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F-14A Fire Protection Test Program

by
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and
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FEBRUARY 1977

Naval Weapons Center
CHINA LAKE, CALIFORNIA 93555

Naval Weapons Center

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R. G. Freeman, III, RAdm., USN .... Commander
G. L. Hollingsworth .... Technical Director

FOREWORD

This report presents results of tests conducted to evaluate the technology proposed for an F-14A Aircraft Fire Protection Modification Testing Program in accordance with AIRTASK A510-5102/008-2/6241-000-436. Testing and evaluation of the proposed fire protection modifications is presented. Conclusions and recommendations relating to the proposed technology are presented.

The work reported was conducted between January and August 1976 by the Systems Survivability Branch, Naval Weapons Center, China Lake, Calif. Additional technical support for materials selection and technical analysis was obtained from NASA-Ames Research Center, Grumman Aerospace Corporation, and AVCO Systems Division, Inc.

This report has been reviewed for technical accuracy by H. W. Drake.

Released by
M. M. ROGERS, *Head,*
Systems Development Department
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(U) *F-14A Fire Protection Test Program*, by George E. Monesko and Herman J. Hoffman. China Lake, Calif., Naval Weapons Center. February 1977. 114 pp. (NWC TP 5942, publication UNCLASSIFIED.)

(U) A test program conducted to evaluate the technology proposed for an F-14A aircraft fire protection modification is reported. The modification included fire containment barriers and a fire extinguisher system employing Halon 1301 fire extinguishing agent. Various formulations of rigid polyurethane foam, flexible silicone foam, ceramic felt, and silicone ablative materials were tested to determine the thermal qualities of candidate fire barrier materials. The use of intumescent paints as a fire barrier gap filler and the effect of barrier penetrations in a fire environment were also investigated.

(U) The proposed fire protection modifications were installed in an F-14A aircraft which was surveyed for use as a test article. Fire extinguishing agent concentration tests were performed to optimize discharge nozzle location in the test article. Full-scale fire tests were conducted on the test article under a simulated 250-knot flight condition with rupture of an engine case by a fan blade being simulated by appropriate mechanical openings and introduction of air simulating in-flight conditions. The damage caused by a thrown fan blade was simulated by puncture of the main engine fuel line, fuel ignition by metal sparks, and puncture of the nacelle top deck. The nacelle fire was extinguished by the on-board fire extinguisher system. Nacelle fuel vapors re-ignited from fire on the test pad and the test article was subjected to a 20-minute endurance burn. Post-test examination of test data and films showed that both the fire containment and fire extinguishing system were effective. It was concluded that the fire relight was an abnormal condition since an aircraft in flight would not be exposed to fires from fuel which had spilled overboard.

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ACKNOWLEDGMENT

The authors are indebted to the NASA-Ames Research Center personnel for their cooperation and assistance in the testing of candidate fire barrier foam-type materials.

SUMMARY

A test program was conducted to evaluate the technology proposed for a fire protection modification to the F-14A aircraft. The proposed modification encompassed both a fire extinguishing system and fire containment barriers to prevent rapid migration of fire resulting from engine failure. The proposed modifications were made to the right-hand side of a previously damaged F-14A aircraft which was stricken for use as the test article for full-scale tests conducted under the subject program.

A group of organic and inorganic materials were evaluated as candidate lightweight fire barrier materials for aircraft. A barrier was required to prevent fire migration into the aircraft upper central trough where the flight control rods are located. Test criteria called for a barrier material which would resist burnthrough and prevent a heat buildup in excess of 205°C (400°F) measured 6 inches (15.2 cm) from the barrier backface when exposed to a fire for 15 minutes duration. Testing of several candidate materials disclosed that a 2-inch (5.4-cm)-thick modified polyurethane foam with thin metal facings on both sides provided the desired thermal qualities. Installation and mechanical problems prevented this system from being utilized. Further testing with silicone ablative material over a stainless steel sheet provided an acceptable system from an installation point of view with somewhat reduced thermal resistance. This system proved adequate in the full-scale testing.

Extinguisher agent dispersion and concentration tests were performed on the test article under simulated 250-knot flight conditions. Extinguishing agent flow visualization tests demonstrated that placement of extinguishant discharge nozzles provided satisfactory dispersion of the test medium throughout the engine nacelle and overwing fairing volumes. A concentration of the Halon 1301 extinguishant in excess of military specification requirements was achieved in the engine nacelle and overwing fairing fire-critical volumes under a variety of simulated flight failure mode conditions.

A full-scale fire test was conducted on the test article to determine effectiveness of the fire containment system and fire extinguishing system. This test was conducted at a simulated 250-knot flight condition which included simulated engine failure with resulting ignition of fuel spilled from a severed fuel line. Test instrumentation and motion picture films taken during the test showed that the extinguisher system did extinguish the fire. However, due to a fuel shutoff valve malfunction with resulting fire on the test pad, fuel vapors inside the engine nacelle were reignited by the external fire. An extended burn time of approximately 20 minutes ensued, subjecting the aircraft and the fire barrier modification to a worst-case condition. Although unprotected areas of the nacelle structure sustained severe damage, areas protected by the fire barrier installation showed that the barrier was effective

in preventing rapid migration of fire damage. The fire relight resulting in the 20-minute endurance burn was not typical of an inflight condition as external fuel vapors would normally be blown off the aircraft by the airstream.

Conditions during the full-scale fire test did not tax the protective materials of the center trough fire barrier sufficiently to permit complete evaluation of their effectiveness. Subsequently, additional fire tests were performed to evaluate this fire barrier and the effect of fire on control rod brackets in the aft nacelle areas. It was found that the center trough barrier was capable of defeating fire propagation into the trough. Ablative-covered control rod brackets were seen to offer superior resistance to fire damage as opposed to nonprotected members.

INTRODUCTION

Results of a test program conducted to evaluate the technology proposed for F-14A aircraft fire protection modification are presented. The program was implemented under cognizance of the Systems Development Department and conducted by the Systems Survivability Branch of the Naval Weapons Center (NWC). The F-14A aircraft is an advanced two-engine fighter (Figure 1) which utilizes aft-mounted turbofan engines in side-by-side configuration.

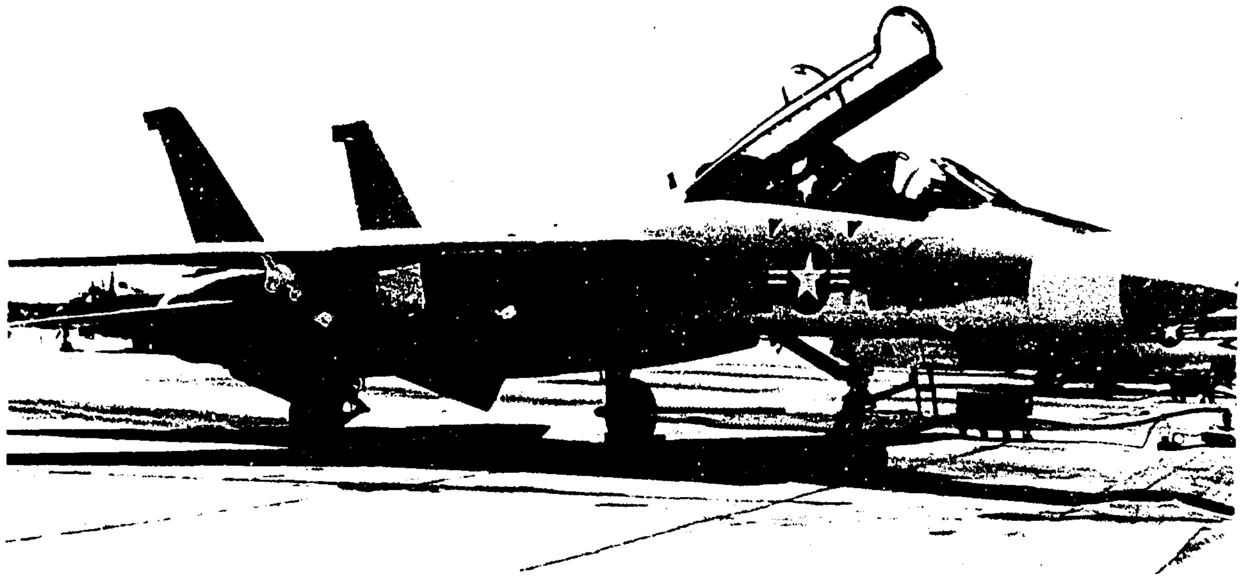

FIGURE 1. Navy F-14A Advanced Fighter Aircraft.

PROGRAM OBJECTIVE

The test program objective was to evaluate the feasibility of the technology proposed by Grumman Aerospace Corporation (GAC) for fire protection of F-14A aircraft. To achieve this objective, testing and evaluation of the proposed fire protection systems in a simulated flight environment was required.

The proposed modifications encompassed both fire containment and fire extinguishing systems. Whereas the effectiveness of both systems was postulated, there was no substantiating data relative to performance

in an actual inflight fire environment. The use of fire extinguishing systems in aircraft is commonplace, however, each type aircraft requires an individual design approach to achieve effective distribution of the extinguishing agent. Fire containment is dependent upon the effectiveness of fire barrier materials. Integrating a system into an existing complex design which required data on performance of fire barrier materials and extinguishing systems precluded making an objective analysis of the proposed technology based on available information.

PROGRAM SCOPE

The test program encompassed two separate but related fire protection systems, namely (1) fire containment, and (2) fire extinguishing. The complexity of factors affecting each system necessitated an individualized testing approach for each system. Therefore the test program was structured to provide two distinct phases of operation identified as (1) subscale testing and (2) full-scale testing.

The subscale testing phase of the program was directed toward evaluating candidate fire barrier systems. Both the barrier material and its structural configuration affect fire barrier system performance. Materials thermal data and barrier performance data were needed to ascertain overall performance visibility. To achieve this end the subscale test program was designed to obtain thermal data for a variety of candidate materials and subsequently test candidate barrier systems in a simulated aircraft fire environment.

The full-scale test program was directed toward ascertaining the effectiveness of the proposed fire barrier and fire extinguishing systems. Accurate evaluation of system performance required testing under either actual or simulated flight conditions. As actual flight testing of a system of this type is impractical from the standpoint of risk, cost and data acquisition, testing in a simulated flight environment was planned. Testing in this manner minimized both risk and cost factors while maximizing the opportunity to acquire meaningful performance data.

F-14A aircraft No. 2, which had been previously damaged, and from which all usable items not required for this test had been removed, was used as the test bed for full-scale testing. The full scale testing effort was directed toward acquisition of airflow and extinguishing agent data in and about the engine nacelle, and subsequent testing under simulated engine-caused fire conditions. All tests were directed toward data acquisition in a simulated flight environment at a minimum risk to the aircraft structure.

Test plans included provision for making the test site modifications necessary for testing on the scale planned. Site modifications included making provision for mapping airflow in and about the engine

nacelle and providing aircraft restraints capable of holding the F-14A aircraft with one engine operating. Routine preparations included design and installation of instrumentation, remote controls and related support systems unique to the F-14A aircraft.

BACKGROUND

Several F-14A aircraft have sustained fire or potential fire damage due to problems occurring in the engine nacelles. Engine failures involving loss of fan blades have been the major source of these fires. Combat damage can also inflict engine or fuel system failure, resulting in fire. Due to the absence of nacelle fire control systems these fires have migrated throughout the aircraft causing loss of some aircraft.

Although the engine fire sources are being reduced, it is not feasible to totally eliminate all potential fire sources. The aircraft contractor evaluated methods of improving aircraft survivability under a fire environment. As a result of these studies the contractor submitted (to NAVAIR) Engineering Change Proposals (ECPs) covering an F-14A fire protection program. The ECP covering installation of a fire barrier system was deemed the most important since it provided methods for fire control and containment. However, the proposed technology was subject to close scrutiny since supporting data were based on small-scale tests and limited system integration validation.

In view of the financial investment required for implementing the ECPs it was recommended that additional testing, preferably on a full-scale basis, be performed to verify the proposed technology. As a result, NWC/China Lake was assigned the task of preparing, implementing and conducting a test program to evaluate the proposed aircraft fire protection technology.

The NWC Aircraft Survivability Test Facility is one of the few capable of conducting an extensive aircraft fire survivability type test. In addition, the high velocity air source required for a full-scale flight simulation test program was available only at this test facility. The facility includes aircraft handling, test pad, instrumentation and control areas required to support a full-scale aircraft testing program.

PROGRAM TASKS

The test program was based upon use of the NWC Aircraft Survivability Test Facility for test and evaluation of the proposed F-14A fire protection modifications. Aircraft handling facilities, instrumentation and control areas, and a dynamic air source (DASH) were available at

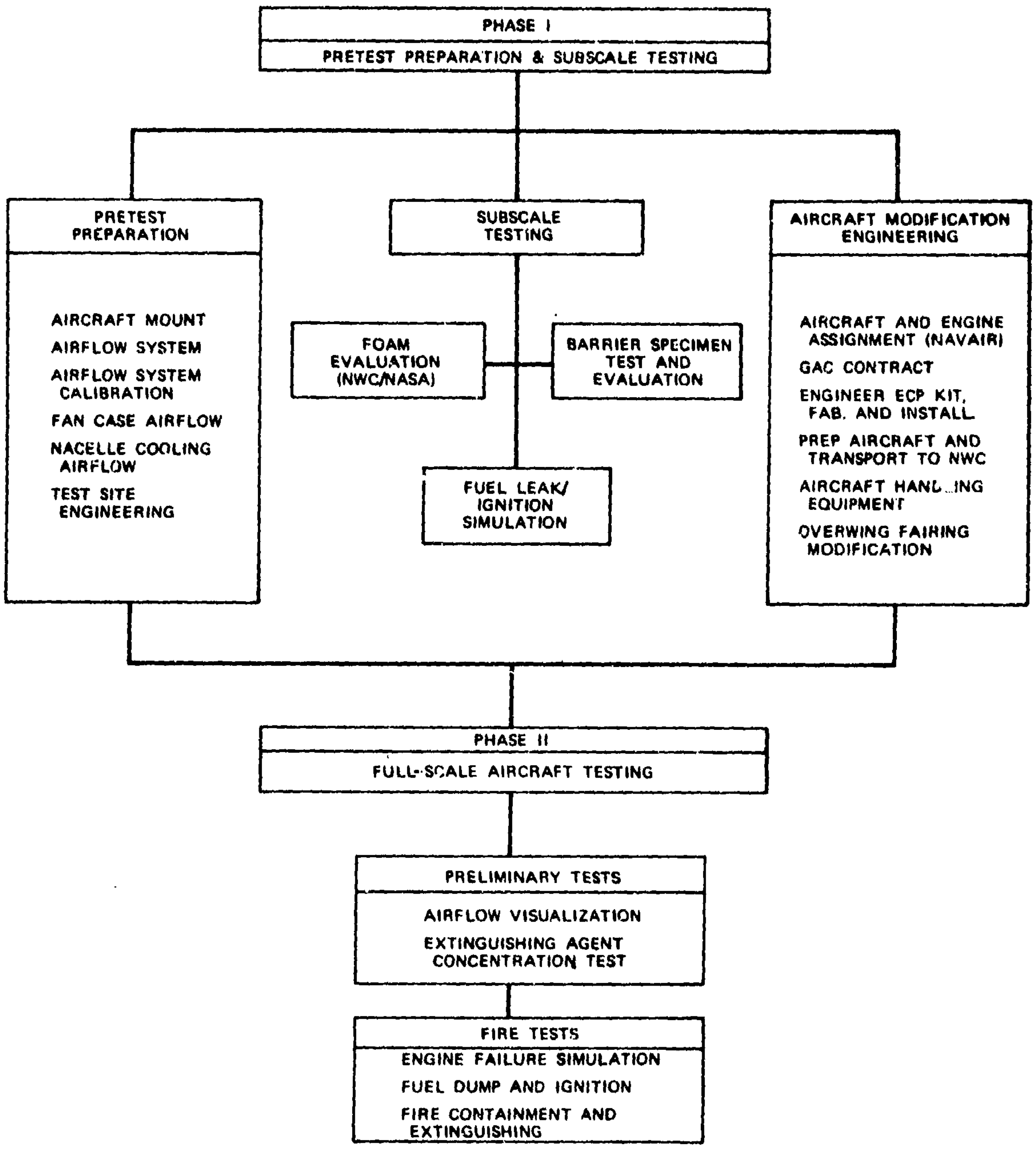

FIGURE 2. Program Tasks, F-14A Fire Protection Test Program.

the facility. The pretest program task effort encompassed test site modifications and preliminary calibration tests necessary to support the full-scale test effort.

During the initial program planning phase it became apparent that the program effort would encompass four major task areas, namely (1) aircraft modification engineering, (2) test facility preparation, (3) subscale testing, and (4) full-scale testing. Each of these task areas was seen to embrace the subtasks delineated in Figure 2.

AIRCRAFT MODIFICATION ENGINEERING

Previously damaged F-14A aircraft No. 2 was stricken and assigned for use as a test bed for the full scale aircraft tests. Although the left engine nacelle area had suffered extensive fire damage, the right side and upper trough were usable for evaluation of the proposed fire protection kit. Both engines and all of the intrumentation and equipment not required for this program had been removed from the aircraft.

Modification of F-14A aircraft No. 2 was performed by Grumman Aerospace Corporation. The work covered engineering and installation of the proposed fire protection modifications, subsequent preparation of the aircraft for shipment, and personnel support during conduct of the full-scale test. Upon completion of the modification/preparation task the aircraft was transported to NWC via the Aero Spacelines Super Guppy aircraft.

To sustain the program schedule, an operable F-14A aircraft for measurement and fit-checking of test items was provided for program use. F-14A No. 24 was assigned to NWC for this purpose. Measurements from this aircraft were used for preparation of the test facility equipment and for verifying the fit of test equipment to be installed in the F-14A test article.

Fire Protection Modifications

The fire protection modifications installed in the F-14A test article are detailed in Engineering Change Proposals 835R1[1] and ECP 853.[2]

[1] Naval Air Systems Command. *Model F-14A Aircraft, Engineering Change Proposal (ECP) GR-F-14A835R1, Fire Protection Program, Code "0"*, by Grumman Aerospace Corporation, Bethpage, N.Y. Washington, D.C., NASC, 13 July 1976. 72 pp. (Publication UNCLASSIFIED.)

[2] Naval Air Systems Command. *Model F-14A (Tomcat) Aircraft, Engineering Change Proposal (ECP) No. 853 Code "0", Flight Control Fire Protection*, by Grumman Aerospace Corporation, Bethpage, N.Y. Washington, D.C., NASC, 17 June 1976. 9 pp. (Publication UNCLASSIFIED.)

ECP 835 covers the design and development of a fire protection program for the following areas of F14A aircraft: (1) Nacelle and sponson area, (2) centerline trough and sloping deck, and (3) accessory area. It also includes installation of a fire extinguishing system. The object of the proposed modifications is to provide fire protection and extinguishment in critical areas of the aircraft.

Modification of the centerline trough and sloping deck area involved sealing of penetrations in the top deck and trough volumes (Figure 3). The use of fire-retardant seals was seen as a means of preventing propagation of combustible fumes (and potential fire) to fire-critical areas. As proposed, modification of the nacelle and sponson area involves the addition of 0.020-inch (0.5-mm)-thick titanium skins to critical surface areas shown in the Figure 4 simplified diagram.

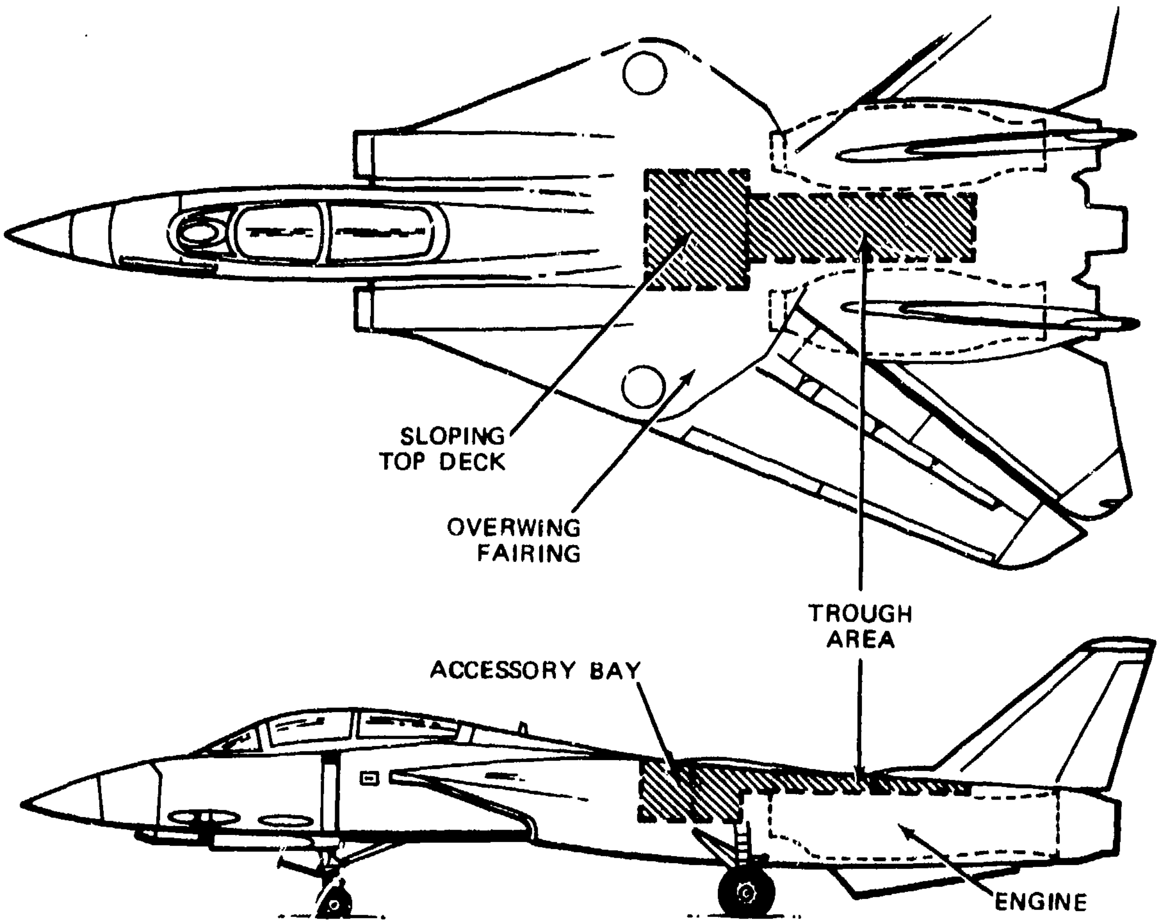

FIGURE 3. Trough and Deck Area Modified for Fire Protection, Simplified Diagram.

