maximum potential of 45 volts was the highest achieved in all

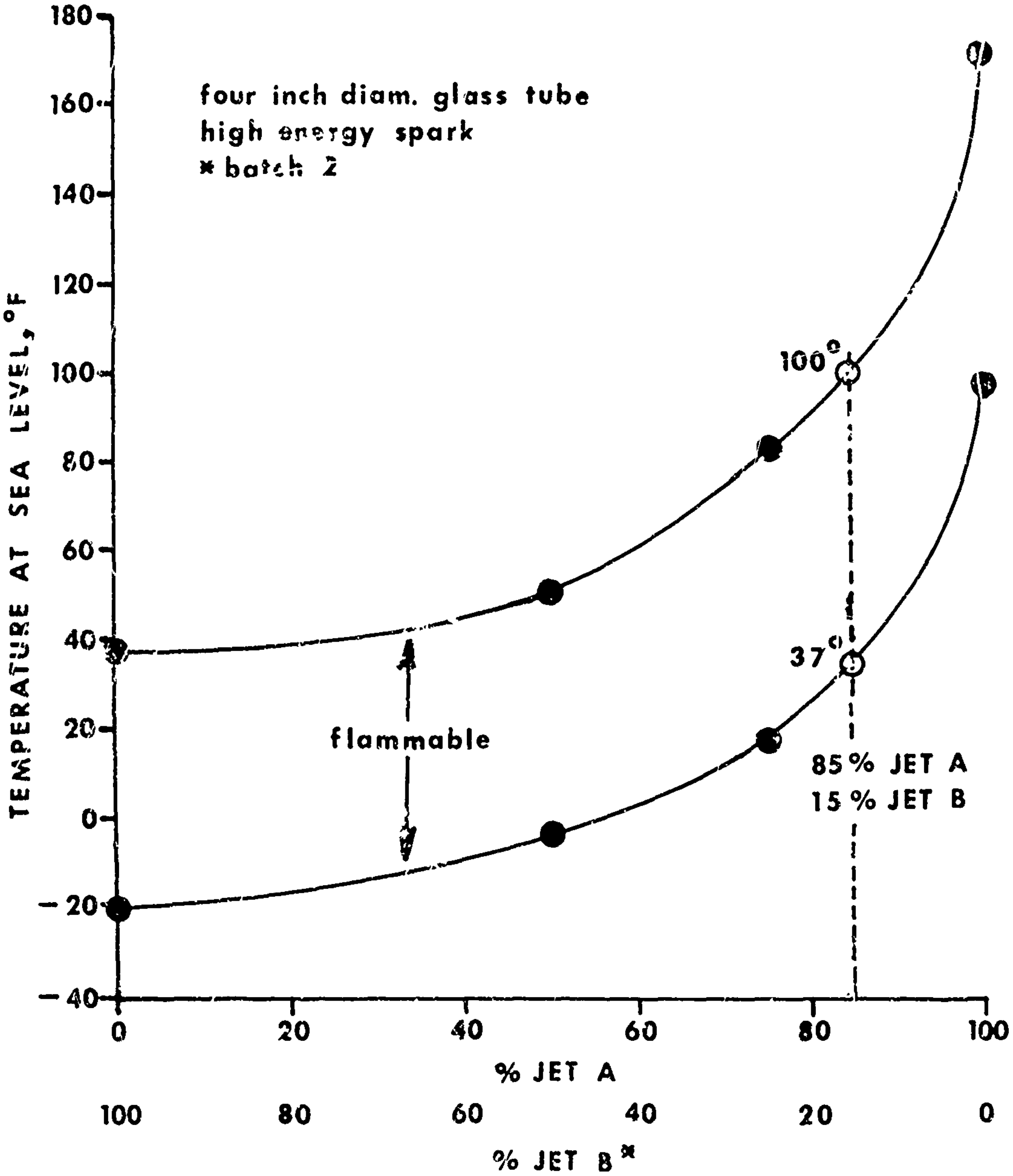


Figure 17. Variation of the Flammability Limits at Sea Level for Blends of Jet A and Jet B

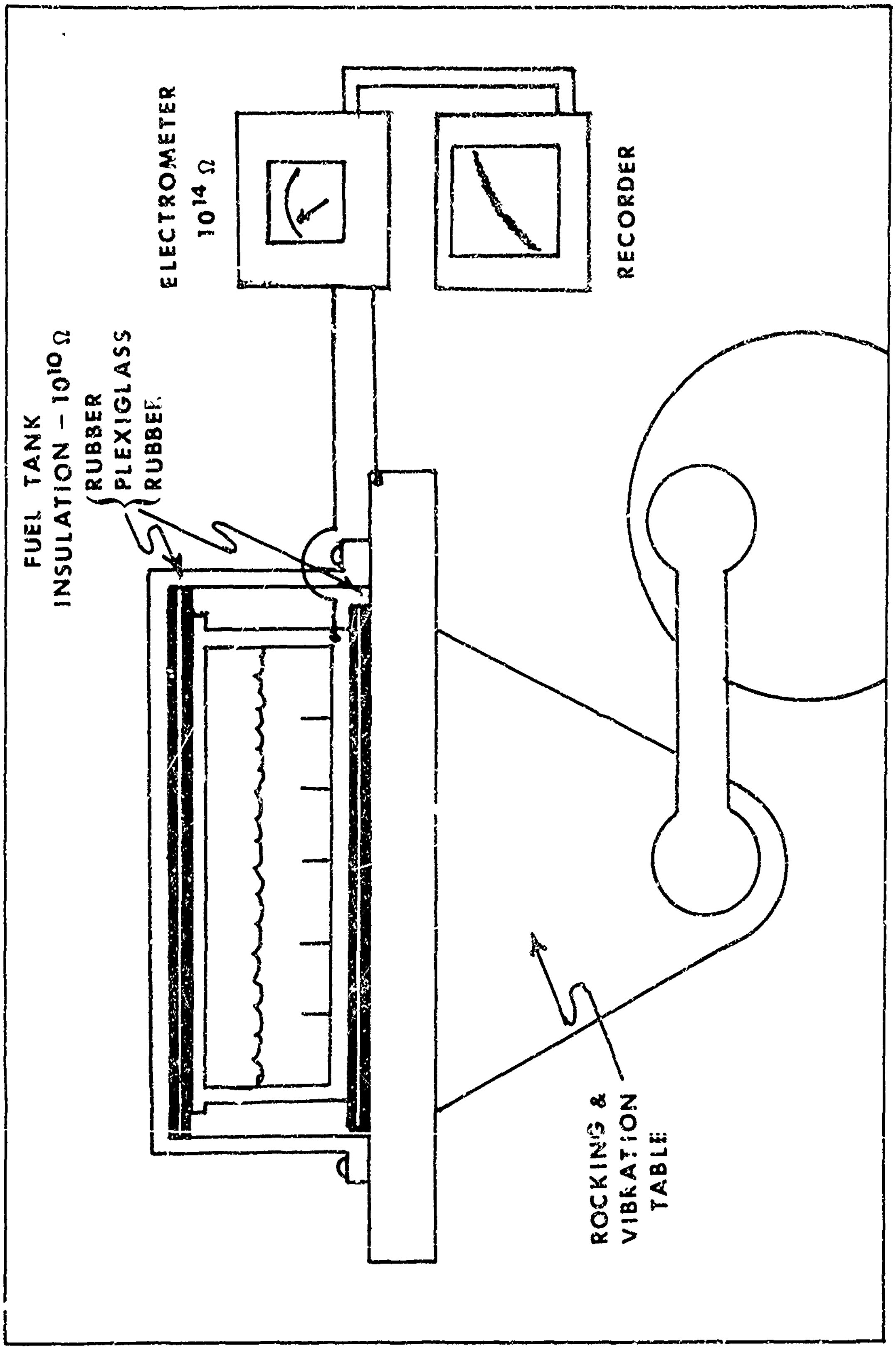


Figure 18. Fuel Slosh and Vibration Test Apparatus

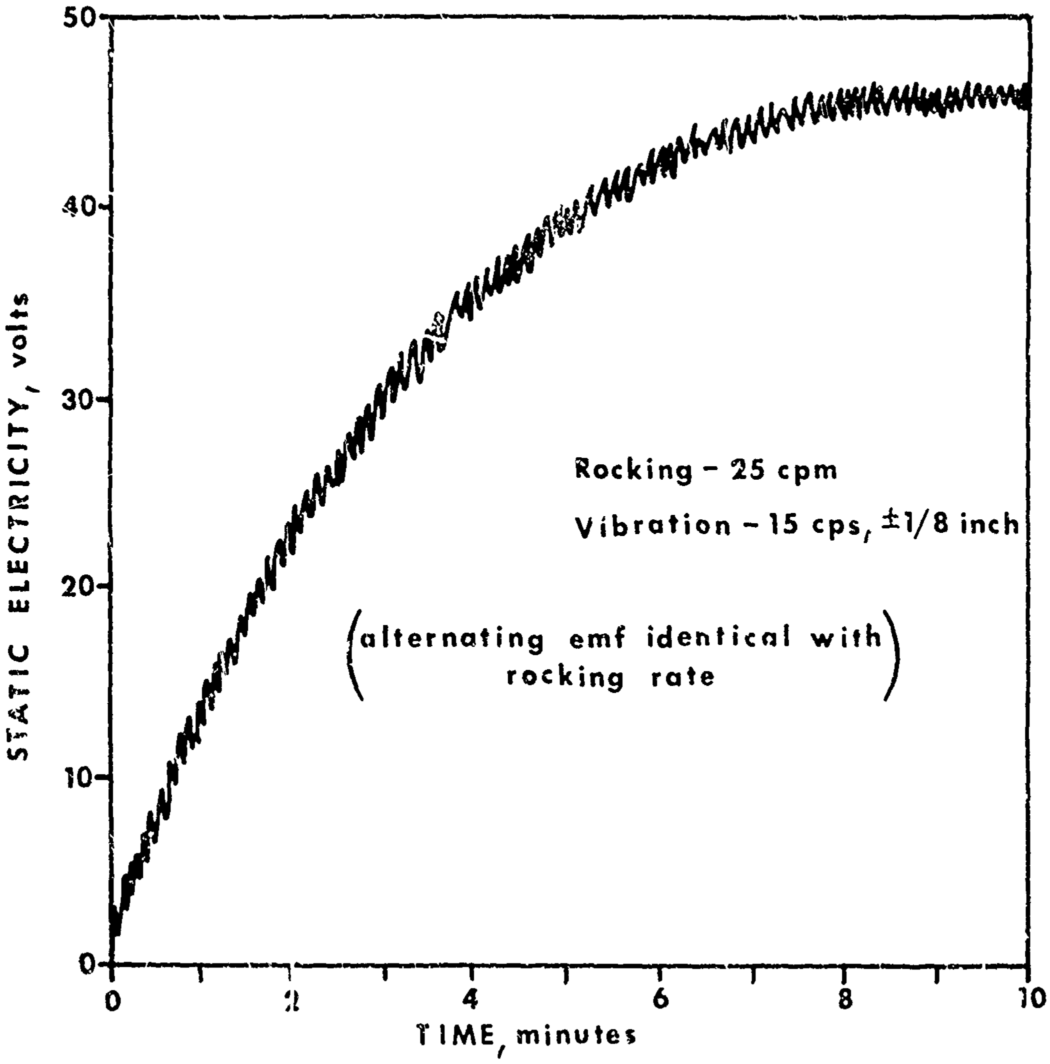


Figure 19. Static Electricity Accumulation in a Rocking and Vibrating Fuel Tank

the tests. This voltage is low compared to the minimum of approximately 350 volts required to produce a static discharge spark (Reference 5). These data indicate that the internal generation of static electricity by rocking and vibration is not a serious hazard as a source of fuel ignition.

## CONCLUSIONS

The equilibrium flammability envelope of the aviation kerosene, Jet A-1, is different from that of the wide-cut turbine engine fuel, Jet B. The net effect is a nonflammable region that exists between the two envelopes. When tank turbulence is sufficient to produce sprayed fuel within the tank and a source of ignition is within the spray, the lean limit of the Jet A-1 will be extended. The extension of the lean limit under these circumstances in this program was so great that the equilibrium, nonflammable region between Jet A-1 and Jet B, was eliminated. When the point of ignition was external to the spray, the combustibility characteristics were identical to those produced under static equilibrium conditions.

Mixing of JP-4 and kerosene in a tank can produce a flammable mixture under conditions where neither fuel alone would be flammable.

## ACKNOWLEDGMENT

This project was supported by the FAA under Project Number 520-002-04X and Agreement Number FA65NF-AP2.

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AFAL-TR-68-290
PART II

# ELECTROSTATIC INTERFERENCE BY FLUOROCARBON COOLANTS

Robert W. Ellison

Martin Marietta Corporation

Electrostatic failures are experienced in missile and space programs and in ground equipment for such programs more frequently than is generally recognized. This is especially true with high-speed digital equipment. The inability to control electrostatic interference seems to be the result of failure to recognize those factors that identify electrostatic situations. The existence of a problem with an inertial guidance system was identified by the following salient facts:

On a recent flight, the digital guidance system of a spacecraft launch vehicle commanded the propulsion to shutdown slightly before the design value of "velocity to be gained" had been achieved. A subsequent review of the guidance data on a bit-by-bit playback revealed that the guidance computer had committed four identifiable errors at a time when the altitude was 88,000 feet. These errors involved the following:

(1) Multiply operation;

(2) Instruction processing;

(3) Accelerometer signal processing;

(4) Vehicle attitude signal.

On the next flight, an accelerometer count error was generated at 58,000 feet; this error took 10.5 seconds to correct.

That the problem was an electrostatic phenomenon was deduced from flight data. No transient detectors tripped, even though the computer had generated a significant error in the accelerometer account. A comparison of voltage thresholds for the computer with those for the detector showed that the detector would trip at voltages below those that could affect the computer provided that the transient lasted longer than a few microseconds. The detector threshold for submicrosecond transients was, on the other hand, far higher than values that would interfere with the computer. Electrostatic discharges were immediately suspected since submicrosecond transients are usually associated with electrostatics. A review of guidance circuits failed to reveal a credible source of submicrosecond, nonrecurring transients. A thorough examination of all telemetry data during the period of flight showed that no transients had occurred at any of the many circuit points monitored; the telemetry system, however, was not designed to monitor transients of submicrosecond duration. All that the telemetry could reveal was that no long transients, such as might result from arcing or shorting, had been picked up. Accordingly, the investigation was oriented to potential sources of electrostatic discharges even though the data did not fully rule out all other possibilities.

The context of this investigation was that many flights had been made with no evidence of guidance anomalies; attention was naturally directed toward any differences in the two vehicles in which the anomalies occurred. One difference was the payload fairing: a nonmetallic fairing had been used on all but three flights. Guidance effects were observed on two of the three flights with the metallic fairing. A metallic fairing would be considered safe, but it was covered with an insulating ablative coating. To rule out the coating as a source of electrostatic interference, small samples were tested in a bell jar at simulated altitudes of 60,000 and 100,000 feet. Not only was the resistance and capacity per unit area very high, but the counter electrode for the measurements was found to flashover to the edge of the sample at altitudes above 50,000 feet (Figure 1). (At low altitudes the coating "punched through" electrically.) It was believed that the anomalies could be attributed to the use of these fairings. In separate tests with the guidance system, the computer made errors when subjected to sparks to the ground plane and equipment cases with energy as low as 565-ergs. It was the concensus that the coating of the payload fairing became charged by encounter with atmospheric particles (ice crystals or dust particles) and discharged as altitude increased, since the voltage for flashover decreased with increasing simulated altitude. Nevertheless, it was not readily explained how a flashover at the payload fairing could propagate to the computer, several feet aft and totally enclosed by the missile skin, past other equipment without affecting anything but the computer. Further, the altitude was a bit too high for meteorological charging.

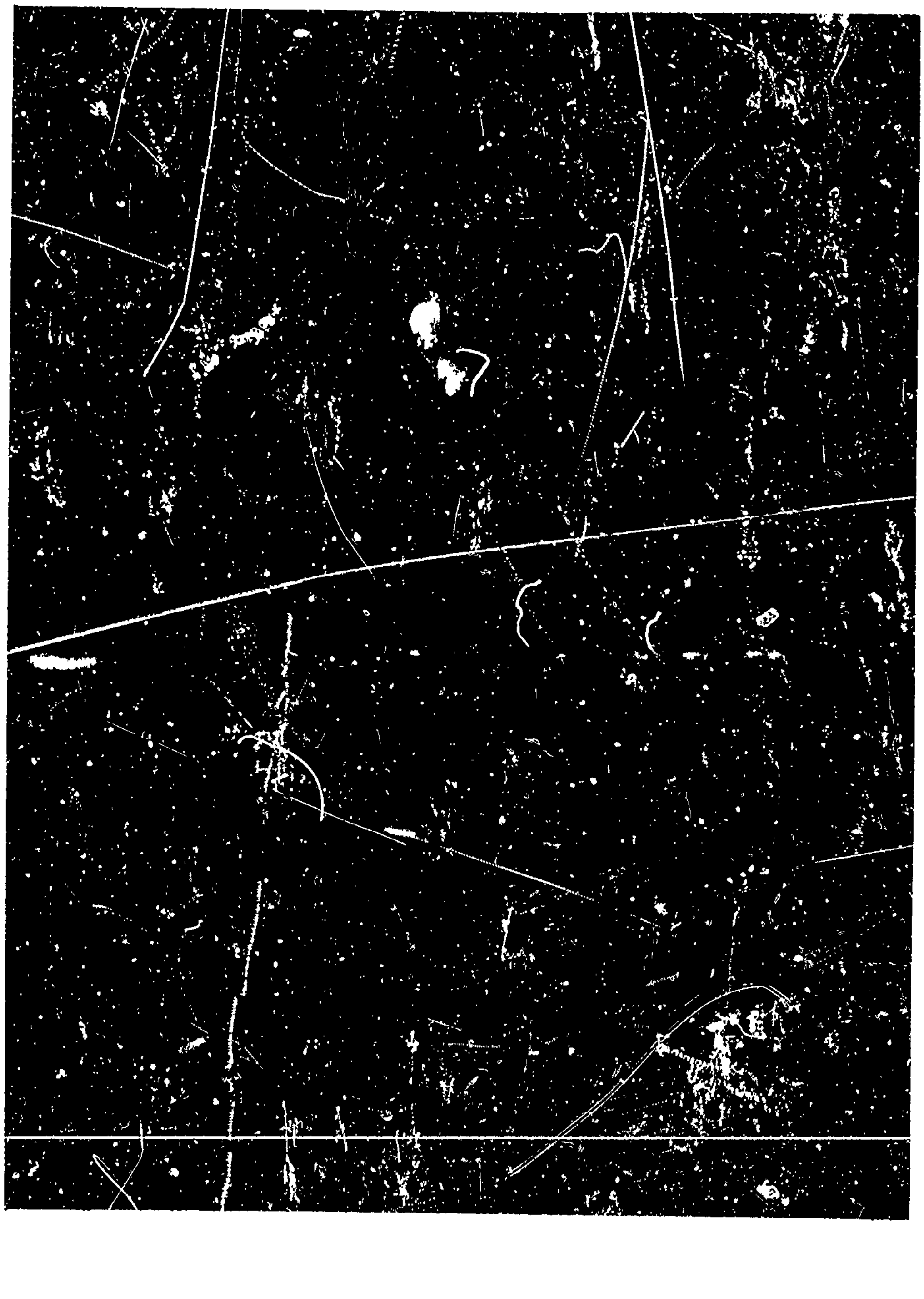

Figure 1. Flashover of a Thermolay-Coated Panel at a Simulated Altitude of 60,000 Feet

A series of tests was conducted on large panels of the payload fairing to gain further insight into this charging mechanism. These tests were conducted at the temperatures and altitudes of the two flights at the time the anomalies were recorded. The results clearly discounted that any significant charge could have existed on the payload fairing coating on either flight at the time of the anomaly; the resistance of the coating at the existing temperatures was too low for even the most extreme meteorological current to generate a voltage that would be within an order of magnitude of the voltage necessary for flashover. Figures 2, 3, and 4 support this conclusion. Some very interesting conclusions regarding the nature of electrical conduction on Thermolay-Lacquer at altitude are given in Figure 5.

With these results, it was necessary to look for another source of electrostatic discharge. The fact that only the computer was affected indicated that sources physically or, at least, electrically close to the computer should be investigated first.

Now, this guidance computer is provided with a force-circulated liquid coolant. Luckily, at the very time that test results on the panels were discounting the hypothesis of meteorological charging, data on the coolant liquid appeared. The resistivity of this liquid was not available in current vendor data, but was found in the preliminary data. This resistivity was found to be the astonishing high value of $10^{14}$ to $10^{16}$ ohm cm. The first test on electrostatics of liquids described a very simple setup for measuring the tendency of liquids to charge (Figure 6), which was performed on the coolant used for the guidance system.

Now, this test looks very simple, but after many attempts to realize a measurable voltage, we found that considerable insight is required to get any results at all. By using a liquid for which data on charging was already available, we found that the apparatus worked. If the liquid was stored for a time in the upper container, the voltages and flow times were different than if flow were started right after the upper container was filled. Sometimes the efflux was attended by obvious vortex formation, and occasionally no vortex developed. In either case, the charging tendency of the coolant was notably greater than published values for liquids noted for their high charging tendency (Figure 7).

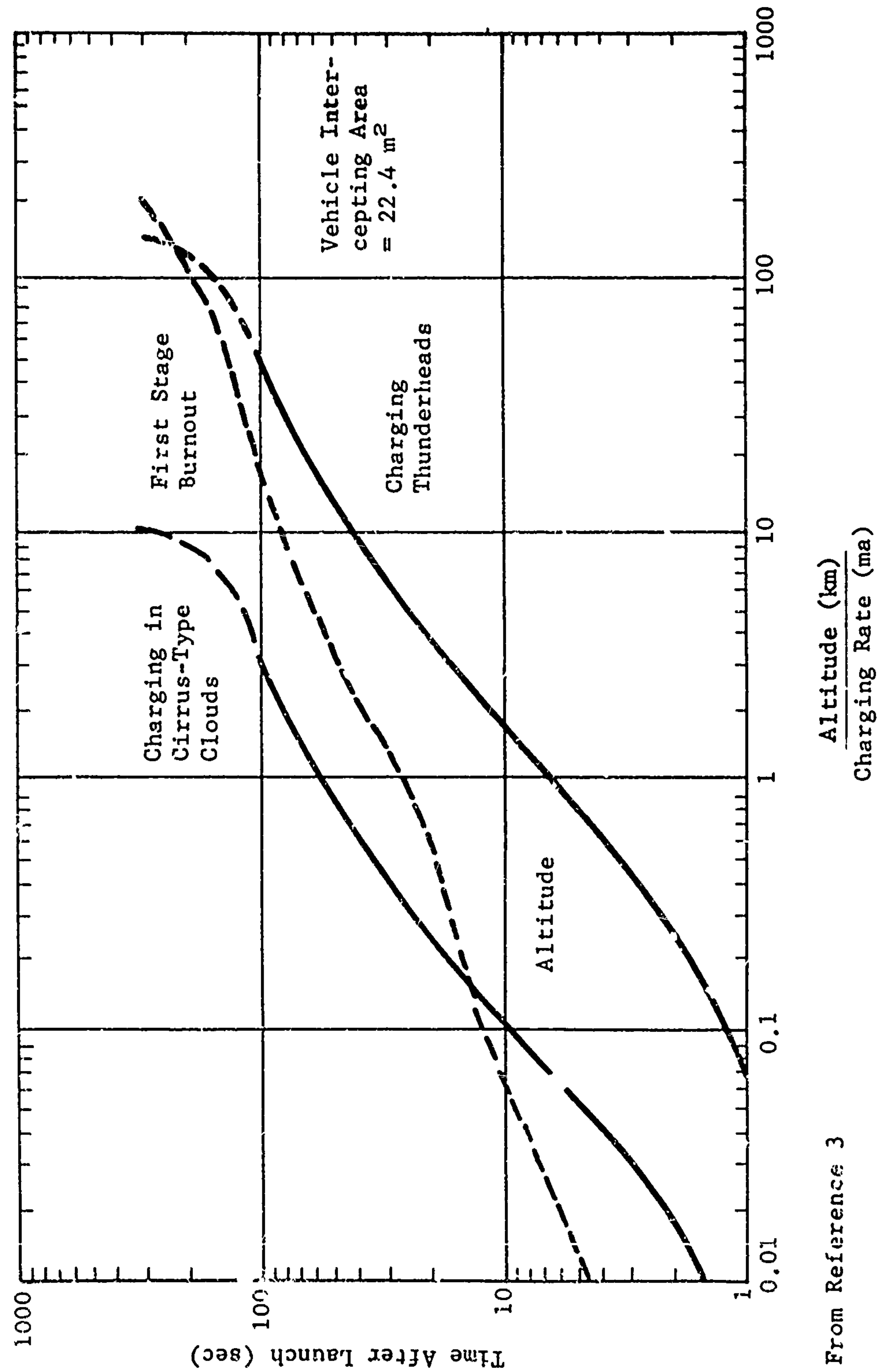


Figure 2. Maximum Charging Rate for the Saturn-Appolo Vehicle in Cirrus Clouds and in Thunderheads

Case 1

| | |
|---|---|
| Time of Anomaly: | $T_o + 71.3 = C_o + 74.7_s$ |
| Altitude: | 58.894 ft = 18 km |
| Maximum Possible Temperature at Sta +23: | 153°F |
| Minimum Possible Temperature at Sta +23: | 125°F |
| DAC Predicted Temperature: | 90°F |
| Maximum Possible Charging Current at 18 km: | $2 \times 10^{-9}$ a/cm$^2$ |
| *Absolute Upper Bound of Charging Current: | $6 \times 10^{-9}$ a/cm$^2$ |

Painted electrode test data combine as follows:

| Panel | Electrode (in.) | Altitude (ft) | Test Temperature (°F) | Resistance (meg) | Worst Case (volts) | *Abs. Upper Bound (volts) |
|---|---|---|---|---|---|---|
| 501 | 4 | 60,000 | 136 | 760 | 40 | 160 |
| 501 | [illegible] | 60,000 | 136 | 80 | 39 | 156 |
| 1 | 4 | 60,000 | 153 | 280 | 15 | 60 |
| 1 | 12 | 60.000 | 146 | 120 | 58 | 232 |
| 501 | 20 | 60,000 | 142 | 60 | 90 | 360 |

Aluminum plate test data were discounted for two reasons:

1) Surface of lacquer contacts a small fraction of lacquer area covered;

2) Temperatures measured on back, heated side are grossly reduced by the electrode heat sink.

*Data added 12/14/67
See Reference 9 for further details.

Figure 3. Comparison of Flight Results With Panel Test Results, Case 1

Case 2

Time of Anomaly: $T_o + 88 = C_o + 91.4_s$

Altitude: 84,960 ft = 25.5 km

Maximum Possible Temperature at Sta +23: 180°F

Minimum Possible Temperature at Sta +23: 165°F

DAC Predicted Temperature, 150°F

Maximum Possible Charging Current: $3 \times 10^{-9}$ a/cm$^2$

*Absolute Upper Bound of Charging Current: $10 \times 10^{-9}$ a/cm$^2$

Painted electrode test data combine as follows:

| Panel | Electrode (in.) | Altitude (ft) | Test Temperature (°F) | Resistance (meg) | Worst Case (Volts) | *Abs. Upper Bound (volts) |
|---|---|---|---|---|---|---|
| 501 | 4 | 100,000 | 200 | 3.6 | < 1. | < 3 |
| 501 | 12 | 100,000 | 180 | 1.2 | 0.87 | < 3 |
| 1 | 4 | 100,000 | 185 | 24 | 2. | < 6 |
| 501 | 20 | 100,000 | 166 | 0.64 | 1.28 | < 5 |
| 1 | 12 | 100,000 | 154 | 40 | 29. | <100 |

*Data added 12/14/67
See Reference 9 for further details.

Figure 4. Comparison of Flight Results With Panel Test Results. Case 2

Panel 501, 70 F

The resistance measured at 500v DC is highly dependent on altitude for any set of electrodes

| $\frac{R_{100,000}}{R_{60,000}}$ | Electrode Diameter (in.) | Edge Dist |
|---|---|---|
| 14 | 20 | 2 |
| 6.7 | 12 | 6 |
| 5.9 | 4 | 10 |

The resistance computer from bleed-off time at 200v DC is independent of altitude for given electrodes

| $\frac{R'_{100,000}}{R'_{60,000}}$ | Electrode Diameter (in.) | Edge Dist |
|---|---|---|
| 1.18 | 20 | 2 |
| 1.00 | 12 | 6 |
| 1.00 | 4 | 10 |

The ratio of resistance computed from bleed-off time to resistance measured at 500v DC is dependent on the altitude but independent of the electrode

| | $\frac{R'}{R}$ | Electrode Diameter (in.) |
|---|---|---|
| 60,000 | > 33 | 4 |
| | 37 | 12 |
| | 44 | 20 |
| 80,000 | 6.9 | 12 |
| 100,000 | > 5.7 | 4 |
| | 5.5 | 12 |
| | 3.7 | 20 |

I propose that conduction at 200 v DC and less is surface conduction in the solid coating, and if this is so, the resistances should be proportional to the electrode diameter. Now the electrodes are semicircularly edged so that the effective diameters at the contact with the coating are less than the nominal diameter by approximately one inch.

| Nominal Diameter | Effective Diameter | Ratios of Diameters | Apparent Resistance | Ratios of Apparent Resistances |
|---|---|---|---|---|
| 4 | 3 | > 3.66 | $> 4 \times 10^{12}$ | > 3.63 |
| 12 | 11 | > 6.35 | $1.1 \times 10^{12}$ | $> \frac{6.45}{5.4}$ |
| 20 | 19 | > 1.72 | $\frac{0.74}{0.62} \times 10^{12}$ | $> \left\{\frac{1.77}{1.49}\right\}$ |

The quotients of ratios of apparent resistance divided by ratios of the log of effective diameters are also significant when surface conduction from an electrode is not infinitely removed from the reference boundary.

$$\frac{\frac{1.77}{1.49}}{\text{Log } 19/11} = \frac{7.5}{6.35} \sim 6.9 \qquad \frac{3.63}{\text{Log } 11/3} = 6.45 \qquad 6.5$$

$$\frac{\frac{6.45}{5.4}}{\text{Log } 19/3} = \frac{8.04}{6.7} \qquad 7.4$$

Figure 5. The nature of Electrical Conduction on Thermolay-Lacquer Panels at Altitude

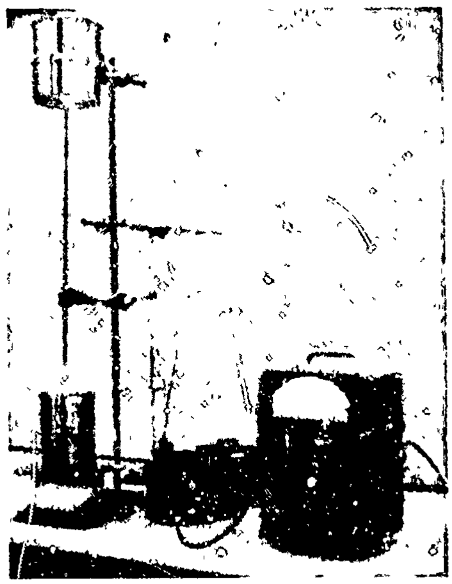

Figure 6. Test Apparatus for Charging Tendancy

If an anodized fitting or a copper braid was added to the lower end of the grounded container and tube assembly, the voltage of the collecting container was instantly almost sufficient to drive the electrometer off scale. The charging property is high, at any rate, and greatly affected by anodized or other surfaces, by turbulence, and presumably by many other parameters. Duplicate tests showed results to be highly variable.

The next step, and a very interesting one, was to measure the voltages, if any, to which flowing coolant might charge an ungrounded hose assembly. Details of the arrangement are shown in Figure 8. Again, electrostatic testing involved some unanticipated subtleties. The charging current was determined to be about 25 picoamperes as follows:

The monitored voltage was increasing at the rate of 30 volts per minute, and the capacity was less than 20 picofarads.

$$\frac{30 \text{ v/min}}{60 \text{ sec/min}} \times 20 \times 10^{-12} \text{ f} = 10 \times 10^{-12} \text{ amp or 10 picoamp}$$

Obviously, the maximum voltage is 300 volts ($10 \times 10^{-12}$ amperes $\times 30 \times 10^{12}$ ohms). The 3000-volt instrument went off scale at 106 minutes, as shown in Figure 9.

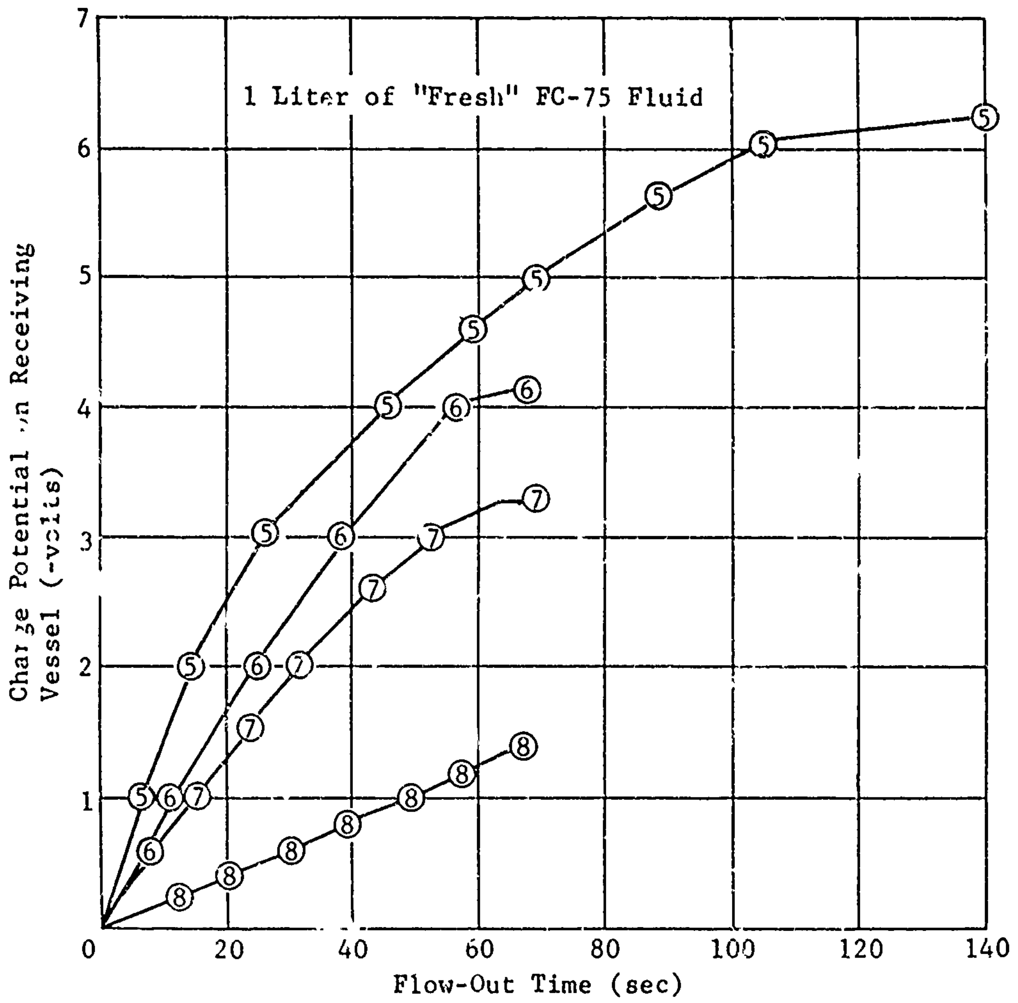


Figure 7. Charging Potential Versus Flow-Out Time for Test Runs 5 Through 8

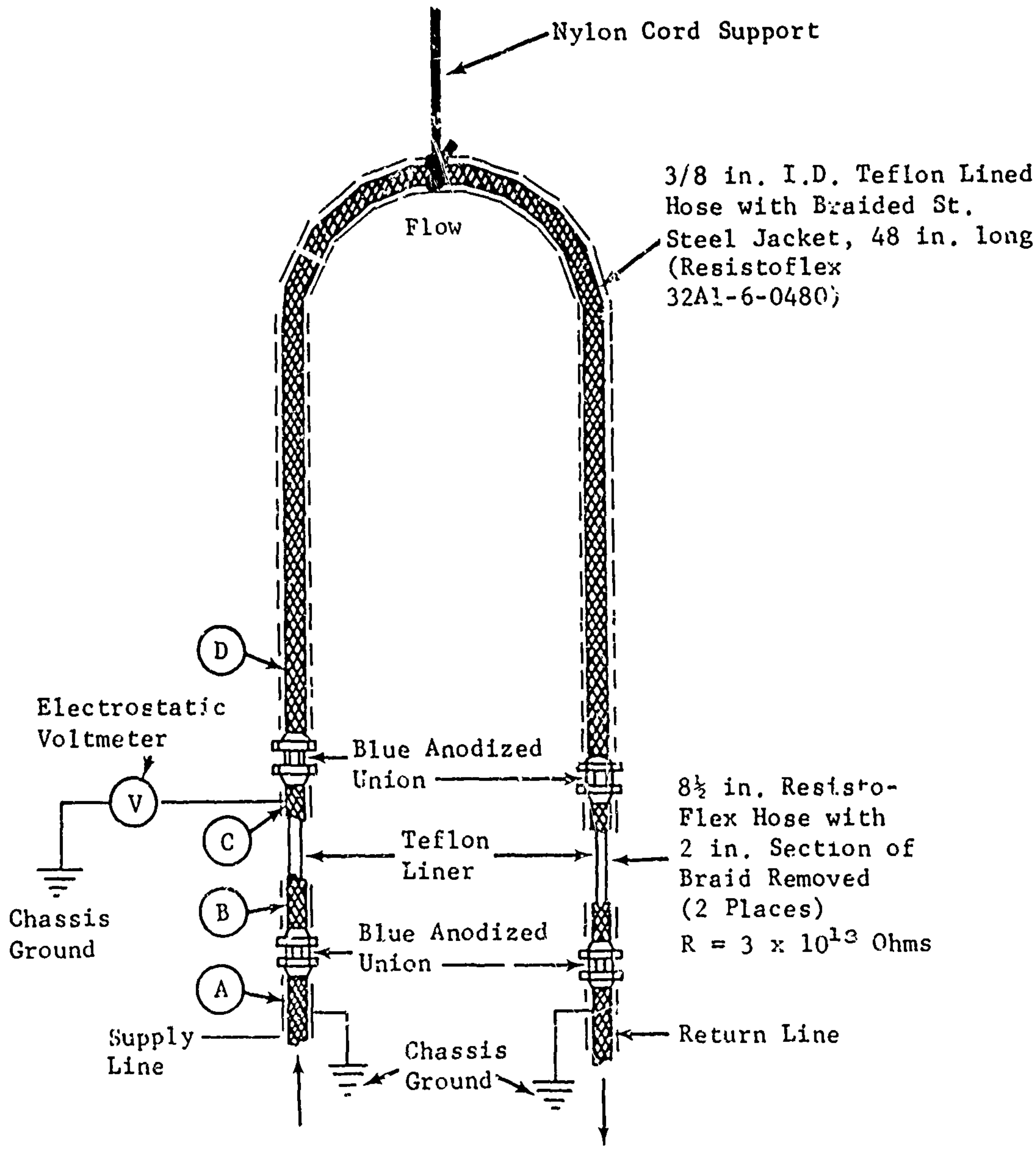


Figure 8. Arrangement for Measuring Charge Potential Under Continuous Flow Conditions

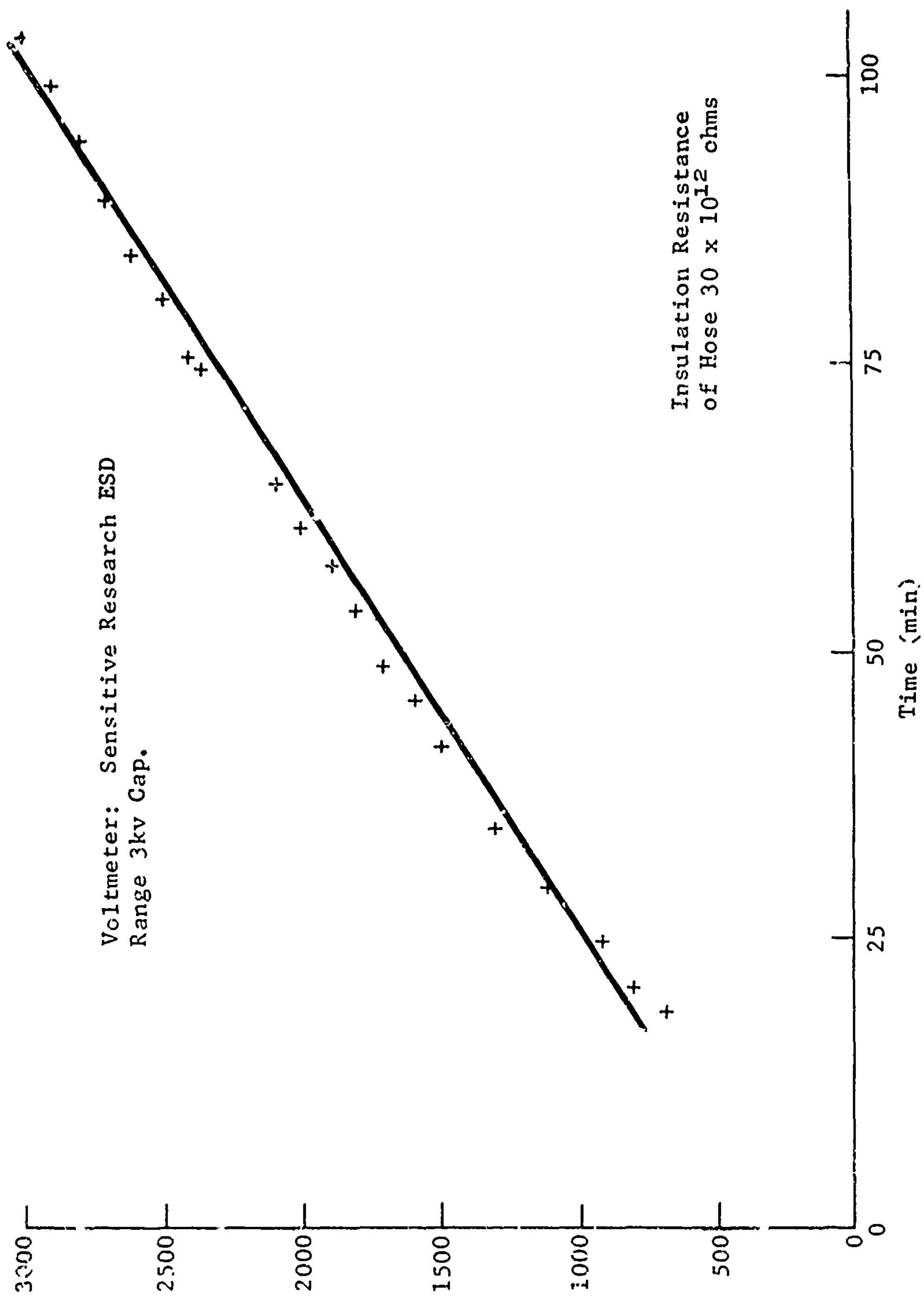


Figure 9. Voltage Generated by FC-75 Coolant Flowing in Insulated Hose

The graph shows that the current is absolutely constant; the very slight curvature is that produced by the increasing capacitance of the electrostatic voltmeter as it is deflected upscale. The obvious conclusion is that you cannot charge 3 x $10^{13}$ ohms above 300 volts no matter how long the fluid is pumped. But the voltmeter did read 3000 volts and the resistance did again measure less than 3 x $10^{13}$ ohms after the test. If we had taken data for, say, 30 minutes and combined it with the resistance reading, we would know that 300 volts was the asymptote for charging continuously.

Ohms law had failed. The only way we can reasonably explain the 3000 volt result is that the resistance must have increased as the voltage increased. Now, many materials do not exhibit a resistivity which is independent of the voltage applied. Most such nonohmic materials exhibit a decrease in resistance as voltage increases, but this material seems to have a resistance that increases with voltage. Is this possible?

> "In 1909, G. Jaffé showed that the conductivity of chemically pure hexane is derived from ions formed by external radiation such as radioactivity. . . . . . The conductivity of pure hydrocarbons does not follow Ohm's law, since upon increase of the field strength a saturation current is approached; the remaining conductivity is estimated to be of the order of $10^{-17}\Omega$ - $M^{-1}$ at a field strength of 20 KV/M." Klingenberg and Van der Minne.

We recall from the beginning of this paper that the particular coolant involved was reported to have $10^{14}$ to $10^{16}$ ohm cm; the upper value, while three orders below the saturation current at 20 KV/M of pure hexane, is conceivably the result of current saturation; in our experiment the gradient of 3000 volts, still increasing linearly with time, corresponded to 40 KV/M, inasmuch as the gap was about 3 inches.

The latest ground tests on the vehicle have shown that many instrumentation effects are created by sparks external to the computer without affecting the computer; levels sufficient to affect the computer create drastic effects to the instrumentation. We take these results to indicate that the sparks which could affect the computer without affecting the instrumentation would have to be internal. Recently tests with prototype and test hardware have verified the hypothesis that FC-75 coolants do indeed charge Teflon hoses, and spark-through holes in Teflon hoses have been located and photographed. For the particular hoses used, a voltage above 20,000 volts was found necessary to puncture radially through the Teflon to the outer braid; when the braid was intentionally insulated from all grounds, the highest voltage measured in such a test exceeded 20,000 volts.

Our purpose in relating this to you is to bring to your attention that electrostatic phenomena are important in space and missile programs, and to describe some of the pitfalls that can occur from a lack of awareness and only a superficial appreciation of electrostatic phenomena. It is hoped that this paper will assist in establishing design criteria to amend specifications so that electrostatic phenomena are covered in design and so that electrostatic behavior will be covered in EMC testing. The accidents and hazards of electrostatic nature that have involved ordnance, propellants, fuels, dust shrouds of plastic, and plastic ties should be sufficient justification for amending EMC specifications; this report, we hope, will further encourage definitive action.

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AFAL-TR-68-290
PART II

# COMPARISON OF CALCULATED AND MEASURED LIGHTNING EFFECTS ON SPACE VEHICLE LAUNCH COMPLEXES

E. R. Uhlig

General Electric Company

(Presented by S. A. Fisher)

## INTRODUCTION

When NASA's Launch Complex 39 was being designed, it was recognized that lightning presented a potentially serious hazard to the safety of the launch facilities, the launch vehicles, and supporting personnel. During the years 1963-1965 the General Electric Company's High Voltage Laboratory was employed on a consultant basis to help analyze the nature of the lightning problem and to assist in providing means of controlling the effects of lightning. The purpose of this paper is to give a brief overview of some of the problems encountered on NASA's Launch Complexes 34, 37, and 39 and to outline the means that were recommended to control the effects of lightning.

## THE MAGNITUDE OF THE LIGHTNING PROBLEM

It is well recognized that Florida has a high level of lightning activity. Over the years the United States Weather Bureau has accumulated statistical data on the isokeraunic level, or the number of days during a year in which thunderstorms are observed or thunder is heard. As can be seen from weather bureau isokeraunic maps, the central portion of Florida has the highest isokeraunic level in the United States, with the Cape Kennedy region running right behind.

As is typical elsewhere, thunderstorm activity is greatest during the summer months at Cape Kennedy. Figure 1 shows the variation in isokeraunic level of the Cape Kennedy region through the year, as determined from weather observations at Patrick Air Force Base.