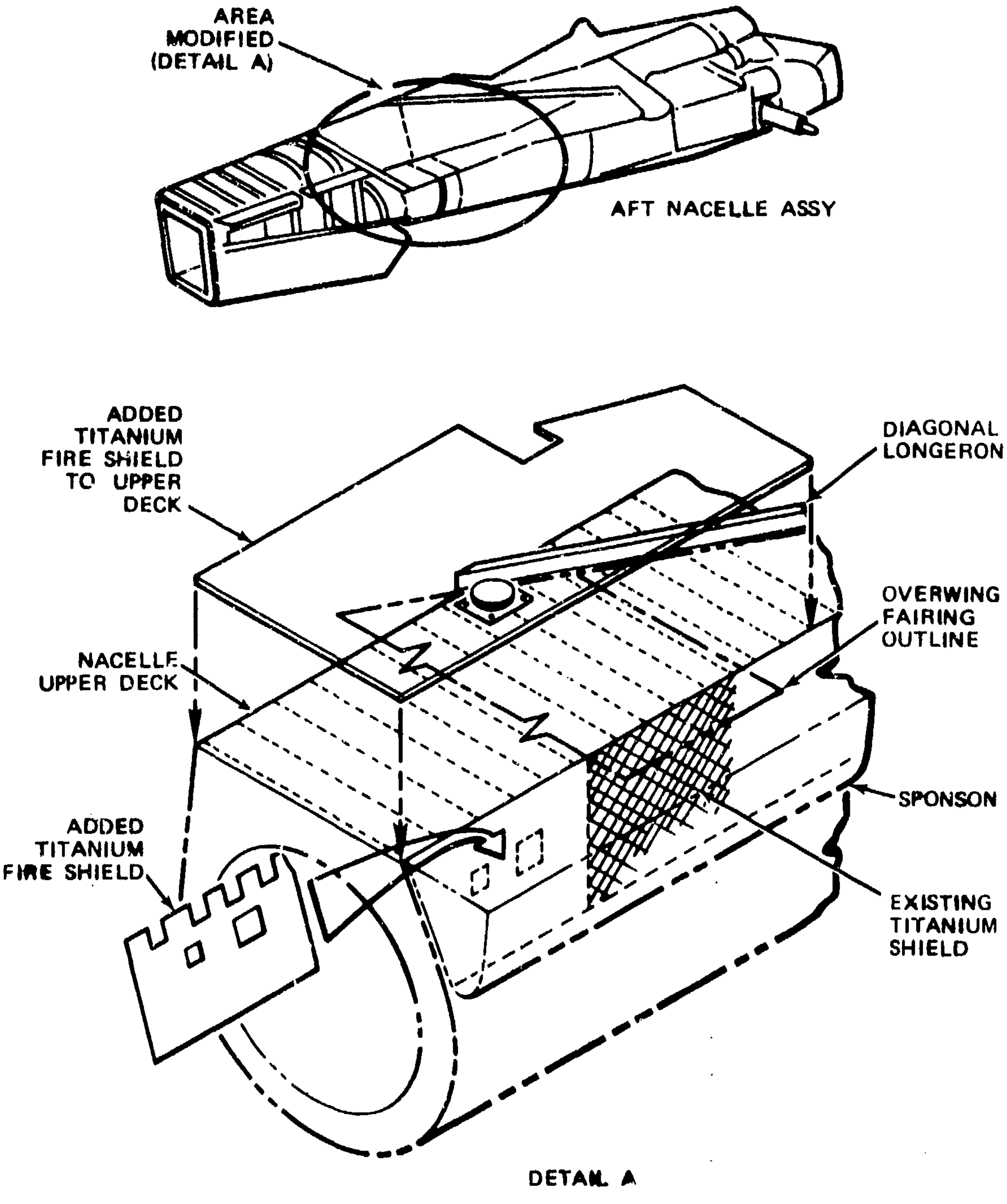

FIGURE 4. Nacelle Area Fire Protection Modification, Simplified Diagram.

The proposed ECP 835R1 (this ECP corresponds to Airframe Change (AFC) 529) entails the addition of a fire barrier to prevent flame penetration to the flight control system. The change in the accessory bay area includes providing a 0.020-inch (0.5-mm) stainless steel fire barrier coated with General Electric (GE) silicone TBS-758 ablator material to prevent flame penetration to the flight control system located in the accessory area. In addition, the change also includes replacing the aluminum lines in the swing actuator bay with stainless steel lines and the application of silicone TBS-758 ablator material to the sloping deck assembly surface of the aft mid module. The cavity separating the wing box beam and fuselage station 534 bulkhead in the accessory area will be sealed by the addition of GM 4107 sealant. All gaps between shielding tubes installed over the wing sweep drive/feedback shafts and the fire barrier will also be sealed to prevent fire migration through barrier gaps.

ECP 835R1 also includes installation of a fire extinguisher system employing Halon 1301 extinguishant. The fire extinguisher system will be capable of discharging into either the right or left engine nacelle and accessory area. The system entails installation of two extinguisher bottles as shown in Figure 5. The 378-cubic-inch (6195-cm^3) bottle is

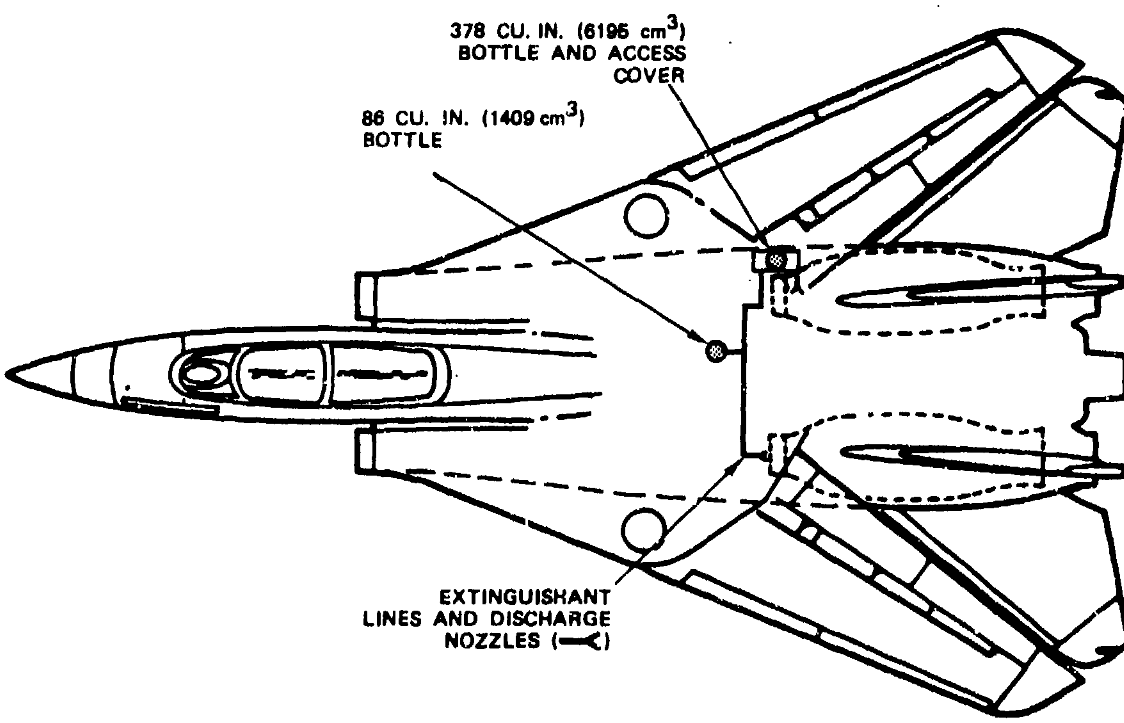

FIGURE 5. Proposed Fire Extinguisher System Installation, Simplified Diagram.

installed in the forward right-hand sponson area with the 86-cubic-inch (1409-cm^3) bottle installed on the right side of the accessory area. The installation includes related cockpit discharge switch controls and status indicators.

The 378-cubic-inch bottle will discharge extinguishant into either the right or left engine nacelle as selected by the cockpit-mounted discharge switch. Simultaneously, the 86-cubic-inch bottle will discharge into the associated accessory bay volume. The agent discharge switches will be installed behind the "L" and "R" fuel shutoff controls. Control installation includes the addition of two advisory lights to the Caution/Advisory Indicator in the cockpit, to monitor status of the extinguishant bottles. The lights will illuminate when the container pressures are below a predetermined level. A fire extinguisher test unit, to facilitate preflight checkout, will be installed to form a part of the aircraft Master Test System.

Additional fire protection measures are covered by ECP 853 (see footnote 2; this ECP corresponds to AFC 530). This modification is to provide fire protection for the directional/longitudinal control system located in the right and left engine nacelles. The purpose of this ECP is to prevent the potential loss of these aircraft controls due to failure of the supporting structure in event of fire. The modification entails the application of GE silicone ablative coating TBS-757/58 to the right and left nacelle frames at fuselage sta ions (FS) 710.65, 726 and 733. These are structural members which support both longitudinal and directional controls. The application of an ablative coating will reduce their susceptibility to failure in the event of flame propagation to these areas.

A third modification, ECP 854,[3] covering fan blade containment was also proposed. This modification encompassed changes in the engine compartments to accommodate installation of the modified TF-30-P412 engine with fan blade containment. Whereas this change does not impact the F-14A fire protection measures, it does increase the potential aircraft survivability. As this modification was not relevant to the subject fire protection test program, the proposed modifications were not included in the test article. (This ECP corresponds to AFC 531.)

The foregoing proposed modifications were incorporated in the F-14A test article by Grumman Aerospace Corporation prior to delivery of the test article to NWC. Modifications were confined to the right-hand side of the test article. The aircraft symmetry permitted adequate testing with the modifications installed in only the right-hand portion of the test article.

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[3] Naval Air Systems Command. *Model F-14A (Tomcat) Aircraft, Engineering Change Proposal (ECP) 854, Code "0", Engine Fan Blade Containment*, by Grumman Aerospace Corporation, Bethpage, N.Y. Washington, D.C., NASC, 12 July 1976. 29 pp. (Publication UNCLASSIFIED.)

Aircraft Engine Assignment

Use of an operational engine to provide actual operating conditions during full-scale testing was deemed important to establish realistic test conditions. The engine would serve as a heat source to vaporize spilled fuel, act as a fire relight source and as a nacelle volume filler to create the airflow conditions required to permit evaluation of the fire extinguisher system.

The possibility of using an older TF-30-P1 engine (in place of the standard TF-30-P412) for the test article had been sought. To determine feasibility of this approach the Naval Air Propulsion Test Center, Trenton, N.J., proposed and accomplished a program to fit a TF-30-P1 engine with a P412 afterburner into a production F-14A airframe. The necessary engine parts were fabricated and the fit between the engine and aircraft was quickly accomplished; no problems existed in fitting, mounting nor interconnecting the aircraft and engine. It was determined that interconnect adaptors for fuel and electrical interfaces would have to be fabricated but space and routing for these were available.

Two TF-30-P1 engines were subsequently assigned for use. Buildup and test of the engines was performed at the Naval Air Propulsion Test Center. Two engines, one afterburner and one diffuser adapter were supplied for the test program.

Overwing Fairing/Nacelle Modifications

To achieve test objectives, it was necessary to accurately map airflow patterns over and within the overwing fairing accessory bay and nacelle areas. As initial placement of the extinguishant discharge nozzles was to be dependent upon the airflow pattern within these areas, internal visibility was desired to assist in determining the flow pattern. In this manner the visual data could be correlated with data obtained from instrument sensors to map airflow patterns.

To provide internal visibility of the area beneath the overwing fairing, the fairing was modified to incorporate Plexiglas windows. Portions of the overwing fairing skin were removed between structural members. These portions were subsequently replaced with Plexiglas of a thickness equal to that of the original skin, to prevent any abnormal disruption of airflow over and within the area. Yellow tufts were subsequently installed on the underside of the modified overwing fairing so the airflow pattern within the volume could be observed.

Visibility of airflow within the engine nacelle area was also desired for optimum airflow patterns determination. Therefore the skin was removed from both the fore and aft portion of the nacelle, and was

replaced with Plexiglas as was done for the overwing fairing. Tufting was also installed in the interior portion of these areas. Installation of the tufting was such that it could be viewed by monitoring cameras during test operations.

The Plexiglas-covered areas are shown in Figure 6. (Tufting used in the interior areas does not show clearly in the black-and-white photos, but was visible during airflow tests.) No significant changes were made to original skin contours in accomplishing these modifications.

Nacelle Airflow Simulation

Full-scale aircraft tests necessitated the need to simulate normal nacelle cooling and fancase rupture airflows based upon a 250-knot airspeed flight condition. Nacelle airflow was to be representative of flight conditions after a fancase engine failure and was to be controlled as to quantity and point of entry into the nacelle. The normal flight condition to be simulated called for an airflow through the forward fixed cowl of the nacelle for engine bay cooling. When an engine malfunction occurs the nacelle airflow is increased, the amount depending upon failure airflow as a function of aircraft velocity. To assist in establishing airflow parameters for test purposes, GAC provided airflow quantity analysis.

Requirements established for nacelle airflow simulation were as follows:

1. Nacelle cooling airflow provided at 0.6 lb/sec (0.29 kg/sec).

2. Engine fancase rupture simulation airflow delivered at 1 to 2 lb/sec (0.45 to 0.90 kg/sec) to the approximate area of fancase rupture.

3. Airflow temperature of ambient ± 20°C.

4. Airflow required for a period of approximately 10 to 15 minutes.

Several possible airflow sources were investigated. These included the possibility of bleeding air from the DASH system or using auxiliary sources such as engine start or airconditioning units. Most of these sources were rejected because of problems in implementation or the inability to meet operational requirements.

Of the potential air sources available, the use of NR-2B and NR-10 airconditioning units offered the most practical solution to the problem. The NR-2B unit was chosen to provide air for nacelle cooling as it is capable of providing a 0.64 lb/sec (0.29 kg/sec) airflow in either a vent or dehumidify mode, and also as an airconditioner. Airflow tests

(a)

(b)

FIGURE 6. Plexiglas Skin Installation, (a) Forward Nacelle Area, and (b) Overwing Fairing.

run on the NR-2B unit verified its suitability for the intended use. It was operated in the airconditioner mode providing airflow of 0.5 lb/sec (0.23 kg/sec) at 50°F (10°C).

The NR-10 airconditioning unit was chosen to provide airflow for fancase rupture simulation. Airflow tests run on the NR-10 unit indicated that the airflow temperature was 60°F (15°C) at a quantity of 0.64 lb/sec (0.290 kg/sec) per unit. It was found that two units provided the required volume within temperature range, therefore two units were used for the tests.

Plans for airflow ducting and installation were subsequently drawn, based on use of the three airconditioning units. The resulting test pad layout is shown in Figure 7. Positioning of the air source equipment was dictated by the requirement to keep the pad area clear and other instrumentation unobstructed. Black iron pipe 3 and 4 inches (76 to 102 mm) in diameter was used for ducting due to its workability and availability. Bolted flanges and flexible coupling were used in assembly of the ducting to provide a degree of flexibility to compensate for thermal expansion and vibration of the aircraft under test conditions.

Engine failure simulation called for conditions simulating a thrown fan blade. Based on failure histories, it was postulated that a thrown fan blade could penetrate the top fairing skin. Since this opening would affect the cooling airflow, placement of the cooling airflow ducting was predicated on a worst-case condition. The largest fancase rupture measured was approximately 12 square inches (7742 mm^2). This was used to predict a 4.5 by 3 inch (114 by 76 mm) oval hole with a total area of approximately 11.25 square inches (7258 mm^2) in the nacelle top deck.

The postulated trajectory of a thrown fan blade and resultant penetration, and related airflow ducting into the engine nacelle are shown in Figure 8. The cooling air outlet was attached to the forward fixed cowl in the approximate area of the aircraft hydraulic oil cooler exit as in the actual installation. Fancase rupture simulation airflow was brought through aircraft fuel cell No. 6, into the engine nacelle as shown in Figure 8.

Fuel Failure Simulation

In the F-14A accident reports it had been reported that in some of the incidents the main fuel line in the nacelle area had sustained penetration. This penetration could be either complete severance by an engine blade or penetration resulting in a gash in the fuel line. The difference in fire consequences for these two types of fuel leaks is the rate at which fuel-air mixing occurs. A gash in the fuel line would have a tendency to saturate the air in the nacelle whereas a severed fuel line would have a greater tendency to puddle the fuel which

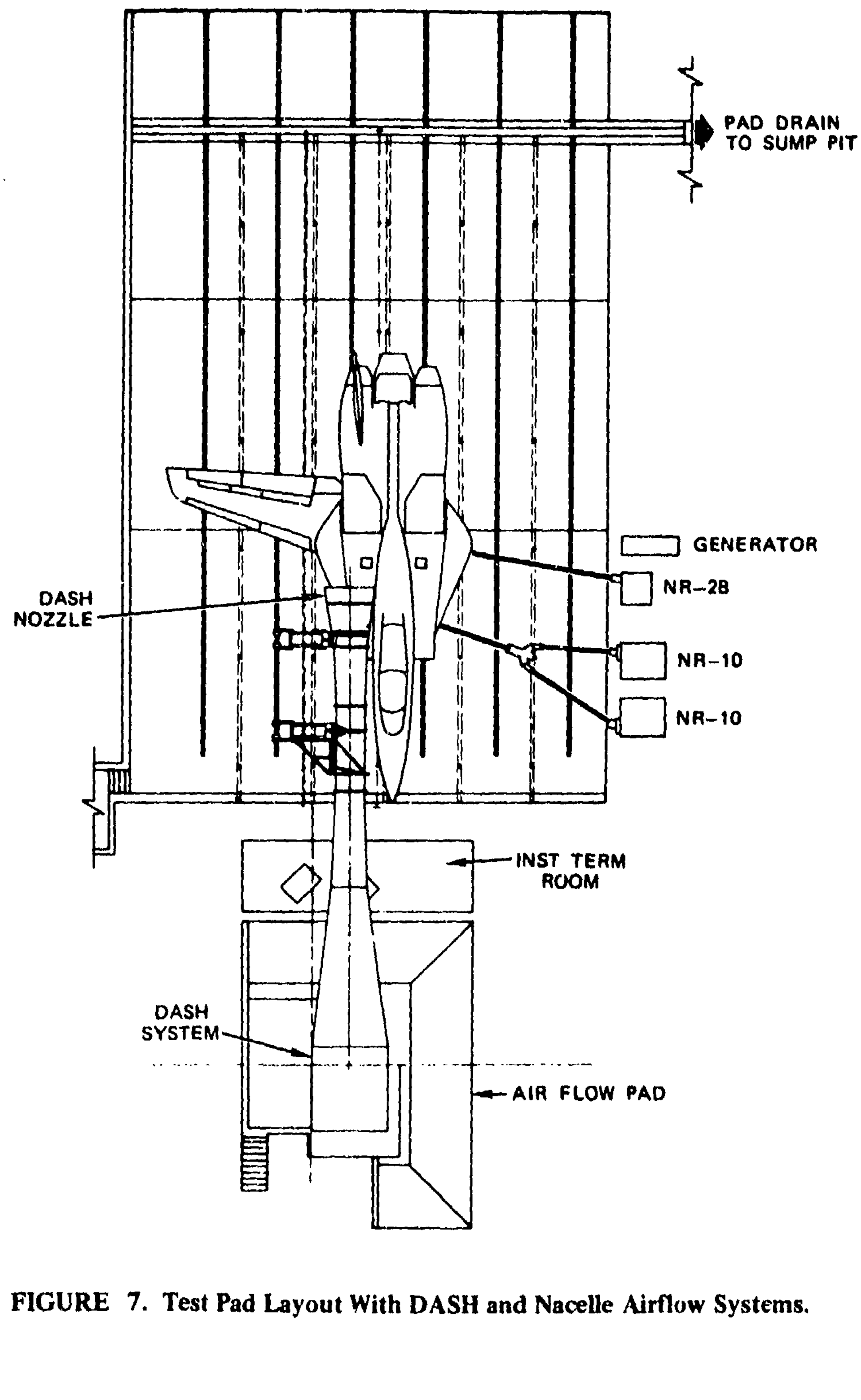

FIGURE 7. Test Pad Layout With DASH and Nacelle Airflow Systems.

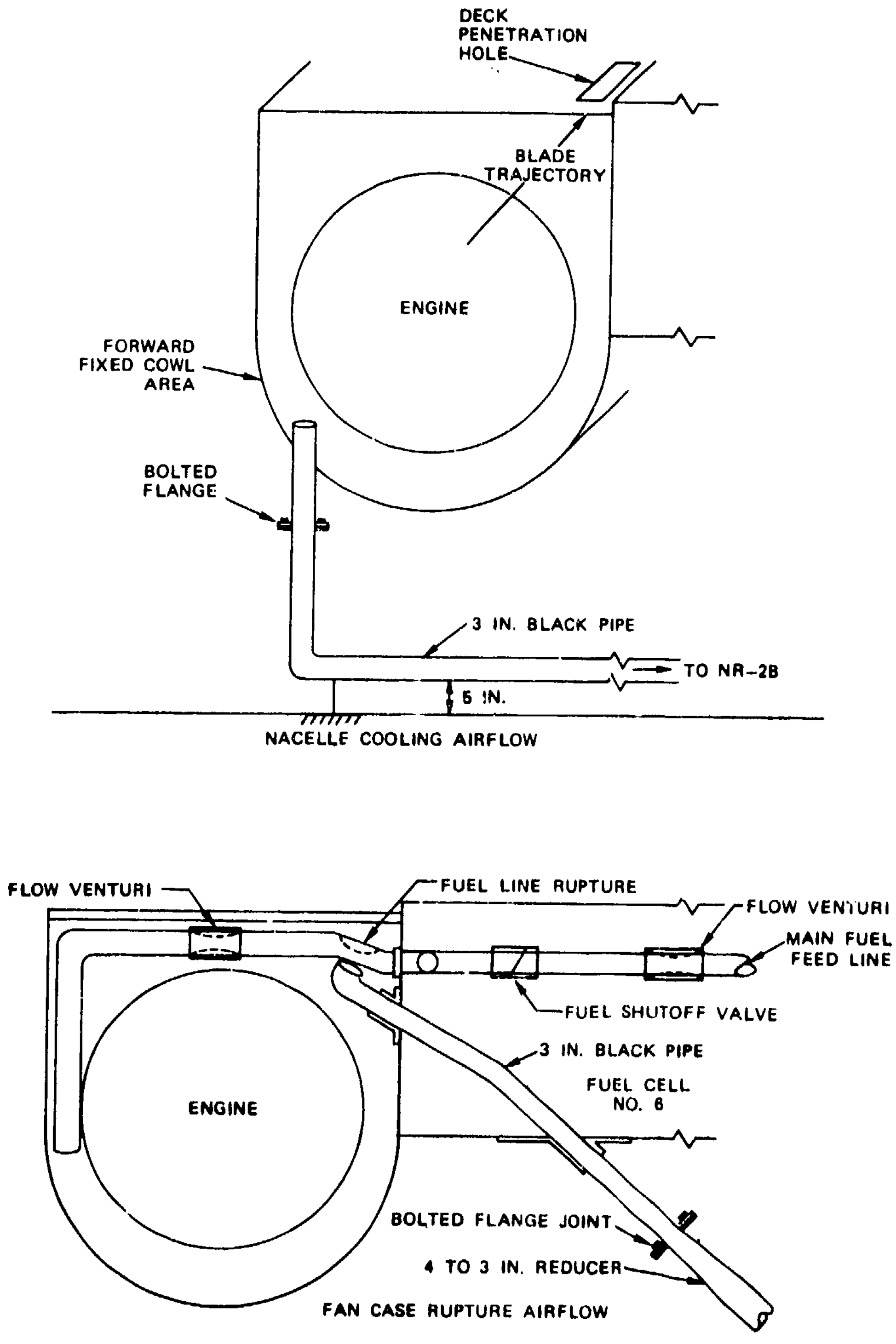

FIGURE 8. Nacelle Airflow Simulation Lines and Rupture Simulation Opening. (View looking aft in right-hand nacelle.)

would then burn at the air entrance interface. It was calculated that at the moment of engine failure the fuel line would be providing fuel to the engine at a pressure of 32 psi (2.25 kg/cm^2) and a flow rate of approximately 28,000 lb/hr (12,700 kg/hr). Failure simulation was based on the assumption that a single fan blade punctured the fancase, the engine fuel feed line and the top of the engine nacelle. Based on actual experience from inflight failures, it was assumed that heat or sparks generated by the ruptured engine ignited the fuel spilled from the fuel line penetration. Therefore the task of failure simulation entailed providing a controlled source of fuel simulating the flow from a punctured fuel line and a controlled method of fuel ignition.

The proposed method of modeling fuel line rupture involved using a linear shaped charge to open a hole in the fuel line (see Appendix A). Use of a metal-based flare was planned to provide fuel ignition, simulating the engine heat/sparks ignition mode. Test instrumentation would be placed so as to determine where hot spots occur and to permit evaluation of the burning experienced. Acquisition of this data was deemed vital for determining effectiveness of the fire barriers and fire extinguisher system.

Because of the variables involved in simulation and measuring fuel flow and ignition time, a test simulator was built to prove out the planned simulation mode. The simulator and fuel failure simulation tests are described in the Subscale Tests section included herein.

Instrumentation Sensors

Aircraft No. 24 was used to identify sensor location. Based upon failure data, 38 thermocouples were placed inside the engine nacelle, overwing fairing accessory bay and center trough to monitor fire progression in these protected areas. Sixteen thermocouples were installed in void spaces and empty fuel tanks adjacent to test areas to monitor fire propagation in these areas which are outside the fire protection area. Placement of the 16 thermocouples was based upon the premise that fire reaching the void areas could not be extinguished by the on-board extingusher, and test facility carbon dioxide (CO_2) would probably be required for fire control. Four thermocouples were mounted in critical areas such as inside the cell where a main instrumentation/control junction was located.

In addition to the thermocouples, 26 data channels were deemed necessary to provide data feedback for test control and monitoring. Other instrumentation sensors included airflow velocity indicators, pressure differential indicators for internal volumes of the aircraft, and engine monitoring equipment.

It was desired to measure the fuel flow rate and the nominal fuel quantity that dumped into the nacelle during the test. A venturi type flow measurement appeared to be the best candidate because of the extremes in flow rate to be measured. In normal midpower engine operation the maximum flow rate in the line to be failed is approximately 24,000 lb/hr (10,800 kg/hr). Pressure in the 3-inch (76-mm)-diameter line is on the order of 30 psi (2.11 kg/cm^2) under these same conditions, thus, at the instant of line rupture, very high flow rates are possible. The fuel pump is coupled to the engine shaft; as the failed engine slows, pressure at the line rupture hole also decreases rapidly.

To measure the fuel quantity dumped into the nacelle after fuel line rupture, two flow measurement venturi were installed in the fuel line. One venturi was installed upstream of the fuel shutoff valve inlet; the other venturi was installed downstream from the point selected for fuel line rupture (see Figure 8 for relative location). Pressure transducers were installed at each side of each venturi to monitor fuel line pressures during testing operations. Provision was made for remote readout of flow and pressure data.

By placing a venturi before and after the fuel line rupture hole a comparison of the fuel flow rate at each side of the line rupture point could be made. Using data recorded during the test operation, the difference in flow rates could be determined. Since the difference in flow rates would be dependent upon the amount of fuel spilled from the rupture hole, the volume of fuel spilled could be readily calculated.

SUBSCALE TESTS

The purpose of the subscale testing program was to establish a test data base on the fire barrier materials response and to verify testing techniques to be used in full-scale testing of the F-14A test article. The test effort in this phase of the program covered thermal optimization of candidate barrier foams, barrier systems screening tests and engine nacelle fuel leak simulation and ignition tests.

FIRE BARRIER MATERIALS/SYSTEM TESTING

A fire containment barrier utilizing polyurethane foam as a basic barrier material was originally proposed in ECP-835 (see footnote 1). Several candidate materials had been considered; however, pertinent thermal and physical data applicable to aircraft fire protection were not available. Therefore, in order to establish a data base for qualitative

evaluation of the proposed fire barrier, several candidate materials and system approaches were evaluated.[4]

The primary objective of the fire barrier evaluation program was to develop a prototype fire barrier system which would provide adequate thermal protection to the aircraft flight control rods in the F-14A upper trough in the event of an engine nacelle fire. Special attention was paid to the problem areas of sealing the gaps required for aircraft structure and equipment penetrations through the fire barriers and maintaining structural integrity of the barrier installation. A secondary objective was to obtain information for development of an aircraft fire barrier specification containing realistic and attainable acceptance criteria.

The test goal was to develop a barrier that can withstand an aircraft fuel fire for a period of up to 15 minutes without allowing temperatures to exceed 400°F (204°C) measured 6 inches (15 cm) from a backface in a closed volume behind the barrier.

To achieve the objective, a twofold approach was used in testing and evaluating candidate fire barrier materials and systems. The initial test phase was directed toward thermal optimization of candidate foam materials; the second test phase involved the testing of many candidate barrier systems in a simulated aircraft fire environment. As a result of tests conducted, it was determined that a foam fire barrier, by itself, would not meet the installation and flight load requirements, after exposure to fire, for an F-14A installation. Further work was performed to evaluate inorganic materials, alone and in conjunction with metal backing. A silicone ablative bonded to stainless steel sheet was selected for final installation in the F-14A test article.

Foam Candidate Materials Tests

The Chemical Research Projects office of the NASA Research Center had completed considerable background work on the development of polyurethane foams for aircraft ballistic protection. NASA continued their effort on investigation of foam materials as possible candidate materials for aircraft fire protection. As a result of this work, the NASA-Ames Research Center prepared a material, process and system specification for a polyurethane foam-based aircraft fire barrier.[5] The following paragraphs summarize the NASA-conducted tests.

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[4] Naval Weapons Center. *Aircraft Fire Simulation Testing of Candidate Fire Barrier Systems*, by Herman J. Hoffman and John S. Fontenot. China Lake, Calif., NWC, November 1976. 40 pp. (NWC TP 5915, publication UNCLASSIFIED.)

[5] National Aeronautics and Space Administration, Ames Research Center. *Preliminary Material, Process and System Specification for an Aircraft Fire Barrier.* Moffet Field, Calif., NASA/ARC, September 1976. (Unpublished preliminary draft, document UNCLASSIFIED.)

The foam materials tested in this phase of the program effort are listed in Table 1, with results of the thermal optimization tests summarized in Table 2. Both organic and inorganic materials were tested and evaluated. Organic materials investigated consisted of various formulations of polyurethane, closed cell, rigid foams. Inorganics tested were flexible silicone and alumina silica rigidized materials. Additionally, metallics combined with both organic and inorganic insulators were tested.

TABLE 1. Polyurethane Foam Fire Barrier Formulations Evaluated.

Formulation parts by dry weight.

Components	1	2	3	4	5	6	7	8	9	10
Mondur MR	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0
Saran 113	16.5	16.5	16.5	16.5	16.5	16.5	16.5	16.5	16.5	16.5
KBF4	...	...	...	16.5	16.5	16.5	16.5	16.5	...	...
Fyrol 2	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
MEG 440	65.0	65.0	65.0	65.0	65.0	65.0	65.0	65.0	65.0	65.0
Freon 11	35.0	30.0	40.0	80.0	80.0	80.0	100.0	75.0	75.0	75.0
"E" glass	25.0	25.0	25.0	25.0	...	...	...	...	...	...
Refrasil 1/8 inch	...	...	...	...	...	6.5	25.0	6.5	6.5	6.5
Refrasil 1/4 inch	...	...	...	...	25.0	18.5	...	18.5	18.5	18.5
DC 195	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0
33 LV	8.0	8.0	8.0	3.5	3.5	3.5	3.5	3.5	3.5	3.5

Ten formulations were investigated in thermally optimizing polyurethane foam. Three potentially significant variables were addressed within these formulations, namely (1) basic foam density, (2) alternate high temperature resistant reinforcing fibers, and (3) potassium fluorborate (KBF4).

Density variation of the basic polyurethane foam (5A43), flight qualified and used to reduce ballistic damage, was easily controlled with the amount of Freon blowing agent used. Silica fibers were chosen to compare with conventional glass fibers because the silica has the advantages of good strength and stability at high temperatures. Effects of adding KBF4 were investigated because it has previously been found to be endothermic at hydrocarbon fuel flame temperatures, and the boron reacted with the carbonaceous char to form a more thermally stable network.

TABLE 2. Summary of Thermal Tests of Polyurethane Foam Variations.

Test No.	Designation	Density		Description	Duration, min	Remarks
		kg/m^3	lb/ft^3			
1	5A43	53.34	3.33	Polyurethane foam, cast reinforced with 2.54 cm (1 inch) "E" glass. Lap joint across center.	4.75	Initial heavy smoke generation. Specimen warpage observed at 2.0 min. Gap warpage initiated at 2.5 min. Gap thermocouple exceeded 540°C at 3.1 min. Escaping flame around corner seal at 3.5 min. Most of specimen reduced to char-like substance.
2	5A43	64.72	4.04	Polyurethane foam, cast reinforced with 1.90 cm (0.75 inch) "E" glass.	6.50	Initial heavy smoke generation. Warpage not as pronounced as during test 1. Some escaping flame around corner seal at 5.0 min, followed by steady rise of backface temperature.
3	5A43	39.56	2.47	Same as test 2.	3.17	Initial heavy smoke generation. Severe warpage with evidence of corner "hot spot" at 1.5 min. Backface temperature rise initiated at 5.0 min.
4	5F14	65.20	4.07	Polyurethane foam, cast reinforced with 1.90 cm (0.75 inch) "E" glass, KBF4 added and catalyst reduced.	6.00	Initial heavy smoke generation. Backface charring observed at 4.0 min. Significant backface temperature rise initiated at 5.0 min.
5	5F14RS	69.20	4.32	Polyurethane foam, cast reinforced with 0.63 cm (0.25 inch) silica fibers, KBF4 added, reduced catalyst.	3.50	Excessive voids noted in specimen prior to test. Noticeable backface temperature rise observed at 1.0 min. Noted formation of white "blanket" of silica on exposed face. Gas flow in oven caused local failures of this "blanket."

TABLE 2. (Contd.)

Test No.	Designation	Density		Description	Duration, min	Remarks
		kg/m³	lb/ft³			
6	5F14RS	64.56	4.03	Polyurethane foam, cast reinforced with mixture of 0.63 and 0.32 cm (0.25 and 0.125 inch) silica fibers, KBF4 additive, reduced catalyst.	8.67	No backface temperature rise until 3.0 min. Minimum warpage. Excellent fire exposed face "blanket," resistant to oven wind forces. Slower temperature rise than previous specimens.
7	5F14RS	45.65	2.85	Polyurethane foam, cast reinforced with 0.32 cm (0.125 inch) silica fibers, KBF4 additive, reduced catalyst. Backface enclosed in box.	5.33	Initiation of backface temperature rise at 2.0 min. Good fire exposed face "blanket," but extremely weak with rapid erosion. At conclusion of test only a thin backface shell remained. Still-air temperature at 5.33 min was 56°C.
8	5F14RS	59.43	3.71	Same as test 6. Backface enclosed in box.	7.00	Initiation of backface temperature rise at 3.0 min. Strong, erosion-resistant "blanket" formation. Still-air temperature at 7.00 min was 52°C.
9	5F14RS	56.55	3.53	Polyurethane foam, cast reinforced with mixture of 0.63 and 0.32 cm (0.25 and 0.125 inch) silica fibers, no KBF4 additive, reduced catalyst. Backface enclosed in box.	8.75	Initiation of backface temperature rise at 4.0 min. Strong, erosion-resistant "blanket" formed. Still-air temperature at 8.75 min was 66°C. Post-test foam damage unrelated to test.
10	5F14RS	64.72	4.04	Same as test 9.	9.33	Initiation of backface temperature rise at 4.0 min. Strong, erosion-resistant "blanket" formed. Still-air temperature at 9.33 min was 85°C.

The exact foam formulations used are contained in Table 1. The ingredients were mixed one at a time in the order given except that the MEG 440 polyol and the Freon liquid were thoroughly mixed together prior to addition to the ingredients preceding them. The final chemical, 33-LV, the catalyst agent, was mixed diluted by an equal amount, by volume, of Freon. Immediately after addition of the catalyst and rapid mixing, the final product was poured into the bottom of a waxed (releasing agent) mold and allowed to free-rise to the top of the mold. The foam was then allowed to cure for 24 hours prior to removal from the mold.

Final prepared specimens were subjected to laboratory-type oven thermal tests. In these tests it was determined that heating of the air space beyond the barrier unexposed face occurs through a combination of convective and radiative heat transfer. Radiation, being a function of temperature to at least the fourth power, rapidly becomes the primary mode of heat transfer providing that burnthrough does not occur. Emissivity and view factors both regulate radiation heating rates. The still air temperature data indicated that 400°F (204°C) measured 6 inches (15 cm) from the barrier materials was not exceeded during the 10-minute test for any of the formulations evaluated. Using information acquired from these tests, additional candidate materials were tested in the aircraft fire simulator at NWC.

Aircraft Fire Simulator Tests

Candidate fire barrier systems were tested at NWC to evaluate barrier system installation design and thermal capabilities with respect to the requirements to protect the flight control rods. For these tests, the fire barrier test simulator shown in Figure 9 was designed and fabricated for use as a realistic screening apparatus.

Installation of the fire barrier materials during an aircraft modification, including the effect of barrier penetrations by various aircraft tubes, wires and actuators, were major considerations in the test and evaluation of system concepts. It was recognized that barrier materials installation for an aircraft modification would entail a building-block approach to materials installation. Therefore the acquisition of data concerning the effect of joint design, fire blockage, and sealing of barrier penetrations was pertinent to barrier system evaluation. The effect of two types of penetrations representative of those encountered in an aircraft were tested. These consisted of (1) an electrical wire bundle, and (2) a thin-walled aluminum tube typical of a fuel vent line.

The fire simulation test requirements were based upon the need to generate a temperature, pressure, airflow and heat flux environment

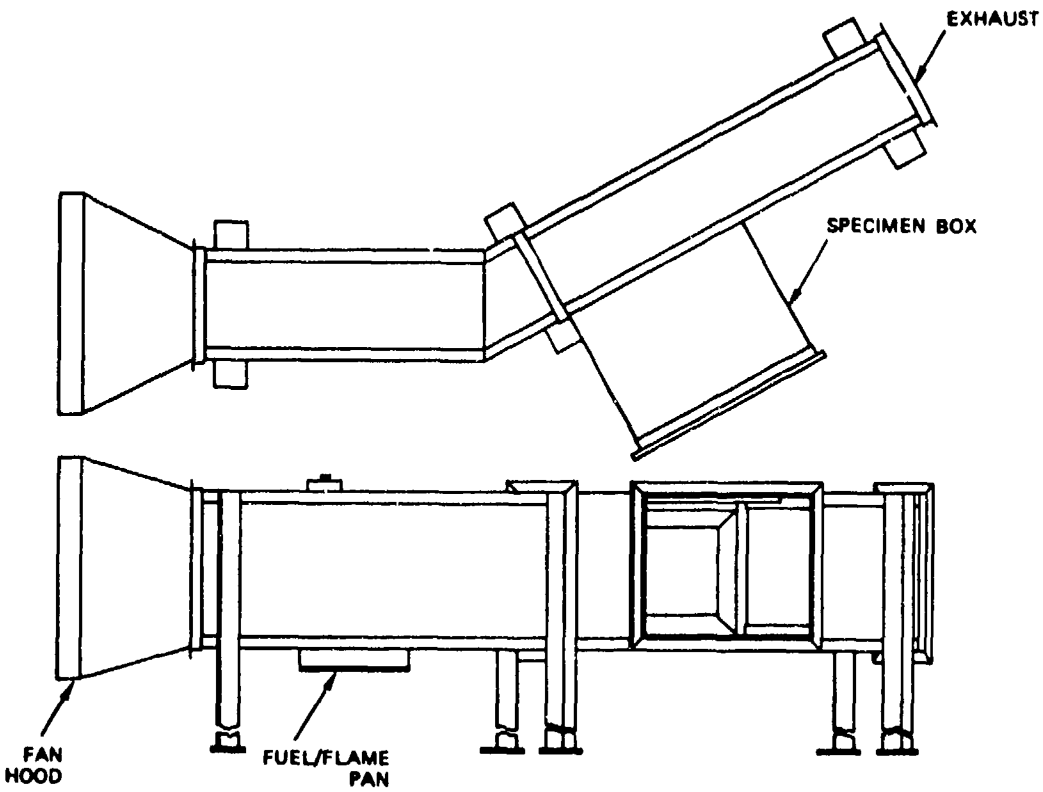


FIGURE 9. Aircraft Fire Test Simulator.

duplicating fire conditions in an inflight aircraft. The conditions established for evaluation of candidate barrier systems were as follows:

1. A heat rate of approximately 11.35 watts/cm^2/sec (10 Btu/ft^2/sec) across the test specimen face.

2. An airflow velocity of approximately 10 knots (5 m/sec) across the test specimen face.

3. A nominally fuel-rich fire typical of that postulated to exist in the aircraft failures to date.

4. Maximum temperature after 15 minutes of 400°F (204°C) measured 6 inches (15 cm) from the backface in a closed volume.

5. A maximum pressure differential of up to 0.5 psi (35gm/cm^2) across the specimen.

Both organic and inorganic materials were tested and evaluated. Organic materials investigated consisted of various formulations of polyurethane, closed cell, rigid foams. Inorganics tested were flexible silicone and alumina silica rigidized materials. Additionally, metallics combined with both organic and inorganic insulators were tested. A detailed listing of all materials and combinations of materials tested is presented in Table 3. A construction technique proposed by AVCO Systems Division was to cast the foam in the desired shape. This gives the foam block a high density thin skin which is thought to strengthen the char against the abrasiveness of the air-driven flame.

Results of all testing are summarized in Table 4. In the first series of tests, continuous specimens were evaluated to determine thermal toughness with no gaps or pressure differential loading applied. Normalizing the results to the installed weight (grams per square centimeter) required for each minute of thermal protection showed that the 5F14RS material provides a given level of protection at the lightest weight for the materials tested. These data are shown in Table 5.

The second series of tests measured specimen strength while exposed to fire. Clearance/gap intumescent seal strength evaluation was included to determine overall barrier system resistance to pressure loading. Intumescent chars of the 477GF and 1600 materials failed at 1.4 and 3.0 in.-H_2O (3.55 and 7.62 cm-H_2O), respectively. The 5A43 and the 5F14RS foam chars failed at 6.0 and 7.5 in.-H_2O (15.24 and 19.05 cm-H_2O) pressure differential, respectively, without reaching the failure point for the 313 intumescent char.

Resistance to burnthrough for gap-filling intumescent char under no-load conditions is shown in Figure 10. The M-30 material was not tested under these conditions; however, results of the load tests indicate that the M-30 material is comparable to the 1200 intumescent flexible sheet. Additionally, several combination intumescent coating schemes were investigated. The 1000 modified applied over 1200 sheet burned through in 6.0 minutes at 2 inches (5.1 cm) foam thickness; 1000 over 1200 sheet burned through in 10.0 minutes at the same specimen thickness.

Of the polyurethane foam types investigated, the 5F14RS foam (basically the 5A43 formulation modified by reduced catalyst, substitution of silica fibers for glass fibers, and addition of a high temperature reacting additive) was clearly superior in terms of thermal resistance. Indeed, it could often prevent burnthrough far in excess of 10 minutes with no supplemental intumescent coating protection. However, it must be noted that this was a special laboratory hand-mixed foam with no production experience behind it. Any of the other tested polyurethane foams could be made to resist burnthrough under the tested conditions with the aid of a proper intumescent coating.

TABLE 3. Materials Evaluated in Aircraft Fire Simulator Tests.

Test specimen description				Source	Remarks
Designation	Thickness, cm (in.)	Density, kg/m^3 (lb/ft^3)	Characteristics		
Organic Materials					
5A43	5.1 (2)	40 (2.50)	Glass reinforced, semirigid (no outer skin)	NWC Mixed	Ingredients supplied by AVCO, mixed to NASA specifications
5A43	2.5, 5.1, 7.6 (1, 2, 3)	50 to 65 (3 to 4)	Spray molded (without skin) (also closed-mold type)	AVCO	...
BX352-P	7.6 (3)	50 to 60 (3 to 3.75)	Glass reinforced, semirigid	Grumman	Spray-molded
5F14RS	5.1 (2)	50 to 65 (3 to 4)	Silica fiber reinforced, semirigid	NASA-Ames	Molded
Inorganic Materials					
WRP-X-AQ	2.5 (1)	320 (dry) (20)	Ceramic felt	Grumman	Density up to 1120 kg/m^3 (wet)
...	5.1 (2)	320	Silicone foam, flexible	Grumman	...
Metallics					
301 CRES + TBS-758	0.64 (0.27)	...	Silicon ablative on stainless	Grumman	...
Fiber glass + TBS-758	0.68 (0.29)	...	Silicon ablative on fiber glass/epoxy	Grumman	Fiber glass on both sides of silicon
321 CRES + 5F14RF	2.5, 5.1 (1, 2)	...	Polyurethane foam on stainless	NASA-Ames	Fireside of foam covered with aluminum sheet
Intumescent Coating					
1000 & 1000 modified; 1010, 1200 (flexible sheet); 1600B; 313				AVCO	...
477 GF				Grumman	...
M-30; Lacquer type, semiflexible sheet				NASA-Ames	Lacquer type is P-nasa salt in nitrocellulose lacquer

TABLE 4. Summary of Experiments Conducted Against Nonmetallics in Aircraft Fire Simulator.

Specimen description					Face coating			Gap coating					Burn-through time, min	Comments
Type	Thickness		Density		Type	Thickness		Gap		Type	Thickness			
	cm	in.[a]	kg/m³	lb/ft³		mm	in.	mm	in.		mm	in.		
WPR Felt	2.54	1	400	24.98	None	...	...	None		...	None		20+	Continuous specimen
Silicon	5.08	2	320	19.98	None	...	...	None		...	...	...	14.0	Continuous specimen
5A43	7.62	3	77	4.8	1200	0.51	0.020	None		...	...	...	13.1	Continuous specimen
5A43	5.08	2	69	4.3	None	...	...	None		...	...	...	4.5	Edge failure
			67	4.2	313	0.25	0.010	None		...	...	...	13.2	Continuous specimen
5F14RS	5.08	2	71	4.4	None	...	...	None		...	...	...	6.7	Edge failure
			75	4.7	None	...	...	None		...	...	...	15+	No burnthrough
5A43	5.08	2	55	3.4	1200	0.51	0.020	9.5	0.37	M-30	5.59	0.22	5.0[b]	15.24cm-H_2O (6 in.-H_2O)
Silicon	5.08	2	320	19.98	None	...	...	Butt[c]		None	...	...	8.0	Failed joint, no load ΔP
BX352-P	7.62	3	60	3.75	1600	0.51	0.020	6.3	0.25	1600	1.5	0.06	5.0	7.62cm-H_2O (3 in.-H_2O)
BX352-P	7.62	3	60	3.75	477GF	0.51	0.020	6.3	0.25	477GF	1.5	0.06	5.0	3.55cm-H_2O (1.4 in.-H_2O)
5F14RS	5.08	2	60	3.75	1200	0.51	0.020	9.5	0.37	M-30	5.59	0.22	5.0	19.05cm-H_2O (7.5 in.-H_2O)
WPR Felt	2.54	1	400	24.98	None	...	...	Butt		Ceramcoat	...	...	10.0	12.7cm-H_2O (5 in.-H_2O)
	2.54	1	59	3.7									4.0	
5A43	2.54	1	62	3.9									4.75	Edge failure
	5.08	2	59	3.7	1000[d]	0.25	0.010	Butt		1000[d]	1.52	0.06	8.0	
	7.62	3	48	3.0									7.5	
	2.54	1	61	3.8									4.5	
5A43	5.08	2	52	3.3	1010	0.25	0.010	Butt		1010	1.52	0.06	5.75	
	7.62	3	55	3.4									8.5	
5A43 (2 ea)	2.54	1	63	3.9	313	0.25	0.010	Butt		313	1.52	0.06	3.0	
													2.5	
5A43 cut (2 ea)	5.08	2	40	2.5	Lacq	0.25	0.010	Butt		Lacq	1.02	0.04	3.5	Lacquer coating
											2.03	0.08	4.0	
	2.54	1	56	3.5									2.75	
5A43	5.00	2	63	3.93	1000	0.25	0.010	6.3	0.25	1000	1.52	0.06	2.9	
	7.62	3	51	3.18									3.5	
	7.62	3	54	3.37	1000								5.0	
5A43	5.08	2	52	3.35	1000[d]	0.25	0.010	6.35	0.25	1000[d]	1.52	0.06	6.0	Edge failure
	7.62	3	58	3.62	1000[d]								8.5	

See footnotes at end of table.

TABLE 4. (Contd.)