Over the years a good deal of statistical data has been accumulated to show how many times structures of differing heights might be expected to be struck by lightning. This data, shown on Table I, was originally compiled for an average

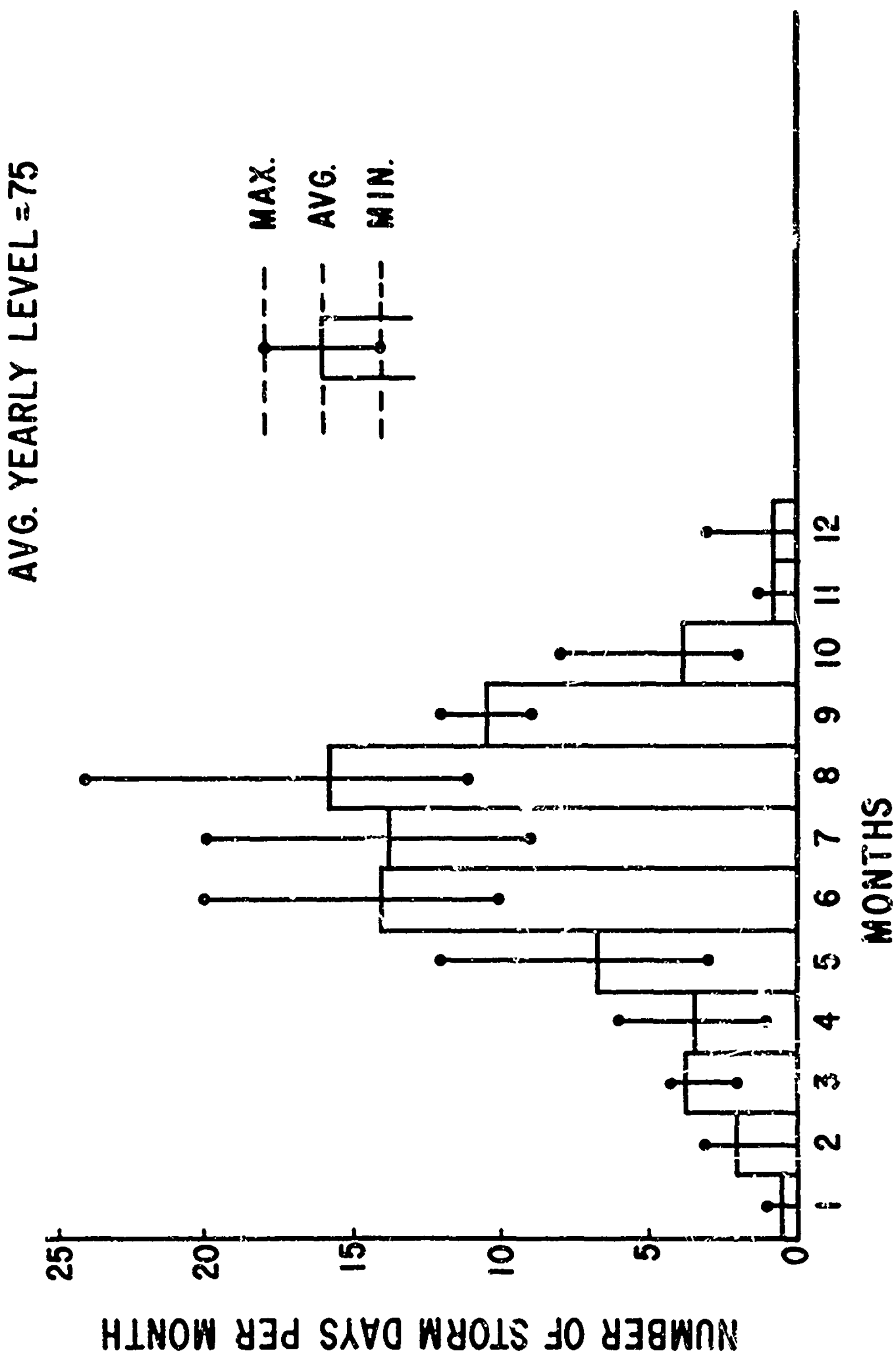


Figure 1. Minimum, Average, and Maximum Monthly Isokeraunic Level at Cape Kennedy, 1957-1962

isokeraunic level of 32, which in the isokeraunic level most representative of northeastern United States. Lightning investigators generally assume that the number of strokes to an object of given height will vary directly with the isokeraunic level. The figures on the right indicate that at an isokeraunic level of 75, which would be typical of that at Cape Kennedy, a structure 400 feet high could be expected to be struck by lightning about 3-1/2 times every year. Heights of this nature are typical of the structures that might be found on a space vehicle launch complex. Table II shows the height of some of the structures in several launch complexes at Cape Kennedy and the expected number of strokes per year. Since these tall structures are good targets lightning strokes to them will by no means be rare occurrences.

Even without regard to the height of the structures, one would expect such large launch complexes to be struck many times during the course of a year. A rough rule of thumb is that one square mile of open land will be struck between one-quarter and one-half times per year for each isokeraunic day. Thus, if the isokeraunic level of an area is 80, each square mile could be expected to be struck by lightning between 20 and 40 times during the course of a year. Many of the strokes would, of course, hit in open unoccupied land and present no hazard, but some of them could be expected to strike areas such as the launch pads or the roadways around the launch complex. For instance, the roadway between the vehicle assembly building and Launch Pad A might be struck between 1.5 and 3 times per year and the 0.2 square mile launch area struck between 4 and 8 times a year, even without considering the tall launching structures on the launch pad. Certainly these incidence rates should be the cause of some concern, particularly when a loaded and fueled launch vehicle could be struck by lightning with disastrous consequences were lightning to ignite the rocket propulsion fuel.

## PROTECTION FROM DIRECT STROKES

Early in the analyses, the conclusion was reached that there was no foolproof method of preventing lightning to the launch complex area and that the only choice was whether to intercept lightning on special shielding structures or on site equipment. Normally, if one wants to shield a structure from direct contact with lightning, he must erect alongside that structure, or preferably over it, another structure that is higher than the structure to be protected. When the

TABLE I

STROKES TO TALL STRUCTURES

| Height (feet) | Number of Strokes per Year from Compiled Data* | Comparable Number of Strokes at Cape Kennedy** |
|---|---|---|
| 100 | 0.15 | 0.4 |
| 200 | 0.45 | 1.1 |
| 300 | 0.90 | 2.3 |
| 400 | 1.40 | 3.5 |
| 500 | 1.75 | 4.4 |
| 600 | 2.10 | 5.3 |
| 700 | 2.30 | 5.8 |

* At an average isokeraunic level of 32.

** At an average isokeraunic level of 75.

TABLE II

EXPECTED STROKES TO TALL STRUCTURES AT CAPE KENNEDY

| | Height (feet) | Expected Number of Strokes per Year |
|---|---|---|
| Launch Complex 39 | | |
| VAB | 525 | 4.5 |
| Umbilical Tower | 450 | 3.8 |
| Launch Complex 34 | | |
| Umbilical Tower | 250 | 1.7 |
| Service Structure | 310 | 2.4 |
| Launch Complex 37 | | |
| Umbilical Tower | 280 | 2.1 |
| Service Structure | 365 | 3.1 |

objects to be protected are something like the mobile launching platform, which is about 450 feet high, one can see that problems would be encountered in erecting alongside it a lightning rod, say, 600 to 800 feet high. Such shielding structures could be made, but they would be very expensive and would be in the way for the normal launch operations. For example, just the scattering of tracking radar beams by any shielding structure would be a nuisance, at least, to launch operations. Some consideration was given to the use of balloon-supported shield wires to protect the launch complex from direct strokes, but it was soon decided that even these would be expensive to maintain and operate and would still be a nuisance to the launch operations on the site.

Actually, the structures on a typical launch complex are massive steel structures that are quite capable of carrying the current in any conceivable lightning stroke. They are, in fact, excellent lightning rods in their own right. Accordingly, it was decided that the most practical method of controlling the effects of lightning would be to use structures like the umbilical tower or the arming tower to intercept any lightning strokes and keep them off the launch vehicle or the ground support equipment. Buildings around the launch complex also had to be protected, but these buildings are no different from any other buildings, so their protection is a routine matter.

The most important problem was what was necessary in the design of the launch structures to ensure that the launch vehicle, itself, would not be struck directly. This reduces to the question of what geometrical relations must be met between a shielding object and an object to be protected. The simplest criteria, and one which meets nearly all practical needs, is the simple cone of protection criteria, as shown in Figure 2. These cones of protection have been substantiated by many years of experience in lightning protection studies. If all points of the protected object lie within a 1:1 cone from the top of the protecting object, then the protected object can be assumed safe from direct lightning strokes. Actually, of course, things are never quite this simple; there is always a small statistical chance of an object being struck even if it is within this 1:1 cone. However, lightning theory indicates that the cone of protection for strokes of the higher current amplitudes is generally greater than 1:1. The degree of shielding is substantial, however, even though the object is not quite within a 1:1 cone of protection. For

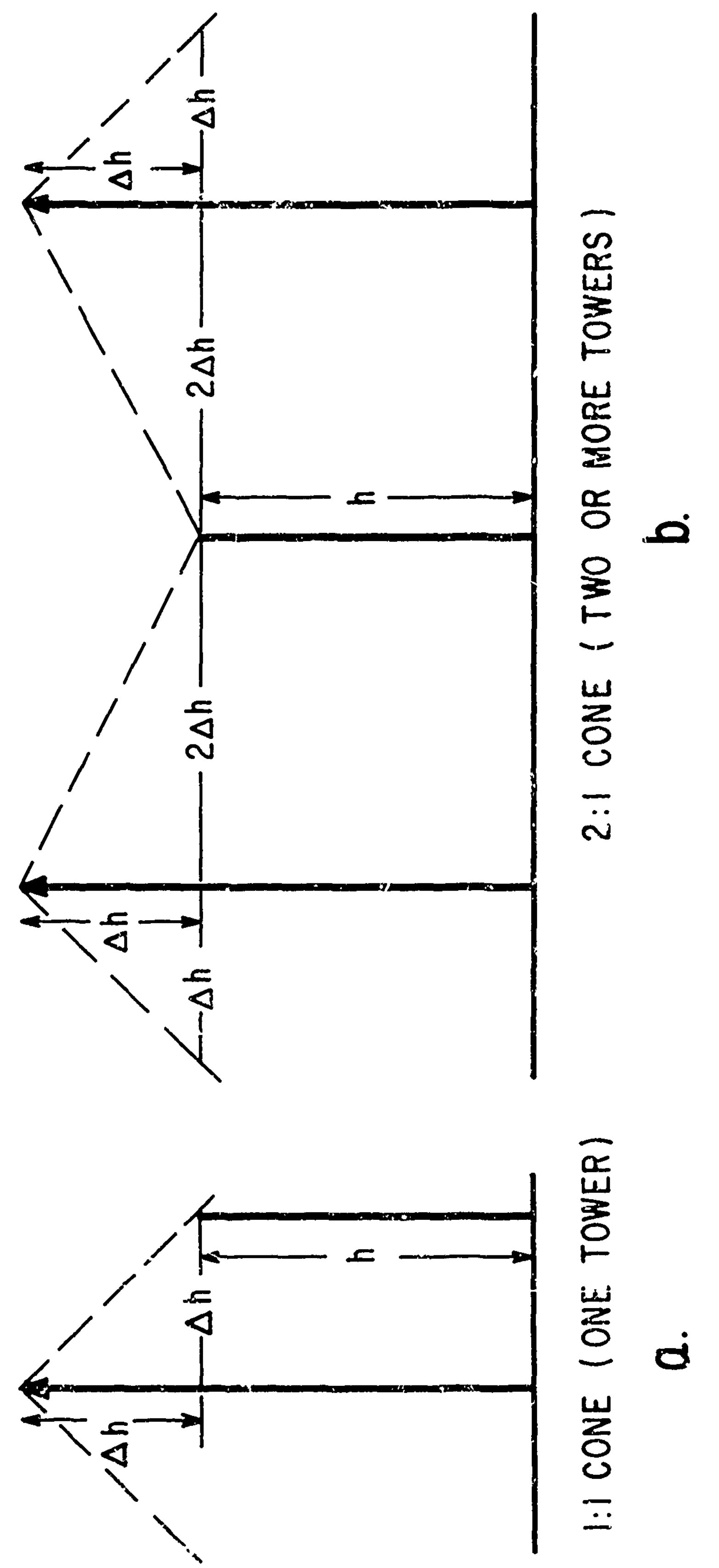


Figure 2. Cone of Protection Criteria

shielding Launch Complex 39, the philosophy was adopted that if the object to be protected were between two or more shielding objects, it would be satisfactorily protected even though it was within only a 2:1 cone of protection.

Figure 3 shows how the cone of protection criterion applies to the Saturn 5 vehicle and its associated umbilical tower. Disregarding the overhead crane, the umbilical tower, by itself, does not provide quite the desired degree of coverage. To remedy this, a retractable mast was added to the tower, which provided protection sufficient that all portions of the vehicle were included in the desired 1:1 cone of protection. Normally, the arming tower is adjacent to the vehicle when the vehicle is being fueled, of course, which provides considerable protection for the vehicle except for a brief period after the fuelling is completed and the arming tower is removed. However, even with the arming tower removed, it is felt that the vehicle is quite safe from any direct lightning strokes, since all strokes to the immediate vicinity should be intercepted by the umbilical tower.

Once lightning strikes the tower, the current must be conducted to ground, where the energy is dissipated. Furthermore, the flow of current through the ground path must not be allowed to produce dangerous differences in potential around the launch pad. To provide low ground resistance and to insure that all points of the complex assume the same ground potential during the stroke conditions, an extensive grounding system has been provided consisting both of ground rods and buried conductors. Figure 4 shows how these ground rods were installed around the launch pad. In the immediate vicinity of the umbilical tower, a number of ground rods were driven under the concrete structures. These ground rods were connected to the reinforcing steel within the buildings and to the pedestals upon which the mobile launcher and the service structure sit when they are in position on the pad. A buried counterpoise was also installed around the periphery of the whole site to provide a grounding system for all of the camera pads. The pipelines and cable trays running to the liquid oxygen facilities, the electrical power substation, and the liquid hydrogen and fuel storage areas were also connected to this ground grid, with ground rods installed along these pipelines and cableways. Any one of the ground rods would be capable of discharging any lightning stroke; however, this wide assortment of grounding

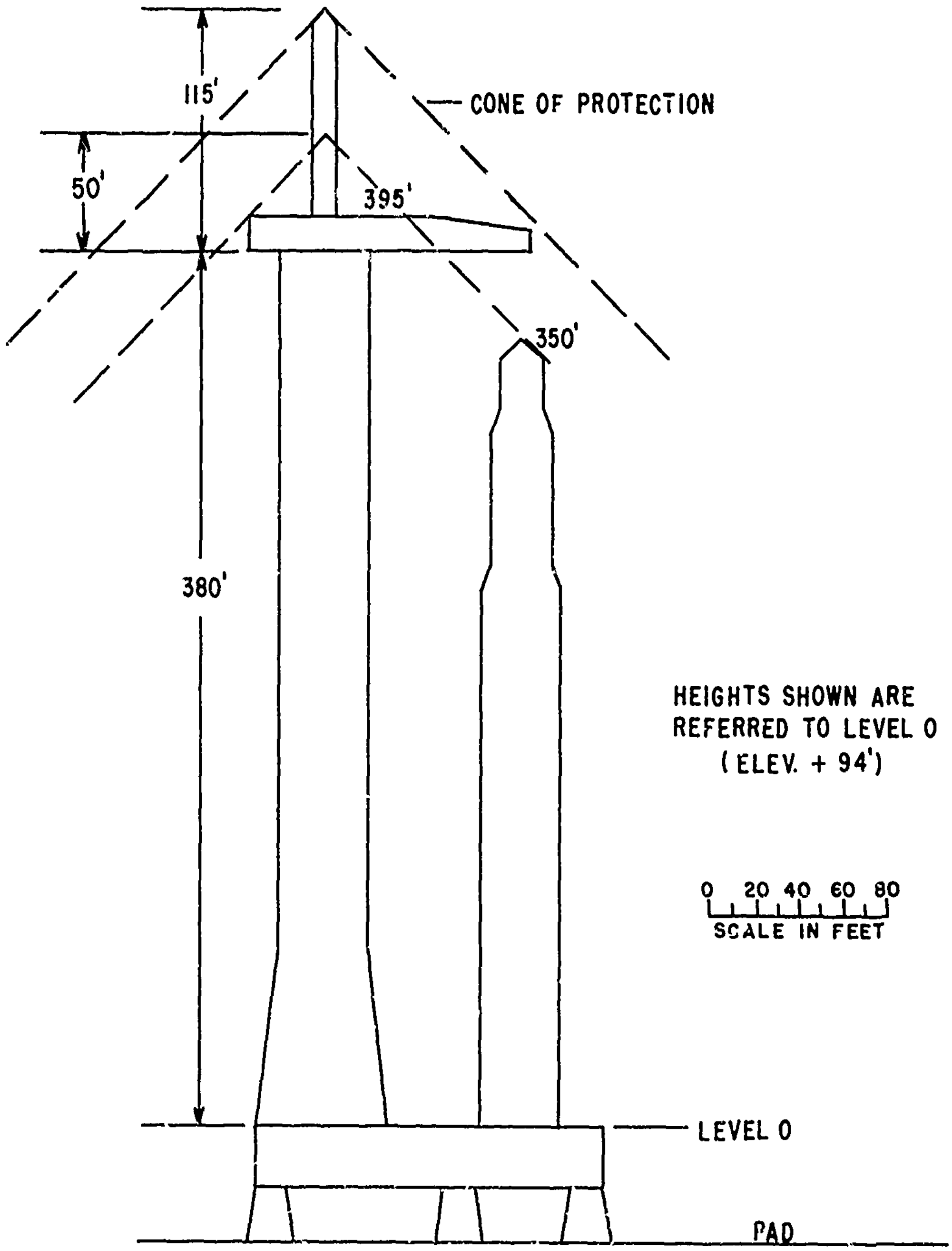


Figure 3. Single Retractable Mast — Elevation View

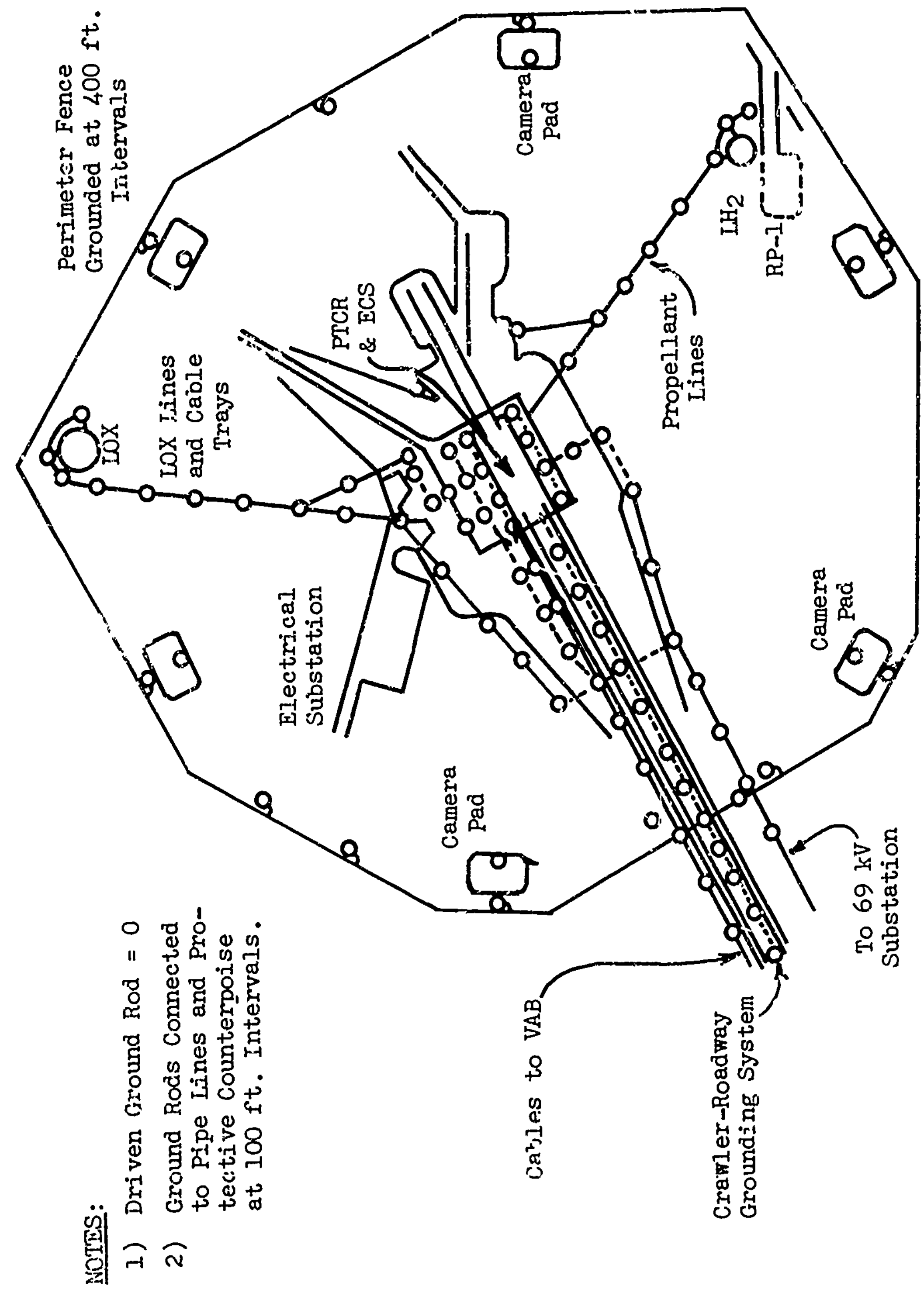


Figure 4. Launch Pad Ground Rod Locations

electrodes was provided to ensure that lightning currents were distributed as broadly and as evenly as possible over the whole area so as to minimize any tendency for a high current flow at one point and create dangerous rises of potential with respect to other points.

These launch complexes have many electrical cable trays running above ground. While the potential damage from a lightning stroke to one of the cable trays would not be as great as a stroke to the vehicle, it could still cause high voltages on the electrical cables and cause extensive damage to the terminal equipment. To prevent a stroke directly to any cableway system, all overhead cable trays were shielded by an overhead shield wire, as shown on Figure 5. The supports for this shield cable were insulated from the supports for the cable trays so that the lightning current could be carried directly to ground. This minimized the inductive voltage drops caused by the flow of current through the inductance of these supporting columns.

Most of the electrical cables around Launch Complex 39 are buried in underground cable ducts. While these cables are protected against any direct lightning strokes, they are frequently subjected to sparking caused by strokes hitting the earth surface nearby. The current tends to flow on the lowest resistance path in the neighborhood before being dissipated and, if the lowest resistance path is a communication cable bank, current may spark through the ground and impinge directly upon conductors in the fibre conduits. To guard against that, a bare counterpoise wire was buried above the cable bank, as shown in Figure 6. This counterpoise wire runs the full length of the cable bank and is connected to low resistance ground rods at about 100 foot intervals. Its purpose is to provide a conductor with lower resistance and an easier entrance by the lightning current than the underground cable bank. Depending on the width of the cable bank, there may be one or two counterpoise wires.

One very interesting problem was this: How do you ground the mobile launcher when it is being moved along the roadway by the crawler? The mobile launcher is as much a target for a direct lightning stroke when being moved as it is when set on the launchpad. A stroke to the launcher presents no hazard to the launcher, itself, or to anyone in the launcher or crawler. Furthermore, the vehicle is not connected to any ground support equipment, so no control of induced voltages

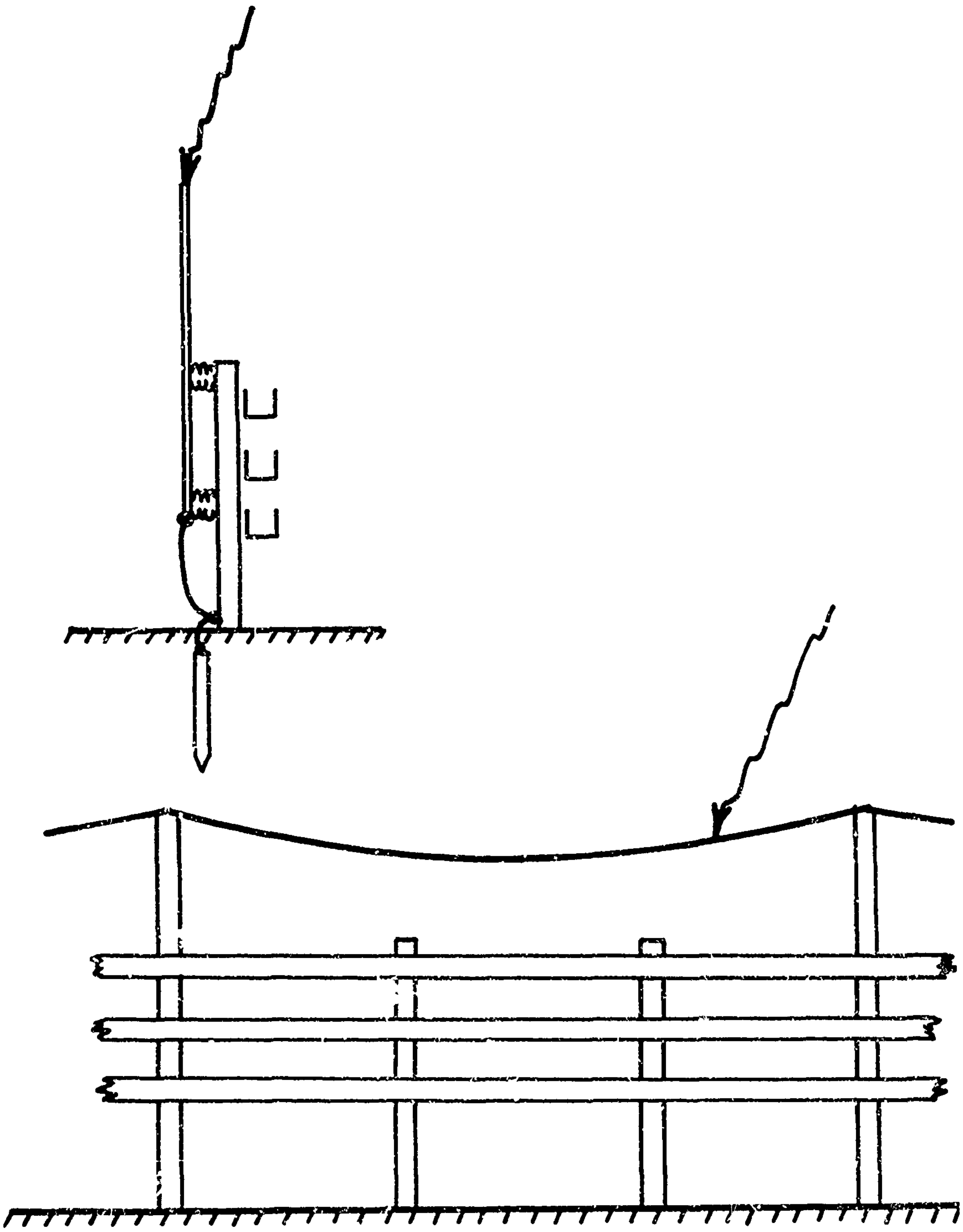

Figure 5. Direct Stroke Protection for Above-Ground Cable Trays

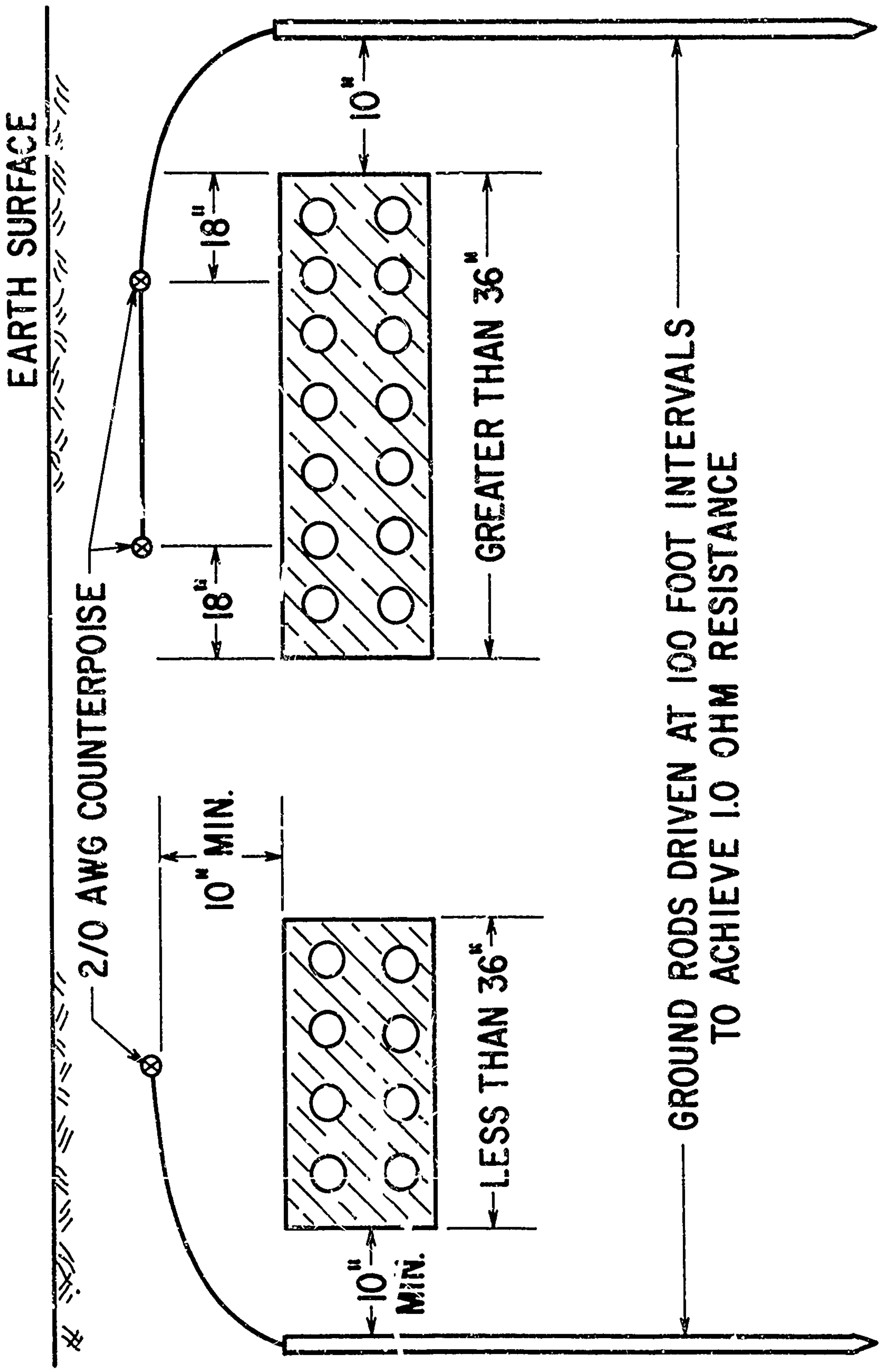


Figure 6. Underground Duct Grounding

requires the launcher to be grounded. The hazard is to anyone on the ground beside the launcher. Many possible ways of grounding the launcher were considered, such as a trolley ground wire or grounded railroad rails with a sliding show; none of these provided grounding sufficient to eliminate all safety hazards, however, and all of them would hinder maneuvering of the crawler. Consequently, it was decided that the crawler could not be grounded sufficiently to make it perfectly safe to people nearby, but to ensure that any side flash from the base of the crawler did not do any harm.

The protection finally adopted was to bury a counterpoise in the center of the roadway about 18 inches below the ground surface, and connect it to ground rods at 100 foot intervals. A large chain is then hocked to the crawler and dragged on the ground surface behind the crawler. This chain is not intended to provide low resistance grounding but to ensure that a stroke that hits the umbilical tower or the crawler is carried to the underground counterpoise and flashes across a path where it will do no harm.

CONTROL OF TRANSIENT VOLTAGES

Control of indirect effects of lightning is as important as protection from direct lightning strokes, particularly on an extensive system employing as much electrical and electronic apparatus as does a typical launch complex. One of the tools used to analyze some of the problems of protecting a launch complex from the effects of lightning was a small-scale geometrical model, as shown in Figure 7. This model has the umbilical tower, the service structure, and the Saturn 2 vehicle of Launch Complex 37, but the results obtained with this model can easily be extrapolated to larger complexes, such as Launch Complex 39 or to smaller complexes.

Pulses were injected into this model from a small pulse generator and the voltage and current distributions in different portions of the model were measured. The model was built to a scale of 100:1; consequently, to preserve the impedance ratios in the model and make them equal to the impedance ratios in the actual prototype, currents injected into the model rose to crest and were maintained for 1/100 of the time of the currents expected to strike the full-scale prototype.

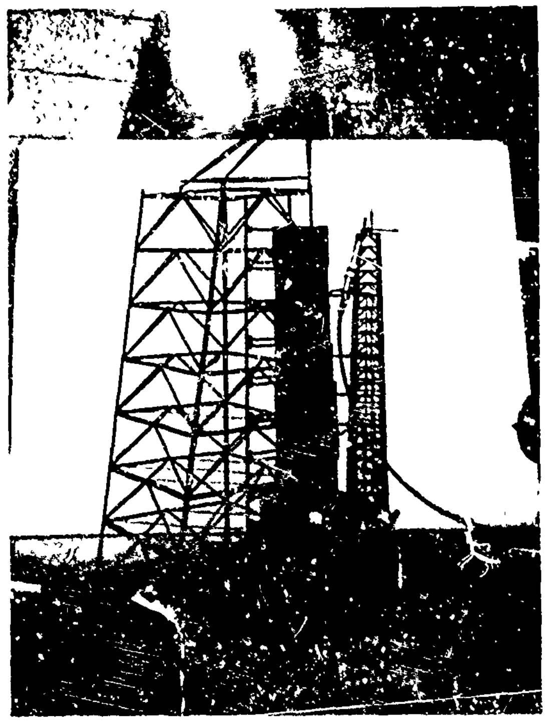

Figure 7. Model With Silo Gates Closed

Figure 8 shows some typical results obtained from this model study pertaining to a lightning stroke to the service structure. The rapidly changing electromagnetic fields set up by the flow of lightning current induce voltages in loops such as between the vehicle, ground, and back up the service structure. These voltages are proportional to the rate of rise of the lightning stroke current and the area enclosed by the loop. These voltages can be surprisingly high. For example, the voltage between the service structure and the vehicle at the 258 foot level, assuming a lightning stroke rising to 100,000 amperes in two microseconds and no interconnections between the vehicle and the service structure, would be of the order of 1100 kv. If a man is standing on a working platform at this level and touches the vehicle at the instant the lightning stroke hits the service structure, the hazard is obvious.

The changing magnetic flux which produces these voltages appears between the vehicle and the umbilical tower, as well, and sets up voltages there even though the umbilical tower is not struck. These voltages are about half those existing between the vehicle and the service structure.

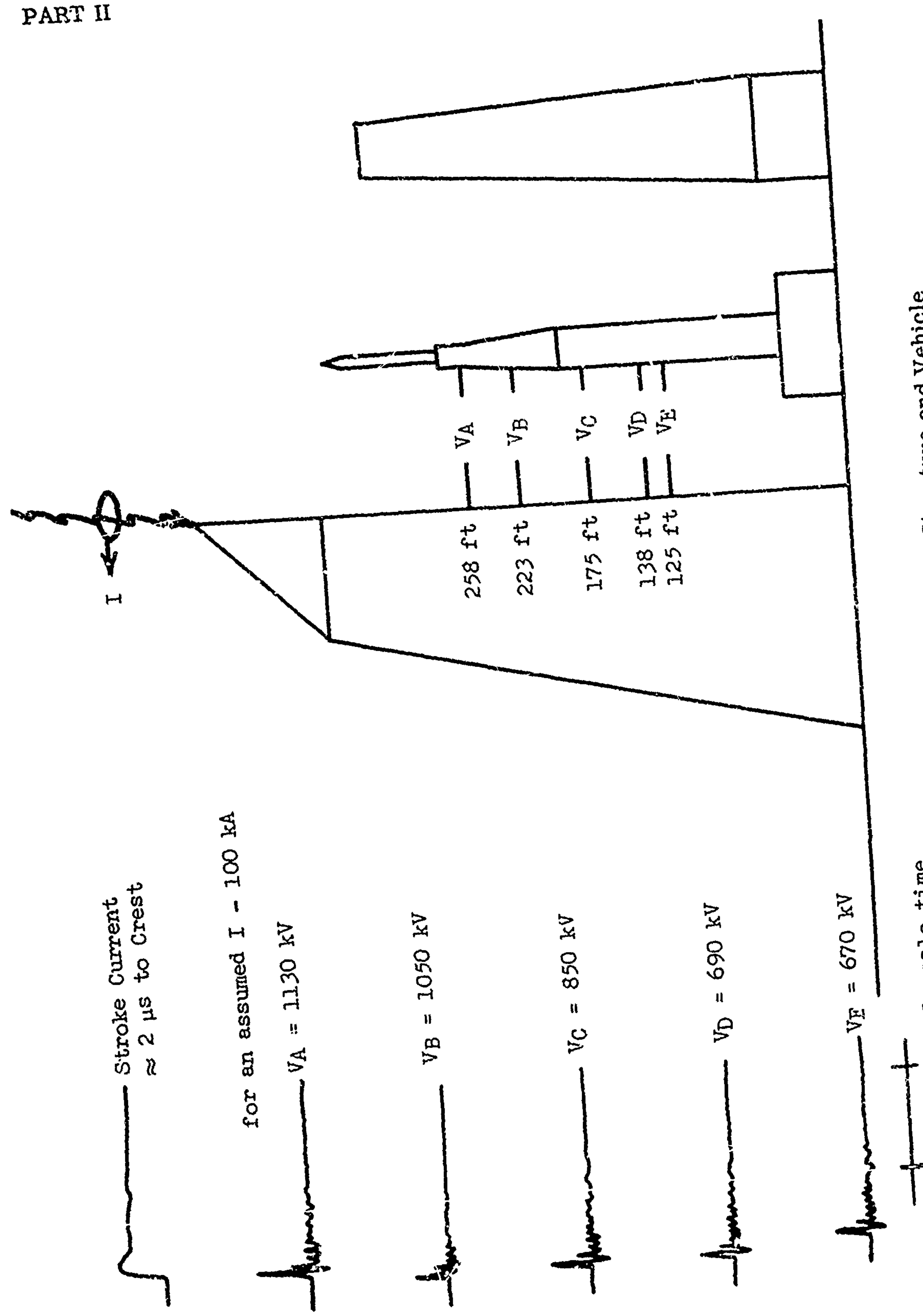


Figure 8. Voltages Between Service Structure and Vehicle

On any launch complex, many electrical conductors run from the vehicle to equipment on the umbilical tower or at remote points on the launch complex. A geometrical model can also be used to determine voltages, at least to the order of magnitude, that might appear on some of these electrical circuits. Figure 9 shows one very simple example in which conductors were run from the vehicle down along the service structure in cable trays to the ground level. The conductors were connected to the vehicle, thus representing a circuit with a low impedance reference to a common ground point in the vehicle. At the ground level the conductors were loaded by the 50 ohm impedance of the measuring circuit. If we consider a stroke rising to a crest of 100,000 amperes in two microseconds, or a linear rate of rise of 50,000 amperes per microsecond, the model data showed that the voltage across the load impedance of that conductor could be as high as 36,000 volts. The magnitude of the voltage depends upon the routing followed by the conductor to the vehicle and on how many other conductors also go to the vehicle. Thus, these figures represent sort of a worst-case example. The 36,000 volts is a line-to-ground voltage; a corresponding line-to-line voltage between two conductors going to generally the same point would be less, about 6500 volts. It is easy to see that such voltages might present a hazard to semiconductor electrical circuits. While the voltages can be greatly reduced by the use of covered cable trays or by the use of shielded rather than open leads, a possibility of dangerously high transient voltages due to lightning still exists.

Designers of large extensive electrical systems who do not consider the effects of lightning are apt to leave themselves wide open to lightning damage because of their natural concern for control of other electromagnetic interference. It is common practice to employ a single-point grounding philosophy and, if shields on electrical conductors are used, to ground these shields at only one end. Such practices may be effective in control of low frequency interference problems, but they can be disastrous if the system is exposed to even remote lightning strokes.

To illustrate this, let us look at Figure 10. In this figure, two separate structures house electronic apparatus. Each structure houses electronic apparatus which is referenced to separate electronic ground systems, but running between the two structures is a shielded multipair cable. When the first structure is struck by lightning, the current flows to ground, but in flowing through the

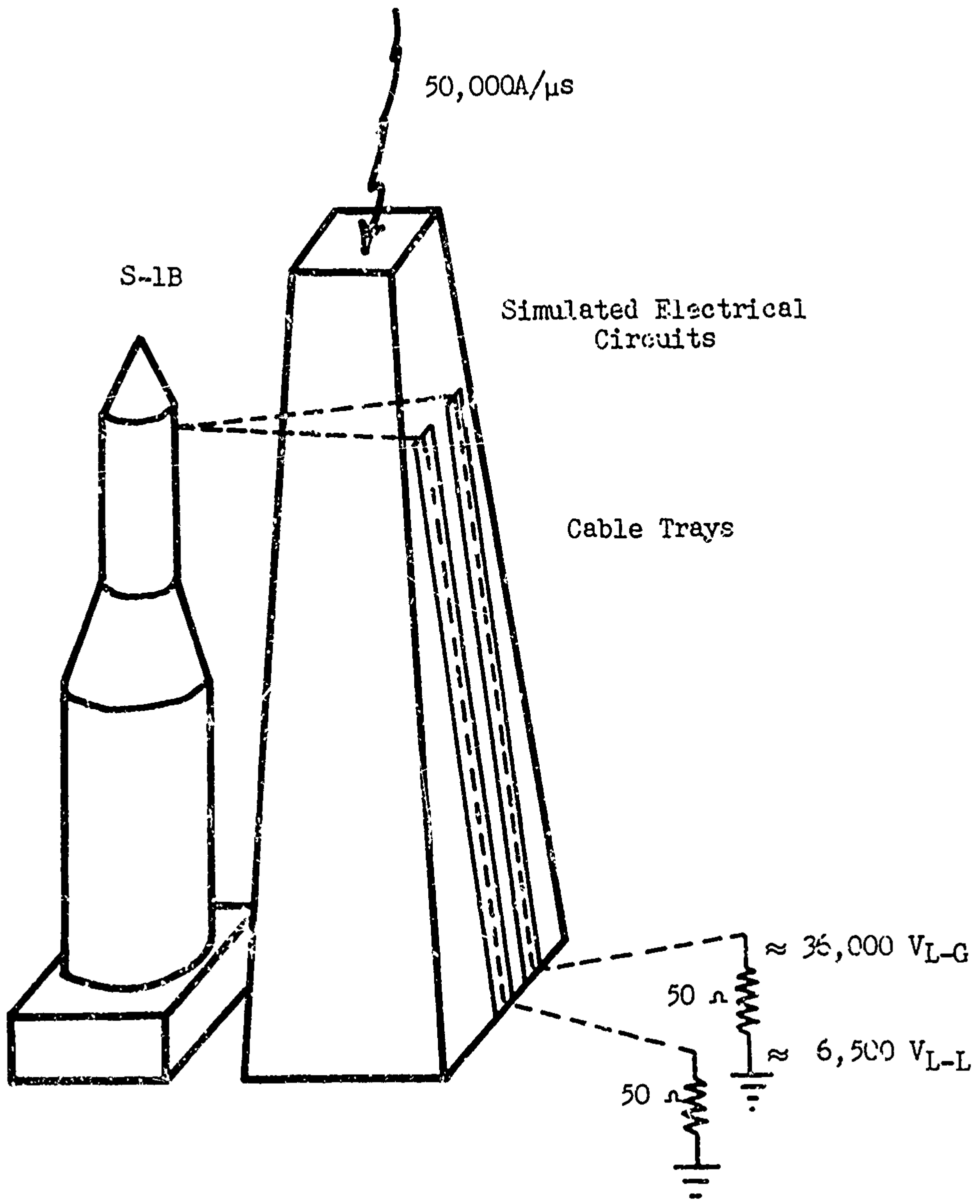


Figure 9. Voltages on Simulated Electrical Circuits

ground with a resistance of $R_1$, it produces a voltage of $IR_1$. Assuming the stroke current is a fairly typical 20,000 amperes and the ground resistance is 5 ohms, then the potential difference developed between the stricken structure and the true earth potential is 100,000 volts. No current is flowing to ground, at the second structure so the potential of this structure relative to ground is zero volts. The shield of the connecting cable and the conductors within this shield are referenced only to the ground potential at the second structure and are thus at the potential of the earth reference. Since a potential difference of 100,000 volts exists between the first structure and the electronic apparatus, two possibilities exist:

(1) If all elements of the electronic apparatus are referenced only to the potential in the second building, this 100,000 volt potential difference appears between the struck building and the chassis of cabinets of the electronic apparatus in it. If a man inside this stricken structure is operating this piece of electronic equipment, he would be exposed to the full 100,000 volts.

(2) If the cabinets or chassis of the electronic equipment are grounded to the stricken structure but the internal electronic circuitry is referenced to the potential of the second structure, the 100,000 volt potential difference would appear on the internal electrical networks. In this case, electrical damage would be almost inevitable.

The surge voltages can be controlled as shown in Figure 10. Basically, the current is allowed to flow on the shield of the conductors and physically-adjacent items are not referenced to separate ground potentials. Note that in Figure 10, the two independent ground systems are combined into one ground system; thus, the chassis and cabinets for the circuits are referenced to the same potential as the structure in which they are housed. Likewise, at Structure A the cabinets and racks are referenced to the potential of the structure in which they are housed. Now the stroke current, as it flows through resistor $R_1$, again tries to build up a potential of 100,000 volts, but the current instead flows along the shield of the cable over to Structure B. Thus, the stroke raises the potentials of both Structures A and B above the 100 potential and so reduces the potential difference between the two. Furthermore, if the current flowing between the structures flows on the outer shield of a conductor, it induces voltages along the conductors which cancel out any remaining difference in potential between the two structures.