Specimen description					Face coating			Gap coating					Burn-through time, min	Comments
Type	Thickness		Density		Type	Thickness		Gap		Type	Thickness			
	cm	in.[a]	kg/m³	lb/ft³		mm	in.	mm	in.		mm	in.		
5A43	2.54 5.08 7.62	1 2 3	61 53 53	3.80 3.31 3.31	1010	0.25	0.010	6.35	0.25	1010	1.52	0.06	3.0 2.5 4.75	
5A43	5.08 7.62 5.08	2 3 2	53 49 73	3.31 3.06 4.56	313 313 1000	0.25	0.010	6.35	0.25	313 313 1200	1.52	0.06	5.0 8.5 4.8	Edge failure
5A43	5.08 2.54	2 1	50 57	3.2 3.56	1000 Lacq	0.25	0.010	6.35	0.25	12/10 Lacq	4.57 1.02	0.18 0.04	10.0 2.25	1000 over 1200 coating Lacquer coating
5A43	2.54 5.08 7.62	1 2 3	58 54 57	3.62 3.37 3.56	1000	0.25	0.010	9.53	0.375	1000	1.52	0.06	3.6 4.75 5.5	
5A43	5.08 7.62 7.62	2 3 3	62 48 71	3.87 3.00 4.43	1000[d]	0.25	0.010	9.53	0.375	1000[d]	1.52	0.06	3.75 6.5 11.75	
5A43	2.54 7.62	1 3	64 50	4.00 3.12	1010 1010	0.25	0.010	9.53	0.375	1010 1010	1.52	0.06	2.25 5.0	
5A43	5.08 7.62	2 3	59 53	3.68 3.31	313 313					313 313	1.52	0.06	3.25 7.25	Gap never closed completely
5A43	7.62 7.62	3 3	53 53	3.31 3.31	1600B 1600B	0.25	0.010	9.53	0.375	1600B 1600B	1.52	0.06	5.75 7.2	
5A43	2.54 5.08 5.08	1 2 2	... 86 57	... 5.37 3.36	1200 1000[d] 1010	0.25	0.010	9.53	0.375	1200 1000[d] 12/10	1.52 2.03 2.54	0.06 0.08 1.00	4.0 6.0 3.5	 1000 over 1200 coating
5A43	5.08 7.62 2.54	2 3 1	... ...	... ...	1200	0.25	0.010	9.53	0.375	1200	2.03	0.08	10.1 20+ 3.5	No burnthrough

[a] Metric equivalent in inches.
[b] Differential pressure applied, values represent load test Δ.
[c] Specimen blocks butted together without gap.
[d] 1000 modified.

TABLE 5. Aircraft Fire Simulator Data on Thermal Resistance of Candidate Materials Without Pressure Differential Loading.

Test specimen description					Face coating			Burn-through time, min	Material required per minute protection[a]	
Designation	Thickness		Density		Type	Thickness				
	cm	in.	kg/m3	lb/ft3		mm	in.		g/cm2	lb/ft2
5A43	5.08	2	69	4.30	None	...	...	4.5[b]	...	
5A43	7.62	3	77	4.80	1200	0.51	0.020	13.0	0.045	0.09
Silicon	5.08	2	320	19.98	None	...	...	14.0	0.125	0.26
WPR felt	2.54	1	400	24.97	None	...	...	20+	0.041	0.08
5F14RS	5.08	2	71	4.43	None	...	...	6.8[b]	...	
5F14RS	5.08	2	75	4.68	None	...	...	15+	0.026	0.05
5F14RS	5.08	2	67	4.18	313	0.25	0.010	13.3	0.026	0.05

[a] Installed unit area.
[b] Edge failure rather than actual burnthrough.

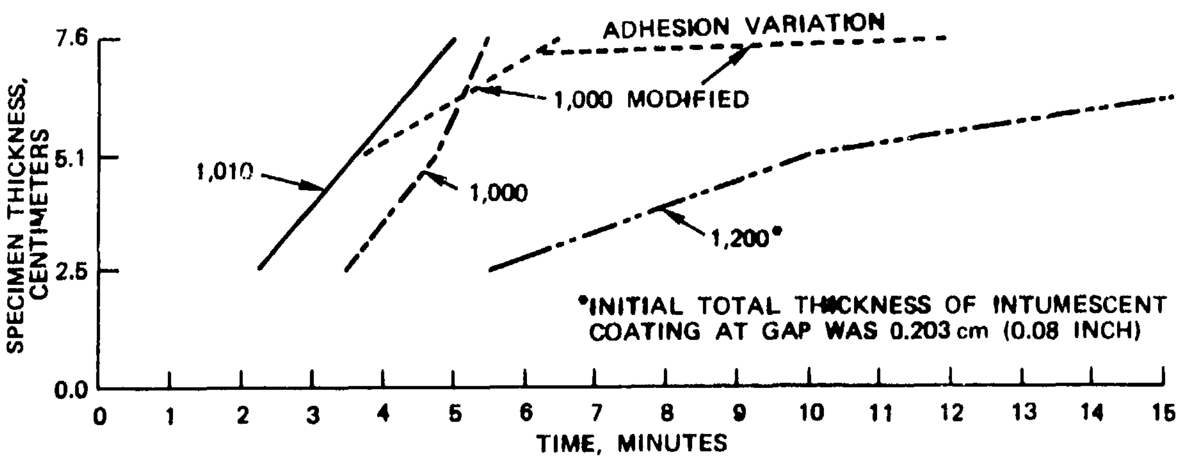


FIGURE 10. Burnthrough Time for Intumescent Materials Filling a 0.40 Inch Gap in 5A43 Foam Specimen.

Limited testing was conducted with two inorganic materials; a flexible silicon foam and a rigidized ceramic felt (designated WRP-X-AQ). Although high in density when compared to polyurethane foams, both materials exhibited high thermal resistance. The silicon foam did suffer from significant distortion prior to burnthrough and continued to burn for a number of minutes after test shutdown. This vigorous self combustion characteristic indicates that the silicon could act as a fire relight source aboard an aircraft. The ceramic felt (visual inspection suggests that it is a dense mat of silica fibers rigidized with a ceramic type binder) proved to be almost totally inert in the thermal test environment. Steam generation during the first 5 minutes of fire exposure indicated that the felt had absorbed a significant amount of water. Later laboratory tests showed that after oven drying the ceramic felt density was 20 lb/ft^3 (320 kg/m^3) but could be increased to 70 lb/ft^3 (1,120 kg/m^3) by water immersion without changing dimensions.

Even with intumescent coating protection, the polyurethane foams burn down to their basic carbonaceous char form within about 5 minutes after exposure to fires of the test intensity. Little actual test or measurement data of physical characteristics of these chars at temperature exist. Their ability to withstand the internal aircraft airflow generated and vibrational type loads is unknown and must be determined prior to any actual incorporation into aircraft. In the limited tests conducted with pressure differential applied loads the foam failed at relatively low levels; this suggests that metallic backside reinforcement would probably be required to meet environmental criteria during an actual fire.

Backside still-air temperature data as functions of time and distance for the TBS-758 silicone coated stainless steel sheet is presented in Figure 11. The 205°C temperature requirement at 5 minutes measured 6 inches (15 cm) from the backface was just met with this barrier configuration. Metal-backed silicon joints, butt and lap joints also were tested. The butt joint failed at about 4 minutes into the burn test. The lap joint (2 inches (5 cm) overlap) failed after 4.7 minutes of fire exposure. One burn test was conducted with the silicon insulator faced on both sides with impregnated fiber glass. This test specimen included a lap joint. Rapid erosion of the fire-exposed fiber glass occurred with joint failure at 60 seconds after start of the burn.

In the test conducted to determine effects of aluminum tubing and electrical wire bundle penetration of a fire barrier, a weak link failure mode predominated. The thin-wall aluminum tube was protected with 0.050 inch, (1.27 mm) of 313 intumescent paint where it protruded into the fire stream. Once fire penetrated a weak spot in the tube, flame propagation from the inside rapidly melted out the rest of the tube thus defeating all other thermal protection, with the tube acting as a conduit for breaching the fire barrier. This phenomenon did not occur with the wire bundle. Although it melted where it was directly exposed to the fire stream, flame failed to penetrate the barrier during the 7-minute burn test.

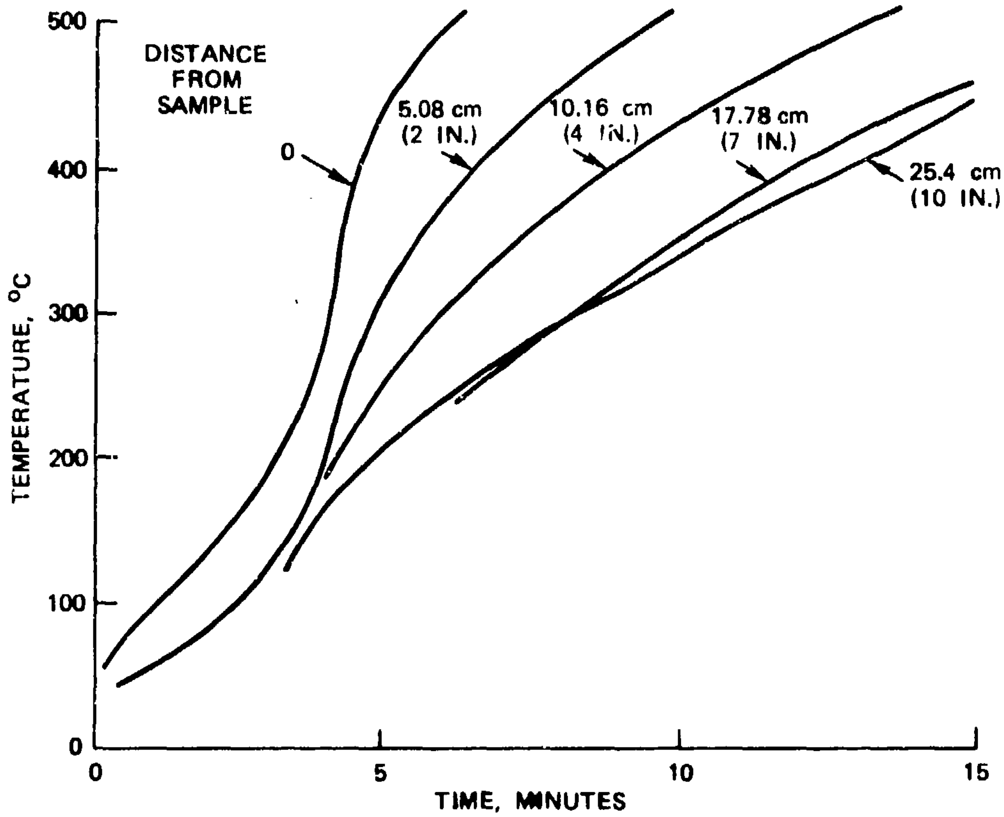


FIGURE 11. Backface Temperature Data for Stainless Steel Sheet Coated With TBS-758 Silicone.

LEAK SIMULATOR/FUEL IGNITOR TESTS

Tests were conducted to verify the techniques chosen for fuel line rupture and fuel ignition for the engine-caused failure simulation. A simulator mockup was designed and fabricated for use in conducting the leak simulator/fuel ignition tests. The basic design of the boiler plate mockup representing the engine fan area of an F-14A right-hand engine nacelle is shown in Figure 12. Fancase rupture airflow used in the test was 1.2 lb/sec (0.54 kg/sec) provided by a 75-horsepower vane axial-flow fan. This airflow was ducted into the mockup. A linear shaped charge (LSC) was used to cut a 12-square-inch (56-cm^2) hole in the fuel line. A 6-second burn time magnesium flare was used as a fuel ignition source.

A mockup fuel line was filled with JP-5 fuel and connected to an approximate 25-gallon (95-liter) fuel supply barrel pressurized to 32 psi (2.25 kg/cm^2) with helium. A flow venturi with a pressure transducer attached was installed in the fuel line rupture. Two cameras with frame rates of 24 and 400 frames/second respectively, were positioned to record test events. The test sequence is given in Table 6. In the first test, airflow was initiated well before the start of the actual test sequence

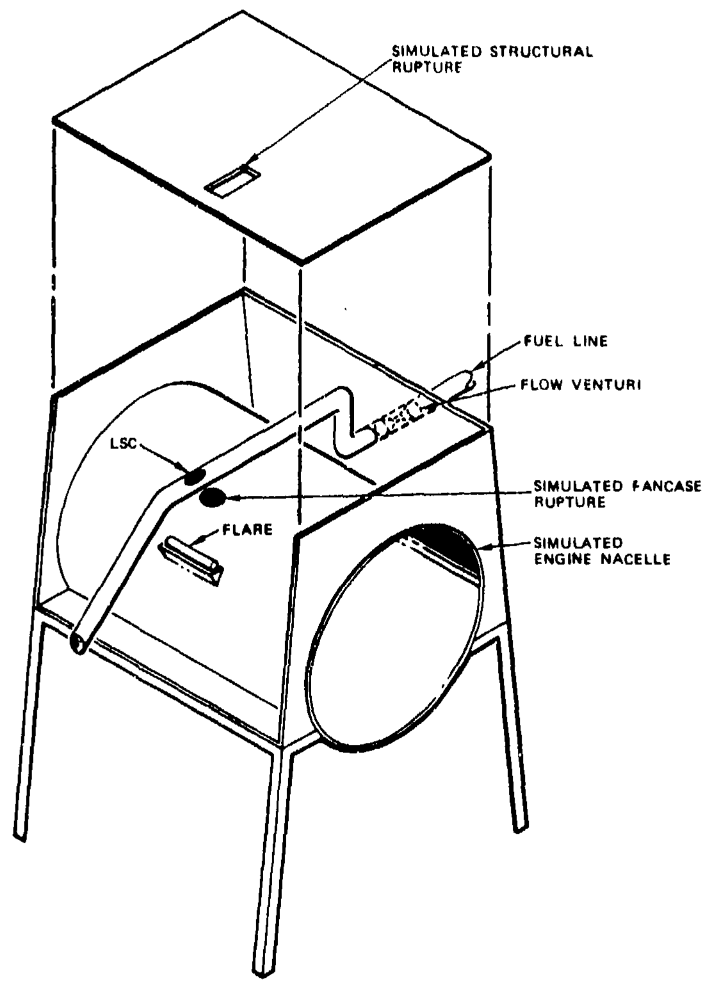


FIGURE 12. Nacelle Fuel Leak/Ignitor Simulator Mockup.

to allow the 75-horsepower fan to come up to speed and to avoid instrumentation transients caused by fan startup. The flare was ignited 0.5 second before detonation of the LSC to ensure a sufficiently stabilized fuel ignition source.

At time of LSC detonation a hole was cut in the fuel line resulting in pressurized fuel being blown out through all openings in the mockup. Ignition of fuel within the mockup occurred immediately after LSC detonation. Film coverage showed that fancase airflow carried ignited fuel through the hole in the top deck. Fire spread rapidly to all areas of the fuel cloud resulting in a fireball approximately 12 feet (3.6 meters) in diameter. Initially, much of the dumped fuel fell onto the pad around the mockup and burned.

TABLE 6. Nacelle Fire Simulator Test No. 1.

Time (T)	Event
T $\geq$ - 60 sec	Initiate water washdown of pad; start 75-hp fan
T - 20 sec	Event sequencer on; start cameras
T - 0.5 sec	Flare ignition
T = 0 sec	LSC detonation; fuel ignition
T + 10 sec	Initiate fire abatement procedures
T + 60 sec	Fire extinguished

The initial surge of fuel was followed by two other distinct surges of fuel from the ruptured line. Each surge produced a new fireball. It was assumed that these surges were caused by the pad fire vaporizing fuel remaining in the supply line after the initial surge, momentarily pressurizing the line and causing a new surge.

Test data from the pressure transducer on the fuel flow venturi was lost shortly after ignition of the fuel due to burnthrough of the instrumentation cables. However, about 56 ms of data were obtained. This data indicated that the LSC was introducing a hydraulic ram overpressure in the fuel line. Subsequent examination of the transducer indicated that the peak overpressure was slightly less than 300 psi (21.2 kg/cm^2).

Based upon these data, it was determined that, in order to protect the pressure transducers used in the actual F-14A test from both hydraulic ram and subsequent fire damage, the transducers should be isolated from the flow venturi by lengths of at least 6 feet (2 meters) of small diameter copper or stainless steel tubing.

Based on initial test results, simulator mockup hardware was appropriately modified. A second flow venturi was installed downstream of the fuel line LSC location and high-pressure, short-response-time transducers were installed on the ends of the tubing in order to more fully explore the effect of hydraulic ram on the test instrumentation. Other changes made in support of the second test included repositioning of the 400 frame/second camera and modification of the test sequence. A second test was subsequently performed using the same test configuration. In this test, the test sequence of Table 7 was successfully accomplished.

The second test provided additional information about the nature of the hydraulic ram imparted to the fluid by detonation of the LSC. It was found that at the end of a 6-foot (2.6-meter) length of 0.0625-inch (1.60-mm)-diameter stainless steel tubing the transducers on both sides of the

TABLE 7. Nacelle Fire Simulator Test No. 2.

Time (T)	Event
T < - 60 sec	Start pad washdown
T < - 60 sec	Start 75-hp fan
T - 15 sec	Start cameras
T - 1 sec	Flare ignition
T = 0 sec	LSC detonation, fuel ignition
T + 10 sec	Begin fire abatement procedures
T + 60 sec	Fire extinguished

LSC installation measured several pressure pulses of approximately 30 psi (2 kg/cm^2) for a period of 2 to 4 milliseconds duration.

The two tests provided information on the proposed methods of simulating fuel line rupture and fuel ignition. It was found that the LSC produced a predictable size hole of desired area in the fuel line without producing a hydraulic ram that would endanger fire-isolated instrumentation. The flare used to cause fuel ignition proved a reliable and effective method of ignition.

The nacelle fire simulation test results verified the techniques which had been proposed for full-scale nacelle fire tests. The simulator tests illustrated the extent to which the fuel could be expected to spread into the engine nacelle structure, the magnitude of the resulting fire, and some indication of the heat produced by the fire.

FULL-SCALE TESTING

Full-scale testing operations utilizing the F-14A test article encompassed airflow tests, fire extinguishing agent flow and concentration tests, and the aircraft fire verification tests. These tests provided the data on which the evaluation of aircraft modifications for fire containment and extinguishment were based.

AIRFLOW TESTS

An overview of the test article in relation to the DASH system discharge nozzle is shown in Figure 13.

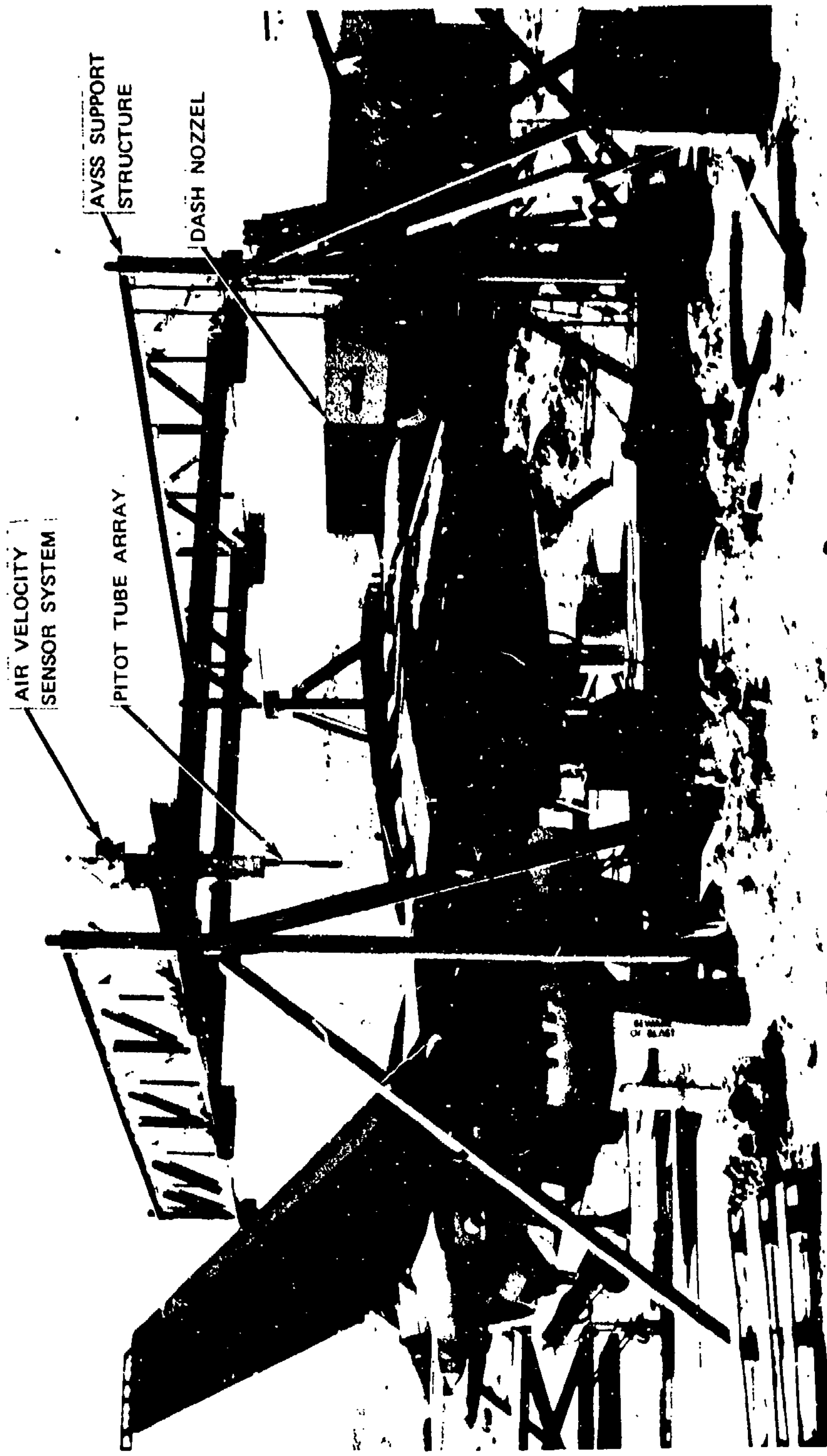


FIGURE 13. Test Preparations for Full Scale Airflow Tests.

The aircraft was jacked to a height which provided maximum coverage of the test article area by DASH system airflow and the pitot tube array. The test article was subsequently leveled and tied down to the test pad. The pitot tube array (Figure 14) was adjusted to provide measurements within approximately 1 inch (25.4 mm) above the skin of the test article wherever possible. Position reference points were established on the upper wing and fuselage surfaces to provide points to which airflow data could be related.

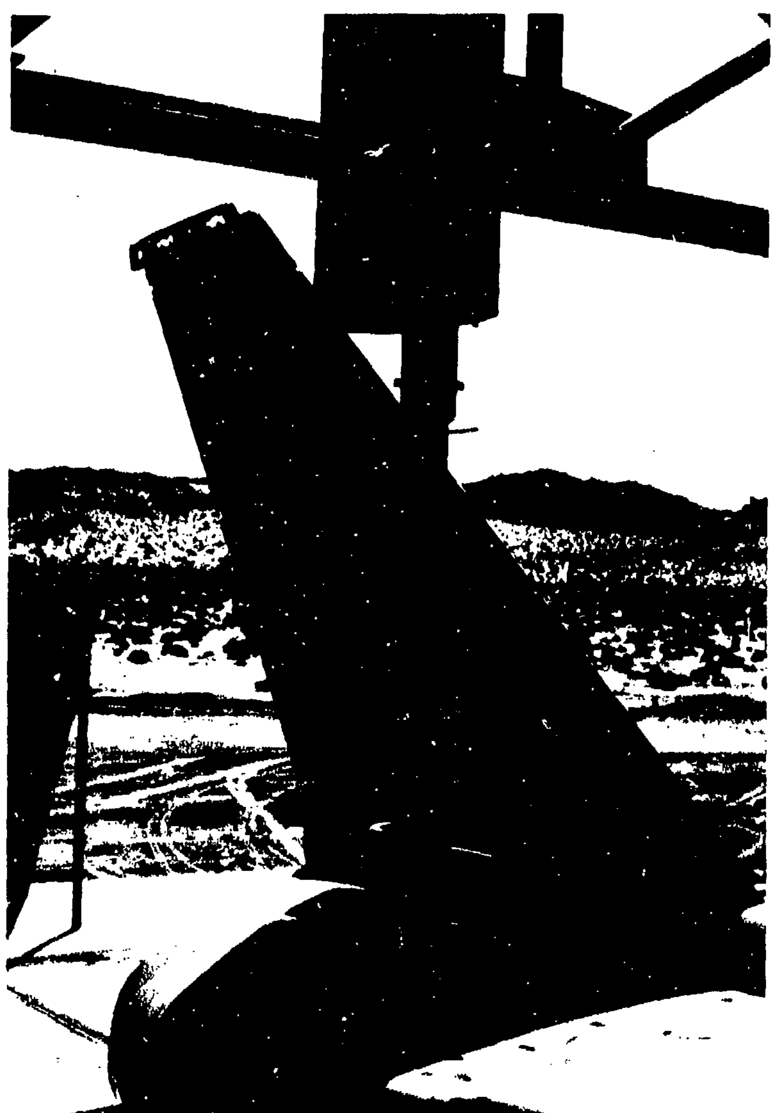


FIGURE 14. Pitot Tube Array Used for Airflow Tests.