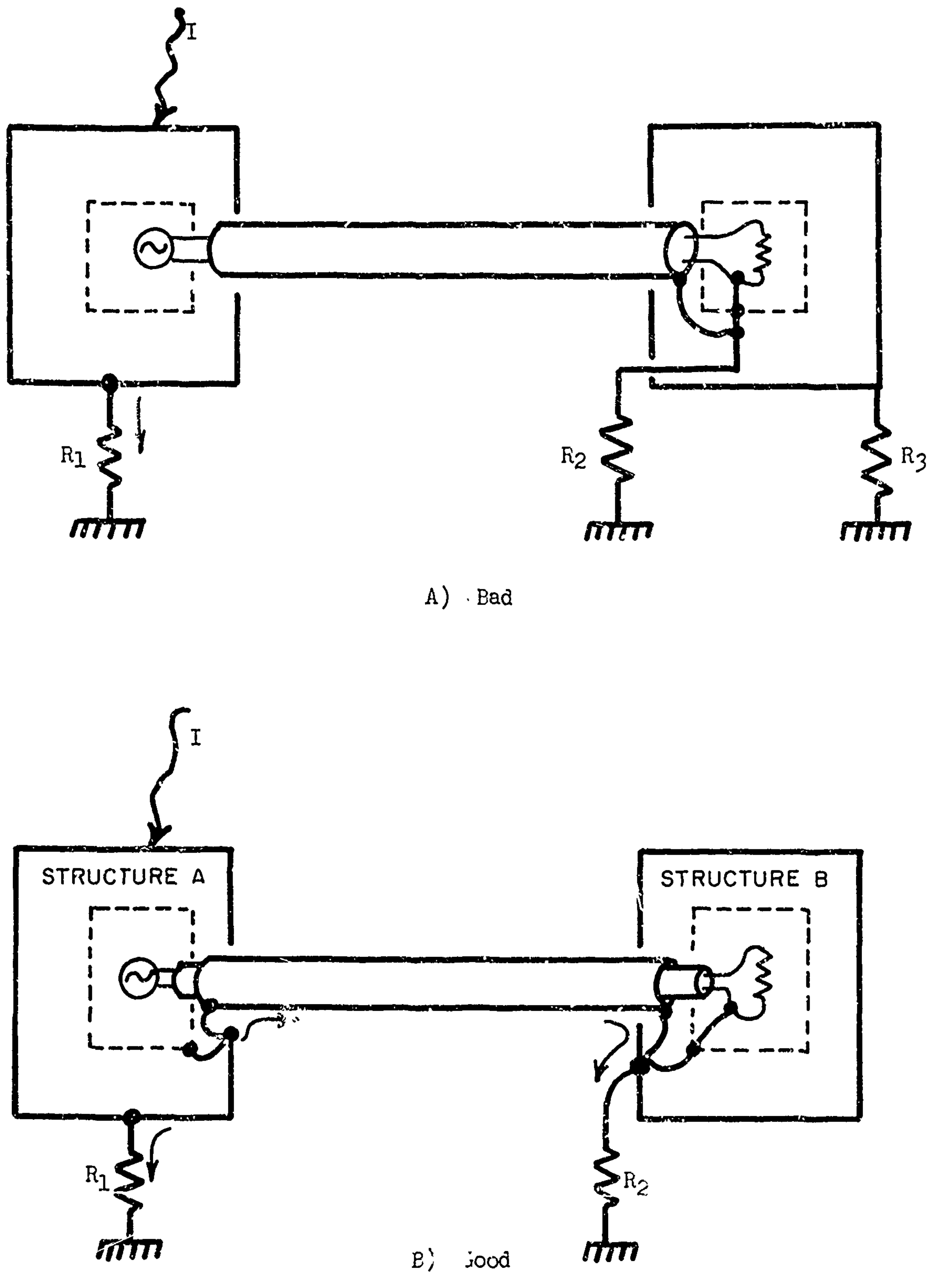


Figure 10. Control of Lightning Induced Transient Voltages

Thus, it is possible for the potential of Structure A to be different from that of Structure B while the potentials of the internal conductors are the same as the potentials of the enclosures in which they are housed.

There are a number of valid reasons for treating shields of some conductors in special ways, such as being grounded at only one end. To accommodate circuit designers in the control of the more mundane low-frequency electromagnetic interference problems for Launch Complex 39, we recommended that all cables have an overall shield which would be grounded to the local ground system at both ends. The conductors within this overall shield could be shielded or not, and the shields of those conductors grounded or not, as local needs dictated.

LIGHTNING EXPERIENCE AT LAUNCH COMPLEXES 34, 37, AND 39

There is a fair amount of information on the characteristics and effects of strokes to the launch complexes at Cape Kennedy and the adjacent Kennedy Space Center over the last few years. Most of the recorded strokes have been to Complexes 34 and 37 and to the numerous tall weather towers around the area.

Figure 11 shows a cumulative probability distribution of stroke current amplitude measured during 1964 and 1965. Current amplitudes were measured by means of magnetic links. While the number of samples is small, they seem to indicate that the strokes have larger amplitudes than are generally observed nationwide. How much reliance one should place on these statistics is a moot point. On Launch Complex 39, mobile launchers have been struck several times in the parking area and at least one time when on the pad with the first Saturn 5 test vehicle. There have been few, if any, reports of equipment damage from lightning. While an absence of reports is not proof of damage, it indicates, at least, that the problem associated with lightning strokes to Launch Complex 39 is very slight.

Some of the recommendations made for Launch Complex 39 were ultimately adopted on Launch Complexes 34 and 37. Very minor electrical damage has been observed as a result of those strokes, although the failure mechanism cannot be reconstructed in all cases. Certainly, the launch vehicle has never been struck directly by lightning nor has lightning caused any major or catastrophic equipment damage on these launch complexes.

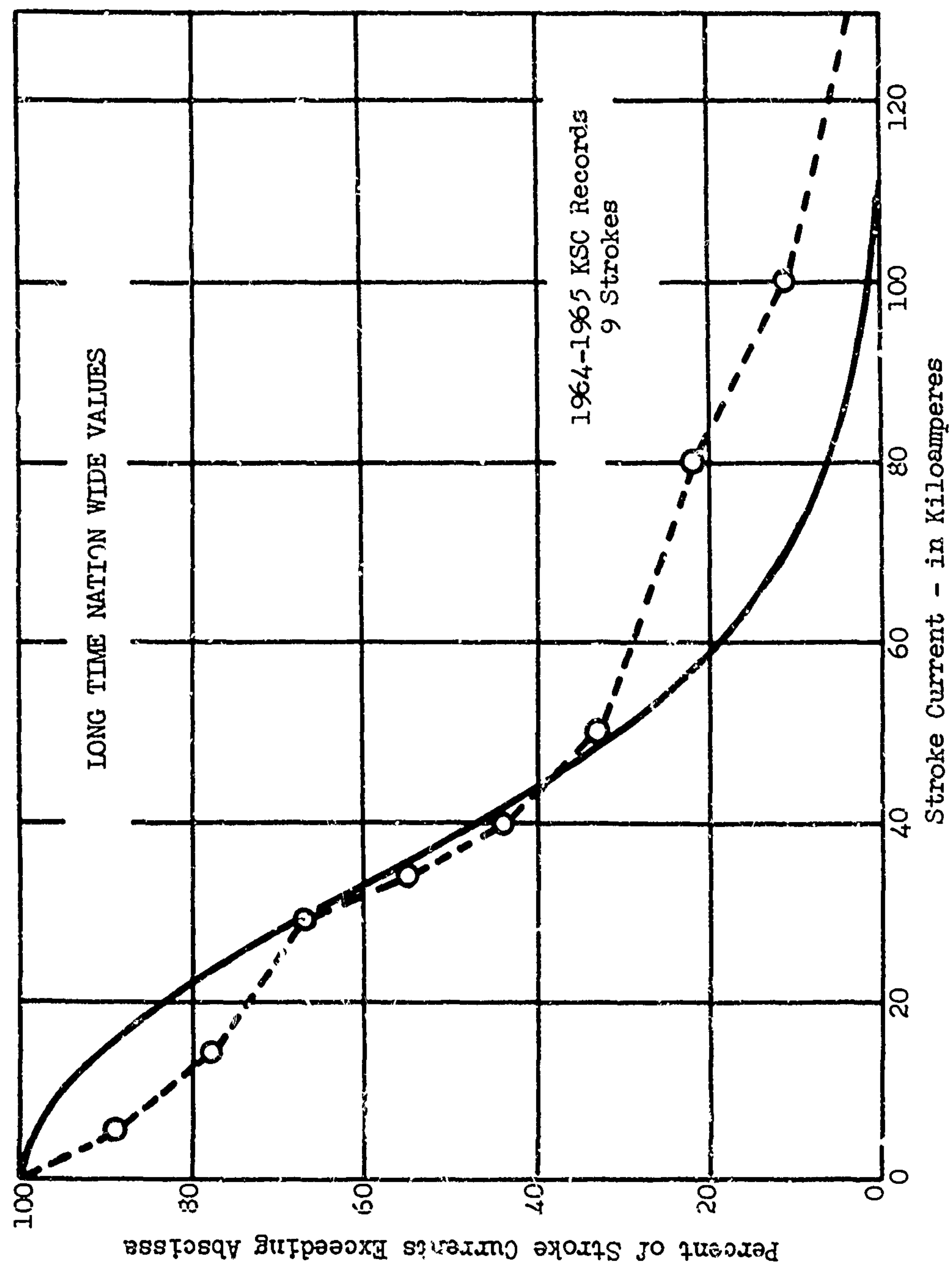


Figure 11. Measured Lightning Stroke Amplitudes

AFAL-TR-68-290
PART II

# ELECTROSTATIC CHARGE EFFECTS ON TFE IN PETROLEUM FLUID SYSTEMS

Douglas R. Barnes

Aeronautical Systems Division

## ABSTRACT

Inflight failures of TFE (Teflon) hose assemblies resulted in an investigation by the Fluid Power Branch, Aeronautical Systems Division. Tests in a simulated aircraft hydraulic subsystem revealed some unusual electrostatic properties of TFE, both virgin and filled, that are not apparent in the usual material sampling tests. Tests revealed that electrostatic voltage builds up to 10 KV or more within the TFE hose. Carbon tracking and hose punctures led to the reported inflight failures.

(Paper not submitted for publication. For reprints, contact author directly.)

# SESSION IV

# GROUNDING AND BONDING

R. A. Peterson, Chairman

Dorne & Margolin, Inc.

# ELECTRICAL BONDING OF ADVANCED AIRPLANE STRUCTURES

L. E. Short

The Boeing Company

The advent of new structural techniques in airplane construction has introduced significant new problems in achieving electrical bonding in the aircraft. Before examining why electrical bonding is required and what the effects of advanced structural techniques are, it is appropriate to define the meaning of electrical bonding.

Bonding is a much misunderstood term and considerable confusion exists between structural and electrical/electronic designers. It may be defined as a binding force which acts to join separate parts together. To the structural designer this means an adhesive or binder which is used to connect adjacent parts of a structure. To the electrical/electronic designer, however, it means a method of achieving electrical continuity or conduction between adjacent components. An aircraft structure obviously must have adjacent parts structurally bonded by adhesives, mechanical fasteners, or both. Likewise, adjacent parts generally must be joined to form an electrically continuous structure. The problem which arises from these two uses of the term "bonding" is that the structural adhesive bond is inherently nonconductive; therefore, when adhesive bonding is achieved, electrical bonding is eliminated without special treatment. The converse may also be true, for if adhesives are loaded with conductive particles to make them conductive, their adhesive properties may be destroyed, so that an electrical bond is achieved but structural bonding is degraded.

## REASON FOR BONDING

At this point, it is appropriate to review why electrical bonding of an airplane is necessary. There are five general reasons why an airplane must be electrically continuous: (1) lightning protection; (2) electrical system current return paths; (3) electrostatic charge dissipation; (4) antenna performance; and (5) electromagnetic interference control.

## 1. LIGHTNING PROTECTION

Lightning protection is required since experience has shown that airplanes are often struck. Commercial airline transport airplanes suffer strikes at the rate of slightly less than one strike-per-airplane-per-year (Reference 1). Lightning strokes release considerable energy which may take the form of cloud-to-cloud, or cloud-to-ground strokes. The point is that strokes never originate or terminate on the airplane, but literally just pass through, as illustrated in Figure 1. There are always at least two attachment points on an airplane for each stroke, with the stroke current flowing through the structure between the two points. The amount of damage produced by lightning can be minimized by minimizing the amount of energy dissipated in the structure between the entrance and exit points. This is done by providing a low-resistance, electrically-well-bonded structure.

Lightning stroke currents may be very large and the onset of current is very rapid. According to MIL-B-5087B, the maximum current may be as great as 200,000 amperes with a rate of rise of 100,000 amperes-per-microsecond. This initial rush of current may be followed by a continuing lower current component of perhaps 250 amperes, which persists for up to two seconds (Reference 2). It is appropriate to note that not all, and in fact very few strokes reach this intensity. These parameters represent the intensity generally accepted as the maximum value for use in design.

The effect of the high-current rapid-rise component is possible sparking at discontinuities through which the current passes. The voltage (v) developed across a discontinuity is a function of the current magnitude (i), its rate of change di/dt, and the inductance (L) and resistance (R) of the discontinuity

$$v = iR + L\frac{di}{dt}$$

Sparking may or may not be serious, depending upon where it occurs and its severity. In an adhesively bonded joint, sparking can lead to carbonization of a portion of the adhesive and reduction of its structural integrity.

The long-term component of a stroke consists of a much lower current than the initial component, but may produce heating in a resistive discontinuity by

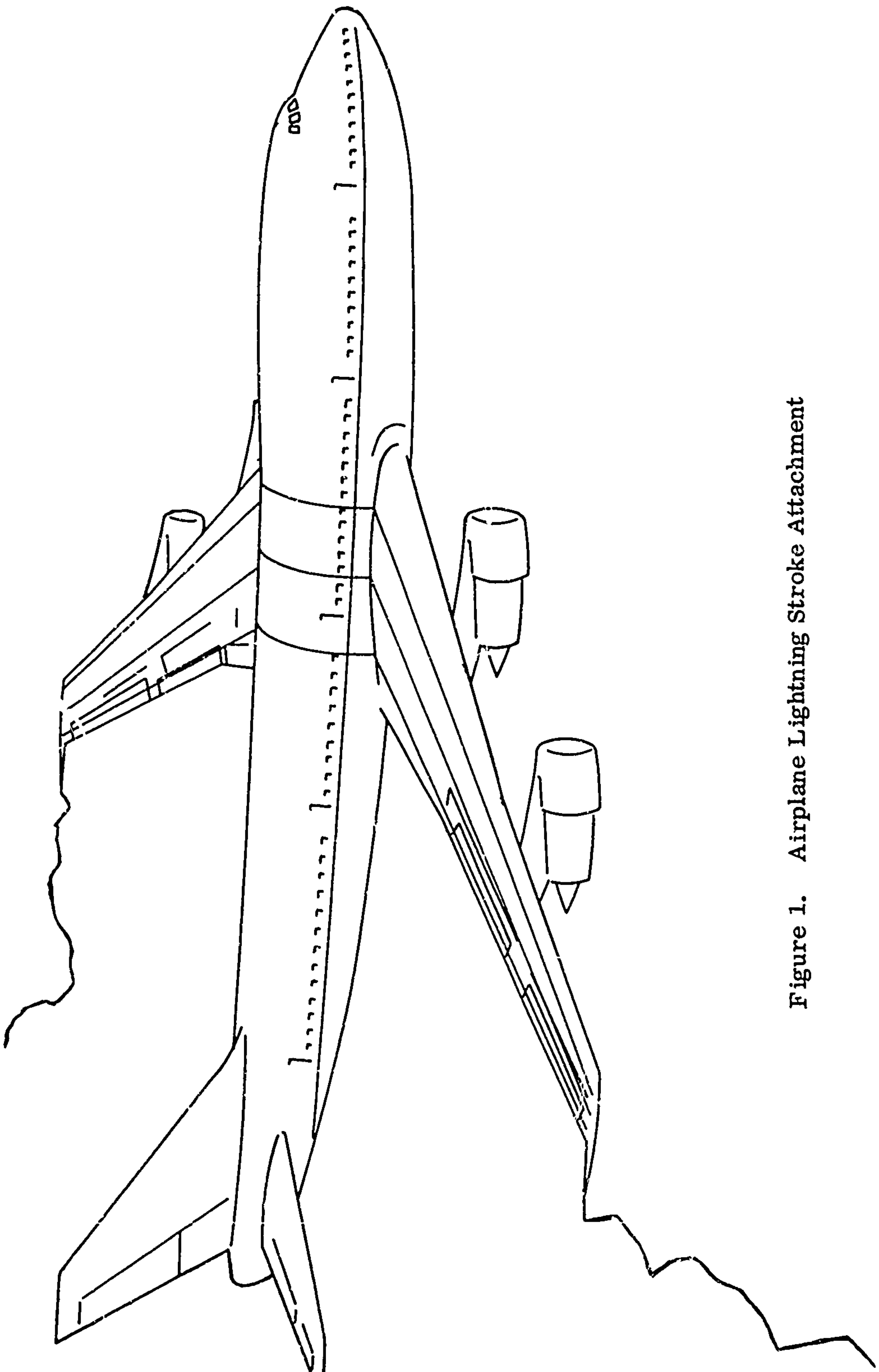

Figure 1. Airplane Lightning Stroke Attachment

virtue of its persistence. This can produce erosion of metals and destruction of non-metallic materials through which the current must pass.

The key to lightning protection is to provide a conductive structure through which the current may flow. This protection has been achieved successfully on modern transport aircraft now in service. The methods by which lightning protection bonding is derived will be discussed later.

## 2. ELECTRICAL POWER DISTRIBUTION

In order to minimize the amount of wire required in an airplane electrical system and thereby minimize weight, the structure is used as the return path for the electrical distribution system. That is, one wire (for a single-phase ac or a dc load) is used to supply operating voltage from the power source and the structure is used to complete the circuit back to the source. The return current flowing in the structure may sometimes reach several hundred amperes (e.g., electrical system faults may cause currents as high as 2800 amperes to flow in the structure); bonding to conduct this current would be of the same order as that required to safely conduct lightning stroke currents. Bonding is particularly critical in fuel vapor areas, because the heat produced by current flowing across a resistive joint must be lower than that required to produce ignition.

To meet this criteria, the resistance between adjacent structural components through which these currents flow must not exceed a few milliohms. For example, if a current of 100 amperes flows across a joint having a resistance of 10 milliohms, it dissipates 100 watts in the joint — sufficient power to light a lamp. Obviously, where power return currents must flow, a low-resistance bond must be provided between adjoining structural elements.

## 3. ELECTRO-STATIC CHARGING

Another requirement for bonding arises from the generation of electrostatic charges in an airplane. In flight, an airplane impacts with airborne particles such as dust and ice crystals. As a result of these collisions a net charge is deposited on the airplane surface. Since this charging is most vigorous when flying through precipitation, it is generally known as precipitation charging. This mechanism is capable of charging an airplane to several hundred thousand volts, at which point portions of the structure go into corona and sparking may occur between poorly bonded exterior parts. These discharges will produce radio noise, which is very familiar to anyone who has used airborne ADF or HF communication systems. As the most vigorous source of this form of radio noise is precipitation-charging, the phenomenon is called precipitation-static, or simply p-static.

Means of dissipating the p-static charge from an airplane have been developed (Reference 3), but their success is dependent upon the exterior of an airplane being electrically continuous. The current deposited on the surface by particle impact must be conducted to the adjacent structure and eventually to the discharger or a large difference in potential between adjacent surfaces and sparking may occur. Since a charge can be deposited on nonconductive exterior surfaces as well as on conductive surfaces, all exterior surfaces exposed to particle impact should be made conductive so that surface sparking or streamering will not occur.

The bonding or conductivity required for control of p-static is considerably less than that required for lightning protection and power current return. Because the total charging current is of the order of thousands of microamps or less (Reference 3), a bond resistance of 500,000 ohms may be acceptable. In fact, surface resistances as high as 10 megohms per square have been found adequate on radomes. Surface conduction and structural bonding must be provided to bleed off these charges, however, since even minute energies radiated from spark discharges can produce crippling radio interference.

Another form of electrostatic charge is generated by fluid movement such as in the fuel system. Theoretically, charging will occur whenever there is movement of fuel, by sloshing within the tanks, by transfer to the engines, or in airplane refueling. In practice, the only significant charging occurs during refueling. Charging currents as high as 20 microamperes have been recorded at refueling rates of 500 imperial gallons-per-minute in a test configuration as shown in Figure 2 (Reference 4). The principle source of charging is the fuel filter; as a consequence, the fuel flowing into the aiplane tanks carries a net

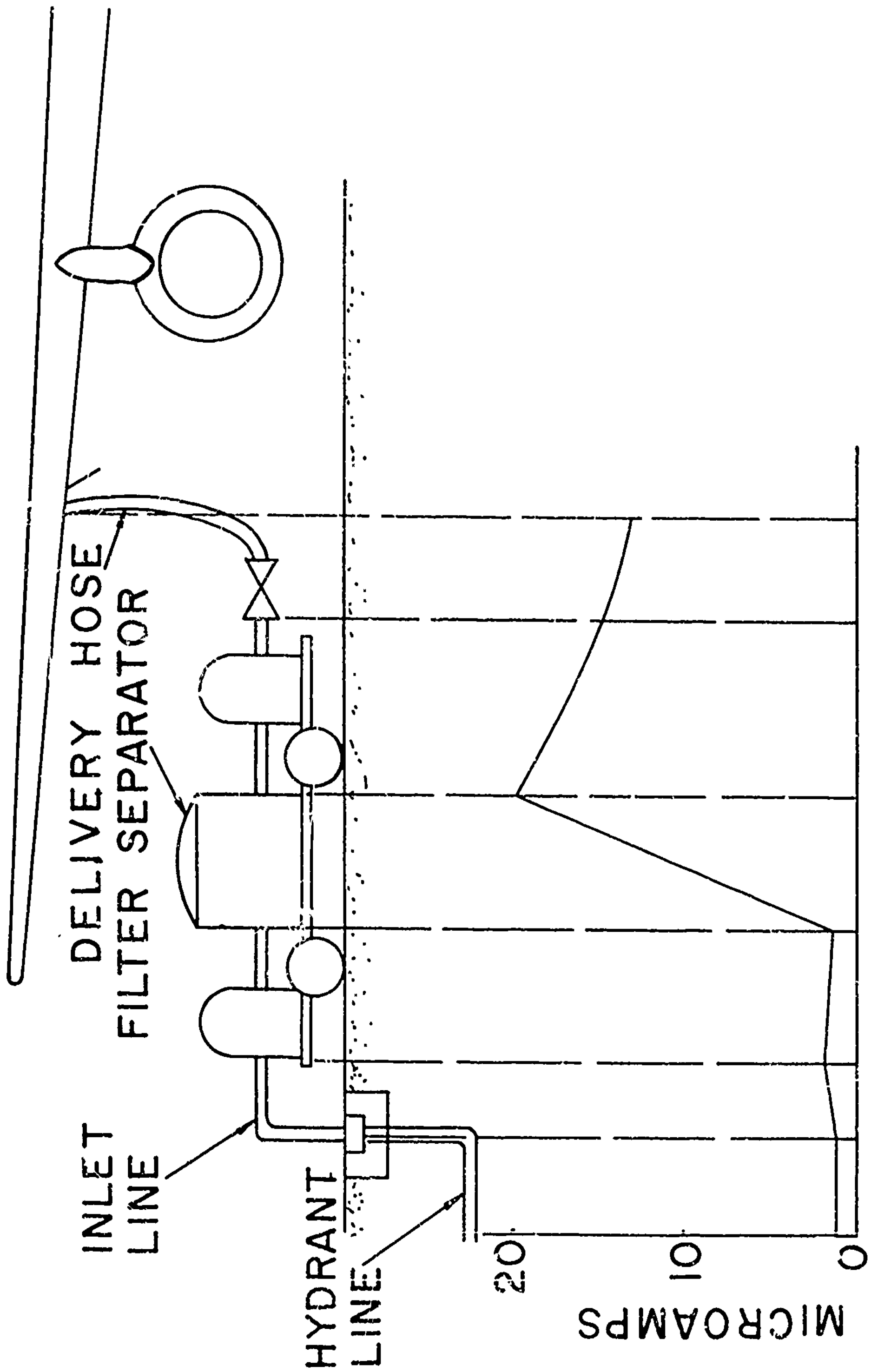


Figure 2. Electrical Charging During Refueling

charge which is eventually deposited on the tank walls. Sufficient conductivity must be provided from the tank-to-fuel surface and then back to the filter so that large voltages are not built up in the fuel system, which includes the airplane fuel tanks and plumbing. Again, only moderate conductivity is required; in fact, coatings of relatively good dielectrics may be tolerated on the fuel tank walls.

## 4. ANTENNA PERFORMANCE

The efficacy of most airplane antennas is dependent upon the airframe being electrically continuous, at least in the vicinity of the antenna. The antenna impedance and radiation patterns are dependent upon the counterpoise or ground plane created by the metal airframe. RF currents must be able to flow in the structure within at least several wavelengths of the antenna. In order to do this, the adjacent structure must be bonded so that both a low resistance and low reactance path are provided. A phenomenon known as skin effect, by which the current tends to flow near the exterior surface of a conductor as the frequency is raised, becomes manifest in bonding for antenna ground planes as well as for lightning protection. The bond between skin panels where the RF currents flow must take into account the skin effect. If the current is forced to flow inward between bonded assemblies, the inductive reactance, hence the impedance of the bond, is raised.

Whenever high frequencies are introduced, the reactive component of a bond becomes significant. Devices such as bond straps and widely spread fasteners may be effective for low frequencies and direct currents, as in current return paths and electrostatic charge bonds, but are not necessarily effective for RF bonding. RF currents may cause dielectric heating of adhesively bonded assemblies. Where high RF currents flow, such as adjacent to an HF transmitting antenna, significant heating of the adhesive can occur. Experimental bonded structures have produced temperature rises of 150°F in 1.5 minutes in an adhesive adjacent to an HF antenna. To prevent this heating, which could lead to structural weakening, it is necessary to examine the electromagnetic field structure about the bond. Electrical bonding which will minimize the magnitude of fields established across the bond line is required.

5. ELECTROMAGNETIC INTERFERENCE CONTROL

An implicit requirement in all bonding is the elimination of electromagnetic interference. As in the case of p-static generation, the primary requirement for bonding is to prevent radio noise from spark discharges between adjacent parts. Many other factors may produce noise in poorly bonded structural elements. Looking to the antenna systems again, a high impedance joint may result in an RF voltage being developed between two skin panels. This voltage may be coupled directly into other radio systems which use the structure as ground; if a non-linear element is introduced which acts as a detector, the voltage developed may even be coupled into an audio system. Other, less obvious problems may be produced in some cases. For example, on an adhesively bonded airplane, significant voltages may be developed by piezoelectric effects across structurally bonded joints if electrical bonding is not provided.

CONVENTIONAL STRUCTURE BONDING

Electrical bonding is inherent to conventional structures and can be taken for granted. As shown in Figure 3, a structure joined by multiple, closely-spaced fasteners provides current paths adequate for all electrical bonding requirements. Even when nonconductive primers and sealants are used at the faying surface the joint may be adequately bonded. Where this type of construction is used throughout an airplane, i.e., multiple metal-to-metal contact surfaces, electrical bonding is achieved.

The situation even with so-called conventional structures is obviously not completely straightforward, and each situation must be examined on its own merits. For example, the bonding of a non-load bearing fuel tank access door on some jet transports has produced several problems which typify the need to examine all bonding requirements. The door is retained by clamping against the wing skin, as shown in Figure 4. In this case the access door is bonded to structure only through the bonding jumper, which affords a resistance of only a few milliohms between the door and structure. This proved adequate for dissipating electrostatic charging and preventing radio interference. When this assembly was subjected to artificial lightning tests intense internal spark showers were produced (Reference 5). Obviously, low resistance is not the only criterion of a

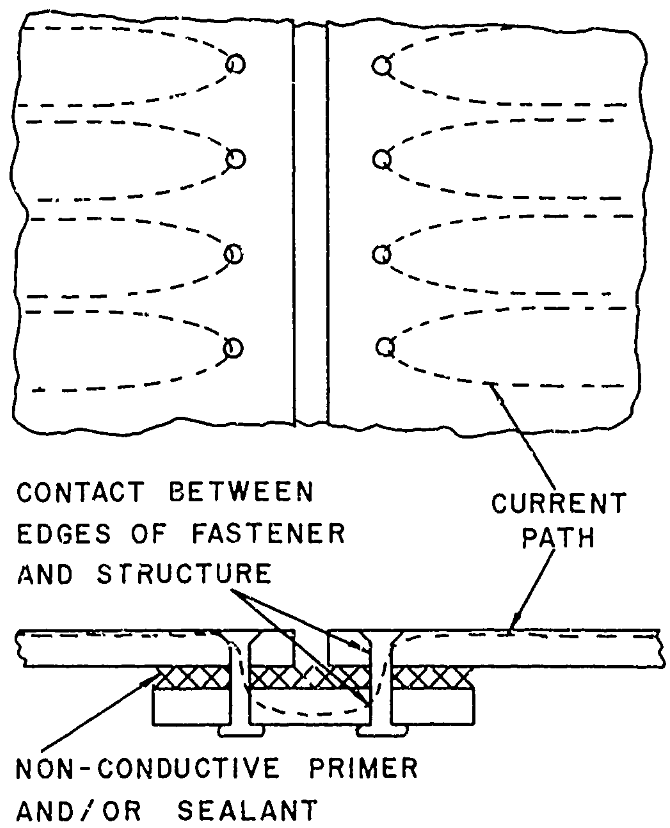


Figure 3. Bonding Through Mechanical Fasteners

good bond; reactance must be considered, and in the case of bonding for lightning protection, tests with simulated lightning must be performed to assure the adequacy of the design. A considerable development program was required to achieve an access door design which is both electrically and mechanically satisfactory. In the final design, the phenolic rub strip shown in Figure 4 was replaced with a knitted wire mesh gasket impregnated with grease. In addition, all finishes, except where an alodine is used, were removed from the faying surfaces of the access door, clamp ring, and wing skin. The mesh gasket provides adequate conduction and affords a pliable surface to absorb the relative motion between wing skin and access door. The grease fills any void space and prevents entry of water.

The practice of environmental sealing of conventional structures does not necessarily interfere with the inherent bonding properties of mechanical fasteners. It is necessary, however, to examine the bonding capability of these

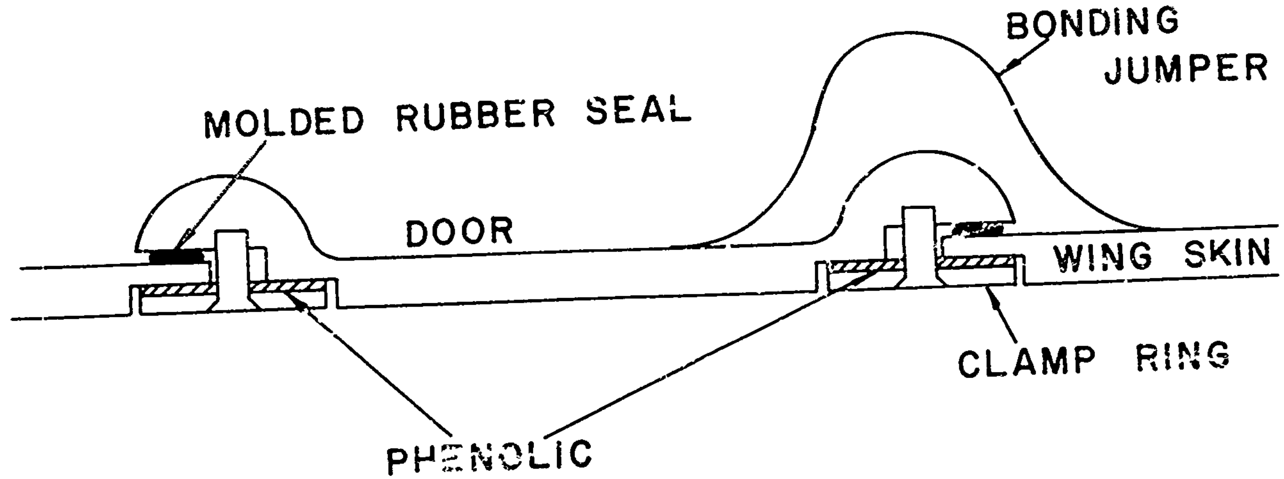


Figure 4. Non-Load Bearing Fuel Tank Access Door

structures, particularly with regard to lightning protection. In at least one contemporary jet transport, titanium taper-lock fasteners are being used in wing skin joints about the integral fuel tanks. In this area where there may be combustible fuel-air mixtures, these joints must pass lightning stroke currents without internal sparking. Even when the fasteners are installed with a wet sealant and environmental sealing is used at the fay surfaces, this structure will pass stroke currents.

A sample of this construction subjected to simulated lightning tests is shown in Figures 5 and 6. This panel was tested by mounting in an opening of an all-metal cabinet, which formed a Faraday cage to shield the interior in the same manner a metal airplane interior is shielded. One-half of the panel was isolated from the cabinet, while the other half was bonded to it. Simulated lightning was directed at the isolated half of the panel so that the entire 200,000 ampere stroke current had to pass through the joint. No sparking was observed in the cabinet during the test, and none was photographed with a camera using ASA 3000 speed film. The success of this particular test cannot be taken as a blanket qualification of environmentally sealed construction. In this sample the fasteners are used in a high density pattern. Being taper-lock fasteners an interference fit is obtained between the fastener and structure so that a metal-to-metal contact is obtained despite the use of sealant on the fasteners. Bonding is achieved as illustrated in Figure 3. Other tests with similarly sealed construction but different fasteners and fastener patterns have produced internal sparking and charring. These tests serve to demonstrate that it is possible to construct environmentally sealed structures which are adequately bonded, even for lightning stroke currents, however, the adequacy of the bond cannot be taken for granted. Obviously, structural and electrical airplane designers must collaborate to satisfy bonding requirements.

## CHALLENGE OF ADVANCED STRUCTURES

New structural techniques using adhesive structural bonding present significant new challenges. The advent of adhesive bonding in airplane construction enables use of composite structures such as honeycomb sandwich construction; thus, nonmetallic skins can be used in high-performance aircraft, which eliminates mechanical fasteners. Since adhesives are inherently nonconductive, the

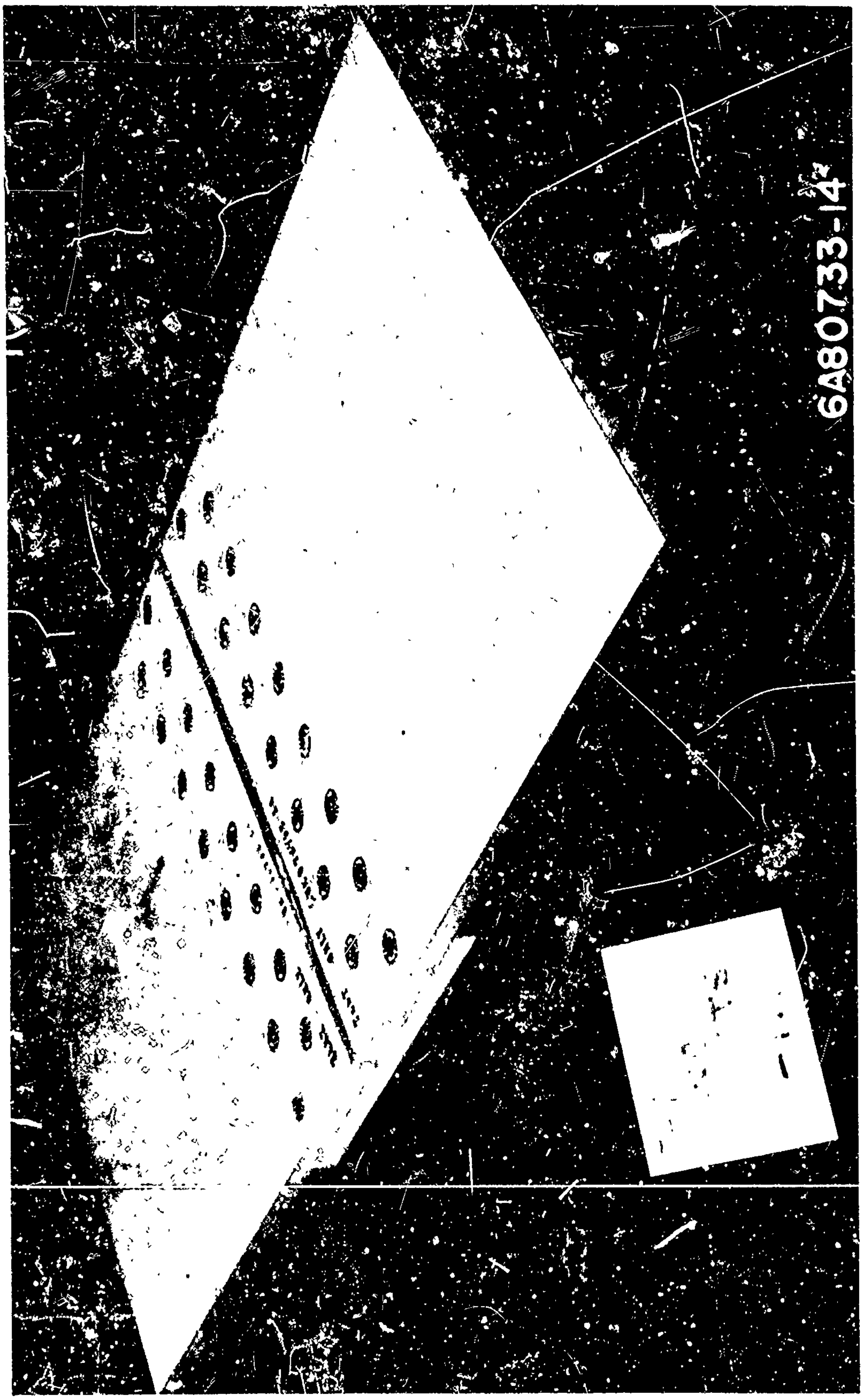


Figure 5. Sample Stringer Splice With Environmental Sealing, Front View

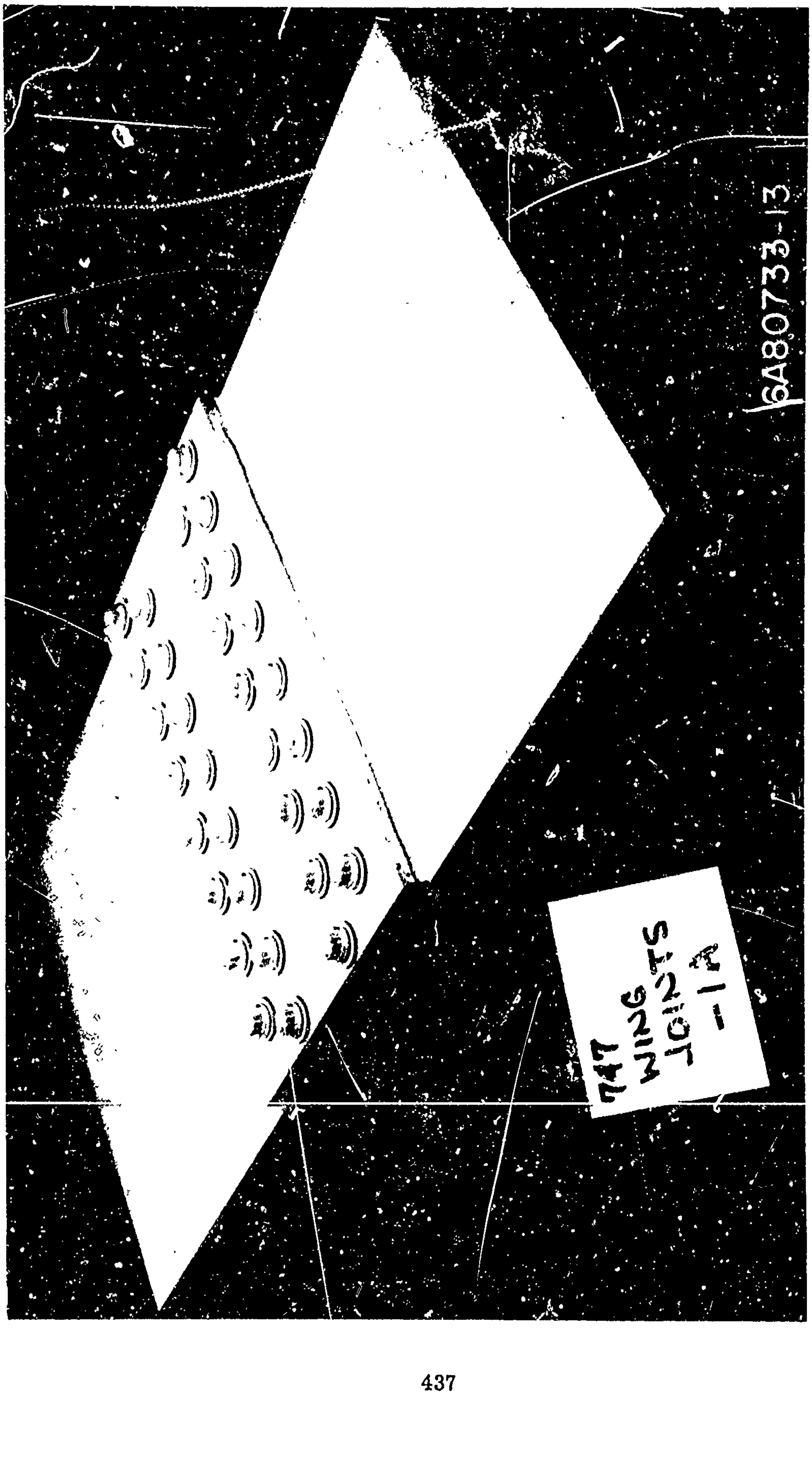


Figure 6. Sample Stringer Splice With Environmental Sealing, Rear View

consequences may be as illustrated in Figure 7. The inner and outer skin panels are electrically isolated by the adhesive which joins them to the core. In turn, adjacent panels are isolated from each other as is the extruded stiffener. If a lightning stroke should pass through this assembly, sparking and dangerously high potentials would occur.

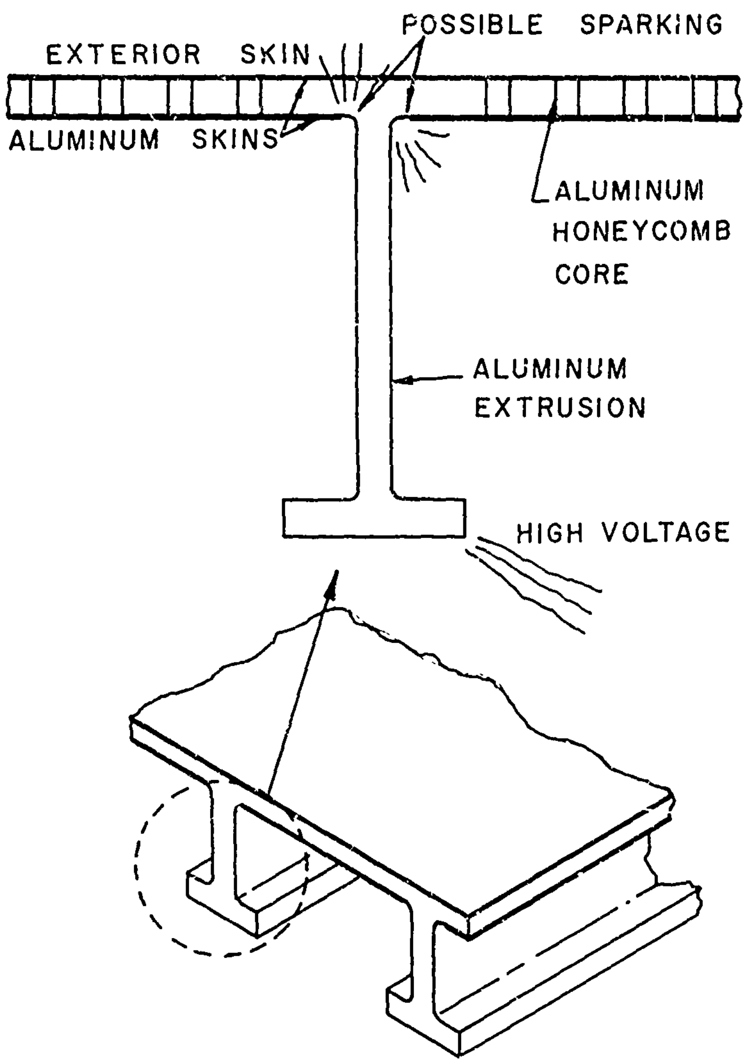


Figure 7. Lightning Stroke Current in an Adhesively Bonded, Modified Monocoque Construction

There is no categorical solution to electrical bonding of adhesively bonded structures. Some possible solutions can be offered for specific types of construction. For example, an adhesively bonded strap joint as shown in Figure 8 might be bonded in any, or in a combination, of the methods shown. The practicality of these solutions will depend upon the structural application and upon the bonding requirement; i.e., must the joint carry lightning strokes, ground return current, or only current associated with electrostatic charging? Obviously, this must be resolved by collaboration of the structural and electrical aircraft designers.

## CONCLUSIONS

While electrical bonding of metal airframes is generally an inherent property of the structure, such is not the case with advanced structures. Special treatment must be given to environmentally sealed and adhesively bonded structures to insure adequate conductivity for lightning protection, return currents, electrostatic charge dissipation, antenna performance, and electromagnetic interference control. The structural designer cannot work independent of the electrical designer, since new structural techniques are inherently nonconductive. Likewise, the electrical designer must recognize the structural requirements, since many of the techniques of achieving electrical bonding may compromise structural integrity.

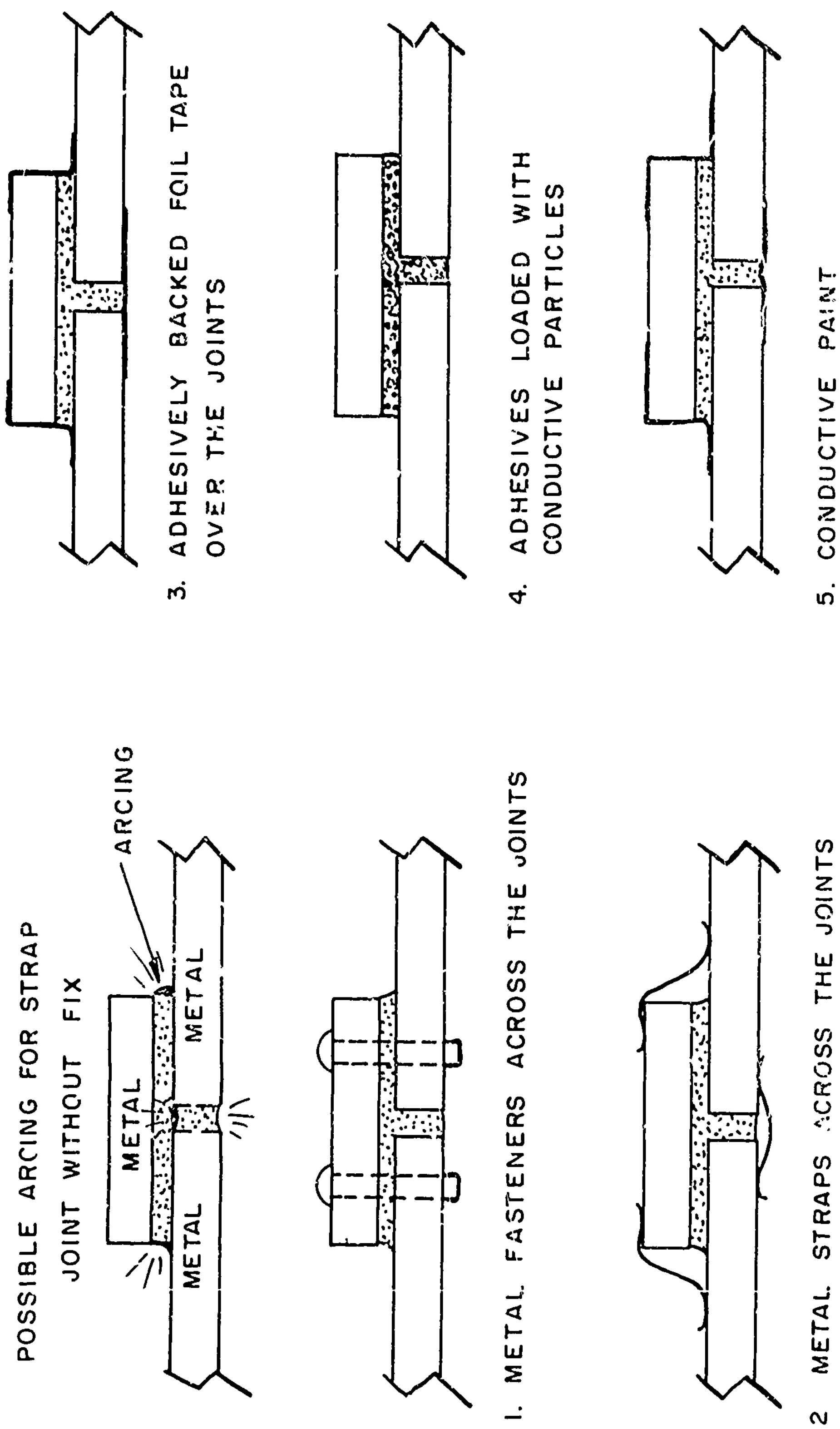


Figure 8. Bonding of Adhesive Strap Joint

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# STATIC ELECTRICITY IN AIR FORCE REFUELING SYSTEMS

J. B. Godwin, Jr.

Kelly Air Force Base

The static electricity phenomenon has been experienced by everyone at one time or another, such as receiving an electrical shock when reaching for a door knob after walking across a wool or synthetic carpet during cold weather. On an average a spark that can be seen approximates 1000 volts. The same condition occurs from sliding across a seat in an automobile. The flow of conventional aviation fuel through a pipe also acts as a static generator. When two different substances are brought into contact with each other and a small difference in potential exists between them, negative charges migrate to one and leave an equal number of surplus charges on the other; static is generated when the two substances are separated. At least one of the substances must be a very poor conductor (or an insulator) if static electricity is to be generated. In the case of flowing fuel, electrons become entrapped in the fuel (a poor conductor) during the separation process and leave a surplus of positive charges in the pipe. Since no electrons are destroyed in electrostatic generation, for every positive charge developed there must be an equal negative charge. High quality fuel is an excellent insulator; while flowing through a pipe, therefore, the charges separate and electrons become entrapped in the fuel, as in an electrical capacitor.

The Air Force has a wide variety of fuel handling equipment including refueling vehicles, pipelines, bulk storage tanks, and filtering equipment. It is the responsibility of the Air Force Aerospace Fuel Directorate to procure and transport quality fuels from the sources to the aircraft through all these systems. In addition to controlling quality, Aerospace Fuels gives guidance on controlling static electricity for all these refueling systems. Electrostatic protection procedures have been complicated by many factors, such as improved refining methods, cleaner systems, nonferrous piping, and new coating systems.

The primary turbine fuel used by the Air Force is JP-4, which technically qualifies as an intermediate vapor pressure fuel. The MIL-T-5624G specification requires a vapor pressure maximum of 3 and a minimum of 2 psi. The majority of Air Force gas turbine engines were designed to this fuel specification, which basically is a blend of kerosene and straight run gasoline. Practically any petroleum refiner can produce JP-4. This fuel is dangerous when transferred in any type transportation system because an ignition source may be introduced at the receiving tank under conditions optimum for combustion; JP-4 is more dangerous than AV GAS 115/145 because of its lower volatility and electrostatic generating tendencies. During aircraft refueling, for example, the air-fuel mixtures are generally in the explosive range, and only a spark will bring on a disaster, such as shown in Figure 1. Maximum precautions must be taken by all concerned to ensure safety to personnel and property. The Air Force emphasizes proper grounding and bonding procedures for maintaining electrical continuity during aircraft refueling; the sequence method serves to reduce electrostatic sparking in the vapor space above the fuel in aircraft tanks.

Technical Orders and aircraft flight manuals outline the procedures for bonding and grounding aircraft. Static grounds and flexible cables with battery clips have been used to maintain electrical continuity. Battery clips of varying sizes and flexible cables of different sizes were attached to convenient locations on the aircraft as shown in Figure 2. These connections sometimes did not provide electrical continuity between the aircraft and servicing equipment because they connected nonconducting parts, and they sometimes damaged aircraft control surfaces. The battery clips would come apart and damage tires of aircraft and service equipment, aircraft surfaces, and gas turbine engines. These problems were magnified in Southeast Asia where quick aircraft turn around required concurrent refueling and rearming. The main deficiencies of grounding were shown to be:

(1) The battery clips were fragile and when broken would cause extensive damage.

(2) The paint on aircraft (usually camouflage) acted as an insulator between the hardware and the airframe and prevented electrical continuity.

(3) Adequate grounding points were lacking on assigned aircraft. Example: Post World War II aircraft A-1E (converted Navy assault Bomber) had no receptacles for grounding, and several century-series aircraft (F-100, F-102)

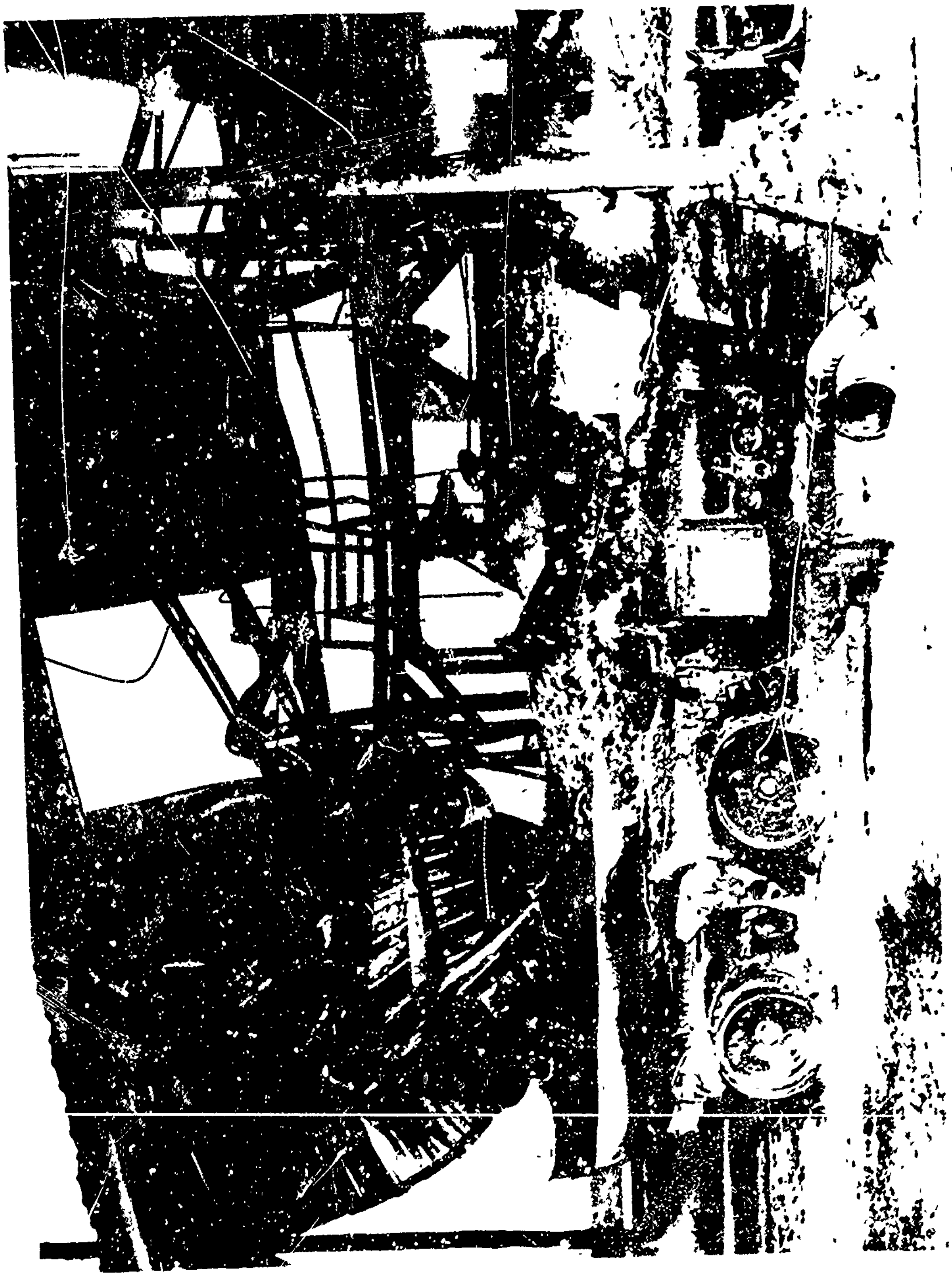


Figure 1. Disaster to a Transport Truck From Static Electricity and Aircraft Fuel

Figure 2. Grounding Aircraft for Fuel Servicing

had only one grounding point so that concurrent servicing could not be accomplished without compromising safety.

(4) Real estate at air bases in SEA is inadequate and aircraft are often parked with overlapping wings. Any ignition of flammable vapor or ordnance by static electricity or improper grounding, therefore, would result in a major accident.

(5) The number, location, and method of attachment of grounding receptacles clearly dictated a need for standardization on all aircraft in the inventory.

Air Force Logistics Command, which has responsibility for modifications to aircraft in the inventory, established a project to standardize aircraft grounding, with prime responsibility assigned to the Aerospace Fuels activity. An official AF Static Grounding Team was chartered in 1967, comprised of scientific and technical personnel cognizant of electrostatic problems. The primary objective of the project was to standardize grounding systems to reduce and/or eliminate static electricity hazards during aircraft servicing, i.e., refueling, rearming, testing, etc.

A female electrical receptacle was designed for mounting on each aircraft and a male plug was designed for each servicing vehicle to be mounted on a 3/32 inch flexible cable. These fixtures are shown in Figure 3. This will eliminate the need for the battery clip. All engineering for retrofit has been completed and aircraft are being modified, generally during inspection and repair, or with field modification kits. Connection sequence is shown in Figure 4. The target date for completion of modifications in December 1969; aerospace ground equipment modifications will be completed earlier. Air Force Systems Command, the activity responsible for aircraft in acquisition, has taken action to amend commercial contracts for aircraft including the C-5A, F-111, A-7, and F-5 per Military Specification MIL-E-6051D, "Electromagnetic Compatibility Requirements Systems."

For the program to be totally effective, transient AF aircraft landing at other military bases and civil airports must be covered by these new grounding procedures. The Army and Navy have initiated a standardized aircraft grounding program using the same fixtures and procedures. (The AF and Navy have

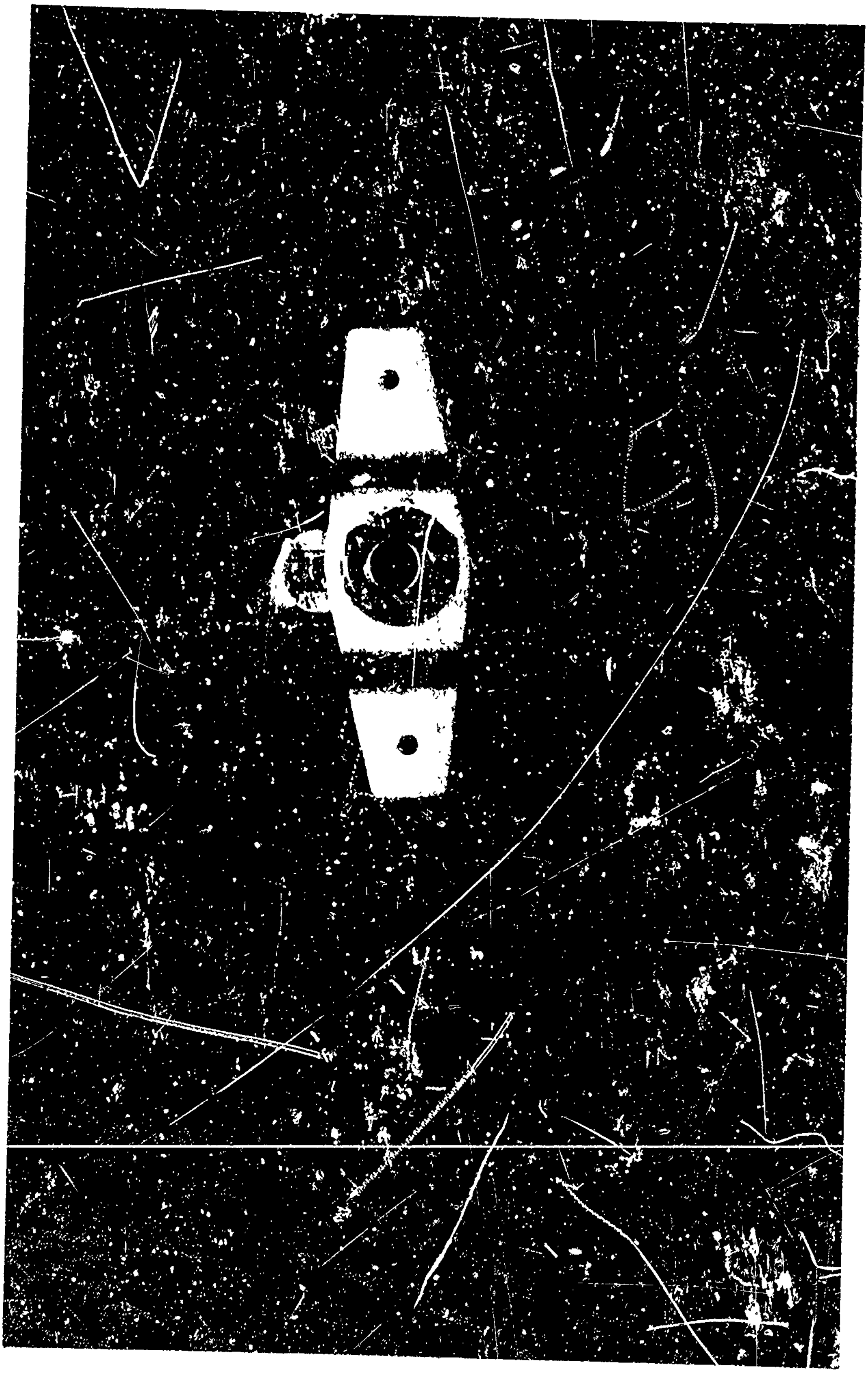


Figure 3. Standardized Fixtures for Grounding Aircraft

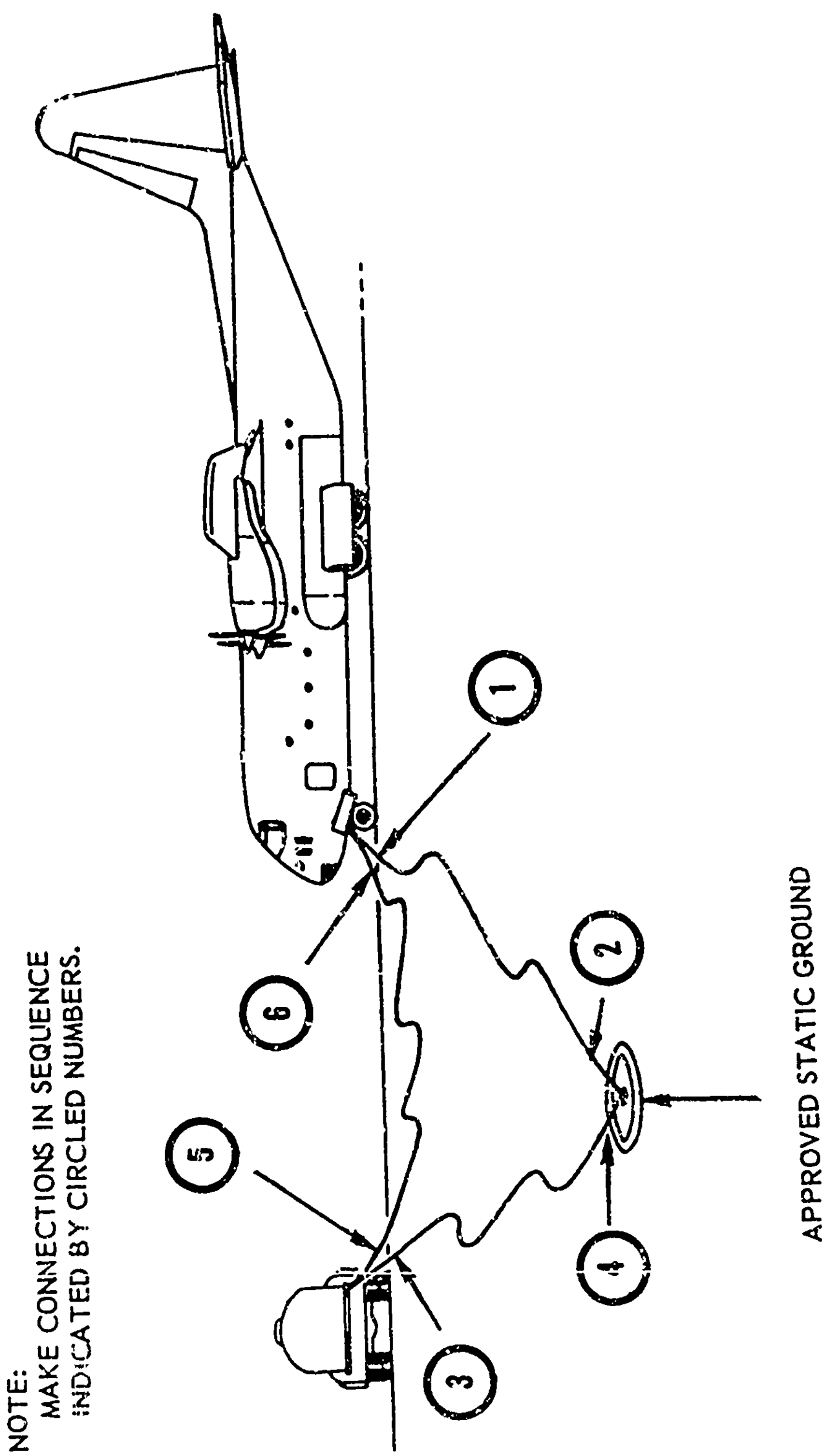



Figure 4. Sequence for Connecting Static Cables for Aircraft and Servicing Vehicle

similar and some mutually common aircraft, whereas the Army will devote their attention to rotary wing aircraft.) The program should have worldwide coordination and expansion to civilian agencies. In the interim, an adaptor will be required to make the system usable at other than AF bases. The next facet of the program will be to include NATO aircraft; decals to identify the static grounding points, as shown in Figure 5, have already been coordinated with NATO for use on aircraft and aerospace ground equipment.

One noteworthy result of the project has been the removal of all landing gear static ground straps installed on Air Force aircraft (Figure 6), which had proved to be of little value in removing static electricity. Their removal has reduced AF costs $637,000 for 1968. The intangible benefits from removing the battery clips from the aircraft system may never be measured; however, if one aircraft accident is avoided by their elimination, the standardized aircraft grounding program will be worthwhile.

This year, both Government and industry have experienced incidents that resulted in loss of life and property, and hazards are being increased in spite of improved fuel handling systems. Desulphurization in refining processes, for example, will increase static electricity hazards by reducing the conductivity of the fuel. Quality control personnel, furthermore, are constantly striving to reduce the moisture and particulate contaminants of the fuel by requiring improved systems using epoxy-coated fuel tanks, which increases static generation and relaxation time. Various types of devices to neutralize the static charge are on the market for systems with low flow rates, but what about flow rates for pipelines and sea-going vessels? And what voltage or charge density is needed to ignite an explosive fuel-air mixture? These questions emphasize the following points:

(1) Little research has been done on conventional and improved systems for bulk handling and storage of aviation fuels.

(2) No standard system exists for measuring electrostatic charges in fuels, such as field strength voltage vs. microcoulombs/$m^3$.

(3) Research and development work is needed in the total electrostatic hazards concept.


Figure 5. NATO Decal

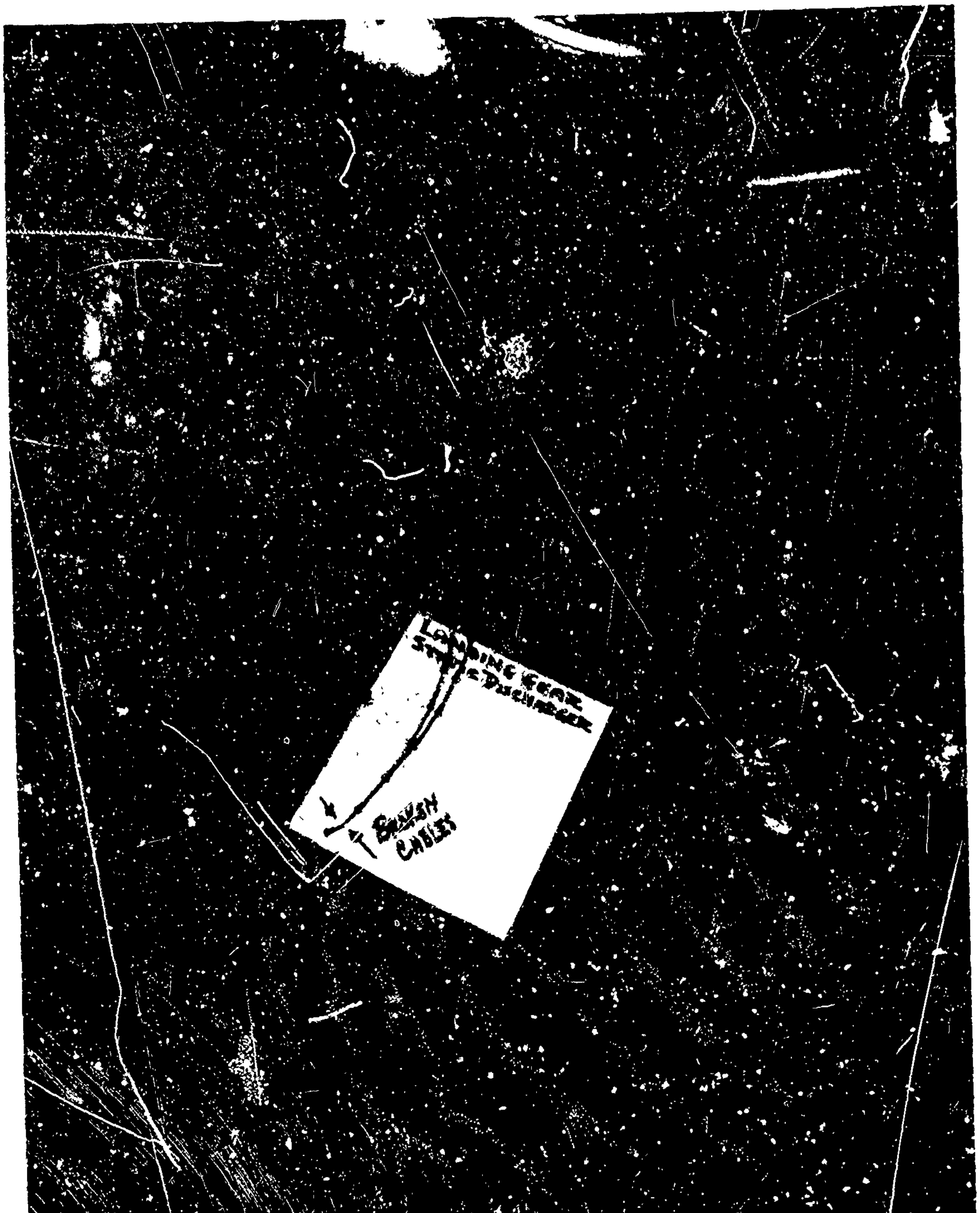



Figure 6. Landing Gear Static Ground Straps

In conclusion, increased refueling rates and improved handling systems will be required for air bus and supersonic type aircraft, which will employ petroleum for fuel and heat transfer. Refueling and defueling procedures will require static charge protection, based upon sound engineering data rather than present state-of-the-art assumptions and individual theories. In my opinion, resolution of the total electrostatic problem requires a joint Goverment-industry effort.

AFAL-TR-68-290
PART II

# A SURVEY OF AIRCRAFT STRUCTURAL BONDING AND GROUNDING FOR LIGHTNING AND STATIC

F. P. Holder

Lockheed-Georgia Company

It is the intent of this paper to survey the design aspects and the significances of aircraft structural bonding and grounding as they relate to lightning and electrostatics.

Often we tend to lose track of the fact that the structure we are designing has anything remotely to do with anything electrical. We have been designing airplanes for years, and we know they have been good airplanes. Certainly very few crash investigations have ever turned up lightning as the probable cause. Then why such emphasis on lightning these last few years? What is changing? A number of things.

## REASONS FOR EMPHASIS

For one thing, we are becoming more and more adventuresome with respect to the weather. Granted that airplanes have historically been able to cope with weather so long as they have stayed out of severe turbulence. And granted that we have more and better radio and nav-aids than we ever had before, and better landing aids. The whole idea is to get more use out of these costly birds by flying them in adverse weather. But just as advancements in communications and in navigation aids are essential for adverse-weather flying, so are improvements in lightning protection.

Another reason for increased stress on lightning protection is that we are getting into the use of materials that we never dreamed of using a few years ago. Presently, at least, these materials may make it a great deal harder to protect aircraft from lightning. Some of these materials when improperly employed, like fiberglass-faced, aluminum-core honeycomb, can react like a small stick of dynamite when struck by lightning, as illustrated in Figure 1. Others, like fiberglass laminate, simply look, as if they weren't there electrically. There was a time when wiring and hydraulic lines and components and people inside almost

Figure 1. Damage From Single 200,000-Ampere Pulse to Fiberglass-Faced Aluminum-Core Honeycomb, With Current Return Through Sparking at Panel Edge

any part of an airplane were well shielded. If lightning were to strike the relatively stout well-bonded, aluminum skin of an airplane, it might burn a few small holes, but the currents would be confined mostly to the outer surface of the airplane, and usually not much of anything would happen more than to give people a good scare, if they were even aware the airplane had been hit. With fiberglass, however, lightning can whip through to whatever is on the inside. Or maybe we're still more advanced and are using some of the exotic fiber-reinforced composites. These fibers are strong as they can be, but they're conductive and they're bound together, usually, with epoxy or some other plastic. But thinly buried conductors like these fibers or the core of fiberglass-faced, metal-core honeycomb may act as a reservoir of charge, which can migrate inward after developing on the skin as a result of friction charging. If every fiber in the composite or every metal ribbon in the honeycomb core is not grounded, sparkover to ground may result and produce radio noise. But what is worse, one good stroke of lightning can produce damage ranging from a hole the diameter of a pencil to loss of an entire panel or radome, especially if wind forces get involved.

Here are several more factors that make lightning considerations more important. There are faying-surface seals and wet-installed fasteners. There's installation of fasteners in dome nuts and the inadequacy of electrical bonding and spark seal-off. There are cantilever-action, clamped-in-place closures for wing openings to avoid stress risers. There is so-called "reverse bonding" of flame-sprayed honeycomb which seems to be great for covering fiberglass with metal but may present problems in trying to provide a really good electrical bond of the flame-sprayed metal surface to the structure or to other flame-sprayed aerodynamic parts. There are new and better finishes that play hob with electrical bonding efforts, and some that are not so new are simply used with more discrimination; we now are careful to put them on everything.

BEWARE OF ANODIZING

The product of anodizing is very hard, tough, and scrub-resistant. It is essentially the same material used as the abrasive for certain high-grade sandpapers. It is a first-rate insulator! Don't use anodizing in a joint intended to carry lightning current! Otherwise, you may expect the current to cause severe sparking in the joint and damage to the contact surfaces. And you should not be surprised if sparking also occurs at some other places. For corrosion control, use conversion coatings such as iriditing or alodining, and make maximum use of fillet seals, post-assembly brushed seals, and the like.

## WEIGHT SAVING VS. BONDING

For weight saving, lighter and lighter weight parts are a perpetual temptation. Magnesium is an interesting material for these but has to be used with discretion and generally should not be used in lightning-vulnerable fuel areas. It will not necessarily set fire to the airplane if struck by lightning, but what about bonding and grounding? The usual: Get good electrical contact in the lightning-current paths while maintaining compatibility of interfacing metals; seal off the bonded areas well to ensure that they will not corrode; and make the bond cross section's minimum area and width greater than for aluminum (which is to be discussed later) by at least a ratio of about 1.6 to 1, the ratio of the conductivity of aluminum to that of magnesium. Hence, the minimum area should be at least 0.16 square inch and the minimum cross-section width about 0.08 inch. Actually, because of the different thermal conductivity in the joint and hence the different heat-sink capability — depending on what interfacing metal is used -- the joint cross-sectional area and width should be modified still further. In any case the joint should be adequately tested. If it is desired only to provide a low-current wire ground to magnesium, the main concern is to make sure of a very thorough sealing job. The wire can be welded to the magnesium if desired.

Titanium is even more in the limelight than magnesium. Again, the requirements for adequate lightning bonding are not well known and any proposed bonding joint should be amply tested. As with magnesium, the electrical and thermal conductivity are so much lower than for aluminum that cross sections should be generously bigger. In favor of titanium as compared to aluminum, however, is its much higher melting temperature — if you can stand the higher temperature. Its melting temperature is about 3000°F, compared to about 1218°F for aluminum.

Before we leave the topic of lightweight parts, let us mention bonding jumpers. Recent investigations indicate that aluminum bonding jumpers are not necessarily taboo, as quick disconnect jumpers are. However, the biggest bonding jumpers presently called out on MS25083 — even copper ones — are inadequate, even in pairs as prescribed by the MIL-Spec., to handle a 200,000-ampere lightning stroke without some assistance from parallel paths due to structure or machinery. A single MS-25093 aluminum jumper wire and integrally formed lugs is good for a lightning-pulse of only a little over 60,000 amperes when new. When two or more jumpers are used to ground a single item, the lightning currents will not necessarily divide equally between them. If the point of contact is much nearer one of the jumpers, the total current-path impedance via the more remote jumpers will be much higher than via the nearby one; therefore, a large percentage of the lightning current may flow through the nearest jumper. Thus, each jumper should be big enough to handle almost all the current by itself.

Why, then, do we need more than one jumper? For several reasons: redundance; to take some of the current and to relieve the first jumper to some extent of the terrific magnetomechanical forces tending to break it or tear it out of its lugs; and to reduce the maximum transient voltage drops appearing between the metal framework and nearby electronic or electrical components and wiring.

NON-ELECTRICAL BONDING

Something now being done extensively to improve producibility as well as to save weight is adhesive bonding. It is not to be confused with electrical bonding. Have you ever checked the electrical resistance between the two faces of different kinds of metal-faced, metal-core honeycomb? This material is usually adhesively bonded. Some of it shows an absolute open circuit unless it is made to withstand high temperatures or the saw-cuts at the edge of the sample have caused burrs that jumper the cement. Be sure to provide adequate electrical bonding across any adhesively bonded joints that ought to be conductive but may not be. With metal-faced honeycomb, you may be able to accomplish this at the edges or closures. Or you may have to use metal inserts or bushings between the faces. An electrical path is needed, whether for lightning, electrical power, static electricity, or whatever.

Here are two cases where static electricity may present a problem because of a little adhesive. Suppose there is an adhesively fabricated stiffening web or access door panel inside an integral fuel tank. Fuel hitting the insulated surface during refueling or sloshing can cause the surface to become electrically charged until it sparks over to grounded structure nearby. Obviously, this is a hazard. On the airplane exterior this can cause radio noise — lots of it. Suppose the airplane has, cemented on its skin, some metallic decorative trim or some lightning shields known as "armor plate." These pieces of metal may be no bigger than a saucer, but under conditions of precipitation-static charging their electrical potential relative to the airplane can climb so high that it sparks over to the airplane and creates radio noise. When this sparking occurs rapidly, the interference may become so severe that it blocks out much of the airplane's radio and navigation equipment. Always ground all such isolated, chargeable metal surfaces.

ANTISTATIC COATINGS

Similar considerations apply to conductive antistatic coatings on radomes and other exposed, insulating structure. Some engineers don't realize that a conductive antistatic coating will not work if it is not grounded; they know the coating is used to bleed off static electricity so as to eliminate corona and avoid radio interference, yet don't realize that if it is not grounded, they may be worse off with than without it. When an uncoated nonconductive surface charges up by friction, with ice crystals, or whatever, an almost infinite number of individual spots all over the surface receive a high potential with respect to the airplane, each other, or the surrounding air. When the number of volts difference per inch of air becomes so great that the air is overstressed electrically, electrical breakdown in the form of corona occurs. As the air adjacent to each little spot breaks down, there is no free flow of charge to the spot from all the other spots over the plastic surface. The current in that particular little burst of discharge therefore, is relatively small. Since all the spots are engaged in the same process in a somewhat random way, the net rate of change of current is much lower than it would be if fairly free charge flow were possible. So, although the radio-noise energy may be substantial at lower radio frequencies, the distribution tapers off so that it is not likely to present a problem above the VHF region — that is, above about 200 megaHertz.

If the surface has a somewhat conductive coating that is not grounded, however, the situation changes. Since all the spots are electrically connected together, they tend to assume identical electrical potentials with respect to each other. Sooner or later, the potential of the whole surface becomes high enough that there is breakdown to aircraft structure. Now there can be a copious flow of charge, and radio noise in the UHF region or beyond, as well as high-intensity noise at lower frequencies. Avoid this by grounding all anti-static coatings.

CURRENT-RETURN PATHS

In general, the best way to accomplish grounding of the antistatic coating on a plastic part is to make the coating continuous around the edge of the part and onto the inside surface where the part is pressed against grounded metal structure. Be sure the metal structure is conductively finished so that the electrical path is complete. Do not assume that merely painting a conductive coating over the heads of metal fasteners is good enough. There are two reasons why it isn't: (1) coatings tend to crack around the edges of such fasteners, because of deformation of the plastic part or dissimilar expansion and contraction, and (2) the fasteners may be removed for quick maintenance and reinserted without touch-up.

Even without cracking or if the coating were run down into the countersunk holes under the fastener heads, it is much better to get a uniform contact all around the edges of the part. Here's why.

Let's assume the coating resistivity to be quite high. Ideally this would be the case for a radome, so as to minimize radio-frequency losses for antennas looking through the radome. The coating works, remember, because it conducts charge to aircraft structure from all points on the radome as fast as the charge develops. If charge is funneled into the fasteners, the current density in these regions is relatively high, so the voltage drop between two points along the current path near the fasteners will be relatively great. Hence, there is a pronounced tendency for the air near the fasteners to be overstressed and for corona to form. And, of course, corona leads to radio noise.

Now back to lightning. The problem of how to electrically bond reverse-bonded, flame-sprayed surfaces to structure was mentioned earlier. Any method you devise is likely to be a little tricky. If the method leads to current concentrations, a single, fairly moderate lightning strike may necessitate the repair of a number of places in addition to the lightning contact point. Let me explain.

Suppose a lightning restrike hits the middle of a flame-sprayed fiberglass panel that is screwed through metal inserts to metal structure at points a few inches apart all around the edges. The metal in the area around the lightning contact point will simply evaporate — explode — for a diameter of up to several inches, and the fiberglass underneath will char and maybe rupture, depending on thickness, configuration, etc. But there will be an end to the area denuded of metal. One reason is that the edges of the burn, being farthest from the core of the stroke, do not receive much heat from it. But the main reason is this: At the instant restrike begins, the metal at the contact point is vaporized and forms a little conductive cloud, which permits the lightning current to spread out and contact the edges of the growing bare spot. As the spot grows, three things happen: (1) the current density in the metal drops rapidly, and since the heating effect of the current in the metal is proportional to the square of the current, the heating effect drops rapidly indeed; (2) the heat sink grows as the area gets bigger, and makes it harder for a given amount of total heat to cause evaporation; and (3) since time is passing, the amplitude of the lightning current begins to fall fairly rapidly.

Now remember, this current has to leave the panel somewhere; we have postulated that the panel is screwed down and gets electrical contact through inserts at all the fasteners. So the currents that spread out at the strike point now bunch together so as to funnel through the fasteners. This raises the current density around each fastener. The heating effect — which is proportional to the square of the current density — shoots up, and metal disappears from around some and maybe all of the fasteners. If the outflowing current had been distributed evenly around the edges, only one big patch would be needed instead of the big one and smaller ones. It would have made little difference if the thin metal surface had been painted, plated, or cemented on; the principle would have been the same.

You may ask, "Why use inserts? Couldn't you just lap the metal around the edges of the panel and clamp it in the joint?" Yes, you could with the cemented foil or the painted or plated coating. But with the reverse-bonding method of flame spraying this lapping presents a problem of its own.

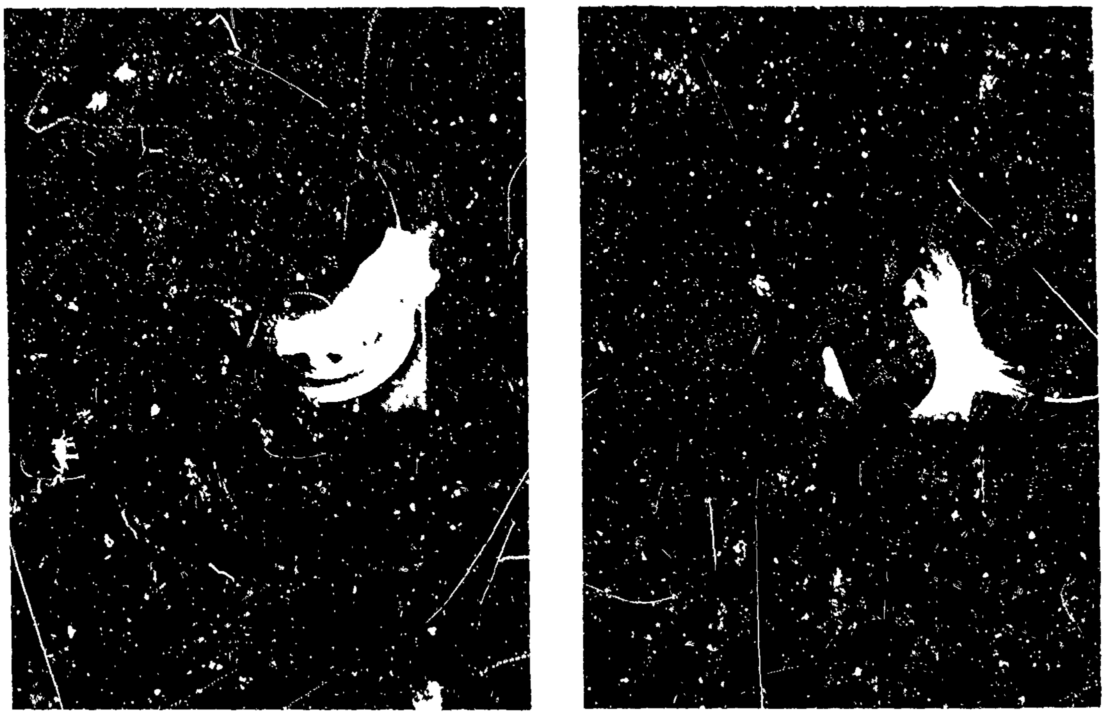
NONMETALS IN COMPRESSION

Also, there is a specification to consider. As you know, the aerospace bonding and lightning-protection document, MIL-B-5087, says "Bonding connections shall not be compression fastened through nonmetallic materials." Exactly what does that mean? Here is an illustration.

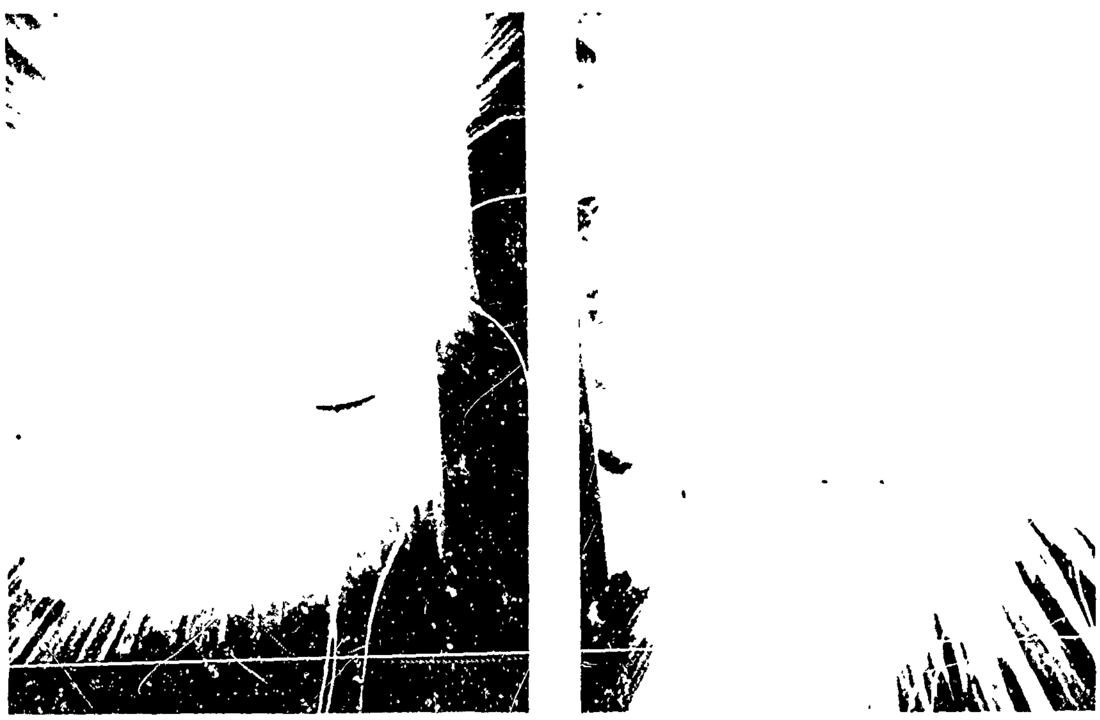
Let's say you are replacing the plug on the cord of your wife's electric iron. For some unexplained reason, you put fiber washers under the screw heads and tighten securely. Everything is fine for a week, when your wife complains that the plug is getting hot. You investigate and find that the wire insulation near the plug is overheated and the screw has turned blue. Why? Because the fiber washers compressed, cold-flowed, or dried out, the contact resistance in the electrical connections increased tremendously, and the electrical heating was severe. If lightning current, as much as it is, were passed through a similar joint, the joint would probably spark, and in lightning parlance a spark may be a blinding flash. (See Figure 2.) The sudden terrific heating vaporizes some of the metal and this white-hot vaporized or molten metal blows outward from the connection.

The same thing can happen in a lightning bond juncture, whether it is at the edge of a metal-faced fiberglass panel or at the wing interface of a fuel-bay access door inside a wing tank, if the tightness of the junctures depends on the ability of nonmetals to withstand compressive force without cold flow, crushing, or the like. The voltage drop across such a joint doesn't have to be very much; a few volts can cause metal sparking. Just remember what happens when you accidentally short-circuit your twelve-volt automobile battery.

Some people say the "nonmetals-in-compression" requirement in MIL-B-5087 doesn't apply to lightning -- that it really refers to electrical power paths, especially since these may have to handle sizable fault currents without overheating or exploding. The answer here is, it depends on who you talk with and who decides whether you have met your contractual requirements. There are almost certainly structural bonding situations for which a specification deviation would be granted if required to resolve an otherwise knotty design problem. An example might be the bonding of a radome external lightning diverter system through the radome fiberglass-laminate wall to the electrical tie-in



a. 100,000-Ampere Pulse to Center of Head



b. 200,000-Ampere Pulse to Retainer Ring

Figure 2. Severe Sparking at Simulated Fuel Probe

to structure. Most radomes are so located that no fuel vapors or other explosives are likely to be ignited by sparking. On the other hand, we might think fuel-area closures are among the cases for which a "nonmetals-in-compression" deviation would never be granted. This is not necessarily true, and this can easily arise because of the necessity to compromise to solve fuel-sealing problems. The rule is, if you cannot avoid a possible sparking situation, make your design such that the sparking will occur in the wind-stream or some other area where there will be no explosive or flammable vapors or other materials.

HARDWARE CONFIGURING

Since we have mentioned fuel areas, let's go a little further and come up with some more rules. Here's one. Avoid having any sharp point, edge, or "outside" corner wherever sparking cannot be tolerated, where the gap is small between the corner and a nearby part that may be at an appreciably different voltage, and where the voltage across the gap may possibly exceed about 300 volts. How high the voltage may become is best determined by high-current impulse lightning tests.

Here's another rule, an empirical one. Always try to be absolutely sure you will have a useful cross section of not less than 0.1 square inch (in the case of aluminum) for any lightning path that involves a region where you cannot tolerate sparking. It is best also, for reasons of mechanical strength, heating, and impedance that this not be a long, thin cross section. The overall cross section can be as long as you like, but, somewhat arbitrarily, it is desirable to have a minimum cross-section width of about 0.05 inch (again, for aluminum). Following this rule you will achieve the 0.1 square inch cross-sectional area criterion about every three inches around the periphery of any fuel-area door, allowing something for nonconformity between the mating surfaces in the contact joint. The purpose is to prevent a strike near the edge of the door from causing local vaporization of metal in the contact area.

Another rule to remember is, always keep lightning-current flow as nearly as possible on the outside surface. That is where the current wants to flow anyway, because of skin effect. Also, always avoid having current flow from a cover plate or retaining ring through fasteners, then outward again to the wing or tank skin. This is especially true if there are internally exposed metal-to-metal junctures that may be part of a lightning-current path, since the metal sparking already discussed may occur, but there is another reason. Whenever there is an "inside" turn or bend — a fillet or corner — in a current path, there is inductance, L. Whenever there is an instant-by-instant change in the current flow — that is, the derivative di/dt — through the inductance, a voltage, e, is developed across the ends of the inductance. The magnitude of that voltage is

$$e = L\frac{di}{dt}$$

The inductance around the corner of a door-mount inside a tank may be very small, but the possible rate of change of current may be extremely high; it is known to get as high as $10^{11}$ amperes per second. The voltage, e, may become high enough from lightning current flowing around the small corner that the air in the fillet area is overstressed, electrically, and breaks down in the form of a spark, especially at high altitudes — and this spark can easily have sufficient energy to ignite fuel vapors. So another rule emerges, namely: stay away from designs that cause current to flow around inside corners in areas where sparking or corona cannot be tolerated.

SPARK SEALING

If you cannot avoid having a current path around an "inside" corner in a fuel vapor area, you should fill the corner with a carefully applied, void-free, nonconductive fillet seal or its equivalent. This can withstand far higher voltages than air can. In general, do not use conductive cements, sealants, or gaskets in joints where sparking matters. The current path formed by such a material is likely to be very unevenly distributed, of relatively high resistance, of inadequate heat-dissipating capacity, or any combination of these. So the material itself tends to spark.

Never depend on a fillet seal as your primary protection from a spark causing fuel-vapor ignition at a bonding joint. If you must have a lightning-current path across the juncture between two faying surfaces in a fuel-vapor area, let there be a very thin fissure at least 1/4 inch across, between the edges of the metal-to-metal juncture and the vapor area. Have this fissure packed with some type of physically and electrically strong insulating material — for example, MIL-S-8802 sealant or a clamped-in-place cork-and-rubber gasket material such as Lockheed has found works well. Such a wide barrier is needed between a spark source and a vapor region because a spark develops tremendous pressures that tend to blow out or through or past a seal. This is illustrated in Figure 3.

This undependability of sealants for lightning protection is one of the main reasons why aircraft floating-type dome nuts that may have to carry lightning currents may be unsafe in a fuel area. If lightning current flows into a floating dome nut through a threaded fastener, sparking may occur in the threads. Or sparking may occur in joints within the dome nut structure itself. Merely looking at such a dome nut without cutting it apart can be deceptive. In one common type of self-sealing dome nut, the current path is found to be from the screw into the thread, then radially outward into its outer shell, through the tiny rivets, to shell bearing area under the rivets, and then to the wing or other mounting surface. Rarely does an aircraft dome nut not spark visibly if it has to handle most of the current of a direct lightning strike to the head of the mating fastener into the nut. If you must use dome nuts, let them be as large and substantial as possible, then butter the mounting cracks and rivets generously with MIL-S-8802 or equivalent sealant, as shown in Figure 4. Then test the arrangement carefully to assure that the dome nut with the sealant to back it up will not allow any sparking to escape. You may find that tedious and stringent quality control is your biggest problem as it is in many lightning-protection designs.

FAYING-SURFACE SEALING

For about five years, now, there has been considerable interest in what faying-surface sealing and so-called "wet" fasteners do to the lightning-strike integrity of an airplane structure, especially in the fuel areas. In 1965 Lockheed and LTRI joined forces in a small program to find out. Here essentially is a summary of that work.