Preliminary airflow tests had been made prior to installing the test article on the test pad, to provide basic DASH system airflow data. After installing the test article on the test pad, a series of tests were performed to map airflow over the top of the test article and to determine flow patterns over and within the engine nacelle. Airflow mapping was performed using the pitot tube array shown in Figure 14 to provide airflow measurements; airflow velocities were subsequently plotted on the planform position map shown in Figure 15 at intervals of 3 inches (76 mm) for a total depth of 24 inches (610 mm). Construction of the pitot tube array and support system was accomplished as part of the pretest preparation task described in Appendix B.

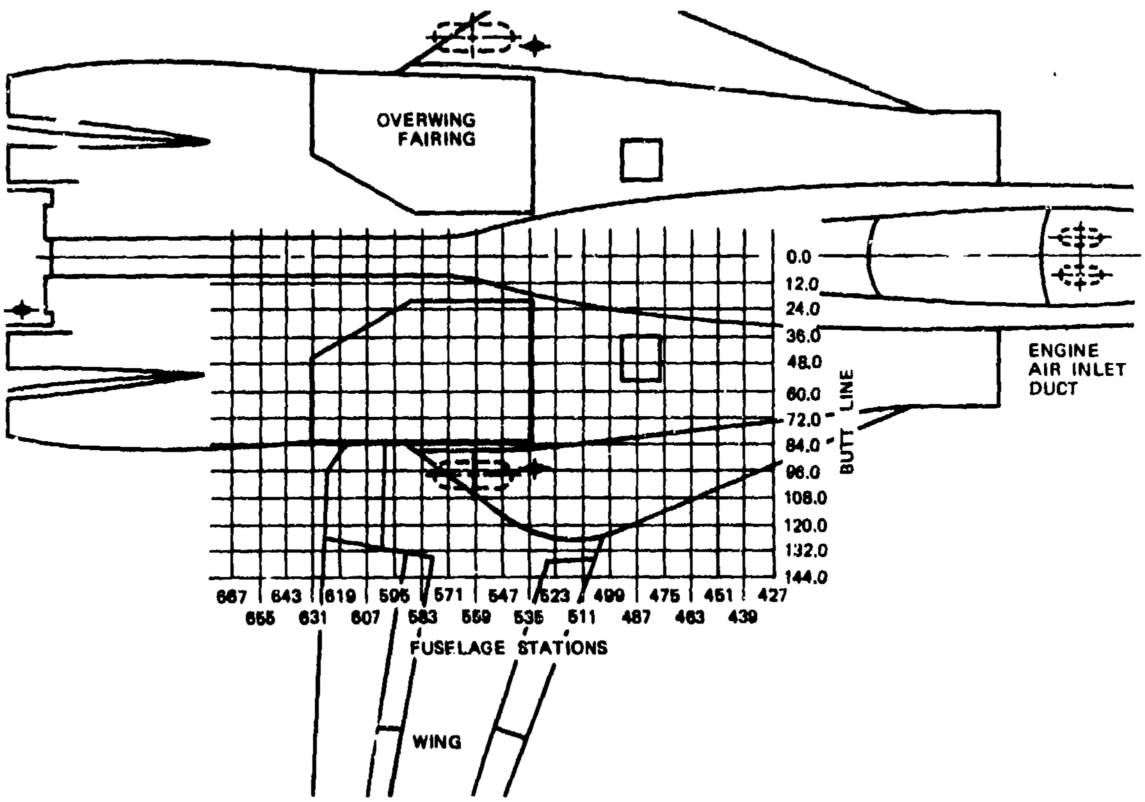


FIGURE 15. Aircraft Planform Used for Airflow Mapping Tests.

Airflow Calibration Test

The purpose of this test was to map the velocity profile of the airflow over the right side, aft portion of the test article. The test simulation goal was to provide 250-knot airflow over the test area. The airflow from the nozzle acts as a free jet; this results in an airflow plume with velocity gradients decreasing with distance from the nozzle. The 250-knot velocity point was therefore chosen to be the aft area of the overwing fairing.

Airflow mapping operations encompassed an area from fuselage station 456 to 655 and buttline 10 to 130. Within this area an airflow depth up to 24 inches (610 mm) was mapped. Airflow mapping was accomplished using the 250-knot airflow condition and traversing the pitot tube array laterally across the wing area. Airflow data, pressure, temperature, and position data were recorded at each 3-inch (76-mm) increment of the traverse. Upon completion of the first traverse, the pitot tube array was set to a new position on the Z-axis and the X-Y traverse was repeated. This procedure was repeated at a total of 11 fuselage stations to accumulate the data necessary to map airflow over the overwing structure.

The airflow data were recorded on the planform view of the aircraft overwing area (Figure 15). Airflow measured by each of the pitot tubes of the pitot tube array was recorded on a separate planform. Data obtained in this manner provided a means of determining airflow profiles at different heights above the aircraft structure. Typical airflow profile maps made from the recorded data are shown in Figures 16 and 17.

The mapping data shows that the desired 250-knot airflow at the aft area of the overwing fairing was achieved with an acceptable airflow quality. There was some turbulence from the DASH system nozzle. This was expected due to the result of a rectangular nozzle blowing over the complex surface of the F-14A test article. In addition to the airflow mapping data, the visual flow pattern evidenced by tufts shows good airflow stability in the portion of the free jet over the test area with ambient air being entrained in ejector fashion around the edges of the nozzle air stream. By positioning the nozzle exit forward of the wing box, the airflow had stabilized by the time it reached the test portion of the overwing fairing. A view of the tufted test portion mapped in these tests is shown in Figure 18.

Flow Visualization Test

The purpose of this test was to establish the airflow paths in both normal operation and damaged configuration in the engine nacelle and overwing fairing accessory bay areas, and to introduce a vapor to see the mixing and distribution of the fire extinguishing agent. This was accomplished by the prior installation of Plexiglas skin over portions of the engine nacelle and overwing fairing, along with tufting the interior areas. Flow visualization tests encompassed the following: (1) The flow of normal nacelle cooling air, (2) flow of fancase rupture simulation air in addition to the nacelle cooling air and top deck rupture, and (3) carbon dioxide (CO_2) agent flow into the fancase rupture simulation environment through the fire extinguishing system nozzles. It was intended that results would be used to help optimize fire extinguisher nozzle positions and aid in selection of gas analyzer sensor placement patterns for subsequent tests.

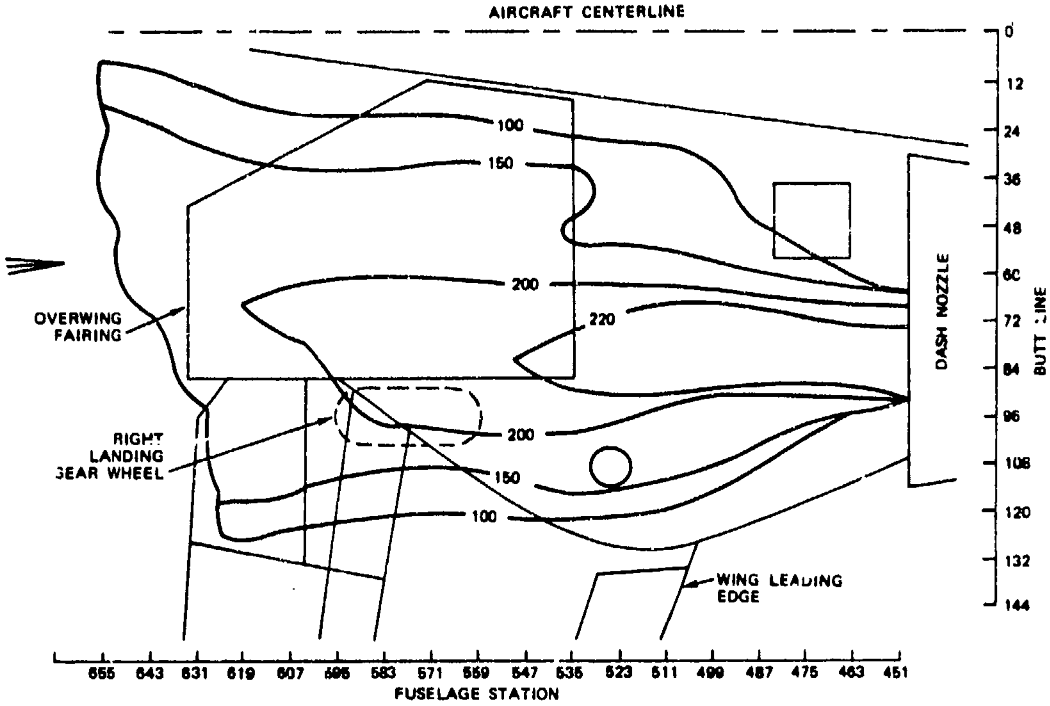


FIGURE 16. Airflow Velocities Measured 1 Inch Above Surface of Test Article.

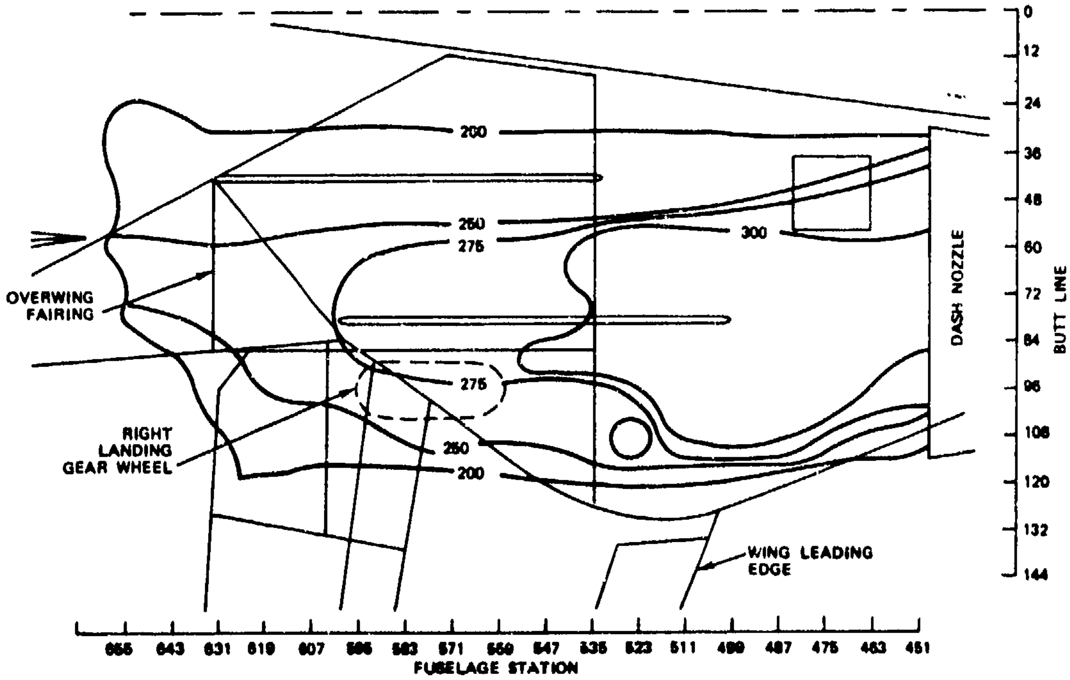


FIGURE 17. Airflow Velocities Measured 7 Inches Above Surface of Test Article.

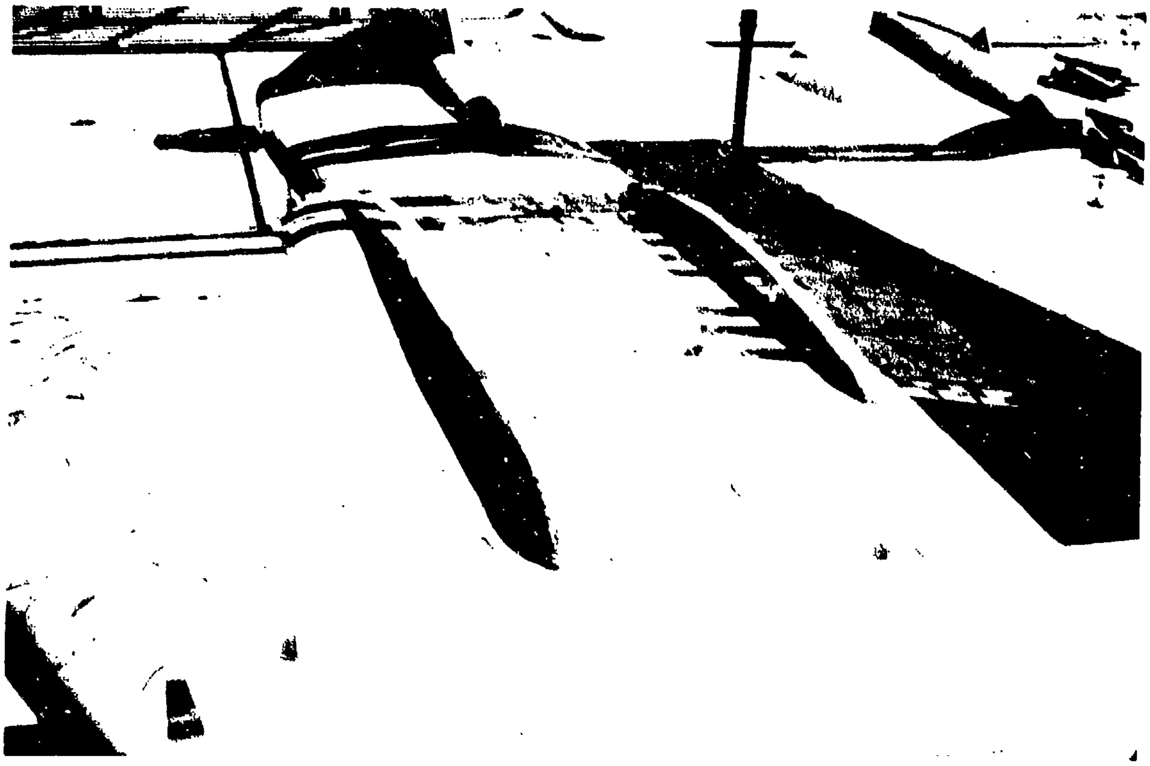
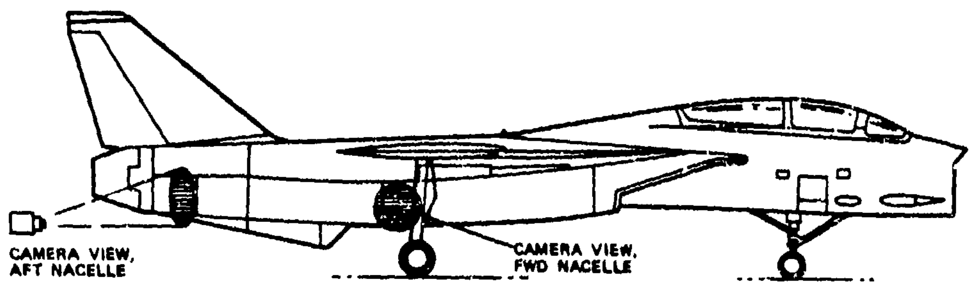


FIGURE 18. Overwing Fairing Tufting for Airflow Tests.

Test visualization was, to a great degree, dependent upon camera coverage to obtain the majority of the data. Primary instrumentation consisted of high-speed cameras (400 frames/second) which were used to accumulate flow visualization data. Placement of the cameras was such as to provide the fields of view shown in Figure 19. Pressure differentials between the nacelle, accessory bay and control rod trough were also monitored.

On production aircraft, two bottles of fire extinguishant are mounted in the aircraft (refer to Figure 5 for system layout). The same bottle sizes were used for the tests reported herein but unique mounting placement was necessary to reduce costs. The 378-cubic-inch (6195.46-cm^3) bottle was mounted outside the right engine nacelle and the 86-cubic-inch (1409.54-cm^3) bottle was mounted to a plate (see Figure 20) cut out of the overwing fairing. Discharge tubing diameters and lengths were the same as those proposed for production aircraft.



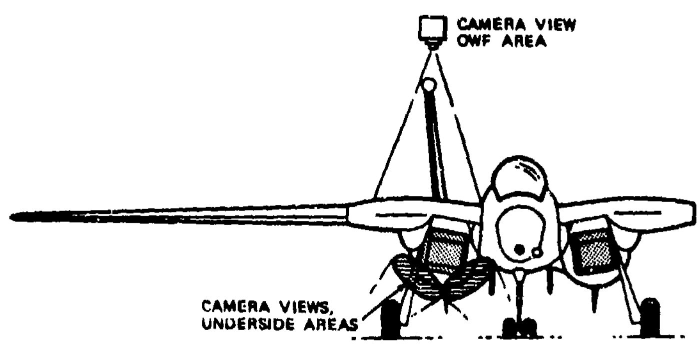


FIGURE 19. Video Camera Placement for Flow Visualization Tests.

FIGURE 20. Extinguisher Bottle Mounting for Flow Visualization Tests.

Tufts of red yarn were attached in the engine nacelle and overwing fairing areas which could be viewed by cameras. Tufts in the aft nacelle area are shown in Figure 21; tufts were also installed in the forward area. Extinguisher nozzle placement in the engine nacelle is shown in Figure 22. The extinguishant discharge loop shown in Figure 22 is located at the nacelle forward wall; cooling airflow is depended upon to carry the extinguishant aft through the nacelle volume.

The bottle containing the extinguishing agent was replaced by a line connected to the test facility CO_2 system for this test. This was used to facilitate visibility as the vapor expands from the nozzle producing a mist. Use of the proposed Halon 1301 agent would be expensive and, as the agent does not produce the same misting condition, its presence could only be sensed as a transient movement in the tufts near the discharge nozzles.

A view of the overwing fairing Plexiglas prior to attaching the tuft pattern was shown in Figure 6. The pretest configuration of the test article is shown in Figure 23.

In performing the flow visualization tests, the DASH airflow was initiated and allowed to stabilize. Subsequently, nacelle cooling airflow was initiated and given sufficient time to achieve a steady-state flow. Camera lights were turned on and, at time T-15 seconds, the data recorders and cameras were started. Engine fancase rupture simulation airflow was introduced at T = 0 seconds. This was followed by introduction of CO_2 into the nacelle at T + 10 seconds. The test cameras, lights and data recorders were shut down after the cameras ran out of film at approximately T + 25 seconds. Subsequently, the support air sources were shut down.

Films of the nacelle showed that extinguishant expanded quickly throughout the forward nacelle volume leaving no voids. Flow was counterclockwise up the outer wall of the nacelle, spiraling aft to expel through the afterburner closure fingers and the nacelle vent. The presence of large quantities of airflow being ejected into the nacelle through the simulated fancase rupture in a direction opposite to the normal airflow circulation did not appear to have significant effect on the dispersion of the CO_2 in the forward nacelle area. It was anticipated that no modification to the nacelle agent nozzles would be required. This was later confirmed by the agent concentration tests.

Flow visualization results in the overwing fairing accessory bay volume indicated good distribution only in the deep forward volume due to the airflow coming into the accessory bay from the nacelle top deck rupture hole. There was no noticeable extinguishant in the aft and inner volumes. Time did not allow continued testing to examine more closely the distribution of agent in this small but critical volume. It was postulated that Halon 1301 agent concentration might be adequate in this area, whereas CO_2 concentration may be difficult to create in

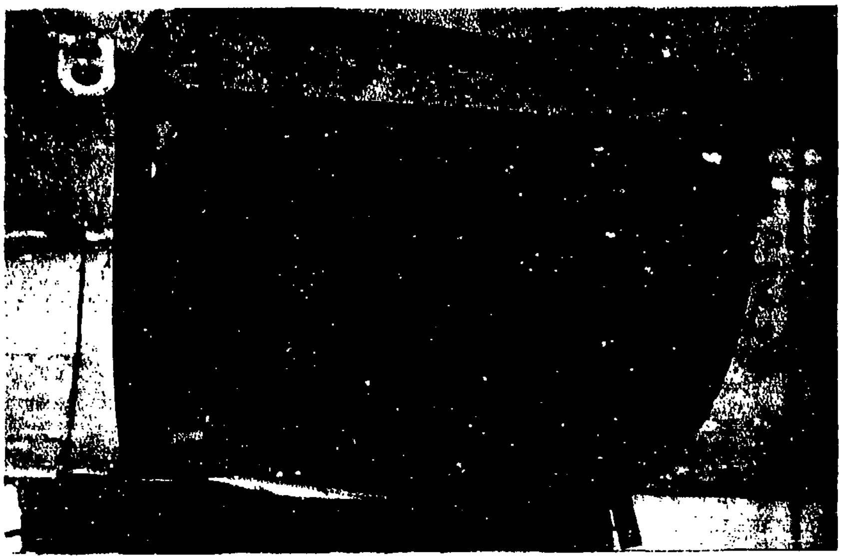


FIGURE 21. Nacelle Tufting Installed for Flow Visualization Tests.

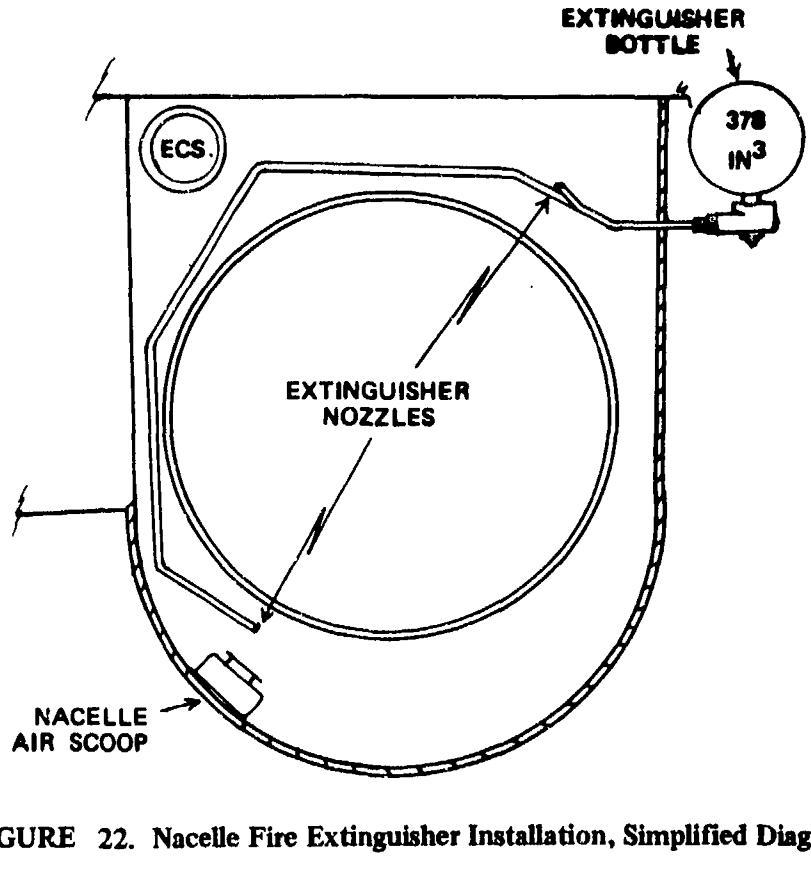


FIGURE 22. Nacelle Fire Extinguisher Installation, Simplified Diagram.

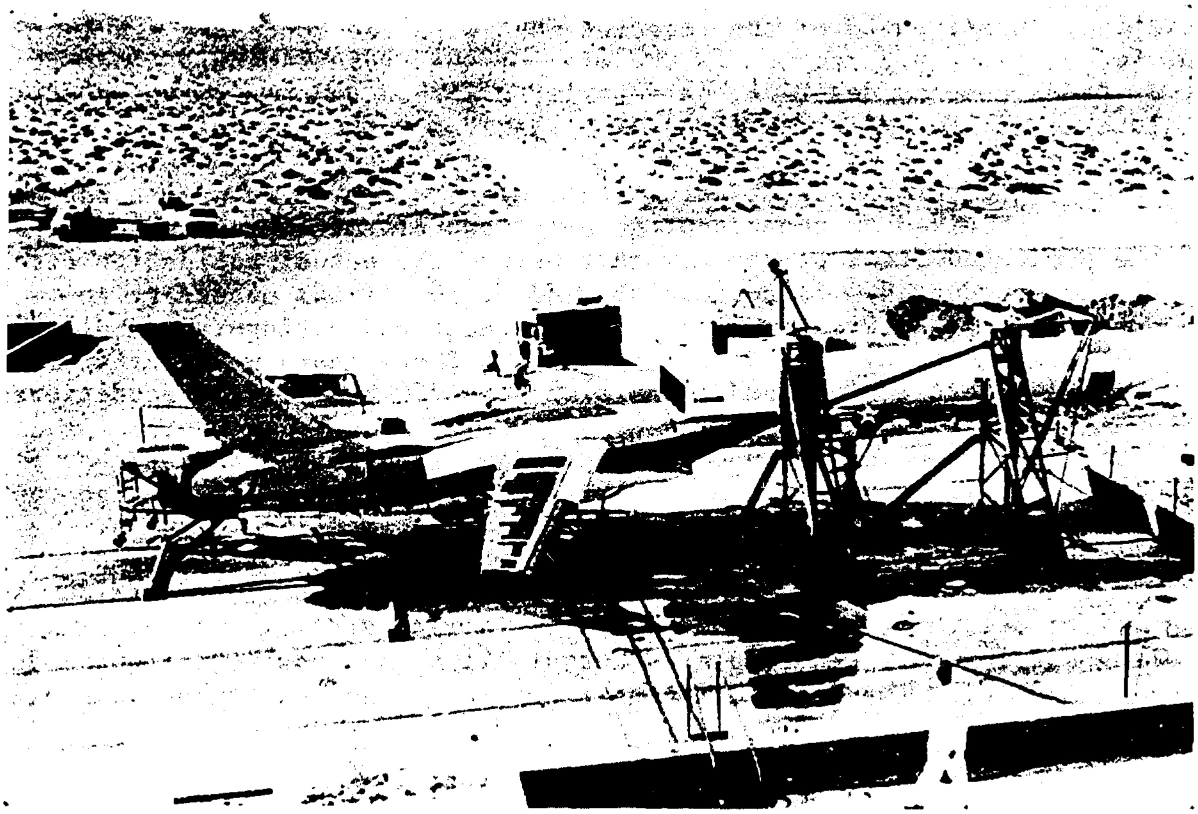


FIGURE 23. Configuration of F-14A Test Article for Flow Visualization Tests.

such confined quarters. For these reasons, no extensive nozzle modifications were made at this time. However, preparations were made to modify these if necessary during subsequent agent concentration testing. The potential areas of low agent concentration due to the airflow paths were identified and gas analyzer probe locations were determined for the next series of tests.

AGENT CONCENTRATION TESTS

This portion of the test effort was directed to initial testing of the fire extinguishing technology proposed in ECP-835 (see footnote 1). System performance requirements defined in MIL-E-22285,[6] specifying an

[6] Military Specification. *Extinguishing System, Fire, Aircraft, High-Rate Discharge Type, Installation and Test of.* (MIL-E-22285 (WEP), 11 December 1959 and Amendment 1, 27 April 1960, document UNCLASSIFIED.)

agent concentration of at least 6 percent by volume in air (15 percent relative) in all parts of the affected zone with the concentration lasting for at least 0.5 second was the minimum test requirement. A report by W. Kidde Company describes in detail the method and equipment used to measure agent concentration along with results of the testing.[7]

The object of the agent concentration tests was to determine whether the system provided an agent concentration capable of extinguishing fire in the protected areas of the aircraft before the actual validation fire tests were conducted. To achieve this objective it was necessary to determine the effectiveness of nozzle placement and extinguishant distribution as affected by the several airflows. These tests were performed on the F-14A test article under the simulated 250-knot flight conditions.

The right-hand engine nacelle and overwing fairing accessory bay of the test article comprised the test area for measurement of agent concentration (refer to Figure 5 for bottle and nozzle location in the aircraft). The tests were conducted with a non-operating engine installed in the aircraft to serve as a volume filler for the nacelle. Airflow conditions for the test included overwing fairing airflow and nacelle cooling airflow for both simulated normal flight and engine failure mode conditions. Five specific test conditions were measured; these were as follows:

1. Nacelle under normal flight conditions.
2. Nacelle with fancase rupture.
3. Nacelle with fancase rupture plus nacelle top deck rupture.
4. Accessory bay under normal flight conditions.
5. Accessory bay with nacelle top deck rupture.

The F-14A test article was tested with two separate fire extinguisher systems (see Figure 5) consisting of a 10.5-lb (4.76-kg) capacity bottle for the engine compartment volume and a 2.75-lb (1.25-kg) capacity bottle for the overwing fairing volume. In preparation for the tests, gas analyzer sensor tubes were installed in the engine nacelle and accessory bay areas at locations shown in Figures 24 and 25.

A Statham Gas Analyzer, Model GA-5b, was used to determine agent concentrations. This unit consists of 12 transducer units mounted into three blocks to which the gas sample (sensor) tubes are attached. A control unit is used to supply power to the analyzer circuits and to translate the signals to the recorder. A vacuum pump pulls in the gases through the sampling tubes to the transducer units.

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[7] W. Kidde Company. *Navy F-14 Engine Fire Extinguishing System Agent Concentration Test*, by R. A. Wasienko. Belleville, N.J., July 1976. 22 pp. (Kidde Service Engineering Report No. 140, publication UNCLASSIFIED.)

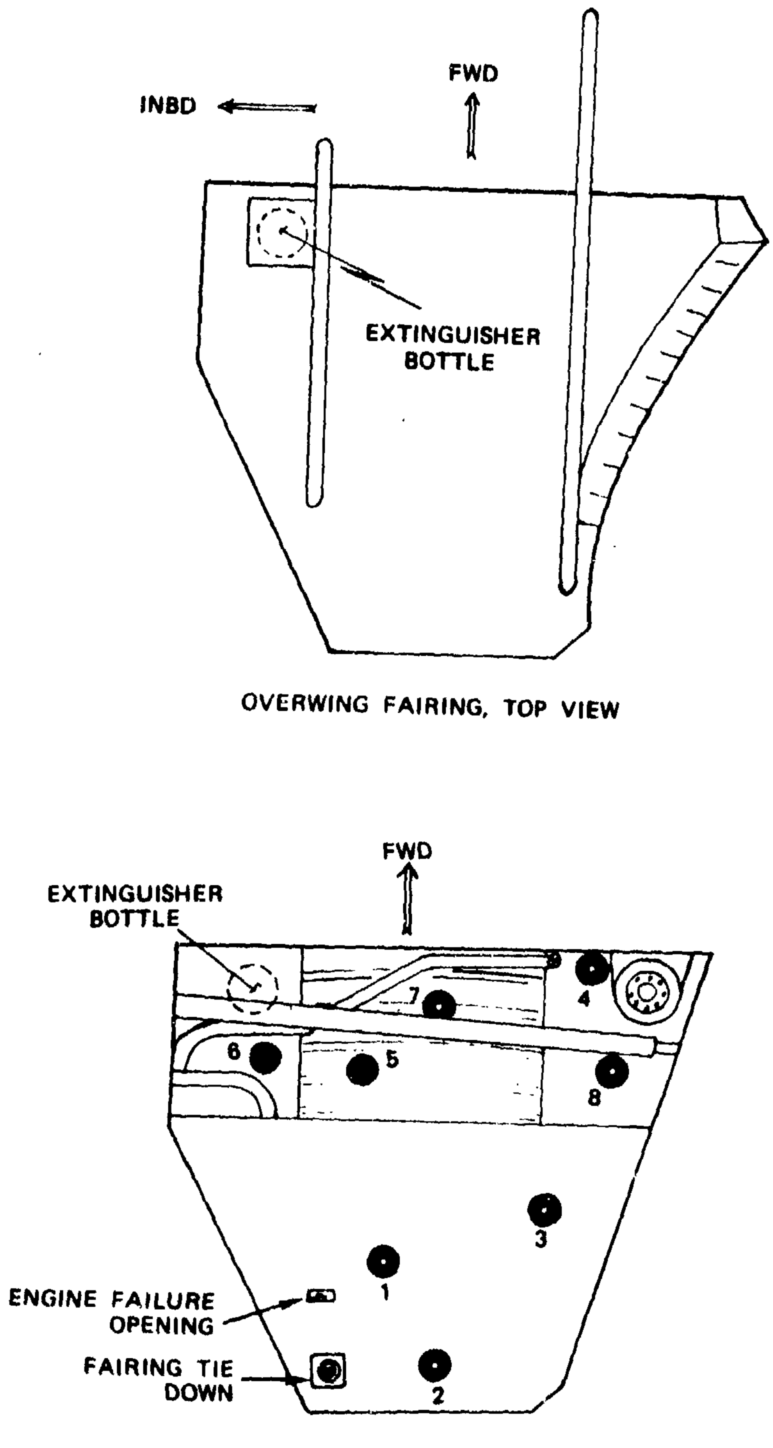

FIGURE 24. Sensor Pickup Locations, Overwing Fairing Volume.

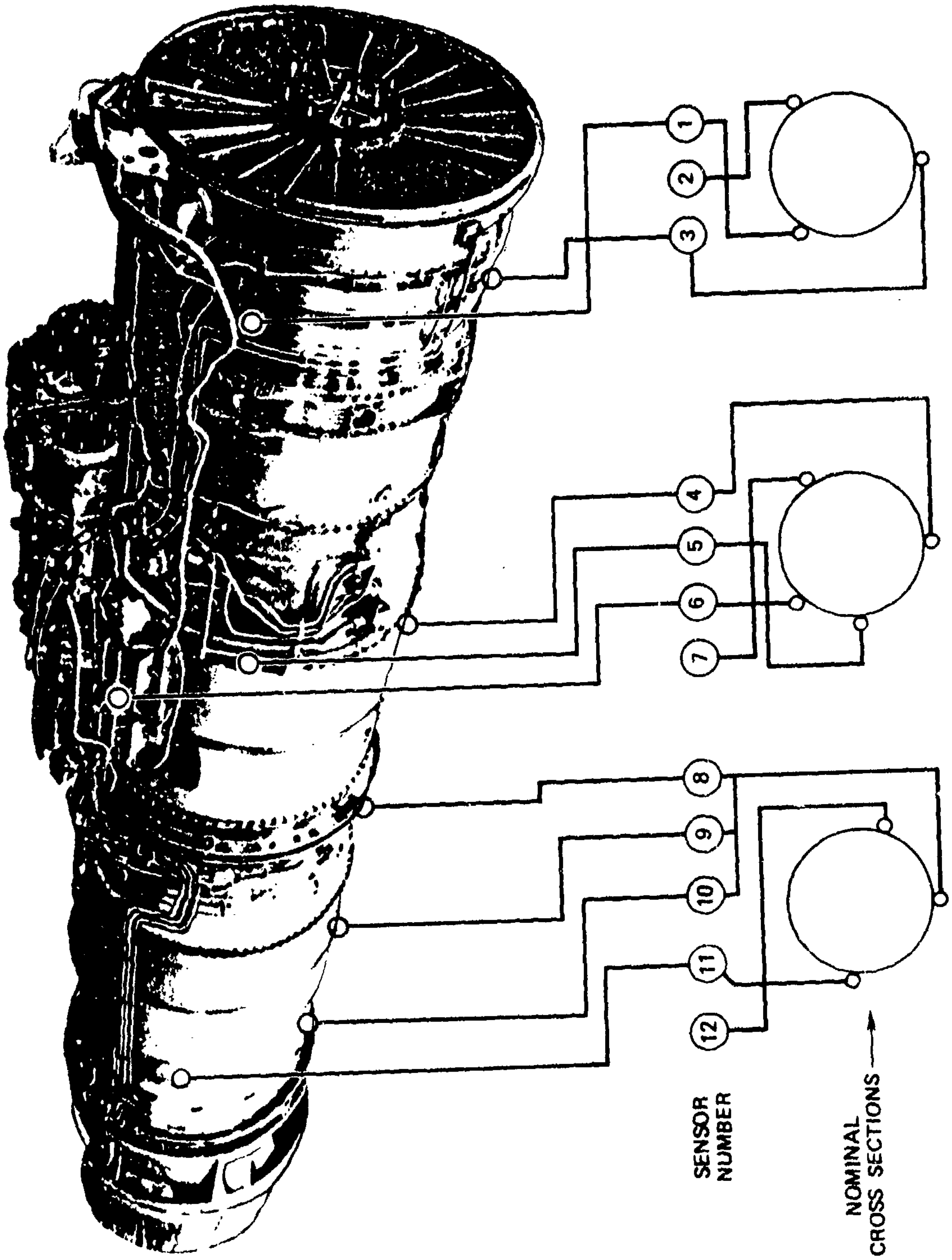

FIGURE 25. Sensor Pickup Locations, Right Engine Nacel.

The raw information produced during the test is seen on the recorder paper as a group of galvanometer deflections. Prior to firing the extinguisher at the test condition, the operator obtains a run of base lines followed by a run with the vacuum pump operating in order to record full air deflections. The extinguisher is then fired and the analyzer system causes the galvanometers to deflect according to the change of gas concentration. The recorder is stopped when it is obvious to the operator that the agent concentration has decreased to a point of no interest.

In reducing the information recorded, air deflection from the base line is measured and compared to mixture deflection for air deflection. These measurements are used to produce the relative gas concentration percentage for a given time as defined by the timing lines on the chart produced by the recorder. This data is normally plotted on curves of percent relative gas concentration versus time. Using the curve found in FAA Technical Development Report No. 403,[8] the relative gas concentration information is translated into percent by volume concentrations.

Eight gas analyzer tests were performed during this phase. The system as originally installed proved to be adequate for the engine nacelle, however the overwing fairing accessory bay nozzle arangement required modification as was indicated by the flow visualization tests.

A summary of the test results is presented in Table 8 for the final nozzle configurations. The corresponding data plots for percent relative agent concentration versus time are shown in Figures 26 through 29.

In the nacelle under normal flight conditions, where only cooling airflow is provided, the agent concentrations are greatest for the longest duration. As more airflow is introduced and the top deck ruptured, these values decrease as would be expected. Under normal flight conditions the agent concentration was maintained above 15 percent for 5.35 seconds which exceeds the test requirement by 4.85 seconds. The maximum concentration for 0.5 second was 47 percent which exceeded the requirement by 32 percent. As test conditions were changed to introduce fancase failure airflow and then add the top deck rupture, time duration of agent concentration above 15 percent and maximum concentration percent for 0.5 second was: 3.10 seconds at 26 percent, and 3.30 seconds at 16 percent, respectively. The last condition is the most severe, and the values were still above minimum specification requirements.

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[8] Federal Aviation Authority. *Aircraft Installation and Operation of an Extinguishing Agent Concentration Recorder.* (FAA Technical Report No. 403, publication UNCLASSIFIED.)

TABLE 8. Agent Concentration Test Summary.

Item	Test condition
Condition	Nacelle under normal flight conditions
Air sources	Nacelle cooling: 0.5 lb/sec (0.23 kg/sec)
Sampler	Right engine nacelle
Extinguisher	Kidde P/N 891490, Agent CF_3BR Weight 10.5 lb (4.76 kg), N_2 pressure 700 psi (49.2 kg/cm^2)
Agent concentration	Maintained above 15% between 1.85 and 7.20 seconds. Maximum concentration for 0.5 sec is 47% between 2.95 and 3.45 seconds.
Condition	Nacelle with fancase rupture
Air sources	Nacelle cooling: 0.5 lb/sec (0.23 kg/sec) Fancase air: 1.27 lb/sec (0.58 kg/sec)
Sampler	Right engine nacelle
Extinguisher	Kidde P/N 891490, Agent CF_3BR Weight 10.5 lb (4.76 kg), N_2 pressure 700 psi (49.2 kg/cm^2)
Agent concentration	Maintained above 15% between 1.90 and 5.00 seconds. Maximum concentration for 0.5 sec is 26% between 2.65 and 3.15 seconds.
Condition	Nacelle with fancase rupture plus nacelle top deck rupture.
Air sources	Nacelle cooling: 0.5 lb/sec (0.23 kg/sec) Fancase air: 1.27 lb/sec (0.58 kg/sec) DASH airflow: 250 knots
Extingusher	Kidde P/N 891490, agent CF_3BR Weight 10.5 lb (4.76 kg), N_2 pressure 700 psi (49.2 kg/cm^2)
Agent concentration	Maintained above 15% between 1.70 and 5.00 seconds. Maximum concentration for 0.5 sec is 16% between 1.80 and 2.30 seconds.

TABLE 8. (Contd.)

Item	Test condition
Condition	Acessory bay under normal flight conditions
Air sources	DASH airflow: 250 knots
Sampler	Overwing fairing accessory bay
Extinguisher	H.T.L. 86 cu. inch bottle, Agent CF_3BR Weight 2.75 lb (1.25 kg), N_2 pressure 600 psi (42.2 kg/cm3)
Agent concentration	Maintained above 15% between 1.70 and 5.10 seconds. Maximum concentration for 0.5 sec is 23% between 3.75 and 4.25 seconds

In the overwing fairing accessory bay, the problem was to get agent into the rear area of the volume. With the discharge nozzle up forward, the aircraft equipment installed in the volume inhibited flow of the agent. The final nozzle design directed the agent forward to deflect off the wing box structure, diffusing around the forward part of the volume, and upward and aft to deflect off the underside of the overwing fairing into the rear portion of the volume. The discharge nozzle design is shown in Figures 30 and 31. This configuration produced a concentration above 15 percent which was maintained for 3.40 seconds, exceeding the time requirement by 2.90 seconds, and a maximum concentration of 23 percent for 0.5 second which exceeded the requirement by 8 percent under a normal flight condition at 250 knots.

When a hole is introduced in the nacelle top deck, agent and airflow will enter the accessory bay. One test was conducted to determine how much nacelle agent entered and how it dispersed. The test conditions are shown in Table 9. In this test, measurements were made in four locations in the nacelle plus the eight locations in the overwing fairing. The object was to evaluate the loss of extinguishant from the nacelle and to determine if the overwing fairing concentration was significant.

Results of this test show that nacelle concentrations were still more than adequate while overwing fairing concentrations were on the order of 4 percent in a small volume. Very little agent went forward of the failure hole (see Figure 24 for location) and none was detected at sensor 6 or 7. This test did not utilize a bottle in the accessory bay as would be used in later fire tests. It did show the additional agent flow to be expected from the nacelle into the accessory bay.

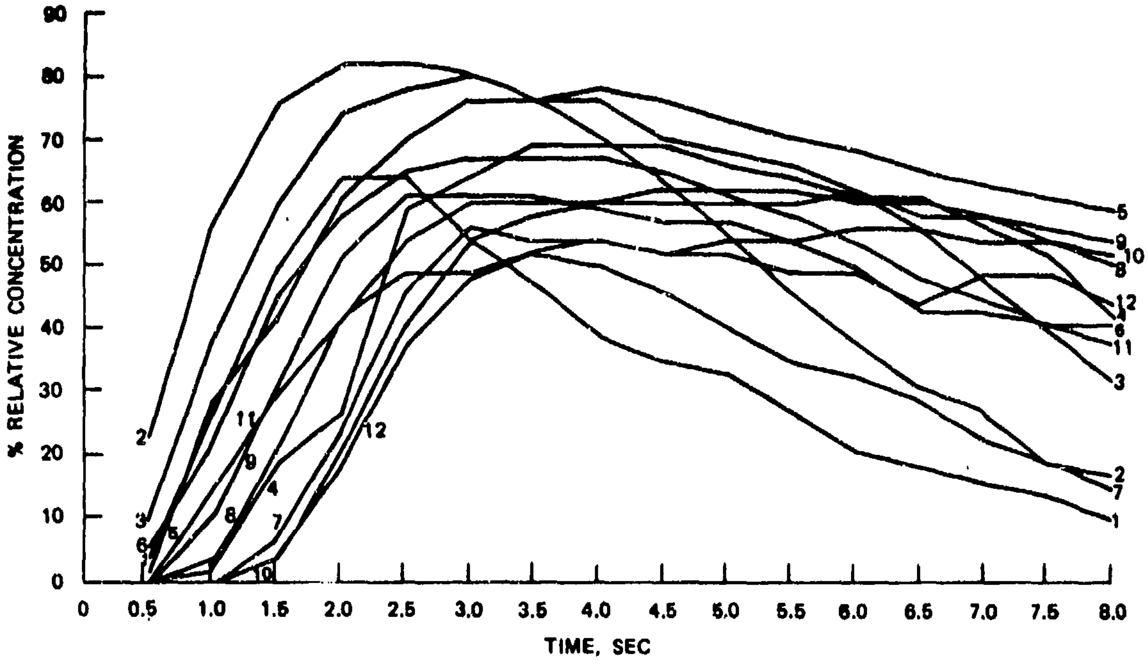


FIGURE 26. Agent Concentration, Nacelle Under Normal Flight Conditions.

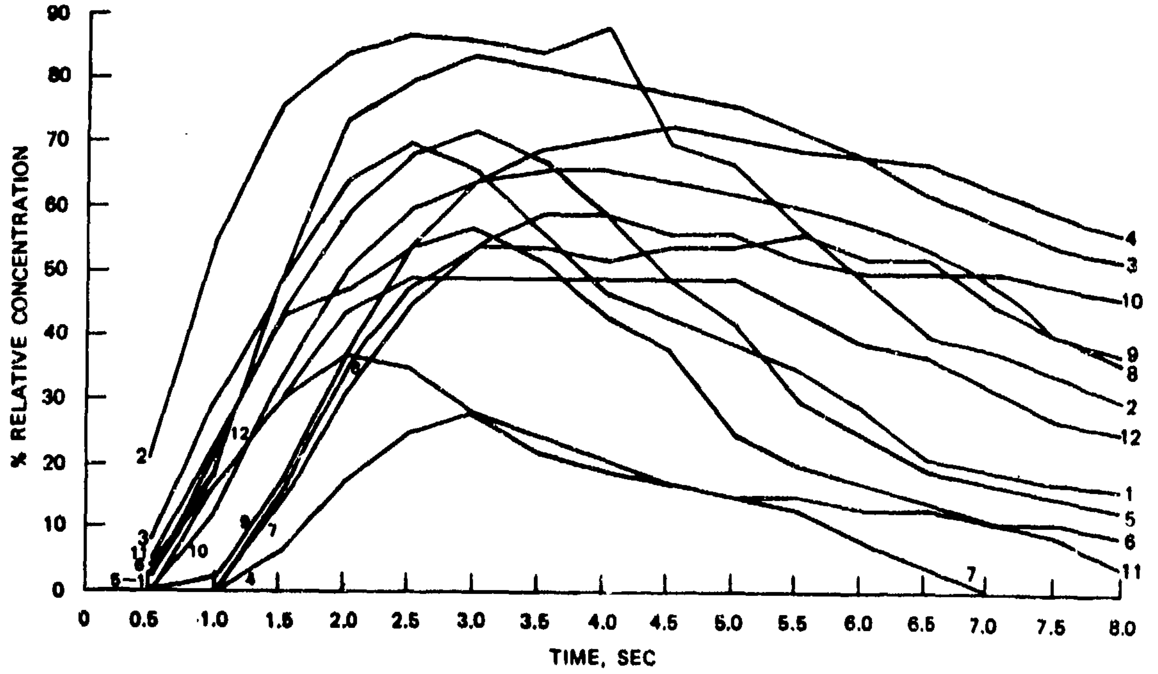


FIGURE 27. Agent Concentration, Nacelle With Fancase Rupture.