"The purpose of this investigation was to determine the effects of environmental sealing practices on the electrical bonding and lightning protection characteristics of airframe joints. Two sets of panels having joints similar to

Figure 3. Surfaces of Simulated Fuel Probe Where Sealant Failed to Contain Sparking

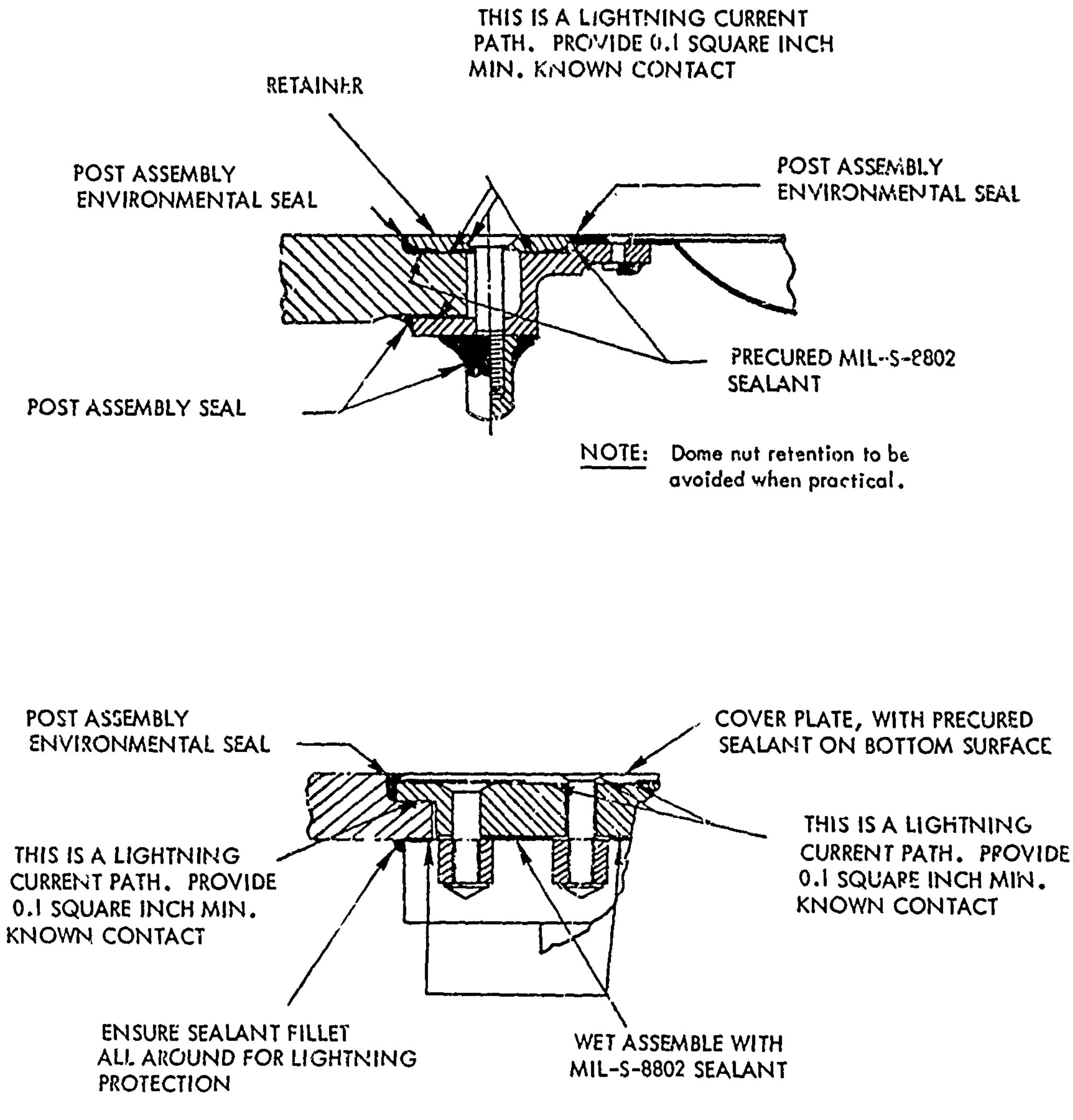



Figure 4. Access Door Retention Using Dome Nuts (Preferably) or Buried Inserts

airframe joints were constructed. One set was sent to the Lightning and Transients Research Institute where the panels were each hit with a simulated lightning strike and the joints observed to see if they sparked. In addition, panels simulating C-141 wing tank joints were tested for joint sparking. The second set of panels was tested by Lochkeed to determine the effect of the environmental sealing on electrical bonding at aircraft power frequencies.

".....Of the first set of 13 panels, all in the 'new' condition, one sparked when hit with a simulated lightning strike. Of the 13 environmentally aged panels, seven sparked when struck. None of the panels constructed without the use of sealant sparked, either before or after aging, and none of the panels simulating the C-141A wing joint sparked before or after aging. The significant facts obtained from the lightning tests are:

1. The use of environmental sealing techniques increases the tendency of skin joints to spark.

2. Environmental aging greatly increases the incidence of joint sparking.

3. The mid-wing splice panels and the wingtip bulkhead to panel joint samples did not spark.

4. Skin joint sparking is not confined to any particular fastener type or sealant application.

5. A thin coat of MIL-S-8802 sealant on the interior surface of the skin did not cause a significant reduction in joint sparking. (In earlier tests on C-141 access doors, heavy sealant did inhibit sparking.)

6. Resistance measurements, ac or dc, do not predict which joint will spark.

".....The significant facts obtained from (the electrical bonding) tests are:

1. The use of environmental sealing techniques did not affect the low-frequency electrical bonding characteristics of the specimens.

2. Aging of the specimens caused their electrical resistances to increase between 10 and 100 percent. The specimens constructed without sealant in the joint and the specimen constructed with conductive sealant in the joint showed the lowest percentage increase in resistance after aging.

3. The current through the joints apparently flows through the fasteners and not through the joint itself. When conductive sealant was used, the current still flowed mainly through the fasteners.

4. The decreased conductivity of the joints after aging seems to be caused by some of the fasteners in the panels ceasing to conduct current, rather than by decreased conductivity of all the fasteners.

"From these results herein described it is concluded that the use of environmental sealing techniques in skin joint construction increases the tendency of the joints to spark when hit with a lightning strike. It should be emphasized that the proper use of sealants in the long run may greatly reduce sparking tendency. The use of environmental sealing techniques does not materially affect the electrical bonding of the joints from a power path standpoint."

STATIC DISCHARGERS

One more small but sometimes bothersome point has to do with bonding precipitation-static dischargers to aircraft skin. LTRI has cited the figure of 8 sq. in. as being advisable for the minimum bond area for a discharger that is to be fastened on with silver-epoxy conductive cement. Where rivets are also used, this figure can probably be reduced. Personally, I have yet to see a discharger with a base area of 8 sq. in. Regardless, if a heavy lightning strike should contact and hang onto the resistance-rod type of discharger, there is a fair chance the mounting base will be damaged and need replacing. So two things are especially important, aside from the manufacturer's application notes. First, try to make your design, including any fasteners, such that complete replacement will be relatively easy. Second, try to make the structure holding the discharger sufficiently rugged that it will withstand a blast that will blow off the discharger when the conductive adhesive under it evaporates. After that, good luck.

RESISTANCE MEASUREMENTS

Let me say something about how we at Lockheed-Georgia use resistance measurements in determining the lightning-current adequacy of electrically bonded joints. You have heard that the mere fact that a joint has a very low resistance is no assurance that it can handle lightning currents. We agree with that, but we say, "If a joint supposed to be conducting does not have a very low resistance it probably cannot handle lightning currents." What we do is this. We make extensive bonding measurements on each test sample before it is to be struck. We make any feasible measurement that could possibly affect lightning integrity. For example, we measure from door panel to retainer ring, retainer ring to wing, door panel to wing, and fasteners to all of these. When the specimen passes lightning tests, we have control information for checking production hardware. If some stray sealant, anodizing, or inadequate fit creeps into production, the change in resistance is usually so dramatic that we can spot a problem right away. In other words, we use resistance measurement for inspection, even of precipitation-static discharger installations. X-ray would usually not do as well.

WORKING PHILOSOPHIES

Throughout this paper you may have detected an underlying philosophy regarding designing for flight safety and maintenance. It is a way of life at Lockheed Georgia — and probably in your companies, too — that if anything jeopardizes safety of the aircraft or its occupants, we do whatever is necessary and available within the state of the art to eliminate the hazard. Certainly we try to select the alternative most compatible with all other considerations, including economics. On the other hand, many questions of lightning protection regarding bonding and grounding do not really involve safety of flight. These questions are harder to answer because we must weigh the probability of lightning or static effects causing some particular problem and the penalty in weight, cost, schedule delay, or decrease in overall reliability resulting from doing something now, against the consequences of waiting to see whether the thing really needs doing. I am referring to those things not already defined by the contract. We have no pat, general solution for this type of dilemma; we wish we did.

So, this paper ends on the note that the problems of structural bonding and grounding for lightning and static prevention are many, often complex, and most certainly varied. The lightning-protection engineers cannot hope to fulfill their obligations without the informed assistance of the structural designers. I feel confident I speak for every lightning-protection engineer in saying that assistance is earnestly solicited.

# PRACTICAL PROBLEMS ASSOCIATED WITH THE EARTHING OF AIRCRAFT

C. W. Cornish, Squadron Leader, RAF

Ministry of Defence

United Kingdom military aircraft are earthed (grounded) for two specific reasons:

a. To provide operator protection when voltages at a lethal level are supplied to the aircraft from an external source. Fault conditions in equipment can create a substantial difference in the voltage level in the airframe relative to the earth lines. A high impedance in the neutral/negative line from the ground power source can give rise to this condition.

b. To afford protection to parts of the weapon system, particularly electrically initiated explosive devices, during loading/unloading operations. The earthing in this instance is directed entirely to the equalising of static charges that are produced on airframe, weapons, weapon carriers, etc., since loading drills preclude the use of all forms of electrical power on aircraft during weapon unloading.

Voltages supplied to aircraft by ground power units can be of the order of 28 volt dc, 112 volt dc, and 200-volt 3-phase 400 Hz ac.

Lethal level voltages as defined by United Kingdom regulations and the Institute of Electrical Engineers are 30 volt RMS AC and 50 volt dc, and in this the Royal Air Force conforms to the civil requirements. For the earthing of aircraft both company earth (that is, the mains distribution earth) and the Service-defined "true earth" are used. Company earth values are, by UK requirements, lower than 5ohms. The defined "true earth value" is a maximum of 10ohms, and this value is aimed at for both static and lethal voltage earthing. "True earth" points are installed at aircraft dispersal points and are made by the installation of earthing rods in a permanent setting. Where aircraft servicing installations have a power distribution pointed (PDC) and earthing cubicle, the common and true earth in the cubicle are strapped together. The strapping of the two earths is to ensure a common value of earth resistance. It is not always a practical proposition to obtain a true earth value of 10ohms, due to soil conditions, and a relaxation in these cases has to be given. Up to 300ohms can be tolerated since all ground power units in the Royal Air Force have voltage sensing protection devices and Institute and Civil Regulations give a 300ohms maximum permitted value of earth where this type of protection is provided.

The major problem that has arisen in the implementation of the earthing policy has been that of circulating earth currents. The majority of ground power units, static and mobile, which supply power to aircraft have the output negative/neutral or one phase connected to trolley chassis or unit frame. When the GPU is connected to the aircraft, the multiple earth loops between servicing equipment and aircraft are provided, in the main, by earths on individual equipments which are fitted to make each equipment intrinsically safe in its own right. These earth loops carry a proportion of the load current from the ground power unit and constitute a serious potential danger to aircraft and servicing personnel.

Earthing of aircraft in the Royal Air Force is mandatory under the following conditions:

a. At all times during loading/unloading of specific types of weapons.

b. If supplied by voltages from ground power units which are at a lethal level.

c. When undergoing maintenance in hangars or in the open, irrespective of voltage supply of ground power source. This is to protect personnel from fault conditions which may occur in mains operated hand equipment.

The various conditions that may arise can be shown by a series of illustrations. For clarity, only the output and earth leads are shown in the illustrations; the power lines to the prime mover have been omitted. In the first four examples, the aircraft is not separately earthed. In Figure 1, DC output negative is connected to trolley chassis and the negative of the aircraft system is connected to airframe. The trolley is connected to earth by the supply cable from PDC. The aircraft, providing nothing else is touching it or is connected to it, normally has only one connection to earth through a conducting tyre, which can be of comparatively high resistance; therefore, the current flowing in the earth loops formed by the conductive tyre, trolley tow bar, and trolley earth connection, which parallel the line between points A and B in fair weather conditions are negligible. In excessively wet weather, however, both tyres and ground are soaked, and the current flowing in the earth loops will increase considerably.

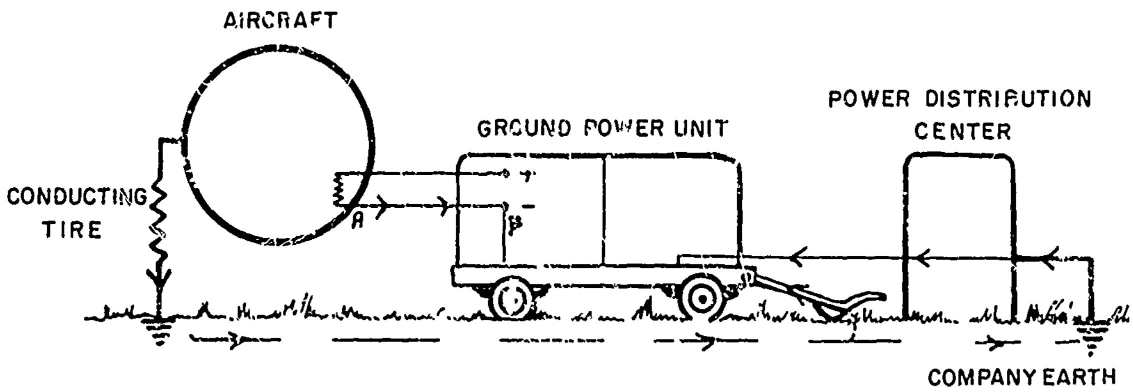



Figure 1. Aircraft Earthed With a DC Electric/Electric Servicing Trolley

In Figure 2, where two electrically operated hand tools are connected to the PDC, low resistance parallel paths to the line between A and B are formed when the tools come in contact with the aircraft skin. The actual current flow will depend, of course, on the resistance of the conductors and the load conditions. Assuming the length of the cables to be the same, with the resistance of the negative line between trolley and aircraft in the order of 0.0086ohms per hundred feet, and assuming the earth line to the hand tool and the trolley to be the equivalent of 37 amp rated cable with a resistance of 0.058ohms per hundred feet, approximately 10% of the load current will flow in the earth line between PDC and trolley and approximately 5% in the earth line to each hand tool. The load current during servicing varies considerably but can be as high as 400 amps; therefore, up to 40 amps can flow in the earth loops. (Under start conditions the load current can peak to 1500 amps.)

The external supply sockets on aircraft are particularly prone to damage because of the nature of their function, with repetative connection and disconnection under all conditions. In these circumstances, the proportion of the load current carried by the earth lead will increase considerably. It is not difficult, therefore, to imagine that using a serviceable flame-proof inspection lamp in the vicinity of fuel tanks could be hazardous under these conditions, where every contact between lamp and aircraft skin makes a circuit carrying considerable current.

For the same conditions described above on aircraft with a mains electrical system which operates at lethal level potential or above, the situation shown in Figure 3 would be impossible.

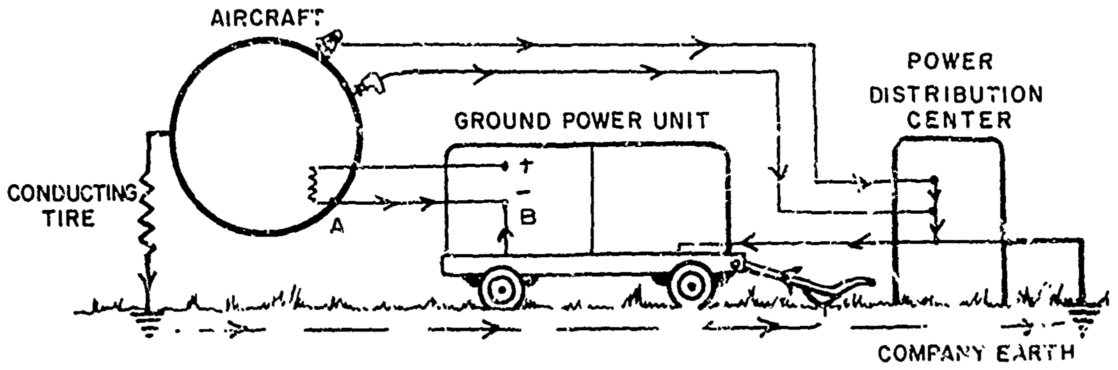



Figure 2. Aircraft Earthed When Two Electrically Operated Hand Tools Are Connected to the PDC

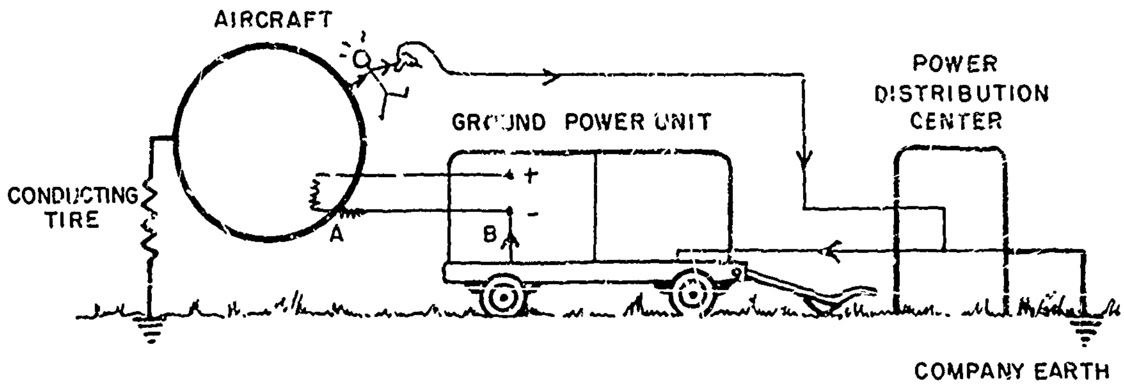



Figure 3. Aircraft With Mains Electrical System at Lethal Potential

On a number of aircraft, electrical power must be on during refuelling to operate fuel cocks, etc. Because the normal standing loads plus refueling loads would discharge aircraft internal batteries, it is common practice to use an external supply. Figure 4 shows the earth loops formed by connecting a refueller to the aircraft while an electrical servicing trolley is connected to the aircraft. The refueller chassis is bonded to the aircraft skin, a conductive strip of fairly low resistance connects the refueller to earth, and the chassis in some instances is also earthed with a lead and earthing spike. The current flow in the line between aircraft and refueller will depend on the weather conditions and the distance between aircraft refueller and service trolley. Normally the earth mass resistance in the loops will make the current flow infinitesimal, but under extremely wet conditions and with a refueller, trolley, and aircraft in fairly close proximity, the current flow may be considerable. A recent example of this hazard has been brought to our notice. A Boeing 707 which had an earth lead connected direct to earth in addition to the rubbing strip on the refueller, was being refuelled. A fault in the aircraft GPU connector resulted in the return current passing down the high pressure refuelling hose bonding wire, via the refueller earth, back to the PDC. Considerable heating of the fuel in the pressure hose occurred before the fault was discovered.

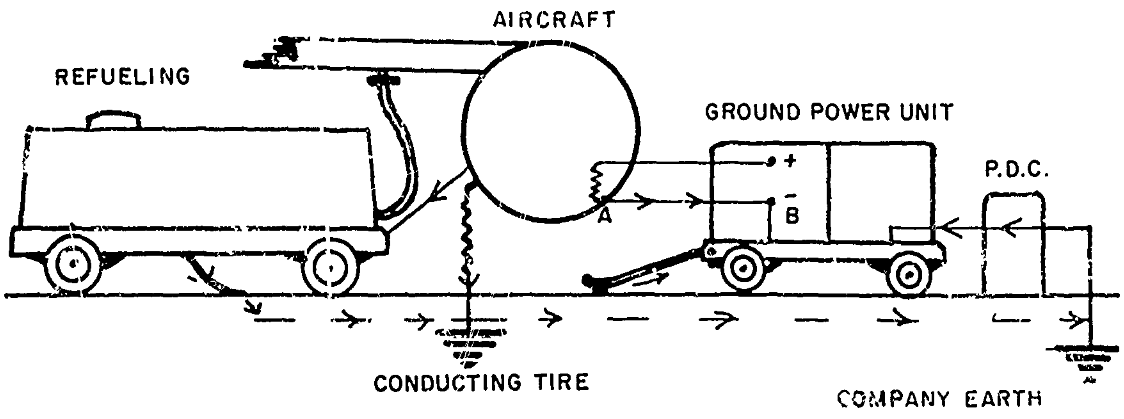



Figure 4. Earth Loop Formed Between Refueller and Aircraft Connected to Electrical Servicing Trolley

Figure 5 shows the earth loops formed when an aircraft is earthed to an earth point and an electrical/electrical servicing trolley is connected; the additional earth loops formed by the use of electrically operated hand tools, shown in Figure 2, also apply in this case.

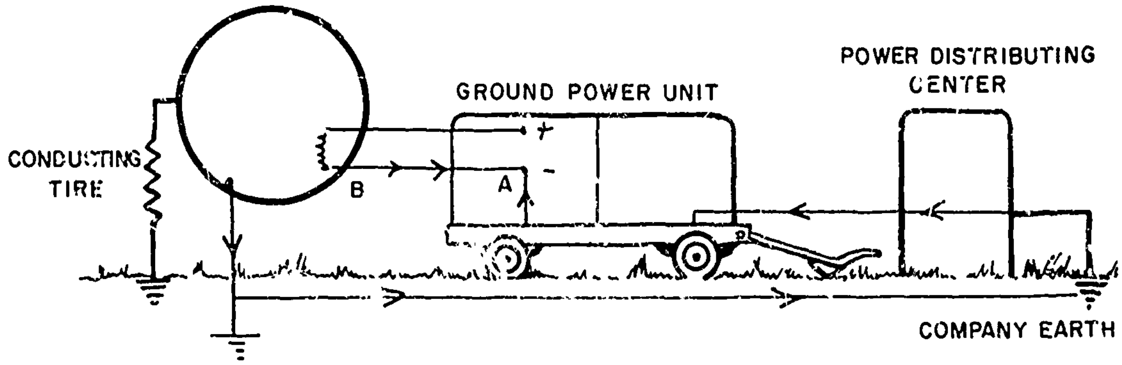



Figure 5. Earth Loops With Aircraft Connected to Earth Point

Figure 6 shows an electrical/electrical-3 phase servicing trolley connected to an aircraft, with neutral connected to trolley chassis and airframe. All the examples of earth loop paralleling the line between points A and B shown in the previous examples also apply to this. Fortunately, the phase loads on most aircraft are fairly well balanced and the current flowing in the neutral line between aircraft and servicing trolley, and thus in the earth loops, is normally small.

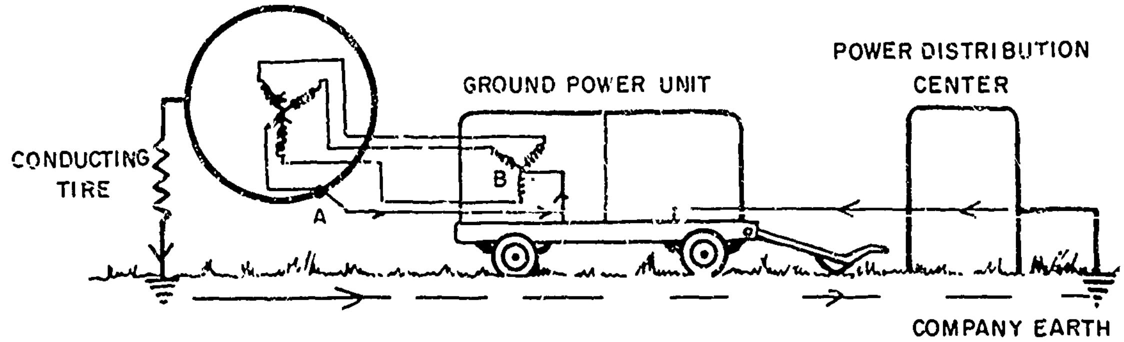



Figure 6. Aircraft With Neutral Connected to Trolley Chassis and Airframe

Figure 7 represents automatic test equipment (A.T.E.) connected to an aircraft with an electrical/electrical servicing trolley connected to the A.T.E. The three-phase electrical system is connected to servicing trolley, the A.T.E. chassis, and airframe. The earth loops will carry a proportion of the phase current. Other earth loops, as shown in the previous figures, would make this situation even more dangerous.

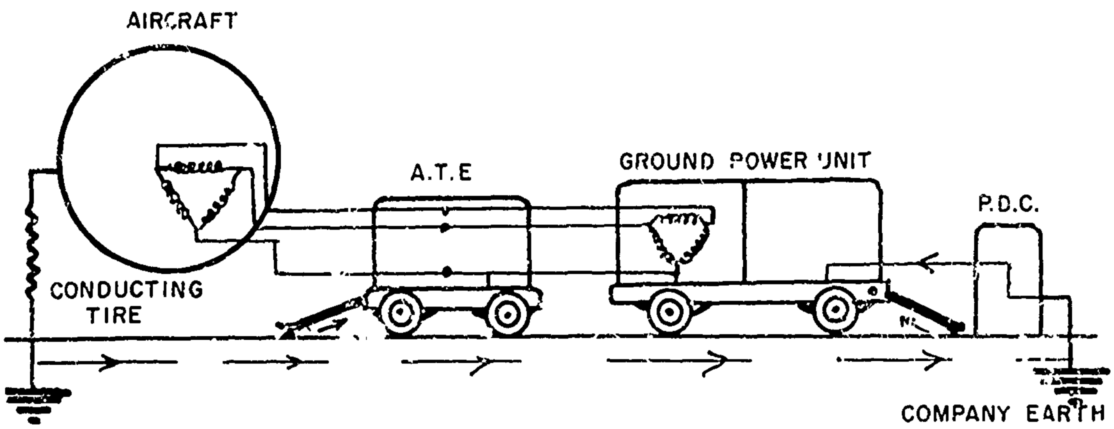



Figure 7. Aircraft With Automatic Test Equipment Connected to Servicing Trolley

The illustrations show just a few of the earth loops which can occur between aircraft and power producing servicing equipment during servicing operations. Many others, including aircraft servicing docks, provide multiple earth loops. It is almost impossible to legislate against this problem without imposing unacceptable restrictions on servicing operations. For example, the use of all mains operated hand tools would have to be stopped while external power is connected to the aircraft.

The long term solution to the problem of the multiple earth path is to ensure that ground power sources have the neutral/negative load isolated from the power unit chassis and to provide a single point earthing for the aircraft and its ground supply. The lifting of the chassis negatives, however, raises the question of operator protection should a line-to-chassis fault occur; this protection can be achieved by the fitting of a detector device to pick up any line-to-earth fault, and trip the power unit supply.

A device, The Earth Fault Monitor, has been developed for use on all types of ground power units, and is undergoing service trials. The device is being developed as a private venture and is subject to patent action; consequently, only brief details have been released for this presentation. The Earth Fault Monitor works on the principle of injecting a high frequency voltage in series with every connection fed out from the ground power unit. Injection is affected by a large toroidal transformer (injection toroid); the secondary windings on this transformer are a straight through feed of all output cables from the ground power unit. The primary of the transformer consists of one turn of wire, fed by the energising oscillator (2.5K Hz oscillator) at approximately 30 volts/turn; i.e., 30 volts is injected in series with each cable, and hence is not detectable in the output. The injection voltage is referenced to earth (chassis) through a 2.5K Hz tuned "accepter" filter. The filters presented a higher impedance to chassis of all other supplies than the 2.5K Hz.

The output cables also all pass through a sensing toroid, which normally experiences a zero net flux condition, as all cables are self-cancelling. In the event of an earth leakage fault in the machine, a current 2.5K Hz can flow through the loops causing a net flux in the sensing toroid at 2.5K Hz. This is detected by the sensing winding, and is filtered and fed to a detector. The detector de-energises a normally operated relay, which in turn opens the output contactor of the machine and also deexcites the alternator.

To build in an extra "fail safe" facility, the power for the conductor is derived from the injection toroid, so that injection must occur before the conductor is operative. As the machine and associated components have a leakage capacity to chassis, which would appear as an earth fault at 2.5K Hz, a capacity compensation circuit is included, to "tune out" the effects of leakage capacity, which will vary on each machine.

It can be seen from this description that any earthing that takes place on the output circuit beyond the 2.5K Hz bandpass filters will have no effect; therefore, earthing of the aircraft can take place with no ill effects from parallel earth paths or loss of the neutral connector connections. Indications at this stage show that the system performs extremely satisfactorily and that high sensitivity can be achieved. Sensitivity at the moment is set to 2K ohms, to reduce a "nuisance tripping" under damp operating conditions.

To cover the interim period until all power units are fully protected, after the lifting of the negatives on ground power units, detailed drills have been issued to cover each specific case; under broad headings they can be summarised as follows:

a. Engine driven GPUs, output negative strapped to chassis: The aircraft is to be earthed but not the ground power unit.

b. Electrically driven GPUs where the negatives are not strapped to the chassis: The aircraft is to be earthed.

c. Electrically driven GPUs where the negative is strapped to the chassis: The aircraft earthing lead must be removed before power is applied to the aircraft; the earthing of the aircraft is provided from the negative/neutral line back to the power distribution point.

The provision of true earthing points and the annual inspection at permanent bases is undertaken by the Ministry of Public Buildings and Works. They are, generally, in the form of earth cubicles. At unprepared bases or where permanent earth points have not been installed, a temporary earthing facility, if this is required, is provided by an earthing rod.

No provision is made for lightning protection by earthing the aircraft in the Royal Air Force. Experience has shown that the discharge path, when an aircraft is struck while on the ground, fails to follow the aircraft earthing lead connected to the ground earthing point. The discharge takes place through the aircraft main wheels.

Discharge paths created by the fitting of conducting tyres are relied upon to dissipate static charges generated on the aircraft during flight and on the ground. Where static charges are created during replenishing operations involving hazardous liquids, e.g., aircraft fuels, LOX, etc. the replenishing vehicle is bonded to the aircraft before operations commence. The servicing earth point on the aircraft and the vehicle bonding lead are used to achieve this bond. No separate earthing of the replenishing vehicle to the true earth point is undertaken.

To sum up, while all is done that can be done to create safe conditions for operating aircraft and aircraft ground power, by providing suitable earthing for all equipment, in the final analysis, adherence to strict servicing procedures and the education of personnel in understanding the problems and importance of earthing are our best insurance.

# SESSION VI

# NONCONDUCTIVE NONMETALLIC MATERIALS

J. L. Moe

General Dynamics Corporation

# INVESTIGATION OF LIGHTNING STRIKE DAMAGE TO EPOXY LAMINATES REINFORCED WITH BORON AND HIGH MODULUS GRAPHITE FIBERS

L. G. Kelly, Air Force Flight Dynamics Laboratory

H. S. Schwartz, Air Force Materials Laboratory

Boron-fiber-reinforced plastics and high modulus graphite-fiber-reinforced plastics are two recently developed "advanced composite" structural materials having higher modulus-to-density and strength-to-density ratios than the conventional lightweight metals used for aircraft structures, such as aluminum and titanium. Based on predictions of significant weight savings to be gained by using these materials in aircraft, an advanced development program was initiated in 1964 by the Air Force Materials Laboratory to design, fabricate, and test prototype aerospace structural components such as the wings, fuselage, and empennage from this material. One of these structural components, the horizontal stabilizer for the F-111 aircraft, is scheduled for flight test in June 1969. Other structural components constructed of boron-fiber-reinforced plastic for flight tests are shown in Table I. Flight tests of the three F-111 components have been flight tested for about 18 months, while tests of the other components have been initiated more recently.

TABLE I

BORON-FIBER-REINFORCED PLASTIC AIRCRAFT COMPONENTS UNDERGOING FLIGHT TEST

| | | | |
|---|---|---|---|
| F-111 | Airflow director panel | F-5 | Torque tube access door |
| F-111 | Wing trailing edge panel | F-104 | Fire access door |
| F-111 | Aft main landing gear door | RA-5C | Antenna cover |
| A-4 | Landing flap | F-4 | Rudder |
| C-141 | Wing tip | F-4 | Stabilizer leading edge cover |
| F-5 | Main landing gear strut door | A-6A | Wing fence |

The development of high modulus graphite-fiber-reinforced plastics for prototype aerospace structural components is not as far along as that of boron-fiber-reinforced plastics from the standpoint of number of components, size, and complexity. One of the graphite-fiber-plastic components that has been fabricated for flight tests, however, is a wing leading edge section for the F-5 aircraft. A simulated fuselage configuration of stiffened shell design measuring 4 x 2 feet has also been fabricated and will be structurally tested.

Boron and graphite-fiber-reinforced plastics both have their advantages, property-wise and fabrication-wise, for specific uses, and it is therefore anticipated that both materials will be used in future aircraft structures.

## LIGHTNING STRIKE HAZARD

One of the in-flight hazards for aircraft is a lightning strike. An aircraft structure of aluminum provides a conductive path for the lightning current, so that little structural damage occurs, and what does occur is usually not catastrophic. However, when nonconductive external parts such as glass-fiber-reinforced-plastic radomes are struck, lightning damage can be so serious as to destroy the entire component.

Boron-fiber-reinforced plastics are relatively poor electrical conductors and graphite-fiber-reinforced plastics are only fair; their response to lightning strikes, therefore, is of interest (Reference 1). Tests to determine this response were conducted with simulated lightning strikes by the Ohio State University Electrical Engineering Department and Lightning and Transients Research Institute in cooperation with the Air Force Materials Laboratory, Air Force Avionics Laboratory, and the Naval Air Systems Command. The results of these tests are presented in this paper, we hope that this information will be used as a point of departure for more comprehensive investigations of lightning strike protection for advanced composite structures.

## LIGHTNING STRIKE TESTS ON BORON AND GRAPHITE FIBER REINFORCED PLASTICS

Lightning strike tests were performed as two separate but related programs with two principal objectives in mind: (1) to investigate the response of boron-fiber-reinforced epoxy as discrete laminates and as laminate skins in sandwich construction in support of the F-111 horizontal stabilizer program, and if lightning strike protection is found necessary, to devise or recommend suitable methods of protection; and (2) to investigate the effect of lightning strike damage to fiber-reinforced plastic laminates. The laminates investigated in this program were thin sheets (0.040" thick) of glass, boron, and graphite-fiber-reinforced epoxy, and aluminum sheet was used as a reference material. The first was a more specific, short-term program, and the latter was more general and exploratory.

## LIGHTNING STRIKE TESTS IN SUPPORT OF THE BORON HORIZONTAL STABILIZER FOR THE F-111

The boron-fiber-epoxy horizontal stabilizer (Figure 1) consists of a center box section, a tip, a leading edge, and a trailing edge constructed of aluminum honeycomb core and boron epoxy face sheets tapering from 0.18 inches to 0.016 inches. The stabilizer is designed to both strength and stiffness requirements and represents a 27% weight savings over the conventional aluminum part.

An initial series of tests was conducted by the Ohio State University on sandwich panels fabricated by General Dynamics/Fort Worth. The panels were wedge sections 24 x 30 x 3 inches consisting of 6-ply (0 ± 45) boron-epoxy-skins bonded to aluminum honeycomb (2.5 lb/in$^3$) and employing a glass-fiber-plastic-channel forward closure and a glass-fiber-plastic trailing edge (Figure 2). A 2-inch-wide 20-mil-thick aluminum cap was bonded to the trailing edge for peripheral lightning protection, and a jumper cable was connected between the trailing edge and the forward channel closure to simulate the continuous peripheral metal strip used on the actual part. For the simulated lightning stroke test, the discharge electrode was placed near the center of the panel to determine the response of the unprotected boron epoxy, and a moderate discharge of 50 kiloamperes was applied. The discharge ruptured the boron laminate and an explosion resulted from expansion of the air in the honeycomb cells and vaporization of the aluminum foil. The aluminum honeycomb core was literally melted out of a 6 x 4 inch area, both face sheets were delaminated, and the skin-to-core bond was separated over an area of approximately 30 sq. in. In a subsequent test, a panel with the metallic strip was subjected to a simulated lightning stroke at the same current level and no visible degradation resulted. These initial tests

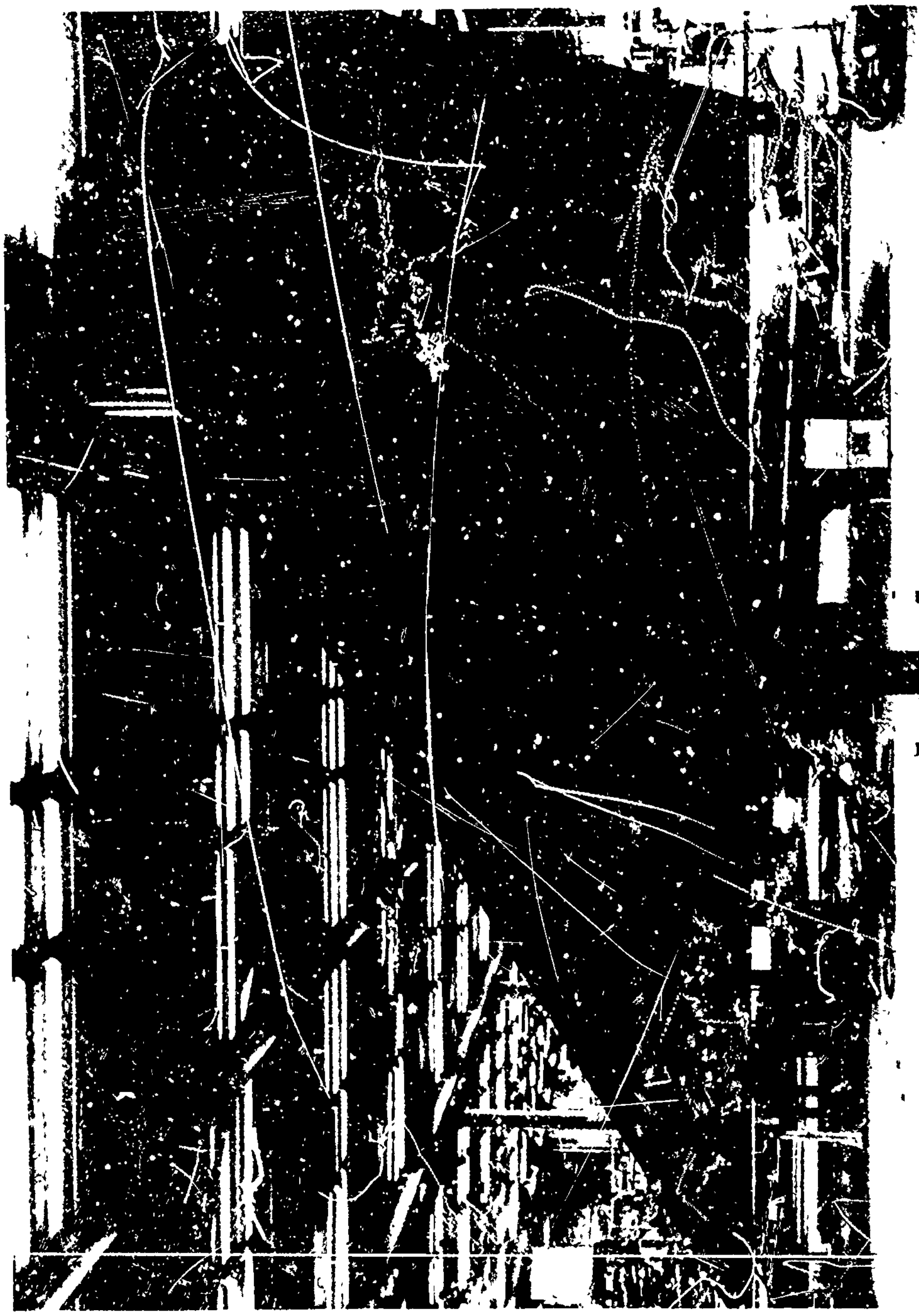


Figure 1. F-111 Horizontal Stabilizer Constructed of Boron Fiber-Epoxy Laminate

Figure 2. Sandwich Panel for Lightning Strike Tests Constructed With Boron-Epoxy Skin and Aluminum Honeycomb Core

demonstrated that even for strikes of low current level (50 kiloamps is considered low, 200 kiloamps is considered high) some form of lightning strike protection was necessary on the boron composite stabilizer.

Additional sandwich panels of similar construction were sent to Lightning and Transients Research Institute for testing at higher current and charge transfer levels, as required by MIL-B-5087B (ASG), the specification for external aircraft sections. The simulated lightning strokes were introduced at the trailing edge aluminum conducting strip. The current levels were 200 kiloamperes and 100 kiloamperes as indicated by Tests 13 and 14, respectively, in Figure 3. Test 14 was a high charge transfer test (110 coulombs compared to 3 coulombs for Test 13). This latter test is characteristic of a cloud-to-cloud discharge having intermediate current levels but longer charge transfer time and thus greater total charge transfer, as opposed to a cloud-to-ground discharge with high current rise rate and high peak current. These tests disclosed that the peripheral aluminum strip provided a conductive path for the lightning current sufficient to prevent degradation of the structural integrity of the panel, although secondary low-current streamers issued across the surface. Thus, an overall surface coating would be necessary to prevent secondary streamers from degrading the laminate.

An overall coating was also desirable for static discharge and antenna ground plane requirements. The coating selected for investigation was a silver-filled epoxy. A boron-epoxy sandwich panel coated with the silver-filled epoxy paint (3 mil) subjected to a discharge of 25 kiloamperes, showed only surface marking of the coating, as shown by Test 10, Figure 4. This sandwich panel was also subjected to a high voltage 100 kiloamperes discharge; damage was as indicated for Test 15.

To ascertain to what extent the aluminum honeycomb core influenced the damage to the boron epoxy laminate, two coated, flat, laminate panels, supplied by Grumman Aircraft, were subjected to simulated lightning strikes of the same high energy. The first panel contained an epoxy paint with silver particle additives integrally molded into the laminate; a coated sandwich panel was tested also, for comparison. The coated sandwich panel suffered extreme damage under high voltage discharge, but the painted laminate suffered only pitting. The second

Figure 3. Effect of Simulated Lightning Strokes on Aluminum Edge Strip on Sandwich Panel


Figure 4. Lightning Strike Damage to Boron Laminate Sandwich Panel Coated With Silver-Loaded-Epoxy Paint

laminate panel contained an integrally cured 6-mil-thick flame-sprayed aluminum coating; this panel suffered only discoloration and slight pitting when exposed to a 100 and 200 kiloampere discharge, as shown in Figure 5. These tests indicate that damage to the sandwich panel is greater because of the presence of the interior metal foil and that the damage mechanism is entirely different from that occuring in sheet laminates.

For the F-111 boron epoxy horizontal stabilizer, we used both an aluminum strip bonded to its perimeter and silver-particle epoxy paint sprayed over the entire surface. This combination is expected to not only provide protection against secondary streamers, but to have sufficient conductive capacity to meet static discharge and antenna ground plane requirements. A weight penalty of 6 lbs is added to the composite stabilizer, but this is 150 lbs lighter than an aluminum stabilizer.

## LIGHTNING STRIKE TESTS ON THIN SHEET PLASTIC LAMINATES

Thin sheet reinforced plastics exposed to simulated lightning strokes consisted of nominally 0.040-inch thick boron, graphite, and glass-fiber-reinforced plastics made with unidirectional plies laid up in a balanced 0° - 90° construction. The as-fabricated panels were 12 x 12 inches, which were cut into 6 x 12 inch test specimens. Details of the composition and construction of the panels are given in Table II. For comparison, an 0.063-inch-thick panel of 2024 T-3 aluminum was also tested.

Each laminate type was tested both "as fabricated" and with aluminum painted surface. Painted panels were spray-coated with a MIL-P-23377 primer and top coated with MIL-C-81352 acrylic lacquer containing 16 oz. of aluminum paste per gallon of lacquer. The primer coat was between 0.5 and 1.0 mil thick and the top coat was 3.0 ± 0.3 mils thick. The painted panels were tested to determine whether the aluminum paint would be conductive enough to protect the reinforced plastic substrate.

The test procedure consisted of clamping the test panels to a ply-wood backing which acted as an insulator, so that the 12-inch dimension was vertical and the bottom clamp, which acted as the electrode for the return path of the stroke current, was horizontal; the test set-up is shown in Figure 6. The stroke was

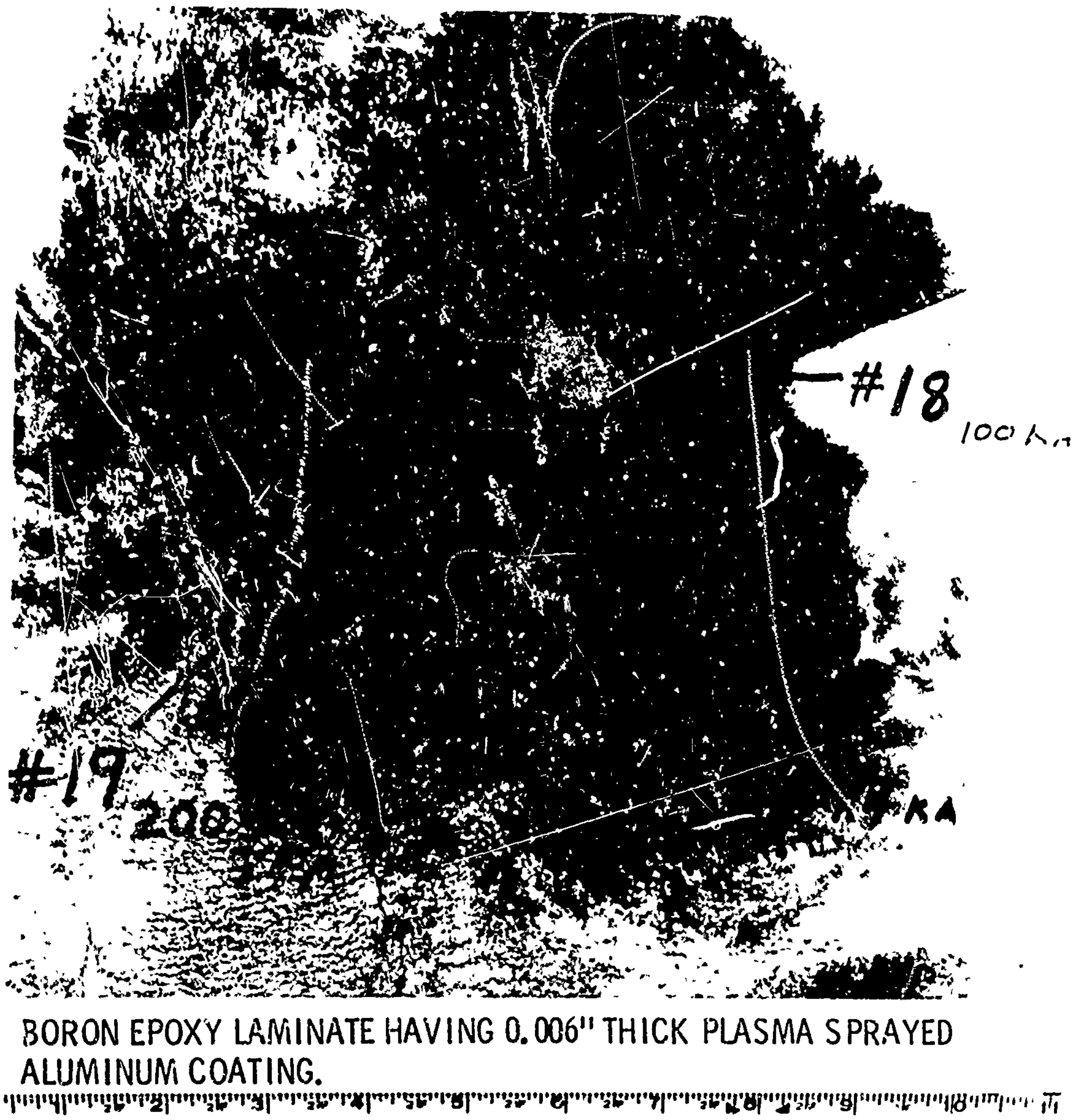



Figure 5. Lightning Strike Damage to Boron-Epoxy Laminate Panel Coated With Plasma-Sprayed Aluminum

TABLE II

COMPOSITION AND PHYSICAL CONSTRUCTION OF THIN LAMINATE PANELS

| Laminate | Lay up Pattern | Reinforcement | Resin | Average Resin Content % by Wt. | Average Fiber Volume % | Average Panel Thickness, in. |
|---|---|---|---|---|---|---|
| Glass Fiber-Epoxy Scotch Ply 1002 | 0°-90°-90°-0° | Glass Fiber Yarn | Epoxy (3M proprietary) | 38.0 | 44.6 | 0.042 |
| Boron Fiber-Epoxy | 0°-90°-0°-90°-90°-0°-90°-0° | 4 Mil Dia. Boron over 5 Mil Dia. Tungsten Core No. 104 Glass Cloth Backing for Boron Plies | Narmco 5505 Epoxy | 30.0 | 50.0 Boron | 0.042 |
| Graphite Fiber-Epoxy | 0°-90°-90°-0° | Thornel 50 w/PVA Size | Union Carbide ERL 2256 Epoxy w/ZZLB 820 Hardener | 41.8 | 51.1 | 0.035 |

Figure 6. Test Setup for Simulated Lightning Stroke Tests

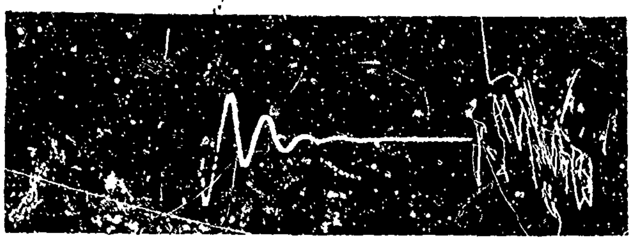
applied to the surface of the panel from a rod-type electrode with the tip approximately 1/8 inch from the panel surface and about 4 inches from the 6-inch edge of the panel. Since the return electrode was a bar about 1 inch wide, the stroke had to travel a net distance of about 7 inches to reach the upper edge of the return electrode.