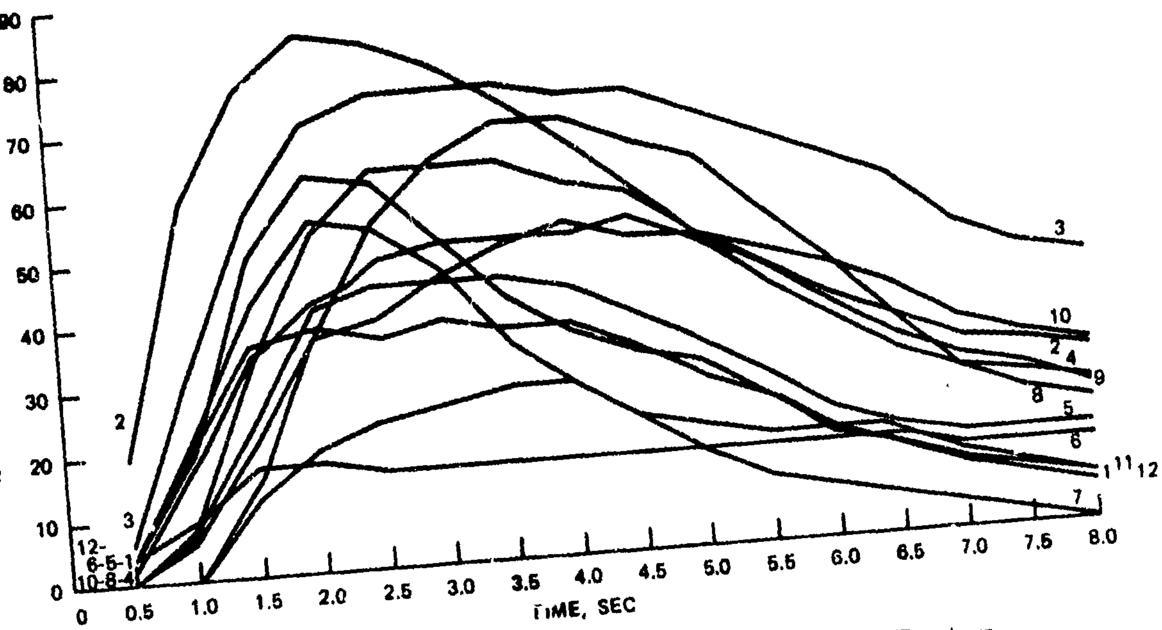


FIGURE 28. Agent Concentration, Nacelle With Fancase Rupture Plus Nacelle Top Deck Rupture.

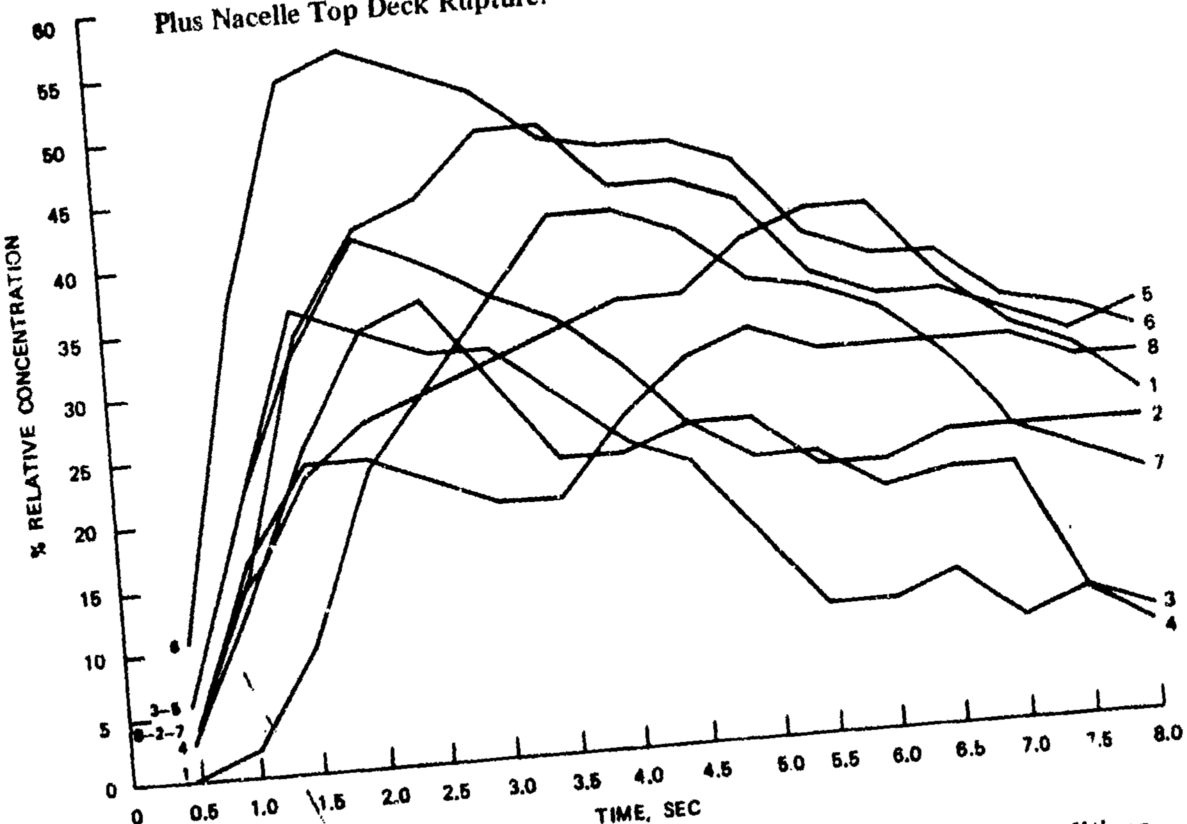


FIGURE 29. Agent Concentration, Accessory Bay Under Normal Flight Conditions.

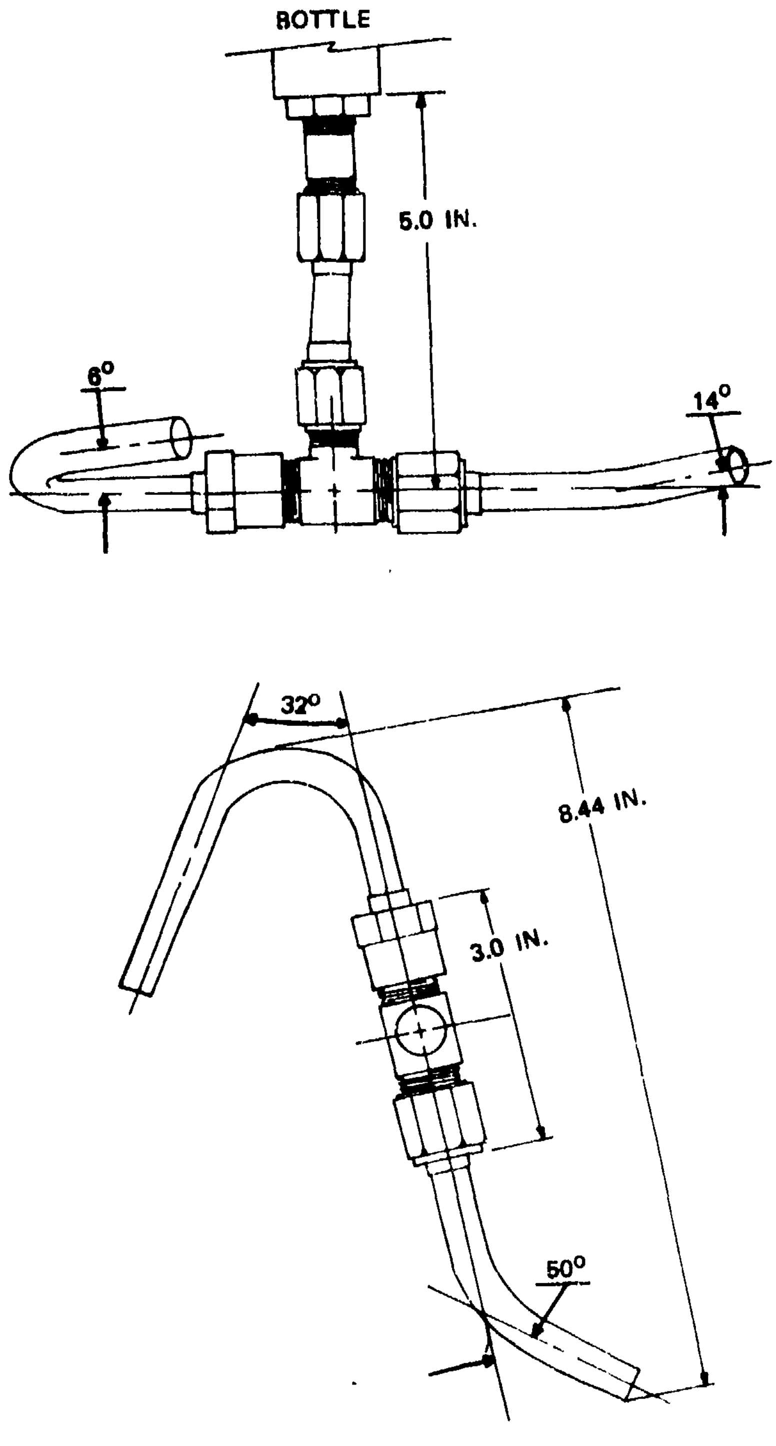


FIGURE 30. Extinguisher Bottle Discharge Nozzle Configuration.

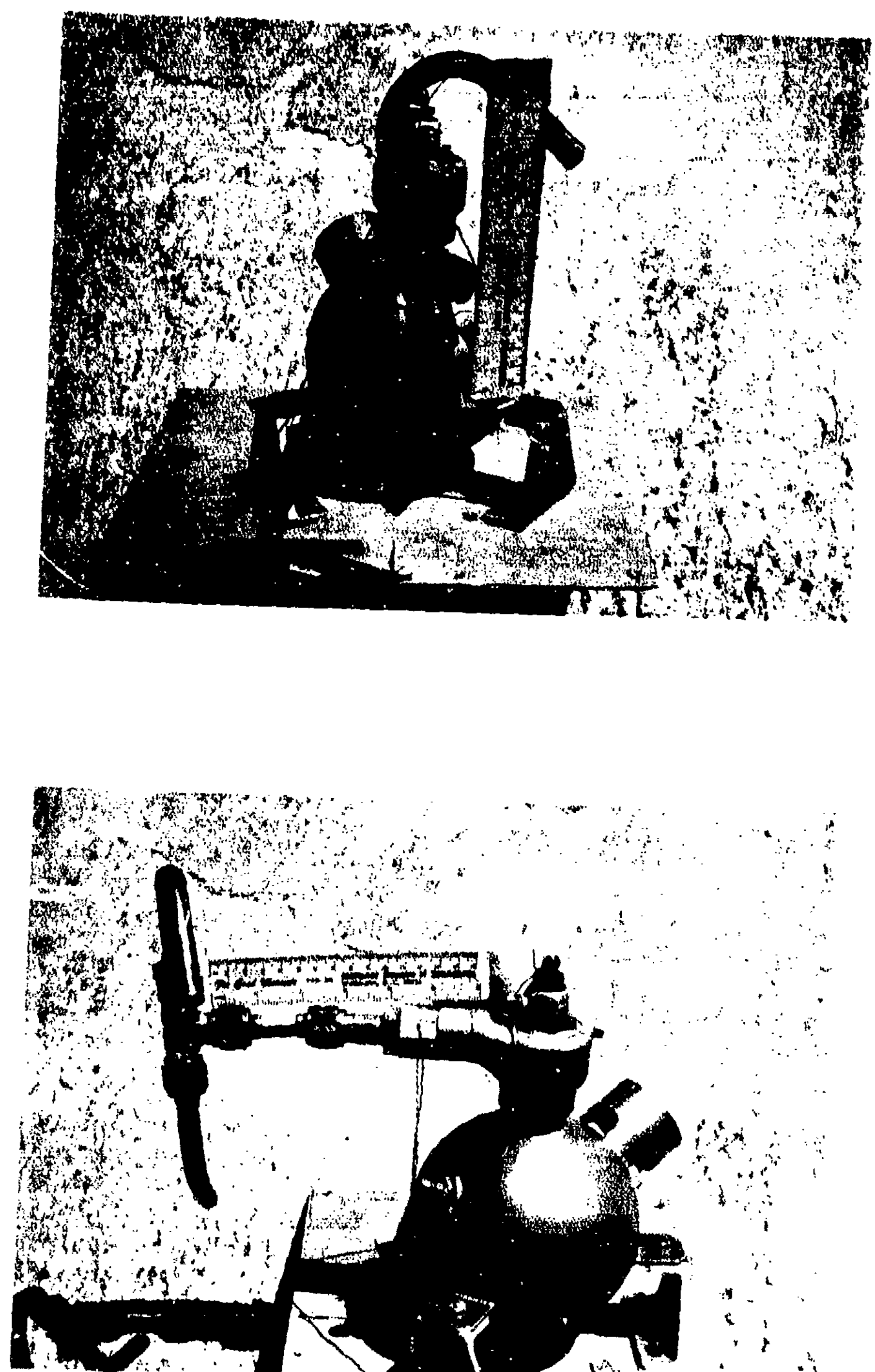


FIGURE 31. Extinguisher Bottle With Mounted Discharge Nozzle.

TABLE 9. Nacelle - Accessory Bay Agent Concentration Test Conditions.

Item	Test condition
Condition:	Accessory bay with top deck rupture
Air sources:	Nacelle cooling: 0.5 lb/sec (0.23 kg/sec) Fancase air: 1.27 lb/sec (0.58 kg/sec) DASH airflow: 250 knots
Extinguisher:	H.T.L. 378 cu. inch bottle, Agent CF_3BR Weight 13 lb (5.9 kg), N_2 pressure 700 psi (49.2 kg/cm2)

FULL-SCALE FIRE TESTS

The purpose of the full-scale fire tests was to validate the effectiveness of the fire protection measures developed and installed in the test article and proposed for incorporation in the F-14A aircraft. Prior subscale tests were conducted to evaluate the effectiveness of fire barrier materials and the performance of engine failure simulation/fuel ignition techniques to be employed in these full-scale tests. Optimum location of on-board extinguisher system discharge nozzles was achieved during extinguishing agent concentration tests.

A series of three fire tests was conducted to evaluate the system effectiveness. The first test was a simulation of a fan blade ruptured engine case with subsequent rupture of the engine fuel line and nacelle top deck. The simulated failure would occur at the selected (simulated) flight speed of 250 knots with a fuel fire initiating in the nacelle. This would be allowed to progress through the nacelle and overwing fairing accessory bay until detected by the fire warning system. At this point the engine fuel line manual shutoff valve is actuated and the fire extinguishing agent deployed.

The second test was a repeat of the basic conditions simulated in the first test. However, in this case the objective was to further test the center trough fire barrier. Therefore, the extent of the fire in the nacelle was limited to fire and fuel coming from the nacelle to the accessory bay through the nacelle top deck rupture which would be caused by a liberated fan blade. Fire in the accessory bay was allowed to propagate in an unrestricted manner until test termination.

The third test was to determine the effectiveness of heat ablative material used to protect the frames supporting the aft nacelle flight control rod brackets. This effort consisted of several small burn tests of protected and unprotected frames in the aft nacelle.

Fire Test No. 1

In preparation for this test the DASH airflow system and auxiliary airflows were started, the test article engine started, and the test pad foamed with a layer of fire retardant. After the engine reached normal thermal equilibrium the throttle was advanced to a 75 percent power setting. At test time - 1.5 seconds (T - 1.5 sec) the fuel ignitor flare was ignited. At T = 0, fuel failure was initiated by firing the fuel line rupture charge, resulting in ignition of the released fuel. Subsequent events are summarized in Table 10.

TABLE 10. Summary of Events, Fire Test No. 1.

Time (T)	Event, actual
T - 1.5 sec	3-second flare ignition
T = 0	Fuel line rupture, fire initiation
T + 1.7 sec	Began retarding engine throttle
T + 7.5 sec	Nacelle fire warning light illuminated
T + 9.5 sec	Fuel shutoff valve actuated to OFF
T + 10.0 sec	On-board Halon 1301 deployed
T + 11.0 sec	Thermocouple temperature decreasing
T + 18.0 sec	Fire relight from external fire
T + 19.0 sec	Secondary explosion; beginning of 20-minute endurance test
T = 20 minutes	Fire extinguished

Approximately 1.7 seconds after fuel ignition, retardation of the engine throttle was initiated with the throttle reaching full off position at approximately T + 3.4 seconds after fuel line rupture. Turbine inlet temperature began to fall in response to throttle movement 2.2 seconds after zero time. At no time during this period did engine response indicate that it had run out of fuel. Supposition is that fuel remaining in the engine lines, pump, fuel control, and the section of feed line downstream of the line rupture was sufficient to keep the engine running until the throttle was fully retarded.

At T + 7.5 seconds the nacelle fire warning light came on. The test control operator subsequently actuated the fuel shutoff valve at T + 9.5 seconds, followed by actuation of the on-board extinguishers at T + 10 seconds. By T + 11.0 seconds the thermocouple sensors began

to show decreasing temperatures in the engine nacelle and accessory bay. This corresponded to absence of visible flame from the nacelle as shown by the films taken during the test. Both the temperature data and the films evidenced that the on-board Halon 1301 system had extinguished the fire.

During the initial burn period, fuel leaking through the nacelle doors spilled onto the test pad and ignited. The foam on the test pad reduced flame size but did not extinguish the fire. As shown by the films, the nacelle fire appeared to be extinguished when, at T + 18.0 seconds, the external pad fire caused a relight which was followed by a small secondary explosion from fuel vapors inside the nacelle. Subsequent analysis showed fuel was still leaking from the fuel line wound. With the airflow still operating the nacelle was pressurized, causing fuel vapors to be forced from the engine bay door edges. It was these vapors which were reignited with subsequent propagation of the flame to the nacelle interior. This was the start of a second fire test which was an approximate 20-minute fire endurance test. In actual flight mode this fire relight would not occur since it would have been blown off by the airstream.

Carbon dioxide extinguisher lines had been routed throughout the aircraft as backup firefighting equipment as part of the test preparations. This system was utilized and kept fire from spreading through the aircraft but it was unable to extinguish the fire burning at the fuel leaking from the original fuel line rupture.

The thermocouple temperature data plots of Figures 32 and 33 show that the on-board fire extinguisher system extinguished the nacelle and overwing fairing fires after the initial 10-second burn. The data plot shows that maximum temperatures were reached approximately 10 seconds after fuel ignition. Correlating the curves with thermocouple position in the test article shows the fire started in the forward nacelle and spread aft in the nacelle, following the airflow circulation from front to rear. At the same time the fuel which was blown through the nacelle top deck rupture by the fancase rupture airflow ignited and burned in the accessory bay. Infrared camera pictures show heat being forced upward along the buttline 10 fire barrier and exiting the overwing fairing as well as being forced out along the outboard (wing pivot) area of the fairing. Temperatures inside the center trough remained approximately ambient. Fire did not cross the center trough fire barrier. The Halon 1301 extinguished the fire in the accessory bay which later reignited (from the test pad residual fire) and was subsequently extinguished by the facility CO_2 system. Concurrent with deployment of the Halon 1301, the temperatures in the nacelle also showed a rapid decrease in temperature indicating the fire had been extinguished.

Analysis of the temperature history in the nacelle shows that the fire started at the failure point then followed the airflow pattern down and aft. It appears that this flow pattern makes just short of one loop

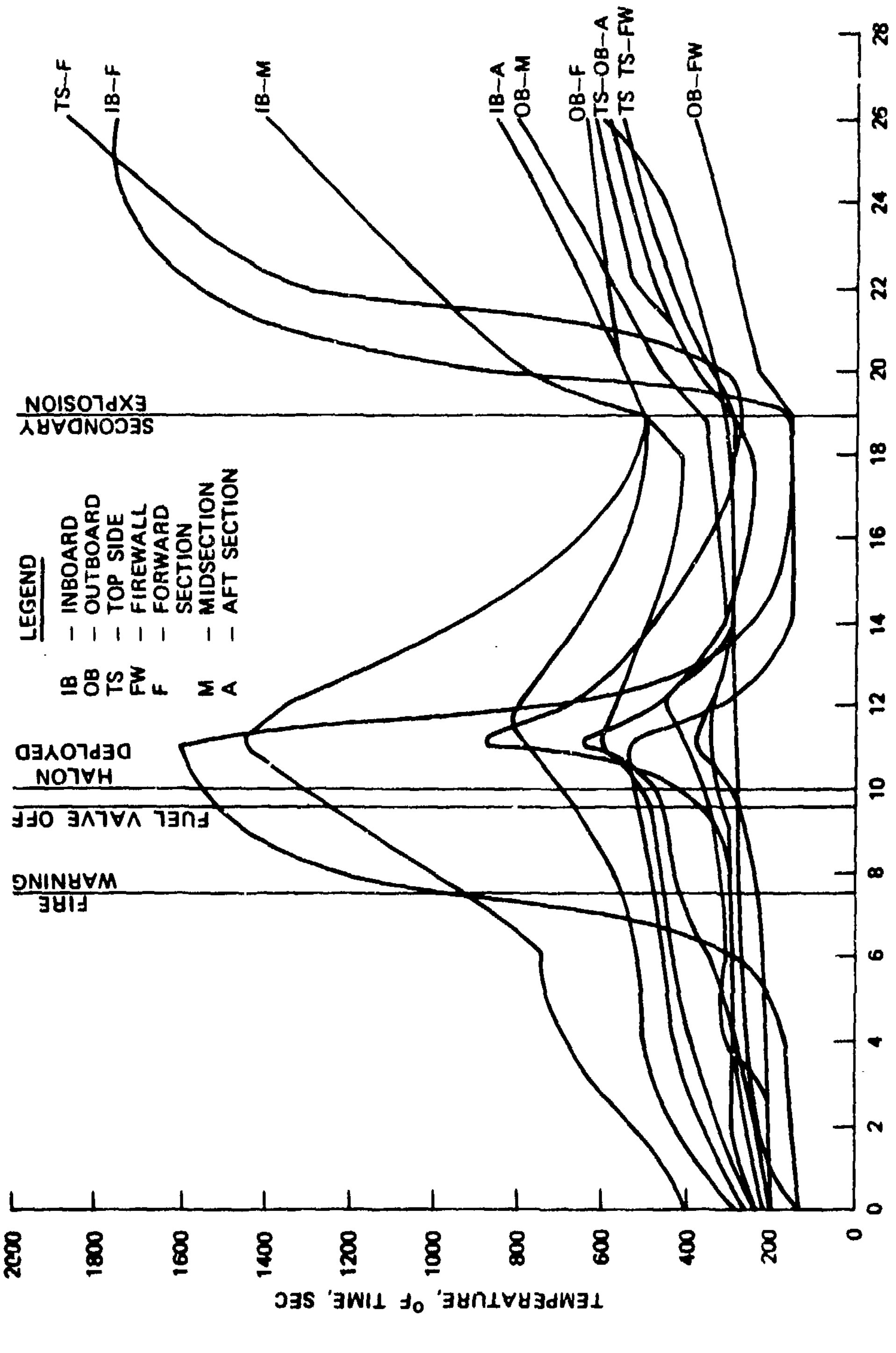


FIGURE 32. Nacelle Temperature Data Plot, Full-Scale Fire Test.