Two basically different discharge wave shapes were used during the tests to simulate very mild and quite severe lightning strokes. Mild strokes were simulated with currents of 25 kiloamperes at 0.024 coulomb, and severe strokes with currents ranging from 60 to 100 kiloamperes at about 1.9 coulombs. Representative wave forms for strokes with different peak currents, as displayed on an oscilloscope, are shown in Figure 7.

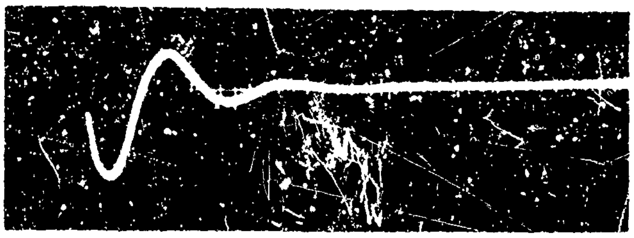
Damage to the panels by the strokes was assessed primarily by visual observation of the front and back surfaces, weight before and after test, and surface resistivity before and after test. In addition, a number of optical photomicrographs and electron beam scanning photomicrographs were taken of the surfaces of the boron and graphite-fiber reinforced plastics in an attempt to define damage mechanisms.

Table III summarizes the visually observed damage to the panels and Table IV lists the weight and surface resistivity of the panels before and after tests. Photographs of front and back surfaces of panels before and after testing are shown in Figures 8 - 16. The aluminum control panel was pitted over an area about the size of a quarter at the point of stroke impingement. The panels with the least damage were the glass-fiber-reinforced plastic. even at current levels of 100 kiloamperes, they exhibited no significant damage.

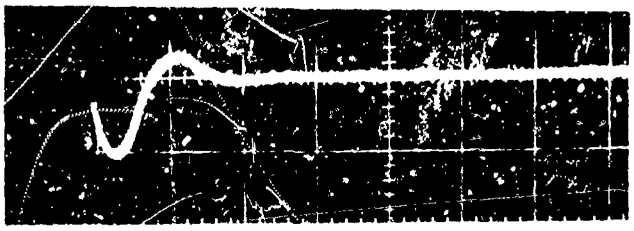
At the lowest current of 25 kiloamperes, both the boron and graphite-fiber-reinforced plastics suffered localized pitting damage at the point of stroke impingement. The aluminum painted panels were damaged slightly more than the unpainted panels. There was no visible damage on the back side of either the unpainted or painted boron panel, but slight damage appeared on the aluminum-painted graphite panel.



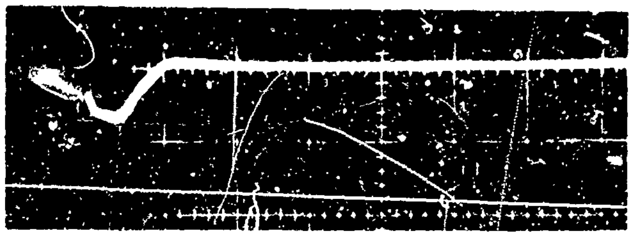
(a) 25 kiloampere crest, 5 μsec/div



(b) 100 kiloampere crest, 50 μsec/div



(c) 85 kiloampere crest, 50 μsec/div



(d) 60 kiloampere crest, 50 μsec/div

Figure 7. Wave Forms for Simulated Lightning Strokes

TABLE III

DAMAGE TO BORON FIBER, GRAPHITE FIBER, AND GLASS FIBER REINFORCED PLASTIC PANELS BY SIMULATED LIGHTNING STRIKES

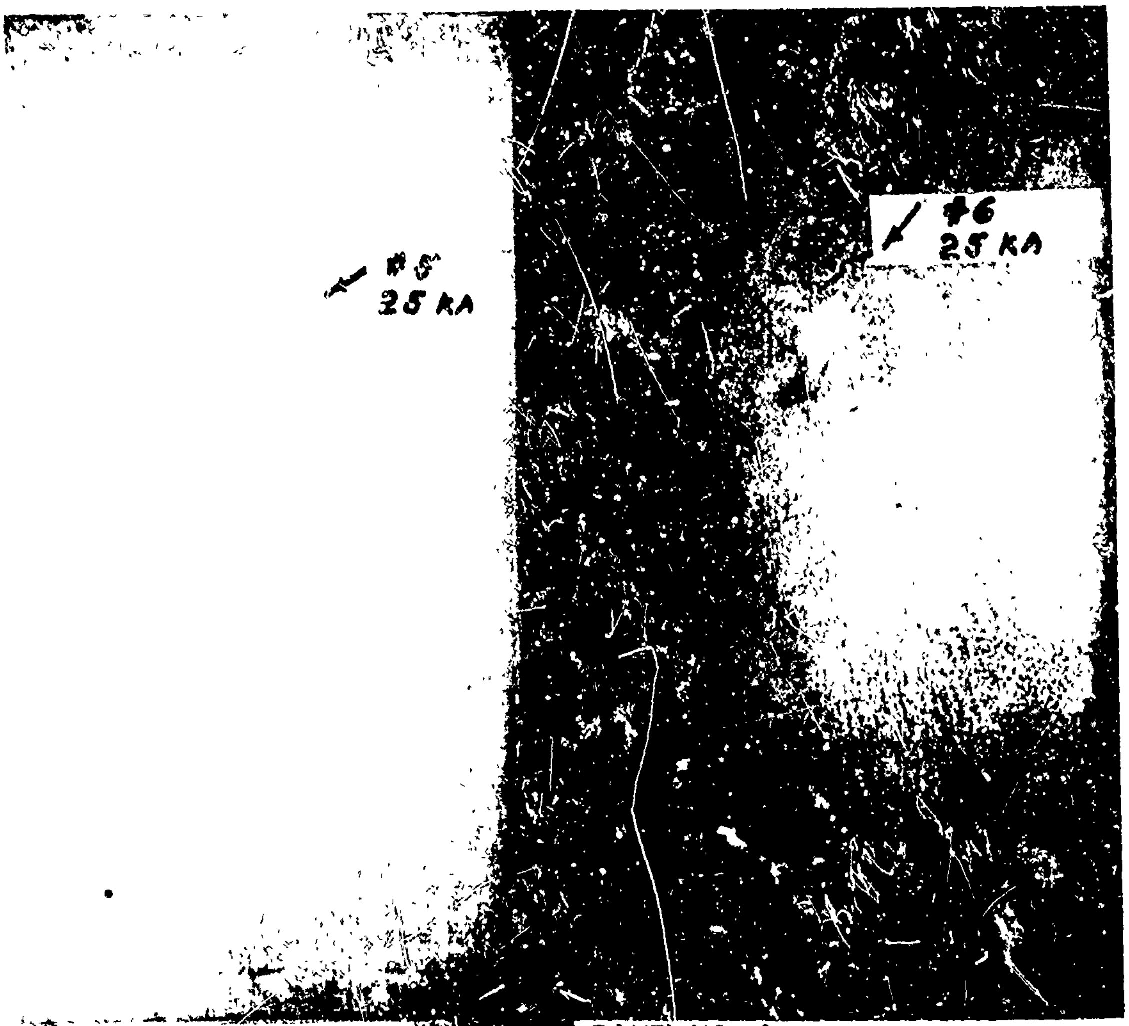
| Test No. | Panel No. and Type | Kiloamps | Coulombs | Visually Observed Damage |
|---|---|---|---|---|
| 1 | 1-Boron, uncoated | 25 | 0.024 | Localized laminate damage |
| 2 | 2-Boron, Al painted | 25 | 0.024 | Localized laminate damage |
| 3 | 3-Graphite, uncoated | 25 | 0.024 | Localized laminate damage |
| 4 | 4-Graphite, Al painted | 25 | 0.024 | Localized laminate damage |
| 5 | 5-Glass, uncoated | 25 | 0.024 | Surface marking only |
| 6 | 6-Glass, Al painted | 25 | 0.024 | Surface marking only |
| 7 | 7-0.063 in, 2024-T3 Al | 25 | 0.024 | Very slight surface pitting |
| 8 | 8-Boron, uncoated | 60 | 1.9 | Extreme laminate damage |
| 9 | 9-Boron, Al painted | 65 | 1.9 | Extreme laminate damage |
| 10 | 10-Graphite, uncoated | 85 | 1.9 | Extreme laminate damage |
| 11 | 11-Graphite, Al painted | 85 | 1.9 | Extreme laminate damage |
| 12 | 12-Glass, uncoated | 100 | 1.9 | Surface marking only |
| 13 | 13-Glass, Al painted | 70 | 1.9 | Surface marking only |
| 14 | 7-0.063 in, 2024-T3 Al | 100 | 1.5 | Slight surface pitting |

TABLE IV

WEIGHT AND SURFACE RESISTANCE OF BORON, GRAPHITE, AND GLASS FILAMENT LAMINATE TEST SAMPLES BEFORE AND AFTER ARTIFICIAL LIGHTNING DISCHARGES

| Panel No. & Type | Weight (as received) (gms) | Weight (After Test) (gms) | Surface Resistance (as received) (ohms) | Surface Resistance After Test (ohms) |
|---|---|---|---|---|
| 1-Boron, uncoated | 97 | 97 | 90 K | 50 K |
| 2-Boron, Al painted | 100 | 100 | 1 K | 3 K |
| 3-Graphite, uncoated | 62 | 62 | 0.25 | 0.15 |
| 4-Graphite Al painted | 69 | 69 | 0.4 | 0.17 |
| 5-Glass, uncoated | 88 | 88 | 7 Meg | 75 Meg |
| 6-Glass Al painted | 94 | 94 | ------ | ------ |
| 7-0.063 in, 2024-T3 Al | 198 | 198 | ------ | ------ |
| 8-Boron, uncoated | 97 | 94 | ------ | ------ |
| 9-Boron, Al painted | 100 | 96 | ------ | ------ |
| 10-Graphite, uncoated | --- | 61 | 0.1 | 0.075 |
| 11-Graphite, Al painted | 66 | 64 | 0.5 | 0.6 |
| 12-Glass, uncoated | 89 | 89 | 10 Meg | 10 Meg |
| 13-Glass, Al painted | --- | 94 | 1.2 Meg | 1.2 Meg |


Figure 8. Damage to 0.063-Inch Thick 2024-T3 Aluminum by 100 KA Simulated Lightning Stroke




PANEL NO. 5
GLASS-EPOXY
UNPAINTED

PANEL NO. 6
GLASS-EPOXY
ACRYLIC ALUMINUM
LACQUER ON FRONT SURFACE

Figure 9. Front Surface of Glass Fiber-Epoxy Laminate After 25 KA Simulated Lightning Stroke

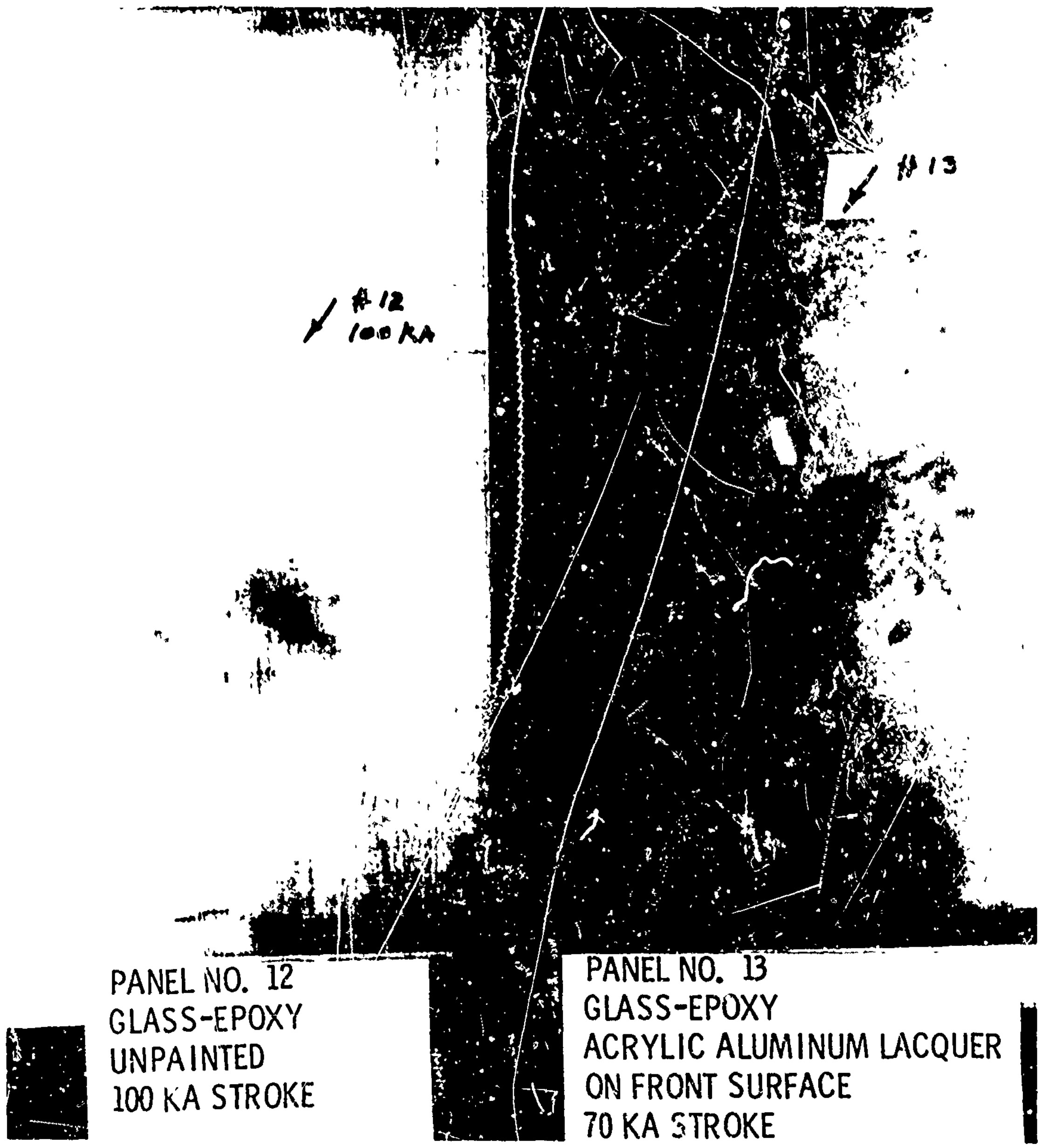



Figure 10. Glass Fiber-Epoxy Laminates After High Current Simulated Lightning Stroke

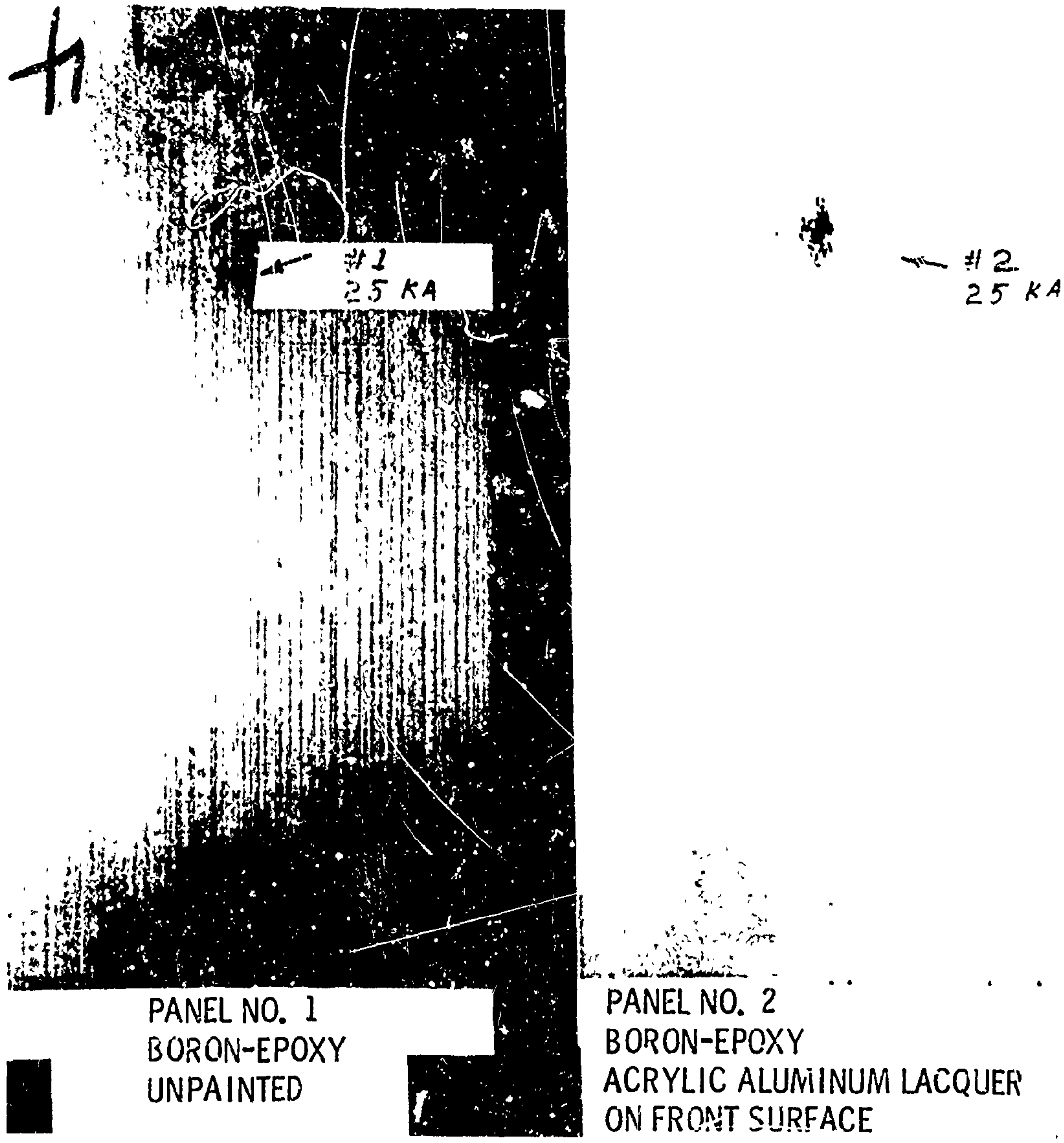



Figure 11. Damage to Front Surface of Boron Fiber-Epoxy Laminates by 25 KA Simulated Lightning Stroke

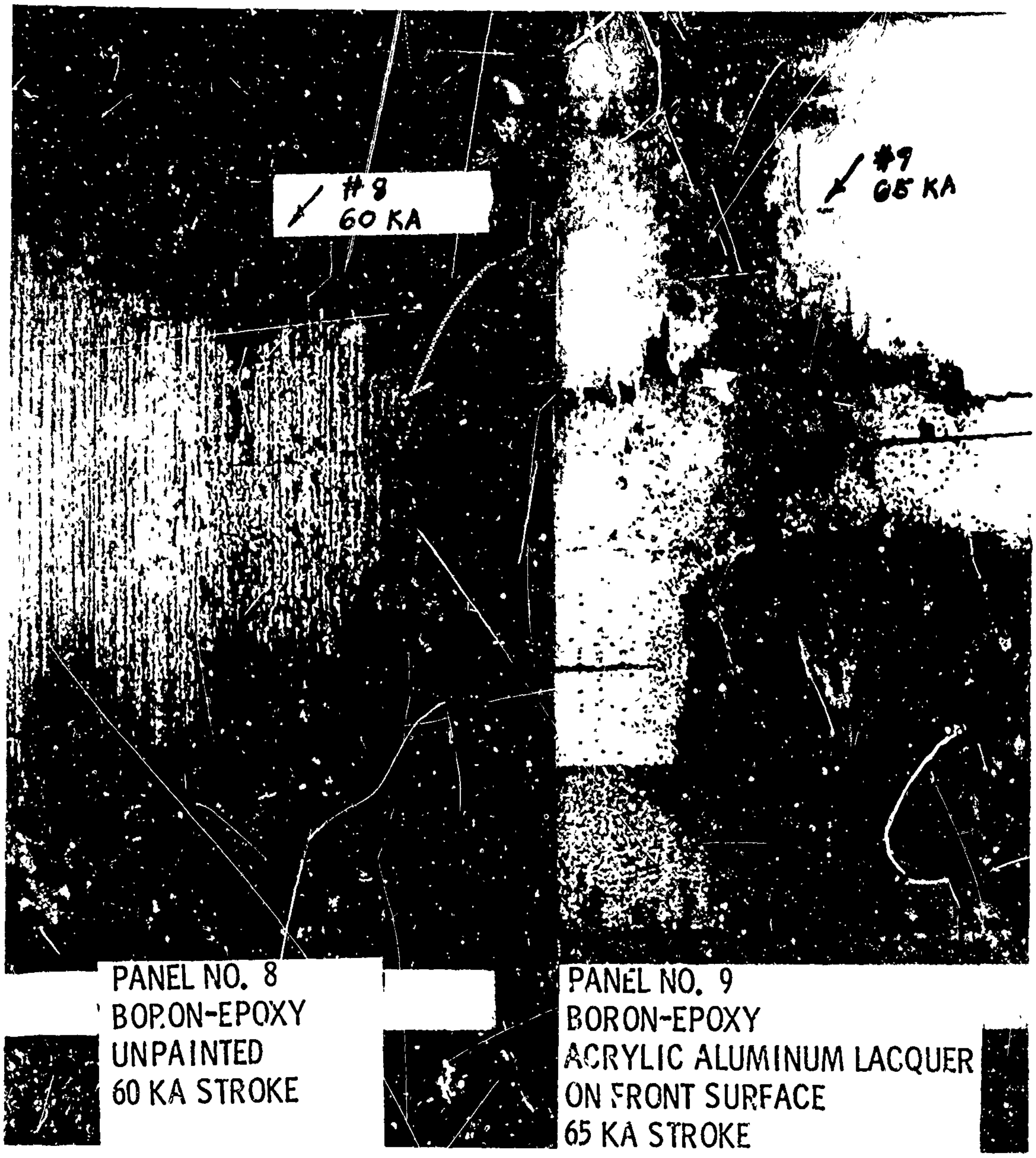



Figure 12. Damage to Front Surface of Boron-Epoxy Laminates by 60 KA and 65 KA Simulated Lightning Strokes

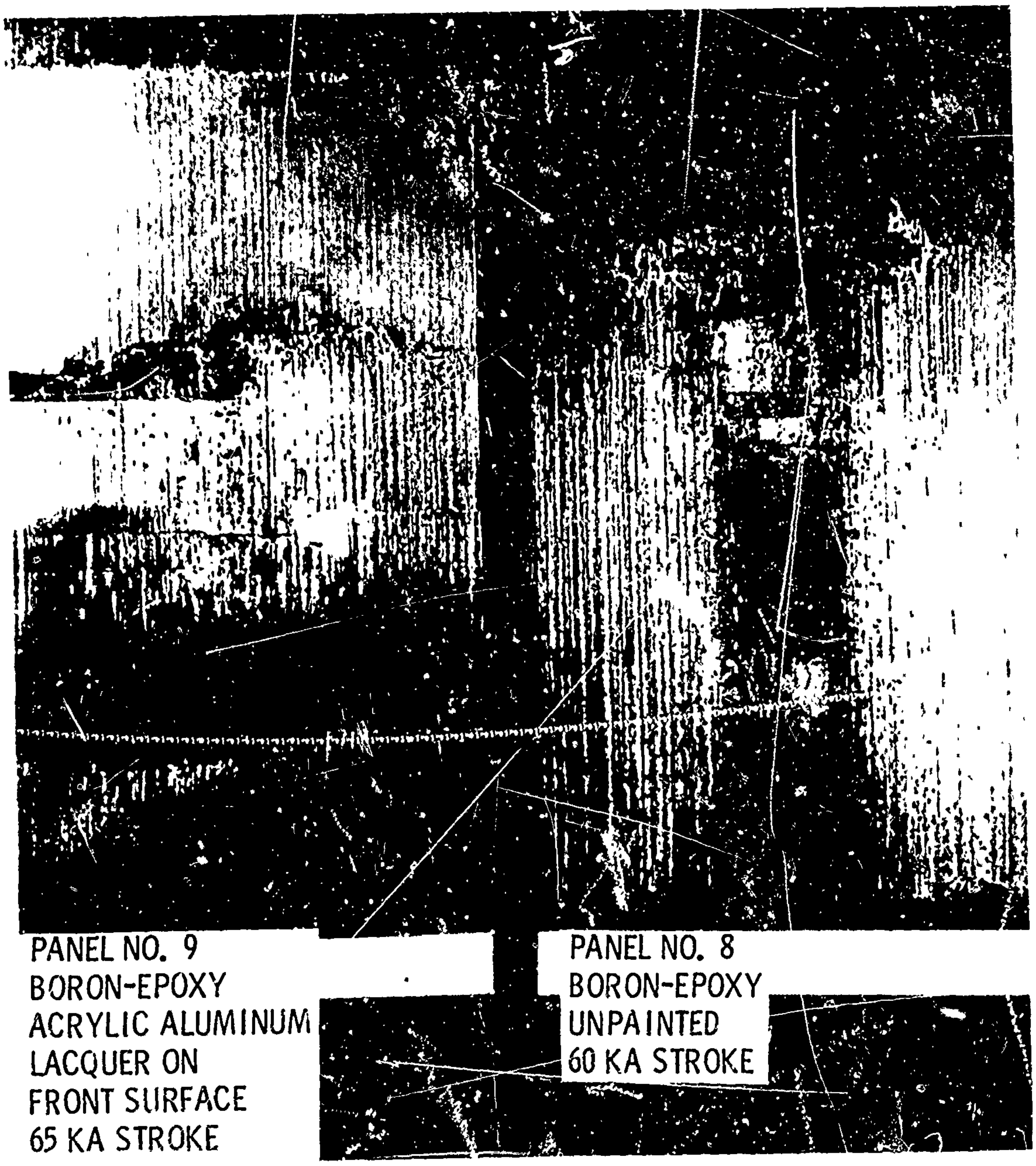



Figure 13. Damage to Back Surface of Boron Fiber-Epoxy Laminates by 60 KA and 65 KA Simulated Lightning Strokes

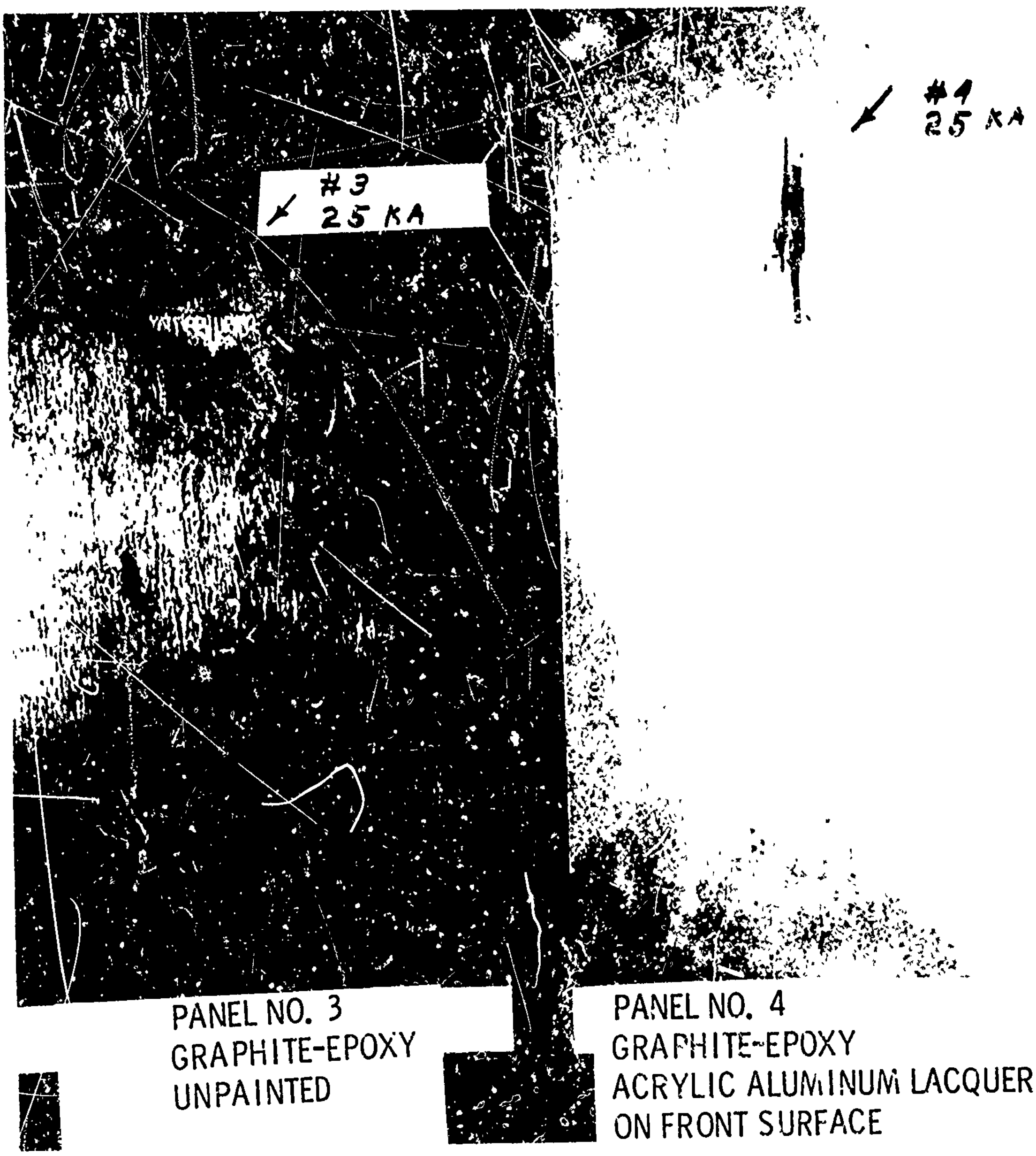



Figure 14. Damage to Front Surface of Graphite Fiber-Epoxy Laminates by 25 KA Simulated Lightning Strike

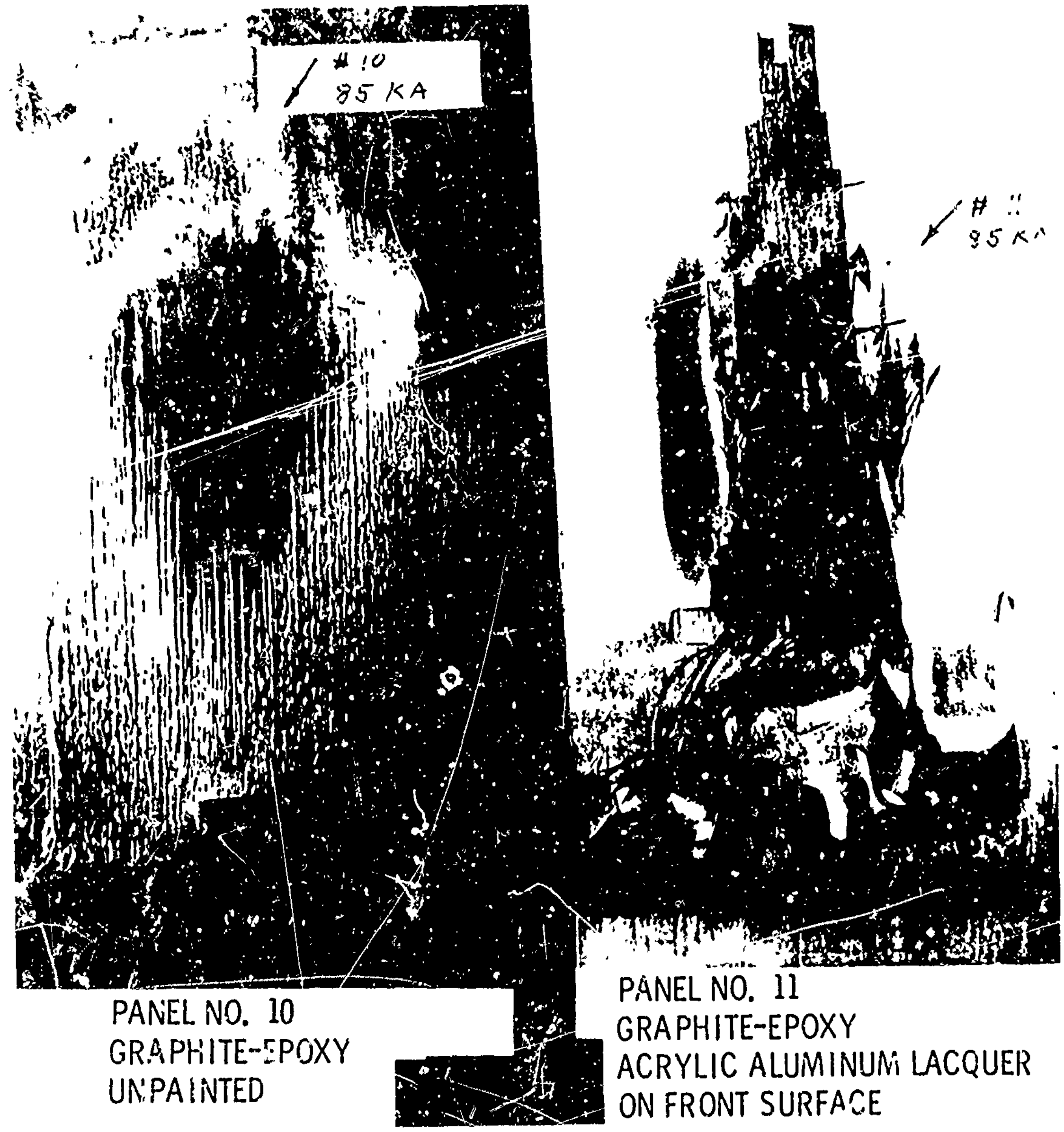



Figure 15. Damage to Front Surface of Graphite Fiber-Epoxy Laminates by 85 KA Simulated Lightning Strike

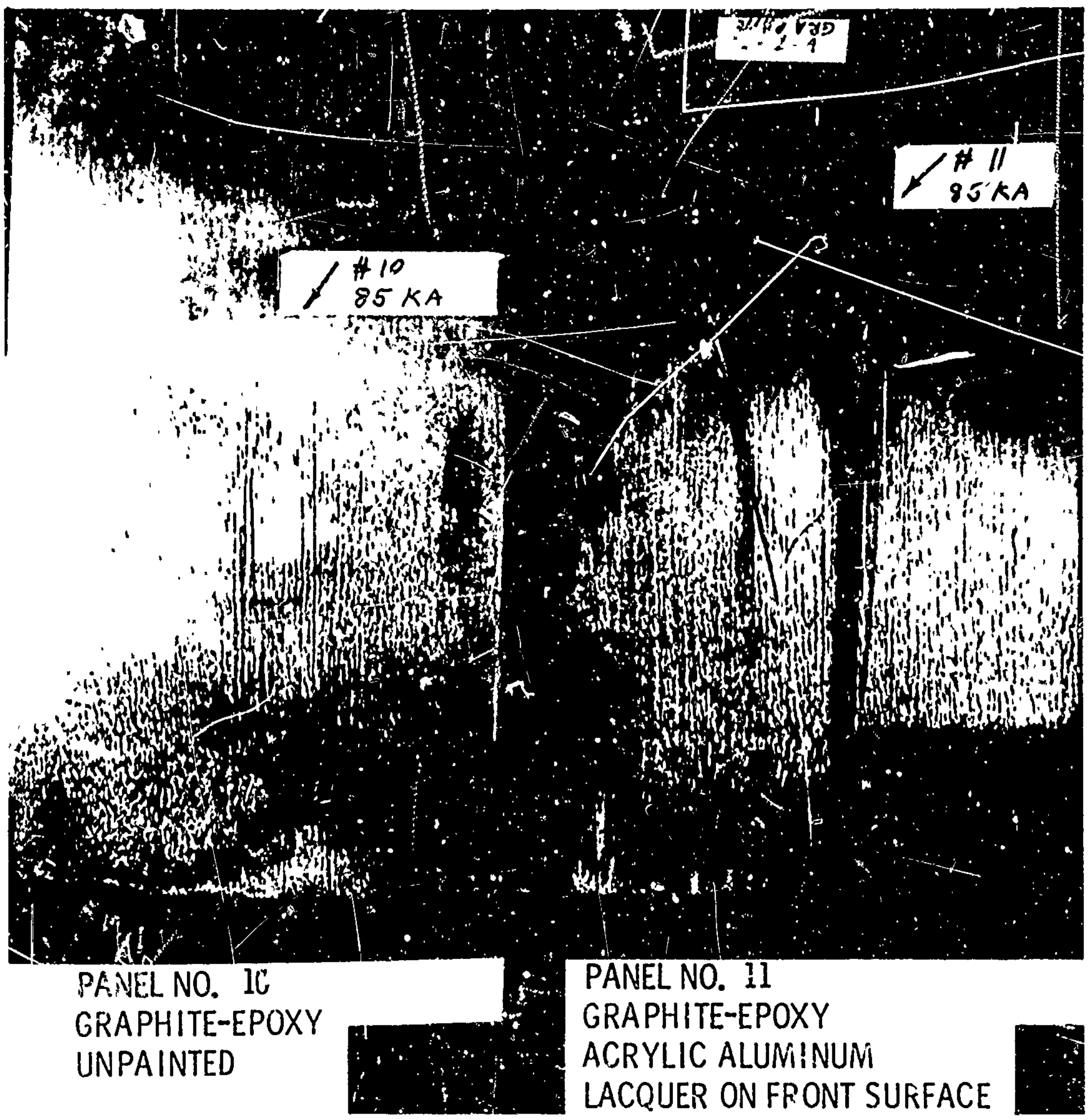



Figure 16. Damage to Back Surface of Graphite Fiber-Epoxy Laminate by 85 KA Simulated Lightning Stroke

An unpainted boron laminate panel exposed to a 60 kiloampere stroke suffered considerable damage; a crack across the width passed through the point of stroke impingement. An aluminum painted boron laminate panel exposed to a 65 kiloampere discharge had six cracks across the width, spaced about 1-1/2 to 2 inches apart; the damage was principally mechanical with only very slight tan discoloration on the back, indicating resin scorching. A number of bubble-type delaminations between the plies could be detected as small white spots in the glass fabric resin cloth in the laminates. A representative photomicrograph of the boron laminate surface is shown in Figure 17.

An unpainted graphite laminate exposed to 85 kiloamperes was damaged considerably. The surface appeared to have bare graphite fiber yarn in the area between the point of stroke impingement and the return electrode. An aluminum-painted graphite laminate exposed to 85 kiloamperes was damaged in a similar manner but more severely than the unpainted panel, with a hole through the panel at the point of stroke impingement.

All of the panels were tested with the fiber of the outer plies parallel to the 12-inch dimension which was the direction of the current path from the point of stroke impingement to the return electrode. For the graphite laminates, the current appeared to have been conducted along the outer plies, since these were debonded and stripped of resin. To determine whether similar damage occurred to the two inner plies, Panel 10 was sectioned parallel to the 6-inch edge. The laminate edges were potted in epoxy resin to facilitate polishing prior to photographing. A photomicrograph of the cross-section of an undamaged part of the laminate is shown in Figure 18 and of a severely damaged area in Figure 19. Figure 19 shows that the two inner plies were relatively undamaged, since the filaments are packed as closely as in the "as-fabricated" laminate; in the outer plies the yarns are completely unbonded and loosely packed. The gray area around the whitish filaments in the outer plies is the potting resin.

Table III shows there is no consistent trend in change of surface resistivity after tests. Also, change in weight was negligible or very minor. The low weight loss of the graphite laminate appeared to be inconsistent with its appearance;

Figure 17. Lightning Strike Damage to Front Surface of Boron-Epoxy Laminate (50X MAG)

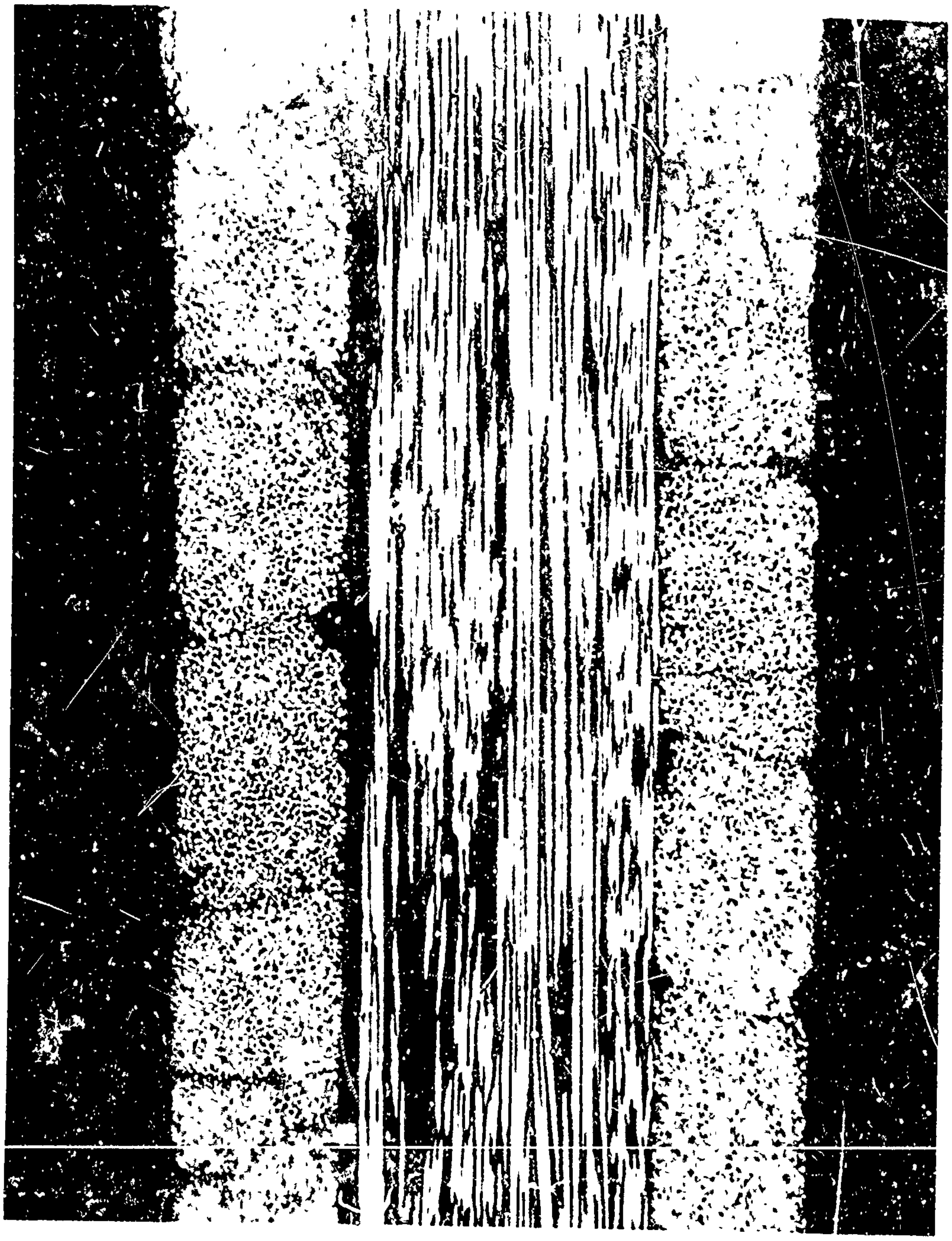


Figure 18. Cross Section of Undamaged Portion of Graphite-Epoxy Laminate (40X MAG)

Figure 19. Cross Section of Damaged Portion of Graphite-Epoxy Laminate (40X MAG)

the resin appeared to have been volatilized, leaving bare unbonded yarn on the surface. The yarns could have become unbonded by mechanical forces, however, without the resin volatilizing.

The amount and condition of resin on the graphite fibers was investigated by making a series of electron scanning photomicrographs across the width of the panel. The photomicrographs covered three areas that were visually different: (1) the center of the panel, near the point of strike impingement where the damage was greatest; (2) near the center, with somewhat less damage; and (3) near the side, with realatively little damage. These photomicrographs show the following: fibers in the center portion of the panel are essentially bare (Figure 20); in the adjacent areas, particles of resin are adhering to the fibers even though the fibers appear bare (Figure 21); along the edges where there appeared to be no damage, some fiber-matrix separation occurred (Figure 22).

The fact that boron and graphite laminates were damaged considerably but glass fiber laminates were not damaged indicates that fiber conductivity is an important materials parameter. R. F. Deacon determined the dc resistivity of boron fibers containing a tungsten wire precursor core to be 2600 $\mu$ ohm-cm (Reference 2). He concluded that most of the conductivity in the fiber was due to the conductivity of the core, and that the boron sheath was a fairly good insulator.

According to Roger Bacon (Reference 3) the electrical conductivity of graphite fibers can be expressed by the relationship $E = C\sigma$, where $\sigma$ is the conductivity in $(\text{ohm-cm})^{-1}$, E is the modulus of elasticity in $\text{lb/in}^2$, and C is a proportionality constant having a value of $5.3 \times 10^4$ $(\text{lb/in}^2)$ (ohm-cm). From this relationship, the resistivity of graphite fibers with a modulus of elasticity of 50 million psi is 1060 $\mu$ ohm-cm.

Although both the boron and graphite fiber reinforced plastics suffered considerable damage by the simulated lightning strokes at 60 kiloamperes and above, the damage mechanism for the boron laminates appeared to be primarily mechanical, while that for the graphite was primarily thermal. Since the core of the boron fiber is the conductive part, thermal stresses may fracture the filaments, and

Figure 20. Surface of Severely Damaged Area of Graphite-Epoxy Laminate (205X MAG)

Figure 21. Surface of Moderately Damaged Area of Graphite-Epoxy Laminate (1020X MAG)

Figure 22. Surface of Slightly Damaged Area of Graphite-Epoxy Laminate (1020X MAG)

the boron sheath may insulate the plastic matrix from the heat in the core. In the graphite laminates, however, the heat is in the graphic fibers, so resin in contact with them would be heated and pyrolyzed or degraded.

## CONCLUSIONS

1. Bare boron-fiber-reinforced plastics and high modulus graphite-fiber-reinforced plastics in thin sheet form suffered considerable damage when exposed to simulated lightning strikes with peak currents of 60 kiloamps and higher.

2. An acrylic base aluminum paint was not effective in providing lightning protection to either boron or graphite-fiber-reinforced plastics.

3. A 6-mil plasma-sprayed aluminum coating on a boron-plastic-laminate panel provided good protection from lightning strike damage at current levels up to 200 kiloamps.

4. A 3-mil coating of silver-epoxy paint provided good lightning protection to a boron laminate at current levels up to 100 kiloamps.

5. Sandwich panels made with boron-laminate skins (both uncoated and coated) and aluminum honeycomb cores were damaged considerably more than corresponding sheet laminates at the same current levels.

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# MECHANISMS OF LIGHTNING DAMAGE TO COMPOSITE MATERIALS

J. D. Robb

Lightning and Transients Research Institute

Composite structures consisting of boron or graphite fibers in an epoxy resin present a new problem in aircraft lightning protection, for unlike fiberglass or metal skins, considerable energy can be developed in the material itself to produce high gas pressures and extensive damage. For lightning protection development and for adequate artificial lightning tests of new material samples, the lightning strike contact mechanisms must be considered. Of equal importance is the way in which the material is to be used, where it is to be used, and its importance in terms of flight safety.

## LIGHTNING STRIKE CONTACT MECHANISMS TO AIRCRAFT

In considering the mechanisms of possible lightning stroke damage to composite material such as the boron epoxy and graphite constructions, it is useful to review the basic mechanisms of stroke contact to an aircraft or an aerospace vehicle of any type A natural lightning discharge initiates from a charge region in a cloud in the form of a step leader which advances at approximately 50 meter steps toward another charged region or toward the earth. In some cases, the aircraft triggers a lightning discharge which would not have occurred otherwise, and in some cases it merely diverts the discharge slightly out of its normal path so that the stroke passes through the vehicle.

As a step leader approaches a vehicle, the intense voltage existing between its tip and the aircraft induces streamers and intense ionization off all the aircraft external surfaces, and particularly from the extremities. Figure 1 shows streamering off a model aircraft subjected to intense cross fields in the laboratory. When the step leader contacts one of the vehicle extremities through the streamer, the vehicle's potential is immediately raised to the extreme potential of the lightning discharge and additional streamering takes place from the opposite extremities of the vehicle to form the step leader for the continuation of the stroke path to another charge region or to the earth.

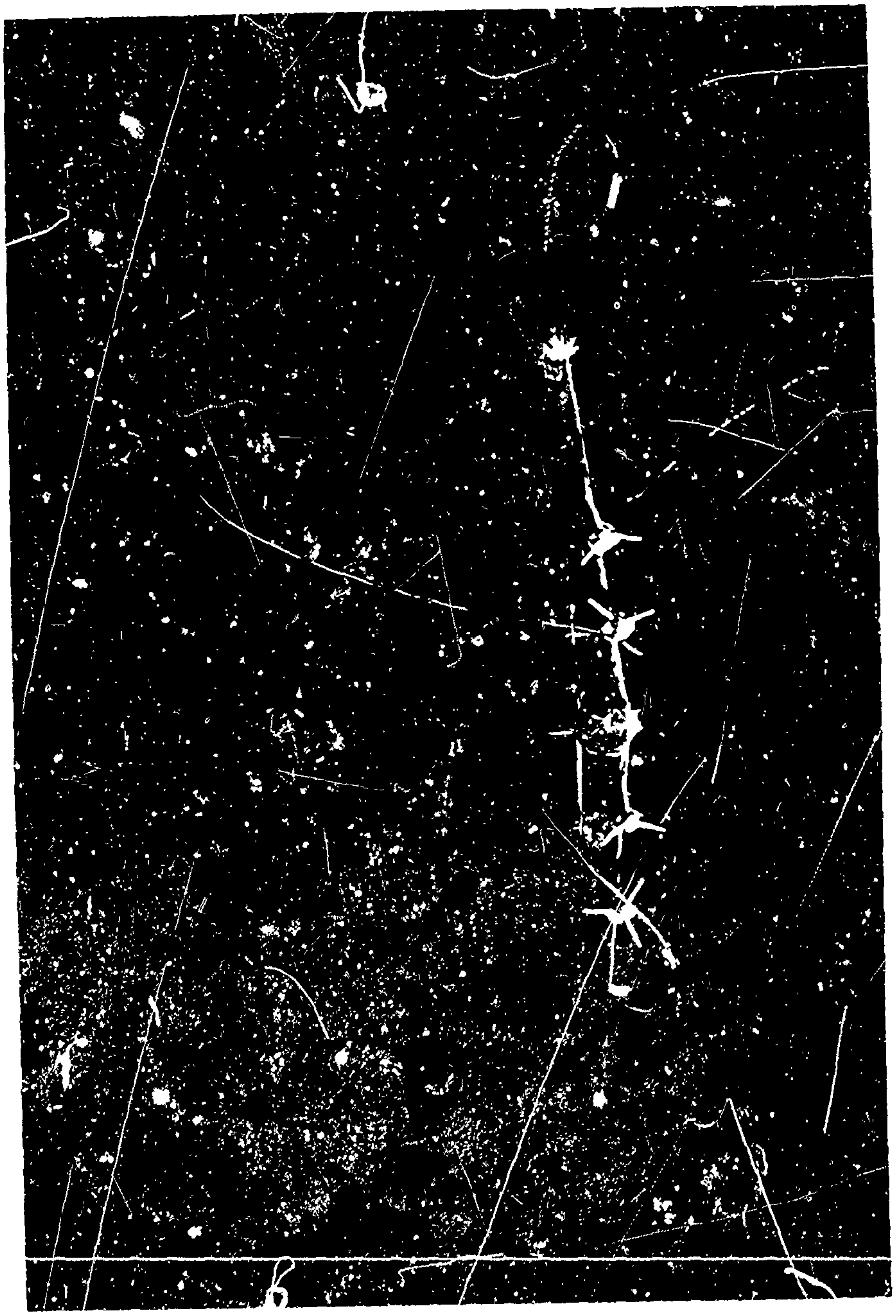


Figure 1. Induced Streamering Such as Occurs in Thunderstorm Regions From Intense Electrical Fields Using Model Aircraft in a Laboratory

When the step leader contacts the earth or another charge region, the existence of an ionized conducting path between the earth and the charge region (or the two charge regions) results in a large surge current in the form of an ionization wave which travels back up the step leader path to the initiating charge region in the cloud. This high-current ionizing wave is referred to as the "return stroke" and consists of a fast-rising high-current surge.

The several phases in the propagation mechanisms of the natural lightning stroke to and through an aerospace vehicle are illustrated in Figure 2. (It should be noted that in any case of significant damage to an aircraft, the discharge passed through the vehicle, and did not initiate from it or terminate on it as may be shown by energy calculations.) The different phases of the discharge are illustrated in Figure 3, which shows a triggered natural lightning discharge to the LTRI Research Vessel THUNDERBOLT taken from a distance of 10 meters; an oscillogram of the associated current is shown below.

Because the vehicle moves past the relatively stationary ionized stroke channel, the stroke may be considered to sweep over the aircraft, permitting contact at nearly any point behind forward strike points. This means, in effect, that nearly any type of lightning stroke component may be expected at a midchord region of a composite material external surface. The mechanisms of stroke contact to composite surfaces are illustrated in Figure 4.

## LIGHTNING DAMAGE MECHANISMS

Lightning damage to metal aircraft skins occurs because the skin at the lightning strike point is unable to carry the discharge current without heating the metal to the melting or vaporization temperatures. The skin may be heated by thermal contact with the arc or by resistance heating of the lightning current flow. The ability of the skin to withstand the high thermal arc temperatures (which far exceed the melting temperature of the skin materials) is therefore determined by the ease with which it can transfer the heat away from the arc contact point; this, in turn, is a function of the skin thickness at the contact point. Because of good conductivity of aluminum, aircraft skins have seldom shown

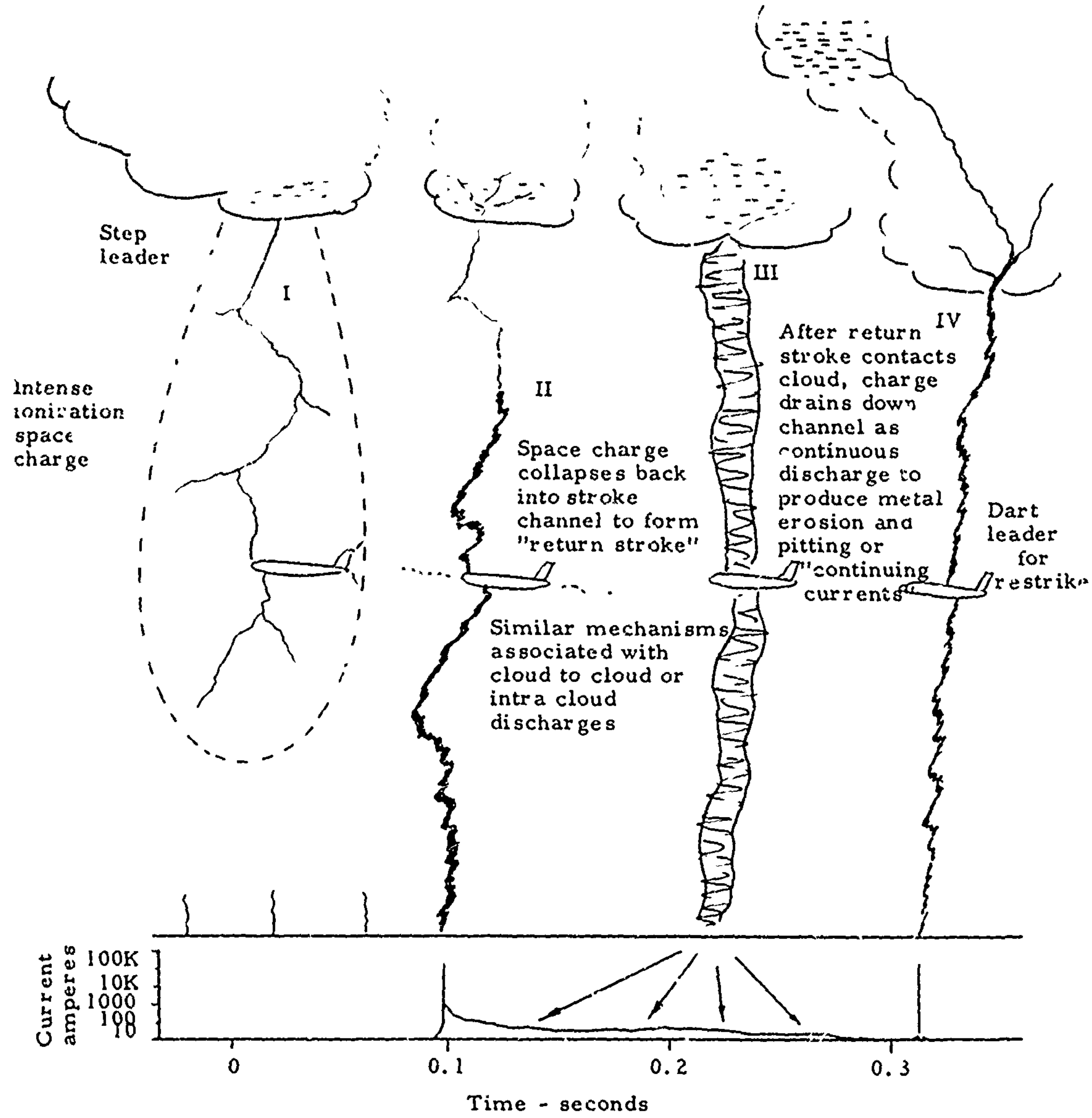
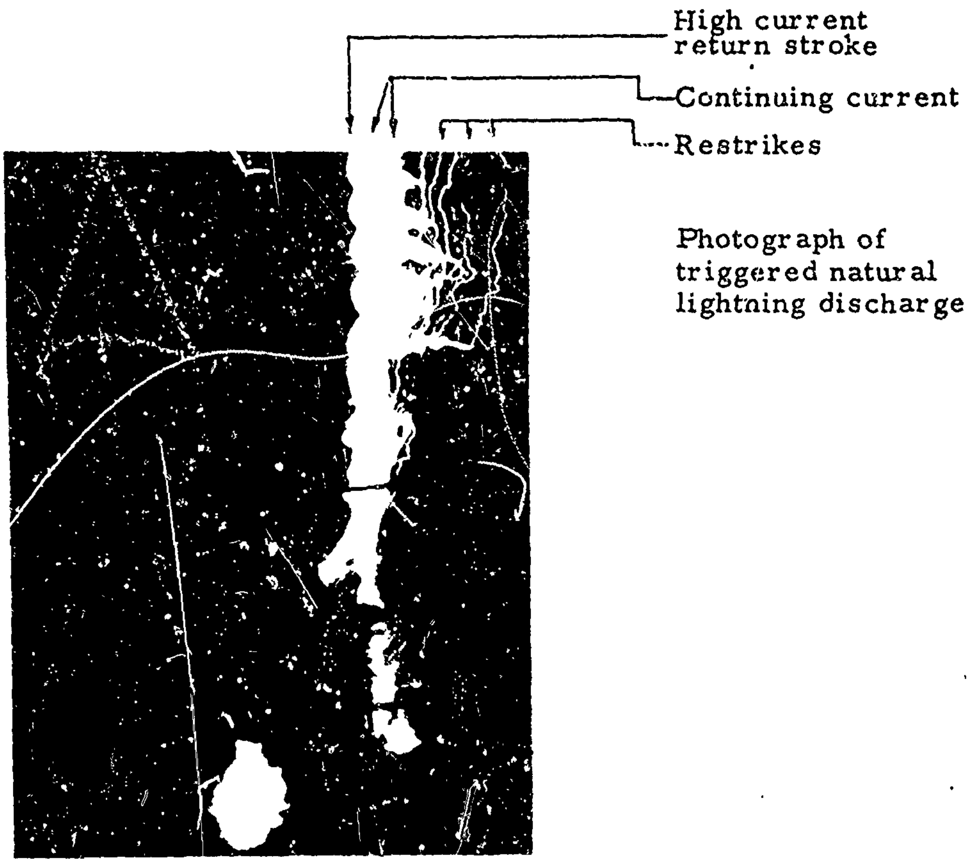



Figure 2. Probable Contact Mechanism and Aircraft Movement Through Lightning Channel; Corresponding Current Flow Indicated Below



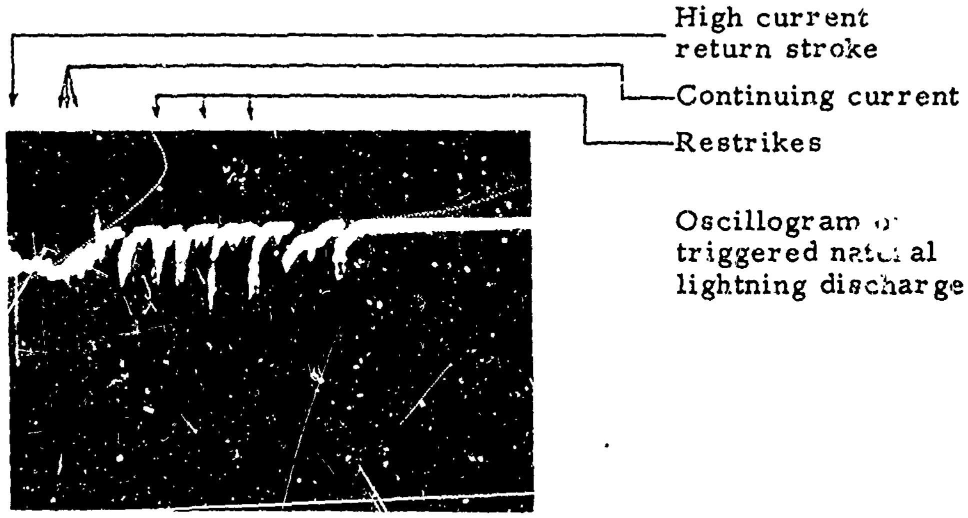




Figure 3. Photograph and Oscillogram of Triggered Natural Lightning Discharge Illustrating Typical High Current Return Stroke, Continuing Currents, and Multiple Restrikes

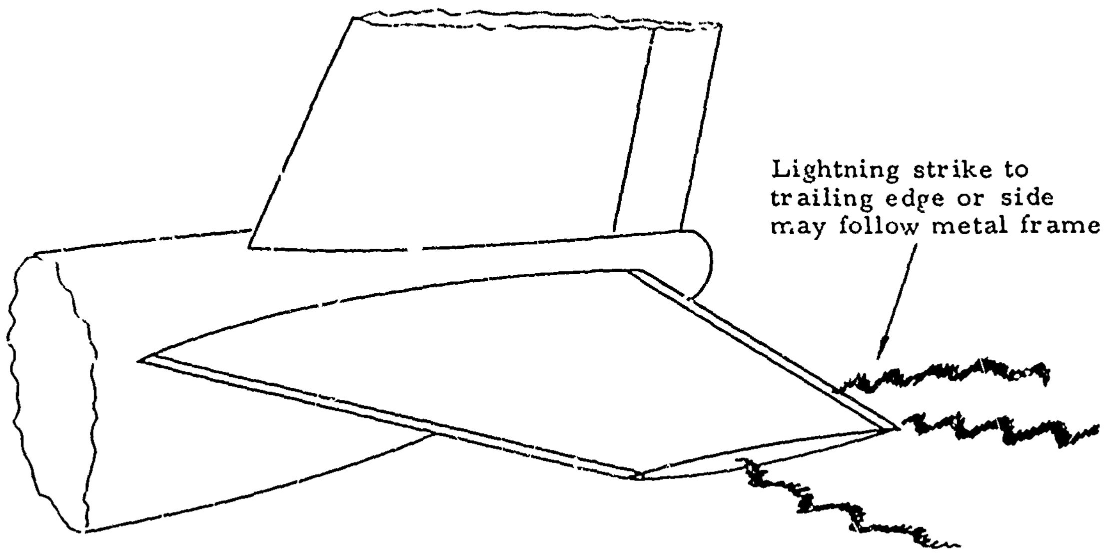



Figure 4(a). Peripheral Metal Frame Around Composite Surface Protects From Stroke to Rear or Side.

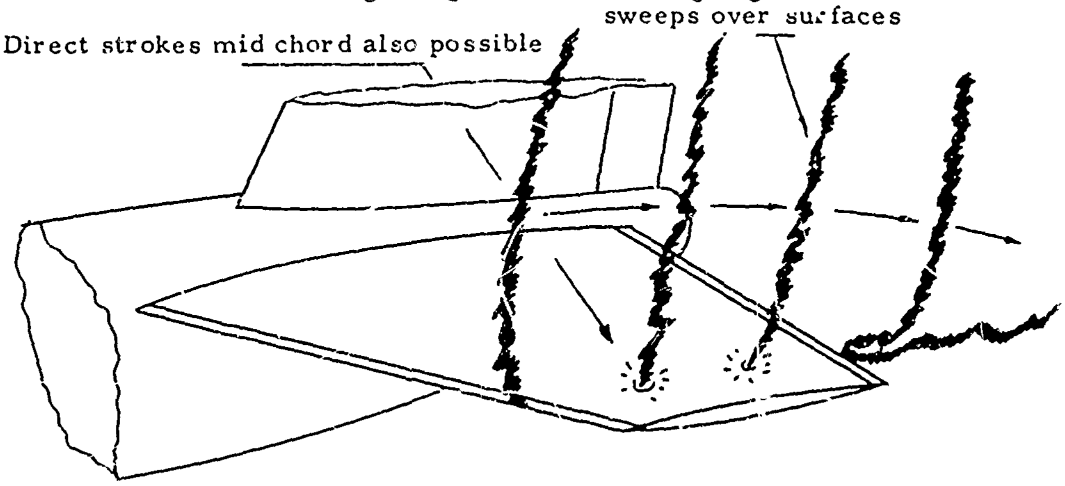



Figure 4(b). Peripheral Metal Frame Does Not Protect From Stroke to Forward Edge Swept Over Section by Windstream, or From Some Direct Strokes

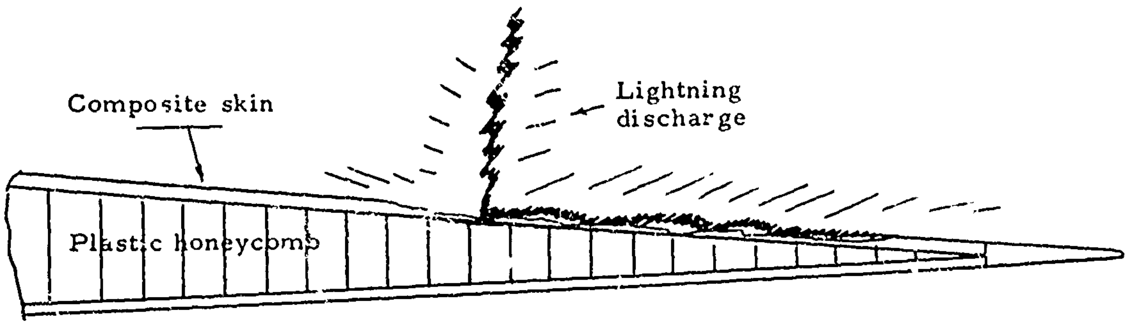



Figure 4(c). Lightning Stroke damage to Section With Fiberglass Core and Composite Skin. (Damage principally in skin.)

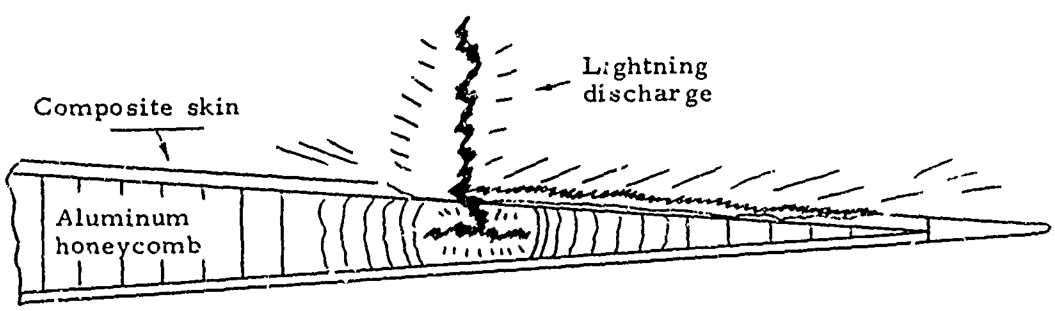



Figure 4(d). Lightning Stroke Damage to Section With Aluminum Honeycomb Core and Composite Skin. (Damage is extensive inside section from "exploding foil" effect as well as in skin.)

evidence of heating except at the arc contact point. The current-carrying capability of various metal skin materials is a function of their resistivity, specific heat and melting temperature, and other more complex factors such as the heat of vaporization, etc.

For conventional fiber glass materials, the resistivity is so high that lightning potentials will generally flash over the external surface unless guided into the plastic interior laminates by metallic end fittings or by poor construction of the plastic boundaries (poor from a lightning point of view). The arc produces vaporization of the resins along the path and intense gas pressures, particularly when confined. In providing lightning protection for plastic surfaces, therefore, the principal problem is to ensure that the lightning is discharged over the outside surface of the vehicle. Lightning discharge arcs inside a vehicle can produce serious structural damage.

Low-magnitude lightning strikes which would not even be reported for aluminum-skin aircraft can seriously damage composite skins because large energies can develop in the material, itself. Calculated values for energy developed in various materials by a moderately severe lightning stroke current are presented in Table I for a section of material approximately 6 inches square by 0.040 inch thick, with the current applied across two opposite edges. For simplicity, the lightning current waveform is represented by the rectangular function of 100,000 amperes crest and 2 microseconds duration. The energy developed in the metals is relatively low compared to that developed in the boron epoxy and graphite materials.

Also of importance in evaluating possible damage is the way in which the material is used. For example, composite material over aluminum honeycomb core can result in puncture of the composite material, with principal current conduction through the aluminum honeycomb core producing aluminum vapor and an "exploding foil" shock wave inside the structure. The exploding foil is similar to the exploding wires used to drive hypersonic shock tubes. Because the resistance inside the section through the aluminum honeycomb core is much lower, most of the energy will be developed internally with internal arcs and severe damage; composite materials are vulnerable to even low current lightning discharges, however, so serious damage would also be produced in the composite skin.

TABLE I

CALCULATED ENERGY IN VARIOUS MATERIALS
FROM 100 KILOAMPERE CURRENT*

| Material | Energy-W |
|---|---|
| Plastics | $W_p \cong 0$ damage from external arc pressures |
| Metals | $W_m \cong I^2 R \Delta t$<br>$= (10^5)^2 \times 3 \times 10^{-6} \times 2 \times 10^{-5}$<br>$W_m = 0.6$ joules |
| Boron Epoxy (Resistivity = 850 ohm-cm) | $W_b \cong I^2 R \Delta t$<br>$= (10^5)^2 \times 8500 \text{ ohm} \times 2 \times 10^{-5}$<br>$W_b = 8500$ megajoules<br>Most of the current will flash over surface after destruction of material |
| Graphite (Resistivity = 0.0025 ohm-cm) | $W_g \cong I^2 R \Delta t$<br>$= (10^5)^2 \times 0.025 \times 2 \times 10^{-5}$<br>$W_g = 5000$ joules |

*For simplicity in illustration, rectangular current waveform used 100 KA, 20 $\mu$ sec.

Also of importance is the location where composite materials are to be used. Where they are not used for primary structures, as in leading or trailing edges, loss of extensive sections may not introduce hazards to flight; the cost of protection in terms of weight and expense can then be traded against the cost of occasional repair. Where the materials are used for primary structures and safety of flight is involved, some type of protection must be provided.

Unfortunately, methods used for protecting plastic sections such as radomes are not very effective for composite materials because of their conductivity.

Conducting strips are used on radomes and other plastic sections to attract lightning discharges to the diverter strips so that the discharges do not puncture the plastic surfaces. This diverter effect is produced because intense momentary electrical gradients exist on the strips because of the lack of conductivity of the adjacent radome material (even with antistatic conducting coatings). The composite materials have good conductivity, so streamers nearly equivalent to those from the metallic strips will be produced, with little diverting action from the strips and very little protection. The degree to which diverting strips protect composite materials is yet to be completely determined, but initial indications are that the diverting action will be poor.

## CONCLUDING DISCUSSION

An initial review of the lightning contact and damage mechanisms for composite structures indicates that they may be struck by any of the major lightning discharge components, including initial high-current impulses, intermediate currents characteristic of cloud-to-cloud discharges, and low-current continuing components, any one of which may produce localized damage. The extensive damage can be done to composite materials even by low lightning discharge currents. Adequate protection can probably be provided if introduced in preliminary phases of a design, but the seriousness of possible problems such as structural damage, pulse coupling into electrical circuits, hazards to fuel systems, and precipitation-static radio interference, which can be introduced if no protection is provided, can hardly be overemphasized.

# THE SUSCEPTIBILITY OF ADVANCED FILAMENT ORGANIC MATRIX COMPOSITES TO DAMAGE BY SIMULATED LIGHTNING STRIKES

W. M. Fassell

A. P. Penton

Philco-Ford Corporation

J. A. Plumer

General Electric Company

## INTRODUCTION

As part of an extensive internal research and development program at Aeronutronic on high modulus graphite — organic matrix composites, the decision was made to ascertain some details of the phenomenology of simulated lightning strikes on this class of composite. Boron filament, epoxy matrix composites were included in the test program to provide comparative data.

To obtain expert technical advice on this part of the program, Aeronutronic invited the General Electric High Voltage Laboratory to join the program to evaluate the performance of selected boron and graphite composites subjected to high current impulses, similar to those encountered in a natural lightning discharge. Three series of electrical impulse screening tests were performed on typical boron and graphite epoxy composites contemplated for aircraft use. Unidirectional boron filament epoxy composites lost up to 89 percent of their flexure strength at impulse current levels as low as 2000-3000 amps. Additional tests to current crest amplitudes as high as 38,000 amps confirmed the degradation of the boron filaments. Typical photomicrographs of damaged boron fibers at the failure planes are shown in Figure 1. Fragmentation of the boron fiber occurs in both the axial and transverse modes.

The mode of failure has been successfully modeled, based on the concept of differential heating of the tungsten-boride core and the boron coating. This mode

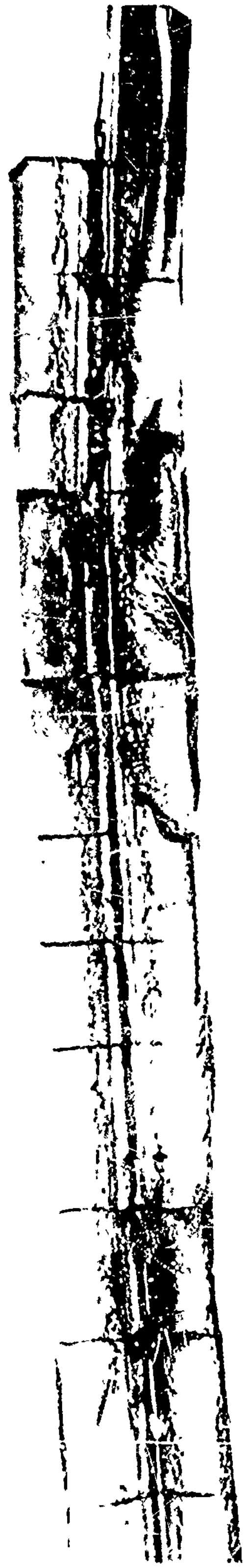


Figure 1. Typical Transverse and Axial Failure Planes of a Boron Filament From an Electrically Impulsed Composite

of failure, the degradation of the mechanical properties, and the measurements of the observed changes in resistivity after each current pulse are all consistent with the proposed model of failure. In essence, the unidirectional boron epoxy composite represents a complex resistor network. The first impulse appears to be carried, based on microscopic observations, by the outer layer of fibers. When axial and radial cracking of the boron occurs, the resistance in this current path increases drastically, forcing the current into the next inner layer. This process continues with each current pulse, causing progressive catastrophic failure of each successive layer.

A possible solution to this problem was discovered during the second test series. The concept involves a method of making electrical contact to all fiber ends, thereby distributing the current more uniformly and avoiding the damage threshold of the individual fibers.

No degradation was observed in the Thornel 40 — epoxy composites, even at current levels of 21,000 amps.

## NATURAL LIGHTNING AND LABORATORY SIMULATION TECHNIQUES

A review of the characteristics of natural lightning which may encounter an aircraft in flight may be appropriate prior to a description of the simulated lightning currents applied in the laboratory for these tests.

All lightning strokes represent a transfer of electrical charge between two or more charge centers. Such transfers, or discharges, are normally unidirectional (for any given stroke); but, due to the difference in rates of charge transfer between the formation of the complete ionized path between charge centers and the subsequent neutralization of the several centers involved in the stroke, a wide range of current amplitudes may exist. During the formative period of the stroke, current amplitudes (rate of charge flow) of only a few amperes may exist. When the complete path is formed, amplitudes may rise to peak values of between 1000 and 100,000 amperes. Such peaks usually exist for only a few microseconds, whereas lower amplitude currents frequently exist for periods of up to 0.5 second. If more than one charge center is involved, a number of high amplitude peaks may occur, separated by periods of low amplitude current flow as the stroke channel is extended to these other centers. In common terminology the high-amplitude, short-duration peaks are referred to as current "peaks" and the low-amplitude, long-duration currents as "continuing currents." The total event is called the lightning stroke.

If a stroke has more than one peak, it is called a multiple stroke. An oscillogram of a typical stroke to the Empire State Building with continuing currents and current peaks is shown in Figure 2. Measurements accumulated over many years indicate that the median amplitude of lightning current peaks is around 40 kiloamperes (kA), although peaks in excess of 100 kA have been measured.

Less data is available from strokes which extend from cloud-to-cloud and do not terminate on grounded objects. Nevertheless, visual observations and measurements of some of the parameters of lightning currents passing through airplanes in flight support a conclusion that a combination of continuing currents and peaks exist as in cloud-to-ground strokes, although characteristic rates of current rise and peak repetition rates may be different. Some recent measurements (Reference 1) have shown that cloud-to-cloud peaks may occur as frequently as one millisecond apart, whereas cloud-to-ground peaks are generally separated by 10's of milliseconds. The same measurements showed cloud-to-cloud peaks (0 - 22 kA) as being of lower amplitude than the strokes to grounded objects.

1. Strokes to Aircraft

Aircraft in flight may encounter both types of strokes. Presumably, aircraft flying at lower altitudes (≈10,000 feet and less) are hit most frequently by strokes to ground, while those flying at higher altitudes, by cloud-to-cloud strokes.

A lightning stroke to an aircraft may not remain at the first point of attachment because the plane moves through the stroke. The stroke appears to be swept across the surface of the plane, leaving a line of separate burned marks, possibly representing the separate current peaks. If the stroke still exists when it reaches the edge of the surface (i.e., trailing edges, rudder, etc.), it will continue to be attached to the edge and the flow of current may burn or melt the skin at this spot.

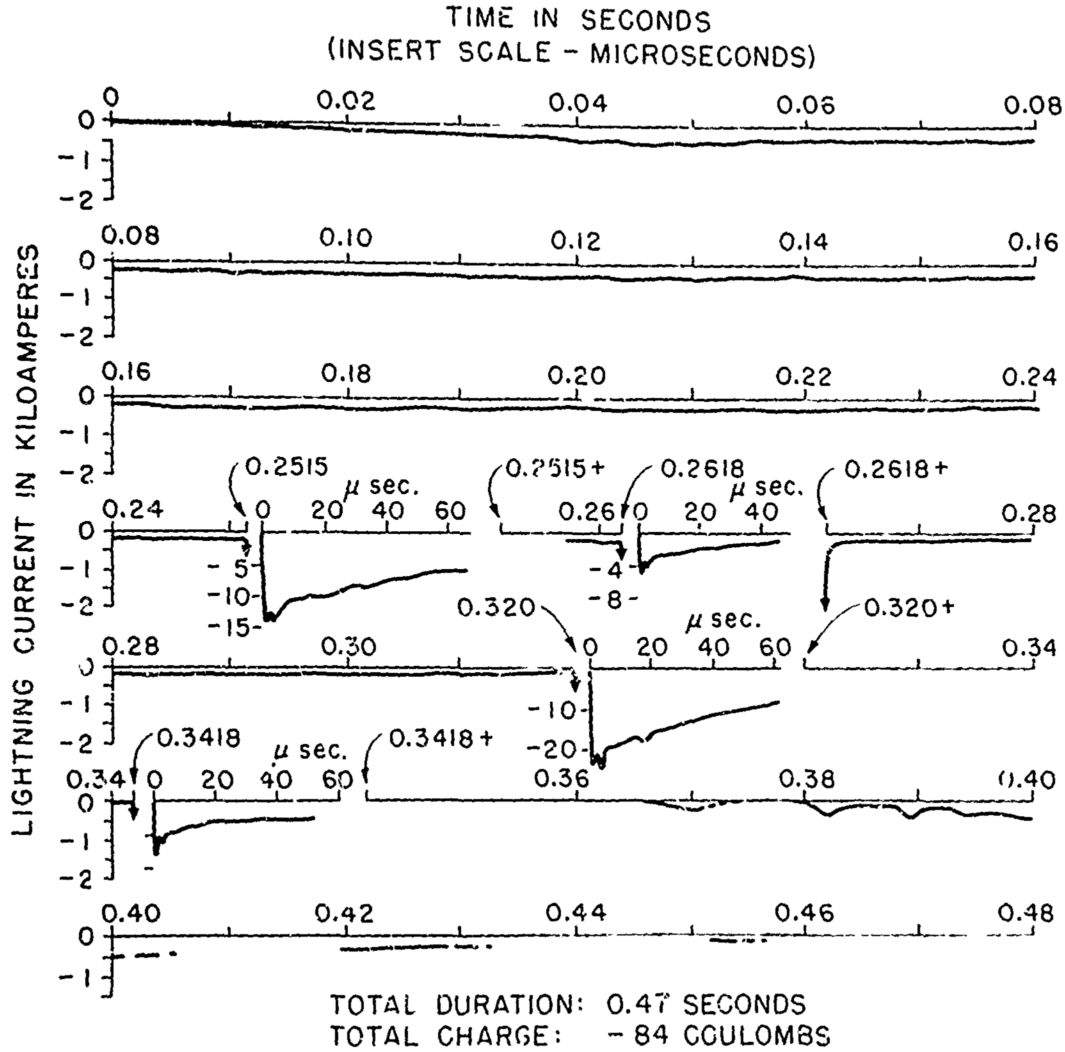



Figure 2. Typical Stroke to Empire State Building Showing Current Peaks and Continuing Currents

Whatever current enters an aircraft must also leave it since the aircraft has insufficient capacitance to store more than a small amount of the stroke's charge. Thus, lightning currents will pass through the structural members as well as the skin of the aircraft.

2. Laboratory Simulation

Since the object of these tests was to determine the effect of lightning current passing through boron and graphite composites, test specimens were not struck by arcs carrying simulated lightning currents. Instead, the currents were conducted into and out of the test specimens by metal conductors (clamps) attached at each end. Thus, burning and blasting effects of high-temperature arcs were not evaluated. The greatest stress would probably be applied to the composites by the current peaks rather than by the continuing currents; since the composite is considered to behave like a resistor, the energy release ($i^2R \times t$) will be greater during a current peak than during the sustained flow. Thus, only the current peaks (impulses) were simulated for these tests.

3. Wave Shapes

Available data shows that current peaks rise to maximum amplitude in a short period of time and decay over a longer period. Many such peaks have rise or "front" times of 1-10 microseconds and decay or "tail" times (measured from start to 50% of crest amplitude on the wave tail) of between 10 and 50 microseconds. A number of such peaks are shown in Figure 2. Accordingly, current pulses within these ranges were applied. Parameters were necessarily different for different current crest amplitudes due to limitations of the impulse generation equipment, but they were consistent for all pulses applied to one or more specimens at a particular amplitude level. Sufficient external (series) resistance was included in the test circuit to prevent specimen resistance changes during current flow from influencing the current wave shape.

4. Test Techniques

a. Current Impulses — Three series of tests were made on the boron and graphite composite specimens, as described in Summary of Investigation. Test Series 1 was a preliminary test of only one specimen of each composite to ascertain the best method of performing more comprehensive tests. In Series 2, groups of specimens were exposed to one or more impulses of identical amplitude and wave shape. The amplitude of the impulses applied to each group was different so that susceptibility of the specimens to both the amplitude and number of impulses could be evaluated. Test Series 3 exposed a reduced number of specimens to higher current amplitudes.

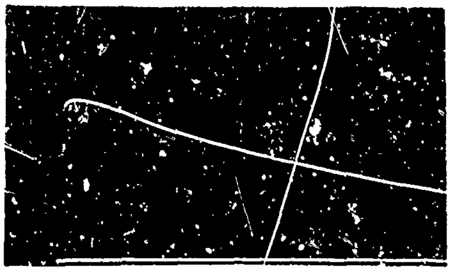
No attempt was made to space the impulses as close together as those occurring in an actual lightning stroke. Successive impulses in these tests were spaced 30 seconds apart, which was the time necessary to recharge the impulse generator. Nevertheless, the technique was considered valid because it shows the cumulative (permanent) effect of several impulses on a specimen and that aircraft surfaces may be exposed to conducted currents from more than one stroke, even though the strokes may have contacted the aircraft at different locations.

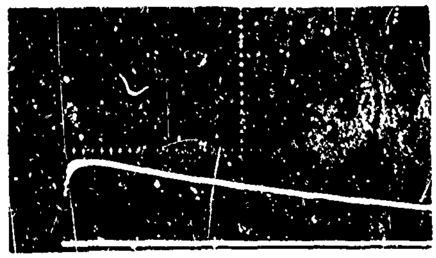
b. Specimen Voltage Measurements — In order to detect any possible transient changes in specimen resistance during the period of current flow, the voltage rise across each specimen as well as the current was measured oscillographically in the second test series. Examples of wave shapes of currents and voltages for a graphite and boron composite specimen are shown on Figure 3.

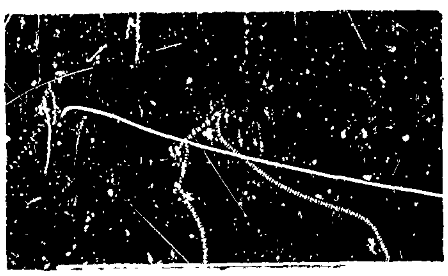
c. Specimen Temperature Measurements — The maximum temperature reached by each specimen was measured by means of Thermopapers. A graded series of papers was taped to each specimen so that the maximum temperature reached could be recorded within ±10°F. It was felt that the low mass and relatively fast response (< 1 second) of these devices would afford a reasonably accurate indication of temperature rise. A limitation of this procedure, however, was that it permitted only the outer surface temperature to be measured and not the temperature of the interior regions of the composite.

SUMMARY OF INVESTIGATION

The results of the three test series, the interpretation of the observations, and a proposed model of the failure mode noted on boron composites are summarized in the following paragraphs.







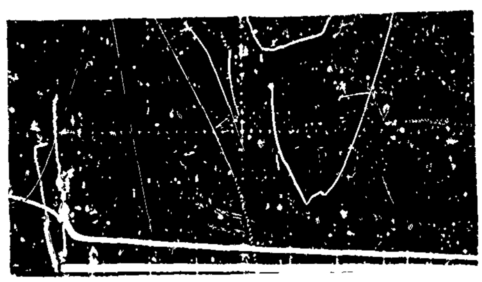


Figure 3. Oscillograms for Current Impulse and Voltage Rise Across a Boron and a Graphite Composite Specimen

1. Series 1 Tests

The first exposure tests were performed on one each boron and graphite fiber reinforced epoxy specimen (Epon 1031 and 828 blend cured with MNA and BDMA). The boron fibers were of the 4 mil type manufactured by Texaco on a 0.5 mil tungsten core. The graphite fibers were Union Carbide Thornel 40. Both specimens had unidirectional fiber orientation and were 6 inches long and approximately 1/2 inch wide. The Thornel 40 specimen had approximately 189 yarns and the boron, 2040 filaments. The impulsive current tests were performed by coating 1/2 inch of each specimen end with a conductive silver-filled epoxy coating. Electrical contacts were made with clamps on each end, as shown in Figure 4. Each specimen was exposed as shown in Table I. A low current amplitude was deliberately used on this first test series so that the progression of any apparent fiber degradation could be observed. Since the specimens remained visually intact following the initial impulses, additional impulses were applied to determine if catastrophic failure would occur. Specimen strength measurements were made after 12 impulses, when the boron specimen was easily broken in two by hand pressure. Visual examination indicated that both boron and graphite specimens showed pitting and flaking at the point of electrical contact, but no change in surface appearance.

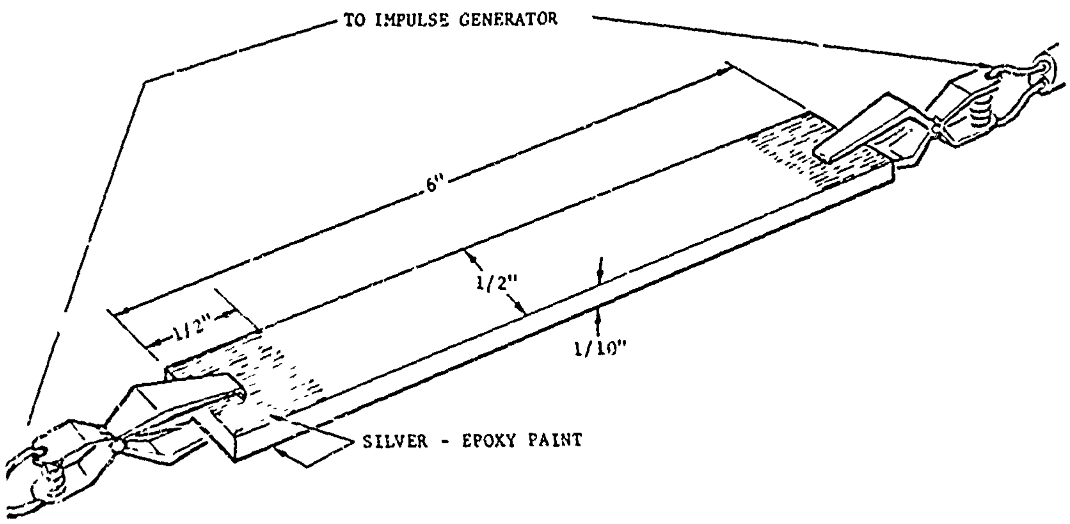



Figure 4. Specimen and Contact Electrode Detail

TABLE I

SERIES 1 TEST CONDITIONS

| | |
|---|---|
| Number of impulses | 12 |
| Total duration of each impulse | 10 $\mu$ sec |
| Peak current for 2 $\mu$ sec | 2 - 3 K amps |
| Specimen temperature rise | Slightly warm to the touch |
| Visual observations | Sparking at contact points |

The boron epoxy specimen was obviously degraded; its unidirectional flexural strength is normally in excess of 200,000 psi, so it would not have failed by slight hand flexing. The temperature rise of each specimen was estimated to be below 150°F, so the boron epoxy degradation and graphite epoxy warpage should not have been due to general temperature rise. The remainder of the boron specimen and the graphite specimen were then tested in flexure and for electrical resistivity, as reported in Table II. These test results indicate significant change for the boron filament composite. The 88.5 percent decrease in flexural strength for the boron composite is further dramatized by the very brittle type failure shown in Figure 5, compared to the typical boron composite flexural fracture shown in Figure 6. The flexural results and the order of magnitude increase in electrical resistivity indicate that some physical change occurred in the boron filaments as a result of exposure to the current impulse.

TABLE II

SERIES 1 TEST RESULTS

| | Boron/Epoxy | | Graphite/Epoxy | |
|---|---|---|---|---|
| | Not Exposed | Exposed | Not Exposed | Exposed |
| Electrical resistivity, ohm-cm | 84.6 | 882 | 0.0025 | 0.0023 |
| Flexural strength, psi | 231,600 | 26,500 | 91,900 | 94,200 |
| Reduction in strength, percent | - | 88.5 | - | - |
| Flexural modulus, psi x $10^6$ | 32.4 | 7.35 | 25.8 | 20.9 |
| Density | 2.17 | 2.15 | 1.48 | 1.48 |

Figure 5. Flexure Failure of Boron Composite After Electrical Current Impulses

Figure 6. Flexure Failure of Typical Boron Composite

To further characterize the boron and graphite fiber epoxy composites, a detailed metallographic inspection was made of the microstructure of each specimen. The appearance of the boron composite was startling. Figure 7 shows a typical boron fiber composite normal to the fiber direction and Figure 8 shows the boron specimen after the current impulse test. Both normal and axial cracks are apparent in the individual fibers. The frequency of occurrence in the cracks normal to the fiber axis is approximately one diameter length. These cracks have disrupted the fibers to such an extent as to produce the gross changes in flexure strength and electrical resistivity. Figure 9 and Figure 10 show photomicrographs of a typical graphite fiber epoxy composite specimen before and after the electrical impulse test. No change is observable. What caused the warpage is uncertain. Since the temperature rise was estimated to be below 150°F peak, this should not have caused the dimensional change, assuming the composite was properly prepared.

2. <u>Series 2 Tests</u>

The second series of tests was planned to study the effect of number of impulses as well as increasing current amplitude on the boron and graphite epoxy composites. At current levels of 2, 4, and 8 kA, both specimen types were exposed for 1, 3, and 7 impulses. In addition, the graphite fiber specimens were exposed for 1 and 3 impulses at 21 kA.

In the preparation of the specimens, one significant change was made in an attempt to improve the electrical contact. At each end of the specimens 1/2 inch was plated with 8 mils of nickel instead of the silver-loaded epoxy coating. Prior to the plating, a 40° bevel was cut with a diamond blade dicing saw to expose more fiber end contact area. Also, the ends of the specimens were carefully sandblasted to remove any resin rich surface, so that the nickel would be in intimate contact with the fibers.

The boron and graphite specimens were auto-clave cured, using the same resin system, fibers, and techniques as in test Series 1. The individual specimens were 6 inches long, approximately 1/2 inch wide, and 0.1 to 0.125 inches thick. The characteristics of the panels from which the specimens were cut and the number of fibers in each specimen are given in Table III. The exact crest

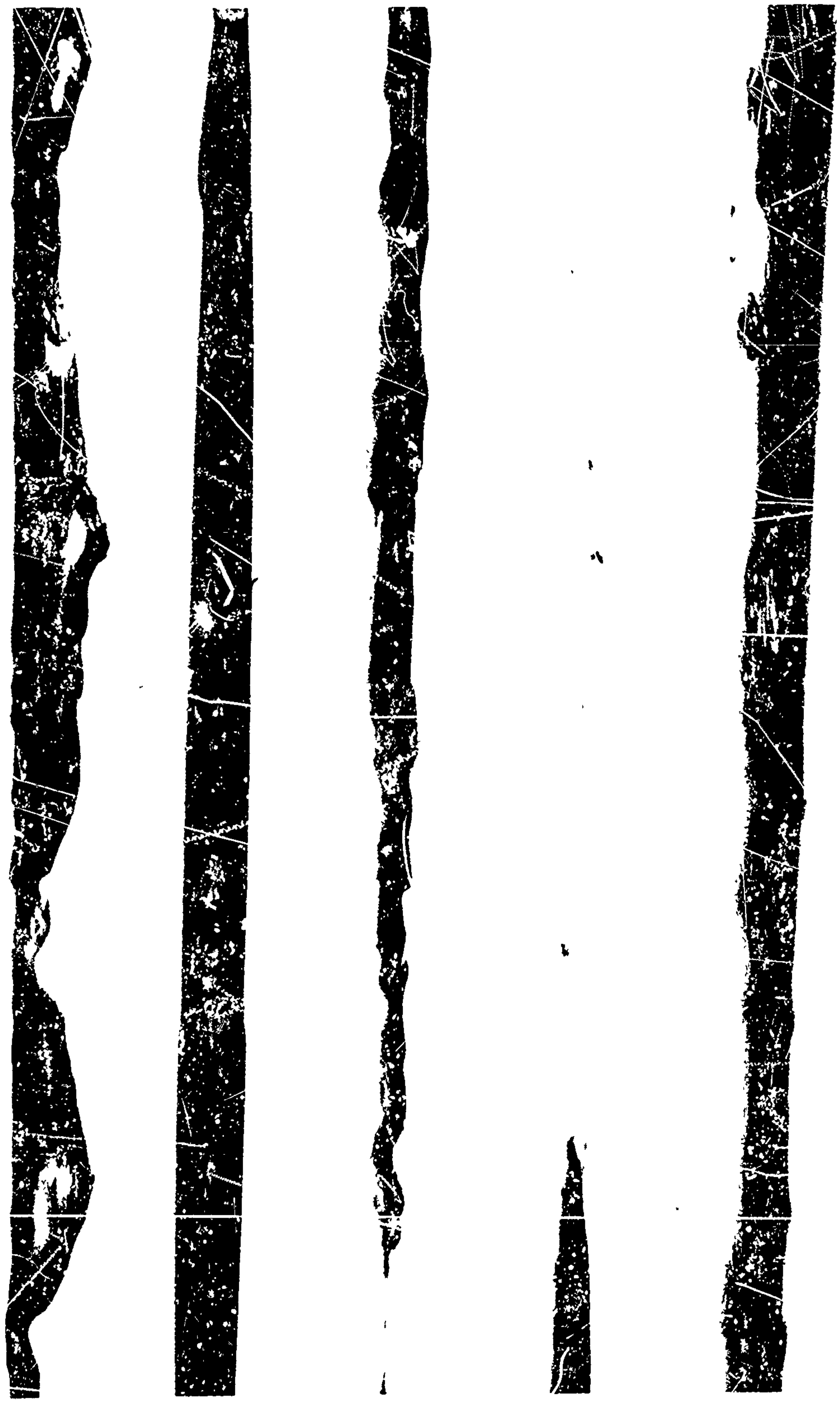


Figure 7. Typical Boron Composite

Figure 8. Boron Composite After Current Impulse Test

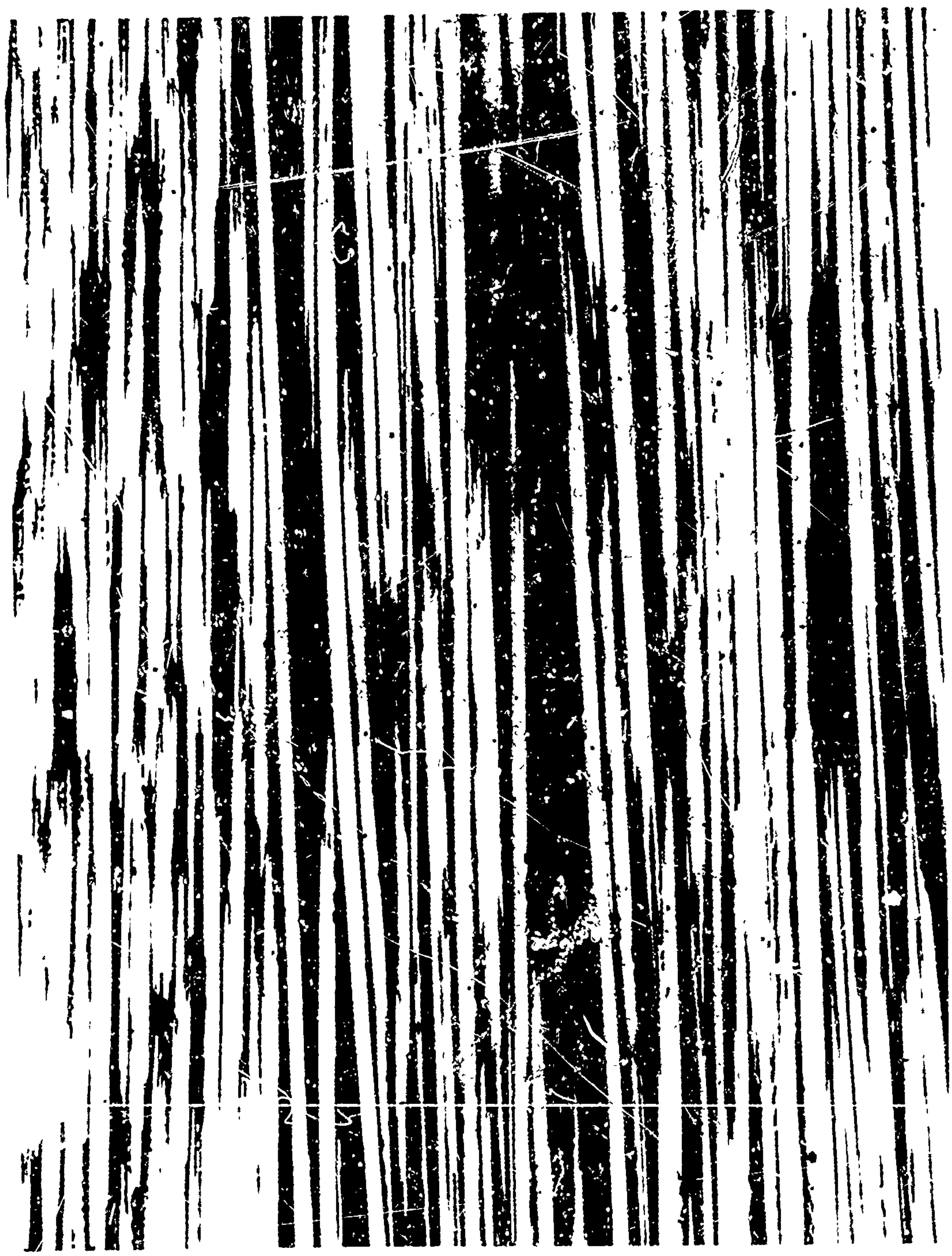


Figure 9. Typical Graphite Composite

Figure 10. Graphite Composite After Current Impulse Test

current amplitude, waveform duration, and number of pulses for the 11 graphite and 9 boron fiber specimens are given in Table IV.

Electrical contacts were made by clamping strips of copper to the nickel plated ends of the specimens. Following the exposures, the resistivities of each specimen were measured and compared to the value before exposure results are reported in Table V. The measured flexural properties of the composite specimens after exposure are given in Table VI; each is compared to the average value for unexposed specimens.

Of the specimens tested, only two exhibited significant strength degradation; the boron specimens exposed to 7 impulses at 4.08 K amps and to a combination of 8 impulses at 2 K amps, 8 impulses at 4 K amps, and 3 impulses at 8.44 K amps. These specimens exhibited strength reductions of only 18 percent and 25 percent, respectively. No significant modulus degradation was noted. The graphite fiber composite specimens did not exhibit any strength or modulus degradation, even with 1 and 3 pulses at 21 K amps. The latter specimen experienced a 260°F maximum temperature, the highest rise noted. There was no specimen warpage.

The most severely exposed boron composite was the only specimen that exhibited a considerable increase in electrical resistivity. This specimen exhibited a 25 percent reduction in flexural strength and a 345 percent increase in resistivity; in Series 1 tests, the boron composite exhibited an 88.5 percent decrease in strength and a 1050 percent increase in resistivity. These two results indicate a 3 1/2 times increase in flexural strength degradation corresponded to a 3 times increase in the measured electrical resistivity. This suggests a correlation between the changes in electrical resistivity and flexure strength.

In the nonstructurally degraded boron and graphite composite specimens, resistivity actually decreased, typically 19 - 27 percent for boron and 7 - 12 percent for graphite. There is presently no explanation for this decrease.

The difference in the measured resistivities of the boron composite specimens in test Series 1 and 2 should be noted. This difference is attributed to the different type of electrical contact — silver-loaded-epoxy coating versus nickel plating. This did not affect resistivity measurements for the graphite composite

TABLE III

PROPERTIES OF SERIES 2 TEST PANELS

| Fiber | Panel No. | Density, $gm/cm^3$ | Specific Gravity | Resin Content, Percent by Weight | Fiber Volume, Percent | Void Content, Percent | Approximate Number of Filaments or Yarns per Specimen |
|---|---|---|---|---|---|---|---|
| Boron | 1 | 1.97 | 1.95 | 24.0 | 58.3 | 5.2 | 4140 Filaments |
| Thornel 40 | 1 | 1.36 | 1.34 | 42.9 | 49.3 | 2.0 | 450 Yarns |
| Thornel 40 | 2 | 1.34 | 1.37 | 35.5 | 56.6 | 2.9 | 450 Yarns |

TABLE IV

SERIES 2 TEST CONDITIONS

| | Crest Current Amplitude KA | Impulse Rise Time $\mu$ sec | Time for Current to Decay to 50% of Crest $\mu$ sec | Number of Impulses Per Specimen |
|---|---|---|---|---|
| Boron Specimens* | 2.00 | 1 | 34 | 1, 3, & 7 |
| | 4.68 | 1 | 34 | 1, 3, & 7 |
| | 8.18 | 1 | 25 | 1, 3, & 7 |
| Graphite Specimens | 2.01 | 1 | 34 | 1, 3, & 7 |
| | 4.16 | 1 | 34 | 1, 3, & 7 |
| | 8.44 | 1 | 25 | 1, 3**, & 7** |
| | 21.00 | 1 | 11 | 1**, & 3** |

*The 3 pulse boron test specimen, prior to the 3 pulse test, had been exposed to 8 pulses each at 2 K amps, and 4 K amps during circuit tryouts.
**Graphite panel number 2.

## TABLE V

## SERIES 2 RESISTIVITY TEST RESULTS

| Current KA | Resistivity Before and After Test (ohm-cm) and Peak Temperature During Test | | | | | |
|---|---|---|---|---|---|---|
| | 1 Impulse | | 3 Impulses | | 7 Impulses | |
| | Before | After | Before | After | Before | After |
| Boron Specimens | | | | | | |
| 2.00 | 0.0112 | 0.0085 (< 130°F) | 0.0139 | 0.0102 (< 130°F) | 0.0116 | 0.0094 (< 130°F) |
| 4.08 | 0.0116 | 0.0085 (< 130°F) | 0.0099 | 0.0079 (130°F) | 0.0126 | 0.0125 (170°F) |
| 8.18 | 0.011[illegible] | 0.0080 (< 130°F) | 0.0096 | 0.0073 (170°F) | 0.0145 | 0.0501* (210°F) |
| Graphite Specimens | | | | | | |
| 2.01 | 0.0026 | 0.0023 (< 130°F) | 0.0028 | 0.0025 (< 130°F) | 0.0029 | 0.0027 (< 130°F) |
| 4.16 | 0.0028 | 0.0026 (< 130°F) | 0.0026 | 0.0024 (< 130°F) | 0.0026 | 0.0024 (< 130°F) |
| 8.44 | 0.0027 | 0.0025 (< 130°F) | 0.0027 | 0.0025** (< 130°F) | 0.0028 | 0.0026** (130°F) |
| 21.00 | 0.0027 | 0.0025 (190°F) | 0.0026 | 0.0024* (260°F) | - | - - |

*The 3 pulse boron test specimens, prior to the 3 pulse test, had been exposed to 8 pulses each at 2 K amps and 4 K amps during circuit tryouts.

**Graphite panel number 2.

TABLE VI

SERIES 2 MECHANICAL TEST RESULTS

| Current, kA | Flexural Strength, K psi / Flexural Modulus, $10^6$ psi | | | | | |
|---|---|---|---|---|---|---|
| | 1 Impulse | | 3 Impulses | | 7 Impulses | |
| | Control | Exposed | Control | Exposed | Control | Exposed |
| **Boron Specimens** | | | | | | |
| 2.00 | 216<br>29.2 | 209<br>29.3 | 216<br>29.2 | 194<br>27.7 | 216<br>29.2 | 211<br>28.7 |
| 4.08 | 216<br>29.2 | 209<br>28.4 | 216<br>29.2 | 224<br>30.1 | 216<br>29.2 | 178<br>29.9 |
| 8.18 | 216<br>29.2 | 217<br>32.2 | 216<br>29.2 | 218<br>32.3 | 216<br>29.2 | 163**<br>23.0** |
| **Graphite Specimens** | | | | | | |
| 2.01 | 81.2<br>15.5 | 80.1<br>16.5 | 81.2<br>15.5 | 84.6<br>15.1 | 81.2<br>15.5 | 81.4<br>16.8 |
| 4.16 | 81.2<br>15.5 | 84.6<br>15.8 | 81.2<br>15.5 | 80.0<br>14.7 | 81.2<br>15.5 | 83.9<br>16.0 |
| 8.44 | 81.2<br>15.5 | 80.3<br>15.9 | 77.5<br>16.0 | 88.6<br>17.1 | 77.5<br>16.0 | 79.6<br>16.8 |
| 21.00 | 77.1<br>16.0 | 77.7<br>17.7 | 77.5<br>16.0 | 76.9<br>16.8 | -<br>- | -<br>- |

*Boron control specimens: 4 specimens; flexural strength range 190-234K psi; flexural modulus range = 26.5-30.6 x $10^6$ psi.

Graphite control specimens: Panel No. 1: 5 specimens; flexural (strength range = 64.1-87.0 K psi (avg. = 77.5); flexural modulus range = 14.5-17.5 x $10^6$ psi (avg. = 16.0). Panel No. 2: 5 specimens; flexural strength range = 75.1-92.5 K psi (avg. = 81.2); flexural modulus range = 15.4-15.6 x $10^6$ psi (avg. = 15.5).

**The three impulse boron test specimen prior to the 3 impulses had been exposed to 8 impulses each at 2 K amps and 4 K amps during circuit tryout.

specimens. This same difference in electrical contact is also suspected as the reason a 2 K amp current level caused degradation to the boron composite specimens in Series 1 but not in Series 2 tests.

3. Series 3 Tests

A third set of tests were performed to determine the effect of higher current levels on boron composites. Two additional unidirectional boron specimens were tested at a current level of 30 - 40 K amps. The two specimens were of the same size and from the same panel as the Series 2 specimens. Timing did not permit the nickel plating at the ends for the electrical contacts; instead, a silver-loaded-epoxy coating, as was used in Series 1, was applied to the beveled ends. The ends of the specimens were sanded prior to coating, which resulted in lower values of resistivity than for Series 1. The resistivity measurements, however, were higher than for the nickel-plated-contact specimens of Series 2.

The test exposures for the Series 3 specimens are given in Table VII. The measured properties of the two boron composite specimens, before and after test, are presented in Tables VIII and IX.

The flexural strengths were degraded 56 percent and 38 percent for specimens 1 and 2. These levels of degradation correspond to increases in electrical resistivity of 805 percent and 318 percent, respectively.

As in the case of the Series 1 tests, metallographic inspections were performed to obtain visual evidence of fiber degradation. Specimen 1 was extracted with dimethylformamide to expose the individual filaments. Figure 11 shows a fiber with the same type of transverse cracks and segmentation observed in the Series 1 tests. Figure 12 is a cross sectional view of a filament at one of the transverse failure planes. Note the axial cracks seen in the Series 1 specimen. Figure 13 shows a cross-section of a filament that has separated along the axial crack paths. Observe the segmentation due to the transverse cracks.

The most severely degraded filaments were on or near the 6 inch by 1/2 inch surfaces of the specimens, and the severity of degradation varied with distance

TABLE VII

SERIES 3 TEST CONDITIONS

| Boron Specimen Number | Crest Current, kA | Impulse Rise Time $\mu$ sec | Time for Current to Decay from Crest to 50 percent of Crest $\mu$ sec | Number of Pulses per Specimen |
|---|---|---|---|---|
| 1 | 30 | 6.5 | 6.0 | 1 |
| | 33 | 6.5 | 6.0 | 1 |
| | 38 | 6.5 | 6.0 | 1 |
| 2 | 36 | 6.5 | 6.0 | 1 |

TABLE VIII

SERIES 3 RESISTIVITY TEST RESULTS

| Boron Specimen Number | Resistivity Before and After Test (ohm-cm) and Peak Temperature During Test | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | 30 K Amps | | 33 K Amps | | 36 K Amps | | 38 K Amps | |
| | Before | After | Before | After | Before | After | Before | After |
| 1* | 0.256 | 0.164 (105°F) | 0.164 | 0.613 (160°F) | | | 0.613 | 2.06 (170°F) |
| 2** | | | | | 0.0120 | 0.0382 (160°F) | | |

*The specimen was exposed to 30, 33, and 38 K amps and permitted to cool to room temperature between pulses.

**This specimen was beveled on the ends and plated with nickel for the after test resistivity measurement and compared to a typical before test plated contact specimen. It was tested with only silver loaded epoxy coated ends.

TABLE IX

SERIES 3 MECHANICAL TEST RESULTS

| Boron Specimen Number and Number of Impulses | Flexural Strength, K psi | | Flexural Modulus, $10^6$ psi | |
|---|---|---|---|---|
| | Control | Exposed | Control | Exposed |
| 1 | 216 | 94.8 | 29.2 | 25.3 |
| One impulse each at 30, 33, and 38 K amps | | | | |
| 2 | 216 | 134 | 29.2 | 29.6 |
| One Impulse at 36 K amps | | | | |

*The control values are average and are the same as previously noted for the series number 2 tests.

from the surface. Some filaments at the surface were nothing but powder, some further in had axial separation, one-diameter length segmentation, and numerous axial cracks. Filaments near the center experienced segmentation on the order of one diameter length as shown in Figure 10, and those in the center were completely unaffected as shown in Figure 13.

For specimen 1, which had a 56 percent reduction in flexural strength, degradation of filaments, based on microscopic examination, was estimated as follows:

| | |
|---|---|
| Severely affected filaments | 15% |
| Slightly affected filaments | 10% |
| Unaffected filaments | 75% |

Again, the severity of degradation decreased rapidly from the outer surface to the interior.

4. Discussion of Test Results

Boron filament composites degrade structurally when exposed to simulated lightning current impulses. Degradation can occur at current amplitudes

Figure 11. Extracted Boron Fiber Showing Transverse Cracks and Segmentation

a. Transverse Failure Plane Showing Axial Cracks

b. Separated Along Axial Crack Plane

Figure 12. Cross Section of Boron Filament

Figure 13. Unaffected Boron Filament From Center of Degraded Specimen

considerably below those of many natural lightning strokes, in some cases at current levels as low as 2000 amps. Unidirectional graphite composites of approximately 0.1 inch thickness, except for warpage in the Series 1 tests, did not degrade at current levels of up to 21,000 amps. The degradation of the boron composites was shown by metallographic examination to be due to degradation of the boron filaments; degradation modes included powdering of filaments, incremental damage (such as severe transverse cracking and filament segmentation), and multiple axial cracking.

Figure 14 shows the relationship between changes in flexural strength and changes in resistivity for degraded boron filament composites. Inspection indicated that degradation levels progressed from occasional filament segmentation, to one diameter frequency segmentation plus axial cracks, to almost complete filament powdering. As would be expected, this damage causes progressively greater interruption of the filament current, which explains the correlation between strength degradation and change in electrical resistivity of the composite.

As previously noted in one of the Series 3 tests, the degree of boron filament degradation was maximum on the 1/2 inch wide surfaces of the specimen and decreased progressively from these surfaces inward. Only about one-fourth of the filaments were degraded, but this corresponds to a degradation in flexural strength of 56 percent. The degraded filaments were in the outer layers, where the stress for flexural test is greatest, so the disproportionate decrease in strength with 25 percent of fibers degraded is expected. In tensile test, the strength reduction is only about 25 percent, which would be expected because the filaments are stressed equally.

The degradation in the outer filaments of the Series 3 specimens is believed due to their electrical contacts. The silver-loaded-epoxy coating would cause the current to enter the specimen through the outer plies due to the high resistance of the resin film between the plies of filaments. The first pulse degrades the outer plies and increases their resistance; the current then begins to flow deeper into the specimen. The specimen acts like a parallel circuit of resistors, or plies, in which the resistance of the outer circuits increases with each pulse. In the Series 3 specimen that had three pulses of 30,000, 33,000, and 38,000 amps,

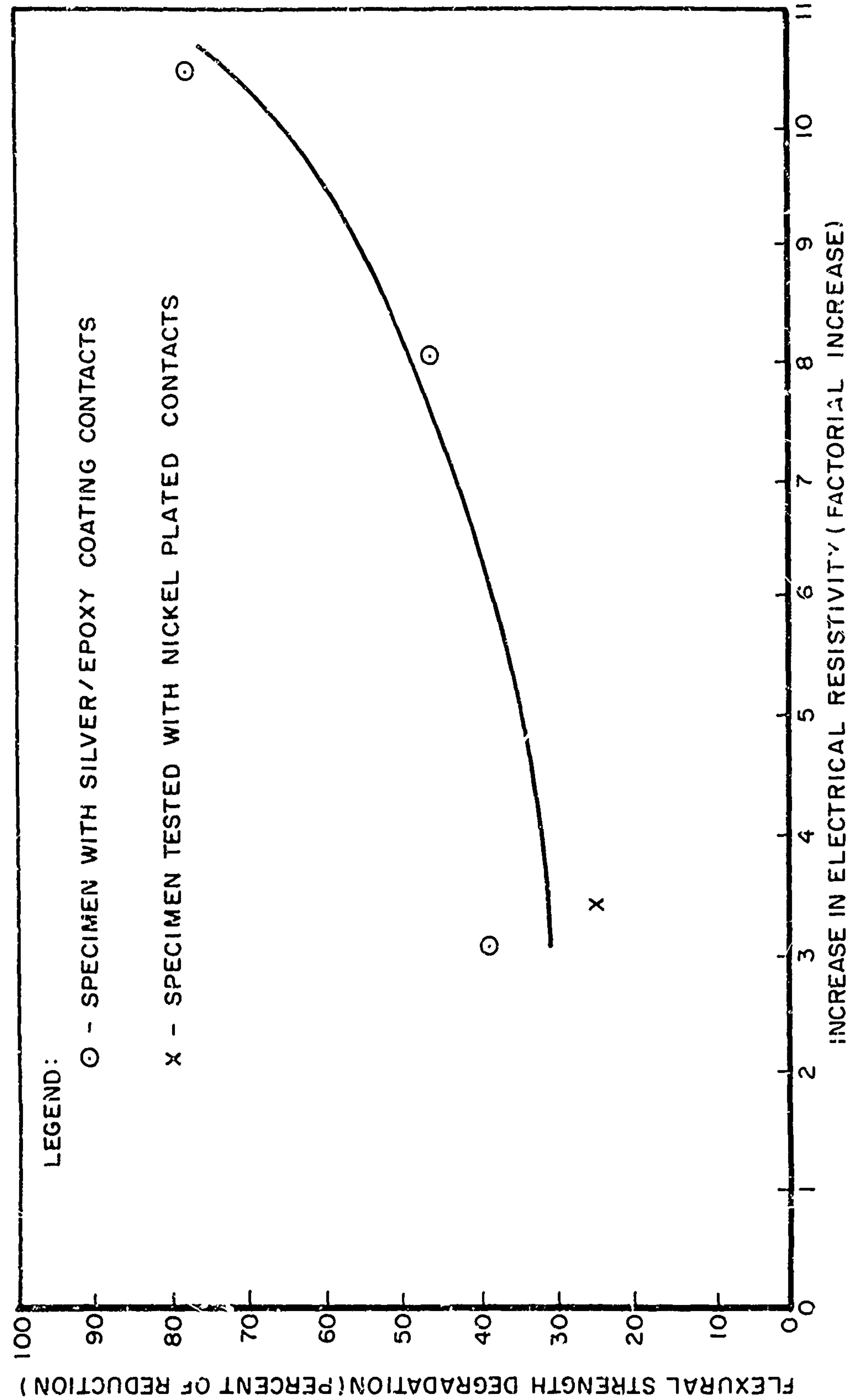



Figure 14. Flexural Strength Degradation Versus Increase in Electrical Resistivity

25 percent of the 4140 filaments (1035 filaments) were degraded to some extent, which corresponds to degradation of 2 - 3 plies on each side. Thus, it might be assumed that each impulse went one ply deeper into the specimen, resulting in three degraded plies on each surface. If this is so, on the first impulse of 30,000 amps, each filament of the outer ply would be carrying approximately 65 amps and for the 36,000 amps impulse, each filament would have carried 78 amps.

The degradation of the outer filaments may also be due to skin effects, which causes short-duration current pulses to travel on the surface of conductor. Further investigation is needed to determine the degree to which each of these factors is involved.

Although not investigated, it is presumed that the Series 1 specimens were most severely degraded at the outside surface, since they had silver-epoxy coated ends. Assuming the outer ply on each side carried the first 2000 - 3000 amp pulse, the current per filament was approximately 8 amps. This would explain the discrepancy in degradation at 2000 - 3000 amp exposure between Series 1 and Series 2 specimens. In Series 2, the current per filament at 2000 amps was only about 0.5 amp due to the improved (to filament) conductivity of the beveled, nickel-plated, end contacts. About 1.0 and 2.0 amps per filament would be experienced at 4080 and 8180 amp levels in Series 2.

The number of pulses or strokes (or possibly the cumulative total current) also appears to be a factor in the degradation of boron filaments. This is significant, since natural lightning is a series of multiple strokes. In Series 2, the specimen showing the most significant degradation was the one exposed to 8 pulses of 2000 and 4000 amps each, and 7 pulses at 8180 amps. The specimens exposed to 1, 3, and 7 pulses at 2000 amps, 1 and 2 pulses at 4080 amps, and 1 and 2 pulses at 8180 amps showed no degradation.

a. Voltage Measurements - The voltage rise across the specimens was measured in Series 2. Since the specimens must be largely resistive, it was expected that the voltage wave shape would be proportional to the current wave shape. From the voltage and current oscillograms, the resistances were calculated (e/i) at several different points on the wave form.

Calculations show that the resistance of the boron specimens remained essentially constant during current flow and identical with the values measured before and after current exposure; however, none of the boron specimens of Series 2 was tested to severe material degradation or catastrophic failure. When a boron specimen did undergo severe material degradation, as in Series 3, a permanent change in resistance was apparent.

Unlike the boron specimens, voltage measurements made of the graphite specimens tested during Series 2 indicated a transient resistance change which occurred as a function of current flow. While the measured values of resistance of each specimen before and after current flow were nearly unchanged, a marked change in resistance did occur as evidenced by the voltage oscillograms (see Figure 3). The relation between current and voltage on this oscillogram taken during a 2 kA exposure, was similar to measurements made at higher currents. In addition, there appeared to be no variation in this relationship between specimens given one or more than one impulse. Observation of Figure 3 shows evidence of two characteristics:

(1) The voltage and current wave shapes were not identical, particularly in the vicinity of the wave crests.

(2) At no time during the current flow is the calculated resistance of the specimen equal to the value measured prior to test (0.0922 ohms).

Values of resistance for the graphite specimen, in ohms, at crest are as follows:

| | |
|---|---|
| Measured value prior to test | .0922 |
| Calculated value during test | .395 |
| Measured value after test | .0922 |
| Calculated value at 50% of crest on wave tail | .214 |

As shown in the table, the resistance of the graphite specimen following exposure was identical to that before. However, the resistance of the specimens during current flow was several times greater. No explanation is immediately evident for this phenomenon, although such a transient resistance increase suggests some temporary change in physical characteristics. Temperature coefficients of resistivity for these materials would not be significant at the temperatures measured during these tests and, in the case of graphite at -0.005/°C, would act to reduce rather than increase, the specimen resistance.

To clearly define this characteristic, tests should be made in which the voltage and current are accurately measured and compared during the first fractions of a microsecond. Such measurements could be accomplished using faster oscilloscope sweep settings than were used for these tests. In this manner, resistance changes could be traced during the initial portion of the impulse from the time the current first begins to flow through the specimen.

This investigation was not intended to evaluate the possible effects on aircraft electrical systems from lightning currents passing through semiconducting materials such as these, or the effects of the reduced electromagnetic shielding provided by such materials. The simple voltage measurements made here, however, do serve to illustrate that substantial voltage rises can occur as lightning currents pass through some segment of composite materials.

b. Summary of Results - The following observations are made as a result of these tests:

(1) Each boron filament has a threshold current at which degradation begins.

(2) The degree of boron composite degradation is dependent on both the number and amplitude of simulated lightning current impulses.

(3) The key to minimizing gross degradation in boron filament composites due to simulated lightning current impulses appears to be to distribute the current simultaneously through as many filaments as possible.

(4) The degradation to boron composites due to simulated lightning current passage can be severe and not be visually detected by ordinary inspection.

(5) These tests served as a method of evaluating the effects of lightning currents on composite materials by permitting measurement of material degradation as a function of the amount of simulated lightning current to which a material specimen is exposed. This type of testing may prove desirable and economical once criteria for composite material degradation are established.

The technique of testing numerous small specimens of uniform dimensions permitted exposure of the material to varied amount of simulated lightning current and facilitated determination of degradation thresholds. Some refinements are desirable, but similar techniques could be utilized to evaluate other questions, such as the susceptibility of composites to continuing currents and to lightning current flow through the composite in a direction of the individual fibers. In addition, the technique should be useful for developmental evaluation of protective measures.

This test method is useful for these purposes, it is not directly applicable to the investigation of some other problem areas. For example, the susceptibility of more complex structures to lightning currents or the burning or blasting effects associated with arc-conducted current entry into a larger panel of such material was not evaluated. These tests indicated that the boron composite may be subject to severe degradation as a result of lightning current passing through it. If such composites are to be used in aircraft, the degree of degradation should be evaluated extensively and protective measures developed.

## 5. Fiber Failure Mathematical Model

The nature of boron filament failure when exposed to excessive electrical current indicates excessive thermal expansion of the filament core. Much of the original tungsten core is assumed converted to tungsten boride during filament processing. Since the boron has much greater resistance than the tungsten boride core, virtually all of the current was assumed to be carried by the core. Resistance heating of the core due to current flow expands it to the point that excessive stresses build up in the surrounding boron sheath.

Assuming this failure model, the degree of correlation with failures noted in the tests was determined by calculating failure stresses at different current levels. In calculating the relationship of current flow and failure stress using the analytical model, the following assumptions were made:

(1) Virtually all of the current flows through the core.

(2) The time duration is sufficiently short to prevent heat transfer between the core and the boron.

(3) All energy dissipated in the core is in the form of heat.

(4) Both the boron and the core are homogenous.

The temperature rise in the core was calculated from the following electrical energy dissipation - heat energy balance as follows:

$$W = I^2 Rt = Q = MC_p \Delta T \tag{1}$$

where

- $I$ = current through the wire (amp)
- $R$ = resistance of wire ($\Omega$)
- $t$ = time duration of current flow (sec)
- $M$ = approximate mass of core material (gms)
- $C_p$ = specific heat of tungsten at constant pressure ($\frac{BTU}{LBM\,°F}$)
- $\Delta T$ = temperature rise of tungsten (°F)

To utilize available property data, additional relationship are incorporated as follows:

$$R = \rho_r \frac{\ell}{A} ; \quad M = \rho_v A \ell \tag{1a}$$

where

- $\rho_r$ = resistivity of the tungsten ( $\Omega$ cm)
- $\ell$ = wire length
- $A$ = wire cross-sectional area
- $\rho_v$ = density of tungsten (gm/cc)

Substituting into Equation (1) and solving for the temperature rise:

$$\Delta T = \frac{I^2 \rho_r \frac{\ell}{A} t}{\rho_v A \ell C_p} = \frac{I^2 \rho_r t}{\rho_v A^2 C_p} \tag{2}$$

The internal pressure due to the thermal expansion was calculated as follows. Since the internal core radius ($R_1$) is much smaller than the external filament radius ($R_2$), the thick walled cylinder model was selected for determining the stresses in the boron. Computation of stresses in a composite cylinder are statically indeterminate (Reference 1), but may be solved by equating the radial deflection of the interface of the two cylinders ($\delta R$).

The radial deflection of the boron is characterized by (Reference 2):

$$\delta R_B = \frac{P_1 R_1}{E_B}\left[\frac{R_2^2 + R_2^2}{R_2^2 - R_1^2} + \mu_B\right] \qquad (3)$$

where

$P_1$ = internal pressure on boron (psi)

$E_B$ = modulus of elasticity (psi)

$\mu_B$ = Poisson's ratio

R's = initial values for the radii

The radial deflection of the boron is composed of two terms, the increase due to thermal expansion minus the contribution due to compression of the tungsten by the boron.

$$\delta R_W = R_1 \alpha_\ell \Delta T - \frac{P_1 R_1}{E_W}(1 - \mu_W) \qquad (4)$$

where

$\alpha_\ell$ = linear coefficient of expansion of tungsten (in./in. °F)

$E_W$ = modulus of elasticity (psi)

$\mu_W$ = Poisson's ratio

Equating the radial deflections and solving for the pressure yields:

$$P_1 = \frac{\alpha_\ell \Delta T\, E_W E_B}{\left[\frac{R_2^2 + R_1^2}{R_2^2 - R_1^2} + \mu_B\right] E_W + (1 = \mu_W) E_B} \qquad (5)$$

The maximum stress in the Boron due to this internal pressure is tensile and derived from

$$\delta_{T_{Max}} = P_1 \left[ \frac{R_2^2 + R_1^2}{R_2^2 - R_1^2} \right] = \frac{\alpha_\ell \Delta T \left[ \frac{R_2^2 + R_1^2}{R_2^2 - R_1^2} \right] E_B E_w}{\left[ \frac{R_2^2 + R_1^2}{R_2^2 - R_1^2} + \mu_B \right] E_w + (1 - \mu_w) E_B} \tag{6}$$

The physical properties and design criteria for the calculations were:

$$\begin{aligned}
\rho_r(25°) &= 21 \times 10^{-6}\ \Omega\,\mathrm{cm}^2 \\
t &= 10^{-5}\ \mathrm{sec} \\
\rho_v(W_2B_5) &= 13.0\ \mathrm{gm/cc}\ (\text{Reference } 3) \\
C_p(B) &= 0.06 - 0.09\ \mathrm{BTU/LBM\ °F}\ (\text{Reference } 4) \\
\alpha_\ell(WB) &= 4.1 \times 10^{-6}\ \mathrm{in/in\ °F}\ (\text{Reference } 5) \\
E_B &= 60 \times 10^6\ \mathrm{psi}\ (\text{Reference } 4) \\
E_w &= 60 \times 10^6\ \mathrm{psi}\ (\text{Reference } 4) \\
\mu_w &= 0.3\ (\text{Reference } 4) \\
\mu_B &= 0.2\ (\text{Reference } 4) \\
R_1 &= 2.5 \times 10^{-4}\ \text{inches} \\
R_2 &= 2.5 \times 10^{-3}\ \text{inches}
\end{aligned}$$

Calculations made for currents of one and twenty amps per filament are given in Table X. Using these data points, a curve of the log of the maximum tensile stress in the boron was plotted as a function of the log of the current per filament (Figure 15). This indicated filament failure would occur at about 10 amps per filament; this appears consistent with failures noted at 8 to 78 amps per filament.

TABLE X

CALCULATED FILAMENT INTERNAL STRESSES

| Current/Filament (amps) | Temperature Rise (°F) | Pressure (psi) | Stress (psi) |
|---|---|---|---|
| 1 | 72 | 9200 | 9400 |
| 20 | 29000 | $3.7 \times 10^3$ | $3.8 \times 10^6$ |

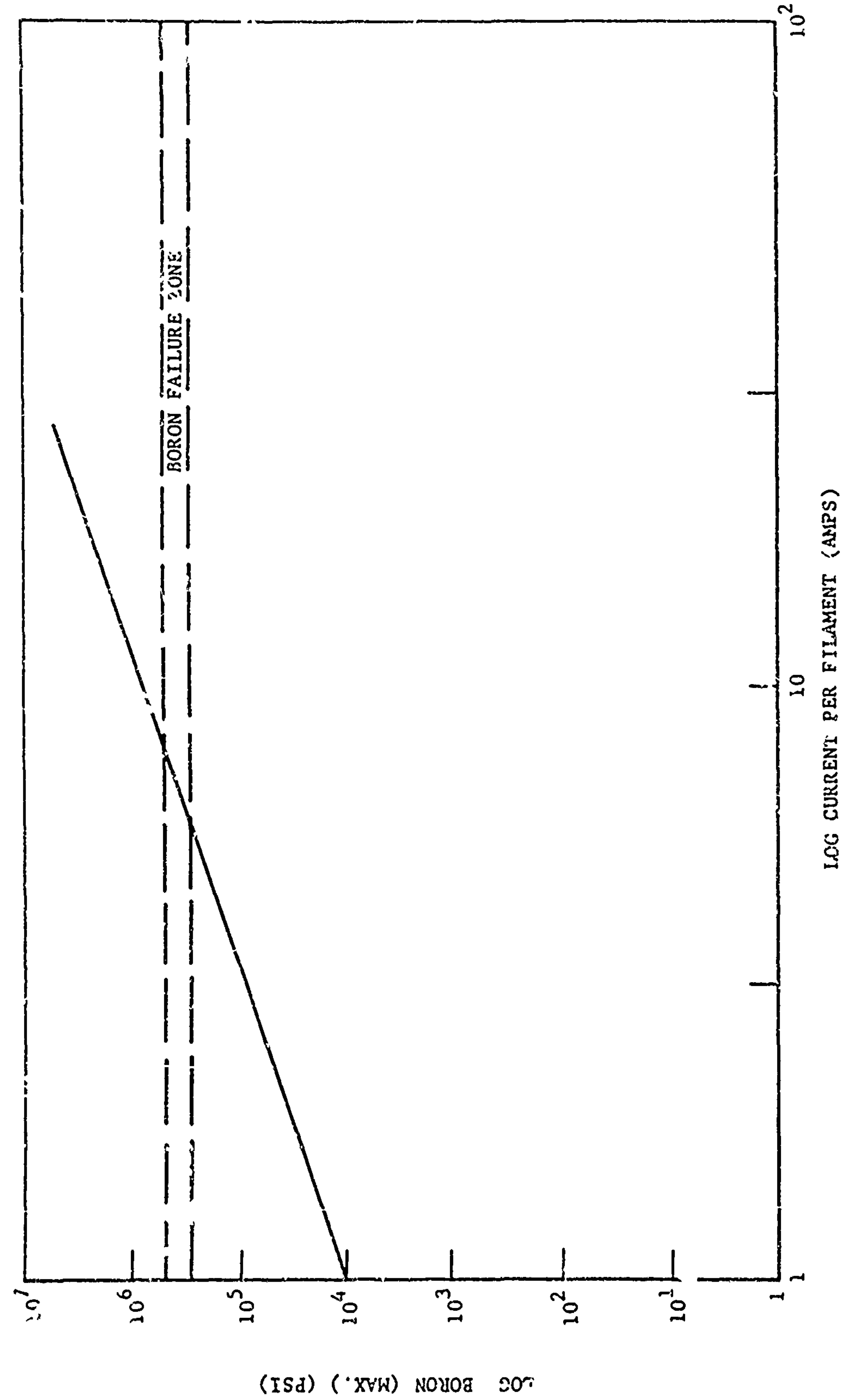



Figure 15. Stress in the Boron Filament Sheath Versus Current Flow in the Filament Core

It is believed that the model is good and that corrections due to dynamic temperature effects on physical properties, the percentage of current running through the boron, heat transfer, and insufficient property data and core composition characteristics will not alter the results by more than an order of magnitude.

6. Current Flow and Degradation Profile Mathematical Model

An analytical model has been envisioned for the case of an electrical current impulse introduced into one side of a composite, or one ply, but it has not yet been reduced to practice. A multi-ply array boron-filament-reinforced epoxy appears to act like a complex resistor circuit. The resistance of filaments in a single ply is much less than the resistance of the interply or inter-filament resin film. Thus, the first stroke or pulse of current is carried by the outer ply of filaments. When the threshold current of degradation of that ply has been exceeded, the filaments fail by transverse and axial cracking, which then interrupts the current flow path of the ply by grossly increasing the resistance. When the filaments fail to the extent that their resistance is greater than that of the inter-ply of interfilament resin film, the current goes to the next deeper filament or layer of filaments. The composite acts like a parallel circuit of filaments and resin films, as illustrated in Figure 16.

When the external filament resistance, $R_1$, grossly exceeds the resin film resistance, $R_F$, more of the current will flow through the next deeper filament, $R_2$. When $R_1$ and $R_2$ combined grossly exceed the resistance of the two resin films, $2R_F$, then more of the current will flow through the ne t level of filaments, $R_3$.

In the case of current introduced at the edge of the panel into filaments that intersect the edge, the filaments act as parallel circuits of equal resistance with equal current flow. The maximum current per filament then is lower. This corresponds to tests performed on unidirectional boron composites with nickel-plated ends. If a composite is bidirectionally reinforced, the current will flow predominately in those layers perpendicular to the edge until their degradation causes tne current to be driven across the resistive resin films and into the nearest fibers that are parallel to the panel edge.

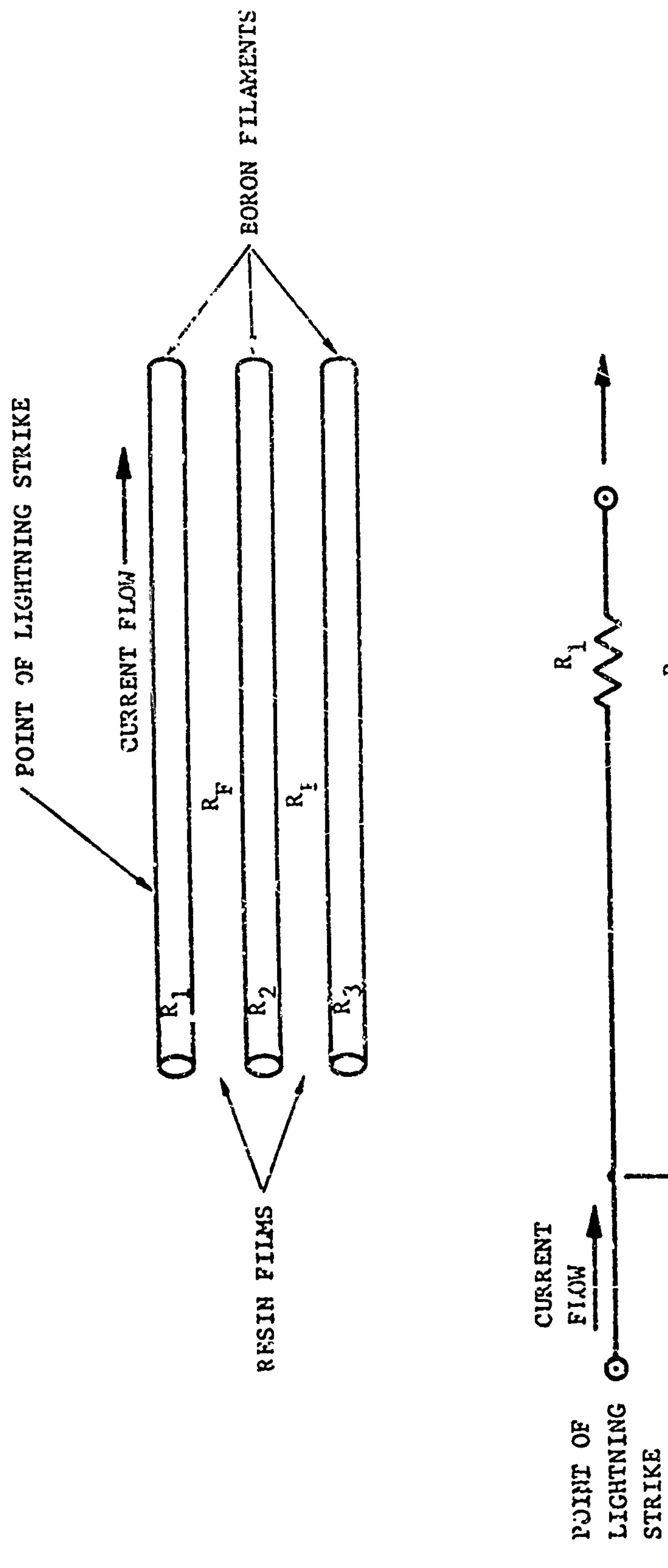



Figure 16. Lightning Strike on Composite Surface - Current Flow Model

As more information on the resistance of the virgin filament and resin films and changes in resistance of filaments as a function of degradation is obtained, the models discussed can be compared with test results and their validity determined. Such models are amenable to available composites program solution.

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R. L. Miller - Materials and specimen preparation and analysis

P. R. Marshall - Fiber failure mathematical model.

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13. ABSTRACT

This report presents the text of the unclassified papers presented at the Lightning and Static Electricity Conference, 3-5 December 1968. These papers discuss problems of lightning and static electricity as they pertain to aerospace vehicles. Research to solve these problems conducted by numerous US and foreign agencies, both governmental and industrial, is discussed. The sessions covered include fluids and fuels, grounding and bonding techniques, survivability of nonconductive materials in a lightning environment, and control of static electricity effects on nonconductive materials.

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