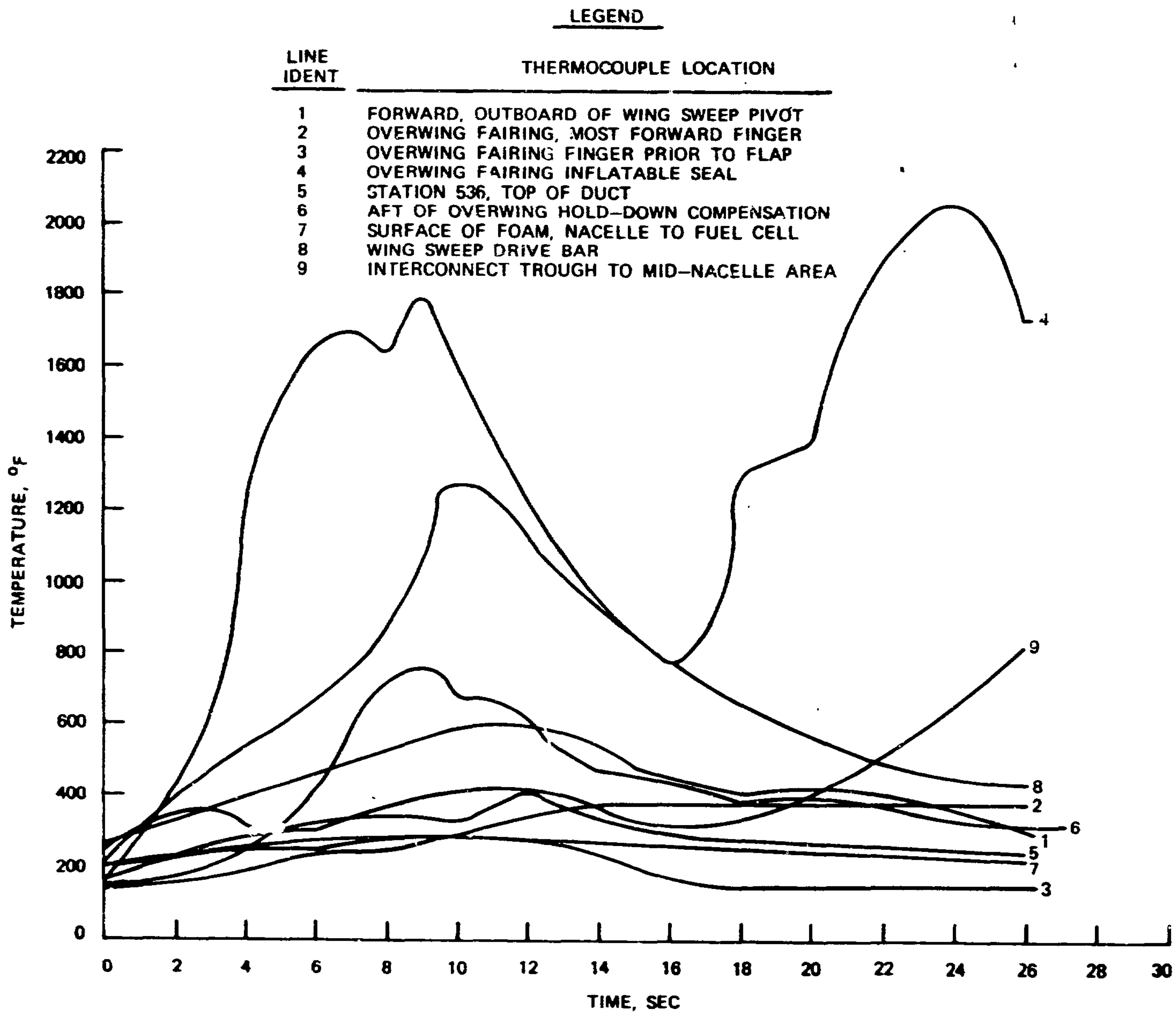


FIGURE 33. Accessory Bay Temperature Data Plot, Full-Scale Fire Test.

axially about the engine before exiting via the aft nacelle vent. The volume of air was sufficient to cause combustion temperatures to be maximum about 6 feet (2 meters) aft of the failure opening. From this point, aft temperatures drop until reaching exit temperature which, in the initial part of the test, never exceeded 400°F (204°C).

Thermocouples in the volume under the overwing fairing show heat penetration first at the aft end then quickly forward from the failure hole past the wing actuator bar and outboard to the wing pivot and the inflatable seal. Temperature of the upper surface of the wing sweep actuator bar reaches 1300°F (700°C) before the extinguishant is deployed. This temperature never increases again during the test. Temperature just forward of the wing pivot reaches just over 600°F (315°C) before the extinguishant released. The sloping deck temperature (top surface of fuel cell 5) never increased much over ambient.

In looking further at the temperature plots of nacelle and overwing fairing temperature data it should be explained that prior to the fire some thermocouples show above ambient temperatures. This was due to the aircraft being at thermal equilibrium with the engine operating, therefore some higher than ambient temperatures should be expected. Further, the rapid decrease in thermocouple temperature was aided by the inflight airflow simulation. Airflow of 1.77 lb/sec (0.80 kg/sec) at approximately 50°F (10°C) was continuously introduced to the nacelle, simulating inflight ram air.

Data obtained from fuel flow venturis installed up and downstream from the line rupture indicated that in the 8.7 seconds between fuel line rupture and the time the upstream venturi ΔP reached near zero equilibrium value, approximately 110 pounds (50 kg) of fuel were dumped into the engine nacelle. This corresponds to approximately 16 gallons (60 liters) of fuel, thus accounting for the excess of fuel which could not be consumed and subsequently poured into the bottom of the nacelle. Consequently, this fuel leaked onto the test pad where it ignited.

The fuel line pressure transducer and downstream venturi ΔP pressure transducer both registered perturbations at approximately 10.1 seconds after fuel line rupture. Fuel line pressure perturbations were again measured at T + 18.7 seconds. These pertubations continued, evidencing failure of the venturi and connecting lines from explosions and fire effects. The downstream venturi ΔP showed signs of sensor connector failure at about T + 20.4 seconds.

The fancase rupture air source and nacelle cooling air pressures both showed perturbations at T = 0. It is calculated that fancase airflow momentarily fluctuated from approximately 1.27 lb/sec (0.53 kg/sec) to 1.09 lb/sec (0.50 kg/sec) and returned to its original value. Ignition of the fuel had discernible but negligible effect on the nacelle cooling airflow rate. A perturbation showed at T + 18.9 seconds; this corresponds to time of the secondary explosion caused by the fire relight.

Smoke paths visible after the test showed that air from the mid nacelle area had crossed through a cable-way into the central trough, then forward to the barrier opposite the accessory bay. This implies that nacelle pressure is higher than trough pressure which in turn is higher than pressure under the overwing fairing. This was expected due to the DASH and nacelle airflow sources and indicated a reduced pressure in the accessory bay. Pressure differences between nacelle and trough did not exceed the sensor channel noise, thus this pressure difference must be low. Photographic coverage of the overwing fairing shows smoke coming from the aft end of the fairing past the inflatable seal shortly after the fire starts. In a few seconds smoke is also seen penetrating through the wing sweep finger seals, and a few seconds after that, smoke issues from the forward seal of the overwing fairing.

Titanium had been added to the forward outboard wall of the engine nacelle to block fire from penetrating in this area and entering the wheel well area (refer to Figure 4). This firewall had temperatures of up to 800°F (427°C) during the initial test, then saw temperatures to 1800°F (982°C) after the relight. Fire did not penetrate from the nacelle through this area although some external damage was sustained because of the test pad fire.

A titanium sheet had also been added to the upper side of the nacelle top deck as shown in Figure 4. Temperatures in the nacelle under this sheet reached 1600°F (871°C) in the initial test, then dropped to less than 200°F (93°C) until relight. Post-test inspection showed that most of the aluminum frames and skin in this area were gone as shown in Figure 34. The titanium was intact. The nacelle inboard heat shield shown in Figure 35 was intact with only several cracks evident.

The high temperature at the beginning of the test would cause some structural damage, but the later endurance test apparently did most of the structural melting. Without the titanium firewall, more of the overwing fairing area would have suffered extensive damage. The fire burned through the cover and most of the support structure for the overwing fairing compensator as shown in Figure 34. This exposed the fire to the hardware access holes drilled in the firewall. These holes then became additional fire paths.

The fire being blown through the top deck simulated blade rupture hole was being partially deflected by the overwing fairing onto the top surface of fuel cell No. 6 (which was empty) and the diagonal longeron. During the second burn of 20 minutes this top deck partially melted away and did severe damage to the diagonal longeron as shown in Figure 36. Viewing the cell from the top of the aircraft, the fire penetrated the deck of the cell and impinged upon the 3-inch (76-mm) fuel line after penetrating the cell bladder. Approximately 12 inches (304 mm) of the line had the top half melted away. This also allowed fuel to drain into the cell. Fire on the test pad penetrated into the bottom of fuel cell No. 6 thereby providing an additional path for burning fuel from the line to feed the test pad fire.

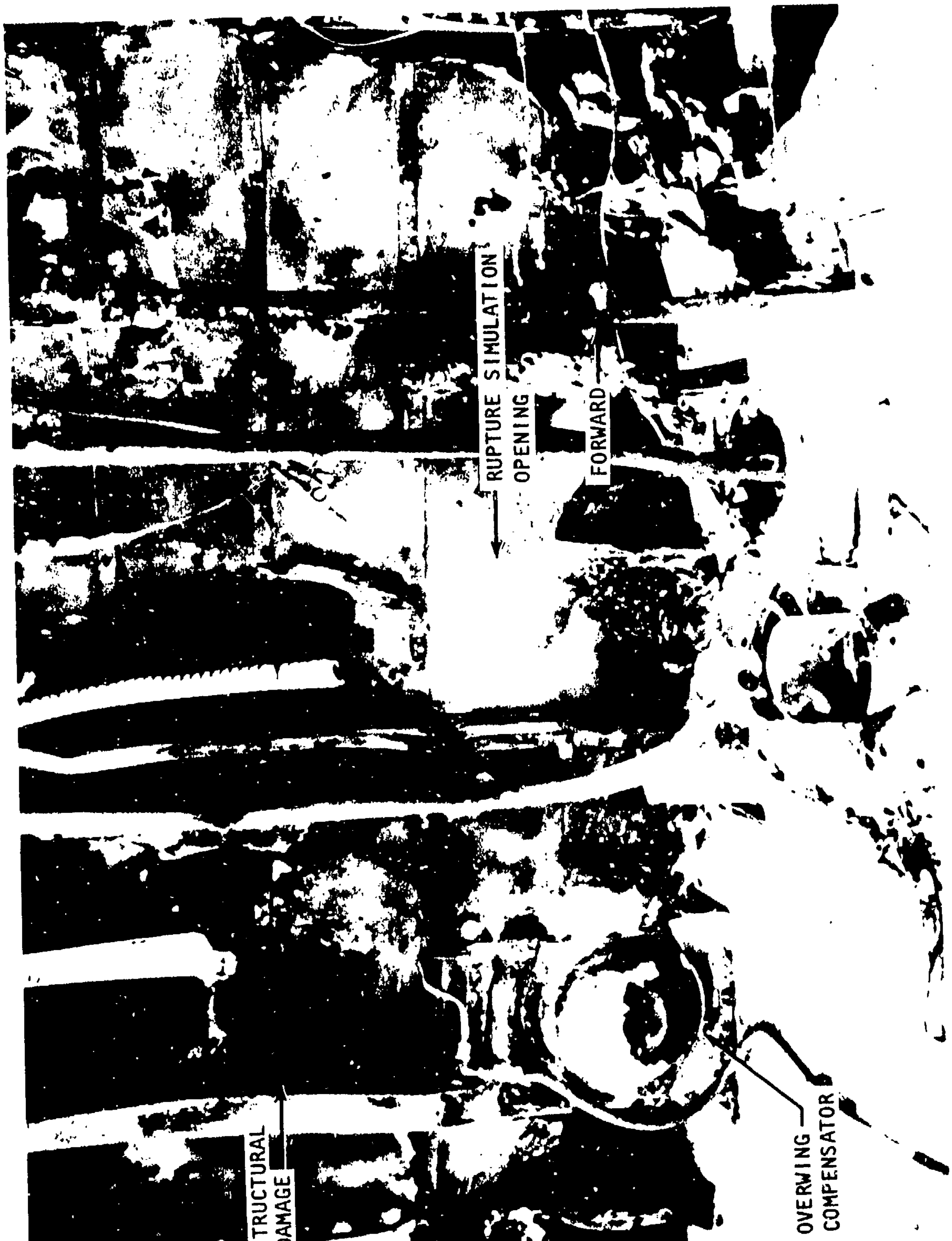


FIGURE 34. Post-Test View of Titanium Barrier and Adjoining Structure.

FIGURE 35. Titanium Heat Shield After Fire Test No. 1.

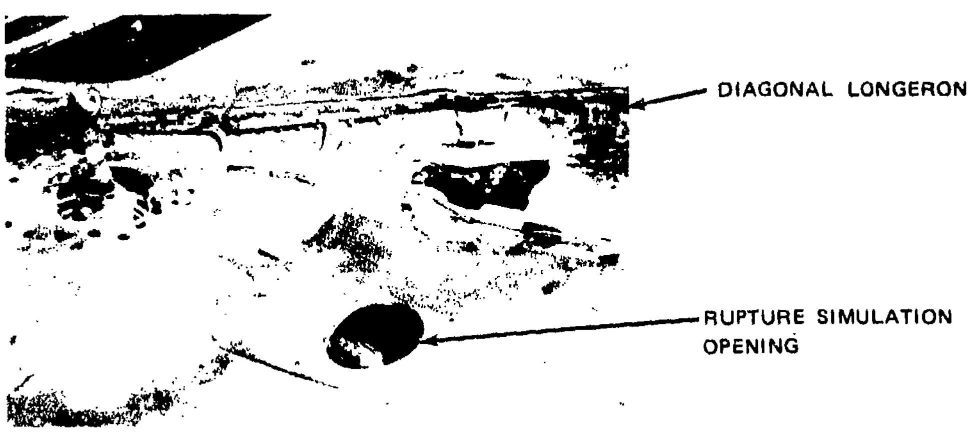


FIGURE 36. Fire-Damaged Diagonal Longeron.

Figure 37 shows a picture taken of the underside of the overwing fairing. The hole position where burnthrough occurred is above the fan blade rupture simulation hole. It appears that fire pushed out of the nacelle through the failure simulation hole was deflected from the overwing fairing long enough to penetrate into fuel cell No. 6, but eventually the fairing melted through and airflow escaped out of the aircraft.

FIGURE 37. Fire Damage to Overwing Fairing.

Table 11 lists the temperature history of fuel cell No. 6 as sensed by a centrally located thermocouple. From this history it appears that initial burnthrough occurred at 4 or 5 minutes. The fuel line that feeds the right engine nacelle passes through fuel cell No. 6. The line enters at the forward wall and immediately enters the fuel shutoff valve. After exiting the valve, the line travels another 14 inches (35.6 cm) aft and makes a right-angle bend into the nacelle. The burnthrough occurred right over the bend in the tube after the 14 inch (35.6 cm) section. Since the temperature decreases from the T + 6 to T + 8.5 minute period, it may be that the internal combustibles in the cell were actually burning out. At 8.5 minutes the fuel line apparently burned through dumping the remainder of the fuel in the tank into and through the cell. Prior to this time the fuel was falling into the nacelle and to the test pad.

TABLE 11. Temperature History, Fire Test No. 2.

Time, minutes	Condition
T + 1.5	Temperature slowly increasing
T + 4.0	Rapid increase in temperature
T + 5.0	200°F (94°C) temperature step
T + 6.0	Maximum temperature
T + 6.5	Temperature decreasing
T + 8.5	Rapid temperature increase
T + 10.0	Temperature decreasing
T + 11.0	Rapid decrease to approximate ambient temperature

The flight controls in the center trough were not damaged by the fire as is evident in the post-test photo of Figure 38. These controls were protected by the silicone-ablative-on-steel fire barrier. As shown in Figure 39, the barrier remained intact throughout the fire.

Following the test, it was determined that the fuel shutoff valve had malfunctioned. It is a butterfly valve and, when actuated to the off position, the butterfly continued past the closed position as shown in Figure 40. Fuel remaining in the wing box and sump tank down to the fuel line level drained out during and after the initial burn phase. During post-test investigations it was determined that approximately 160 gallons (608 liters) of fuel were lost during test duration, draining out by gravity through the partially open fuel valve.

The fuel valve was removed from the nacelle and shipped to GAC for analysis of the failure. Grumman found the mechanical stops that normally inhibit the travel of the valve past shutoff had been severed. This valve failure was the direct cause of the long burn duration resulting from continued fuel leakage from the aircraft. This failure mode had been reported as occurring before, and an engineering change (ECP 854, see footnote 3) was proposed to modify the valve. ECP 854 proposed to change the shaft materials to steel and strengthen the stops, plus several other small changes all designed to prevent the valve failure experienced in this test.

FIGURE 38. View of Center Trough After Fire Test No. 1.

FIGURE 39. Silicone - Ablative Coated Fire Barrier After Fire Test No. 1.

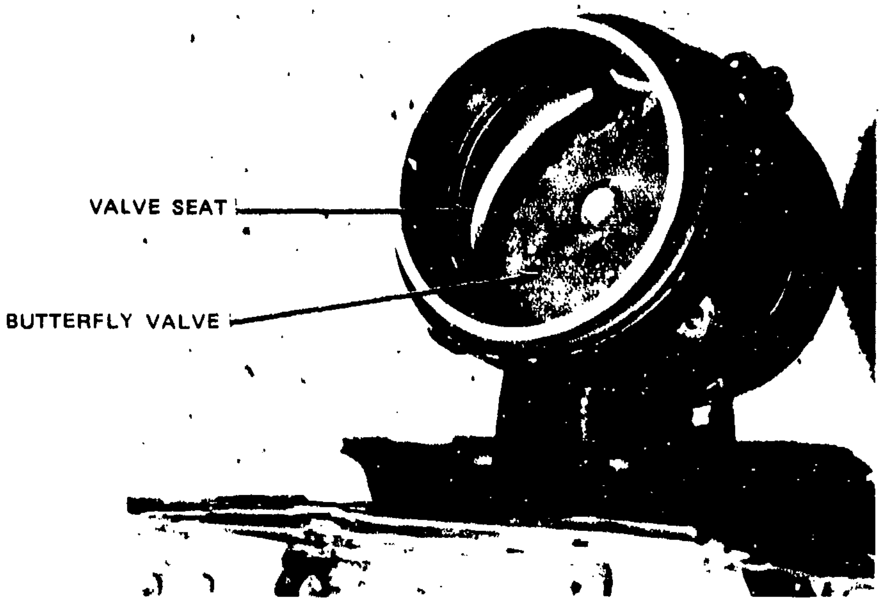


FIGURE 40. Fuel Shutoff Valve Showing Butterfly Overtravel Malfunction.

Fire Test No. 2

This fire test was conducted for further evaluation of the fire barrier in the overwing fairing accessory bay. The object of this test was to subject the center trough fire barrier to fuel fire conditions until either the trough temperature exceeded 370°C (700°F) or significant failure of the overwing fairing occurred. In performing the test, a short-burn-time metalized flare was used to ignite fuel in the accessory bay. The flare was mounted in a small steel cup so only minor damage would be sustained from the flare. Fuel was introduced into the accessory bay through a 0.25-inch (6.4 mm)-diameter line. A second line sprayed fuel into the fancase failure airstream as it entered the overwing fairing. Total fuel flow from the two lines was 1.4 gallons (5.32 liters) per minute.

Prior to the test, additional sealing of the fire barrier was accomplished using red RTV compound. The purpose was to seal the ends and bottom of the barrier liquid-tight. (It was found that raw fuel could puddle in the trough because the trough is the lowest point in the accessory bay area.) It was noted that a large bubble had formed under the ablative on top of fuel cell No. 5 where hydraulic fluid had dripped on it. RTV compound was injected under the loose area to assure it would stay in place during the test.

The test was initiated by igniting the 3-second burn-time flare and injecting fuel through the two fuel lines described. The test proceeded normally until at T + 55 seconds the overwing fairing fire extinguishing bottle nozzle failed from over-pressurization, releasing all the Halon 1301. The extinguishing agent did not discharge through the nozzles but through the neck of the bottle. By not being distributed properly, the extinguishing agent did not completely extinguish the fire. As criteria for test termination specified that the test should be terminated if a hole 12 inches square (7742 mm^2) or greater opened in the overwing fairing, the test was terminated at T + 60 seconds because a large opening occurred.

Failure of the fire extinguisher bottle was at the bottle throat. Rather than fail such that extinguishant came out the nozzle, the entire nozzle and valve assembly was blown off the bottle. This created poor extinguishant dispersion and prevented fire extinguishment by the discharge. Those thermocouples at the forward area of the accessory bay showed a decrease at T + 52 seconds when the bottle diucharged but elsewhere in the volume no significant change occurred until the test was terminated.

This particular bottle design was not the one proposed for F-14A incorporation, but was only used in this test. The final design should be assessed to determine that the same failure mode will not occur when the bottle is exposed to fire heating.

On the television monitor viewing the fairing, a triangle of metal was seen folding up at what appeared to be the area above the nacelle fan blade failure hole in the overwing fairing deck (see Figure 41). It did not melt away, but rather folded back like a soft piece of paper. Post-test views of the underside of the fairing showed crystalline fracture of the aluminum ribs and cover plate. Apparently heat from the fire of the nacelle failure had softened the structure and a slight load was applied by the airstream. When the extinguisher bottle blew its charge, the added pressure caused the aluminum to fail.

To reiterate, the intent of this test was to tax the fire barrier and ascertain when heat penetrated into the trough. In the test as conducted, flame did impinge on the barrier as shown by subsequent checking of the ablative barrier (see Figure 42). The test duration was too short for significant heat penetration of the barrier. Temperature of the air 3 inches (76 mm) into the trough from the barrier reached approximately 100°F (38°C) while all control rod temperatures stayed at the ambient of 70°F (21°C). Temperatures at the front face of the barrier ranged from approximately 750 to 1800°F (400 to 980°C).

In post-test assessment of test damage, the inboard rear edge of the overwing fairing damage shown in Figure 42 was found to have peeled back approximately 2 square feet (0.185 $meter^2$) of the top area. The

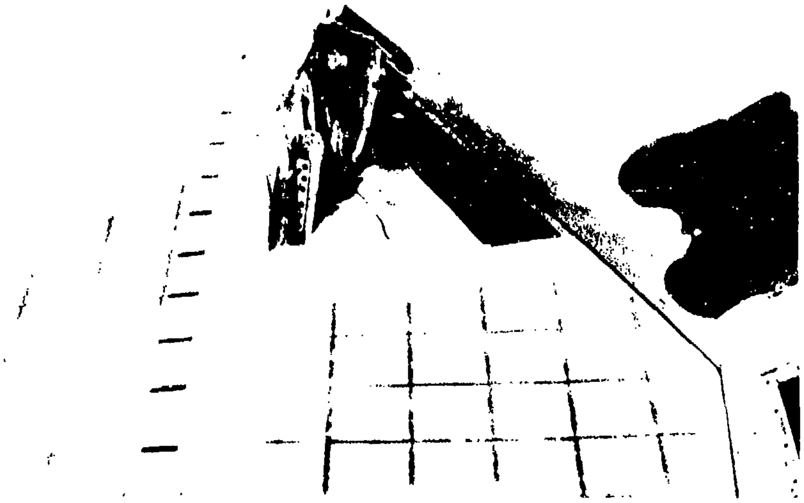


FIGURE 41. Peeled Area of Overwing Fairing.

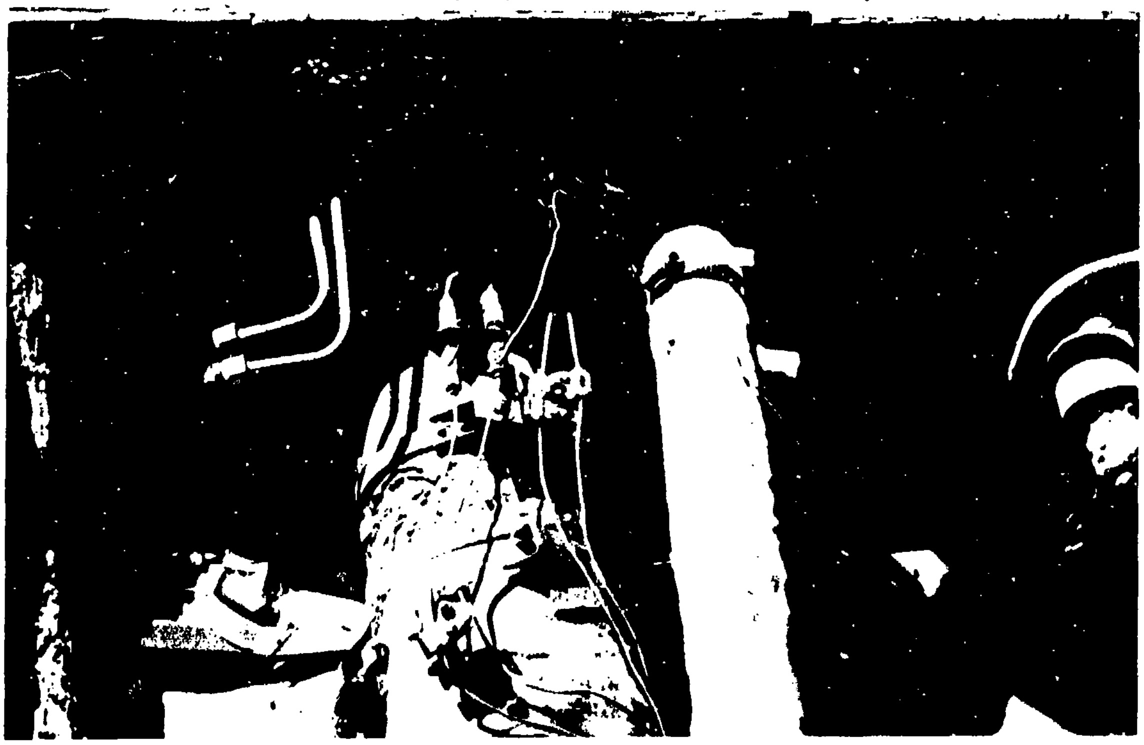
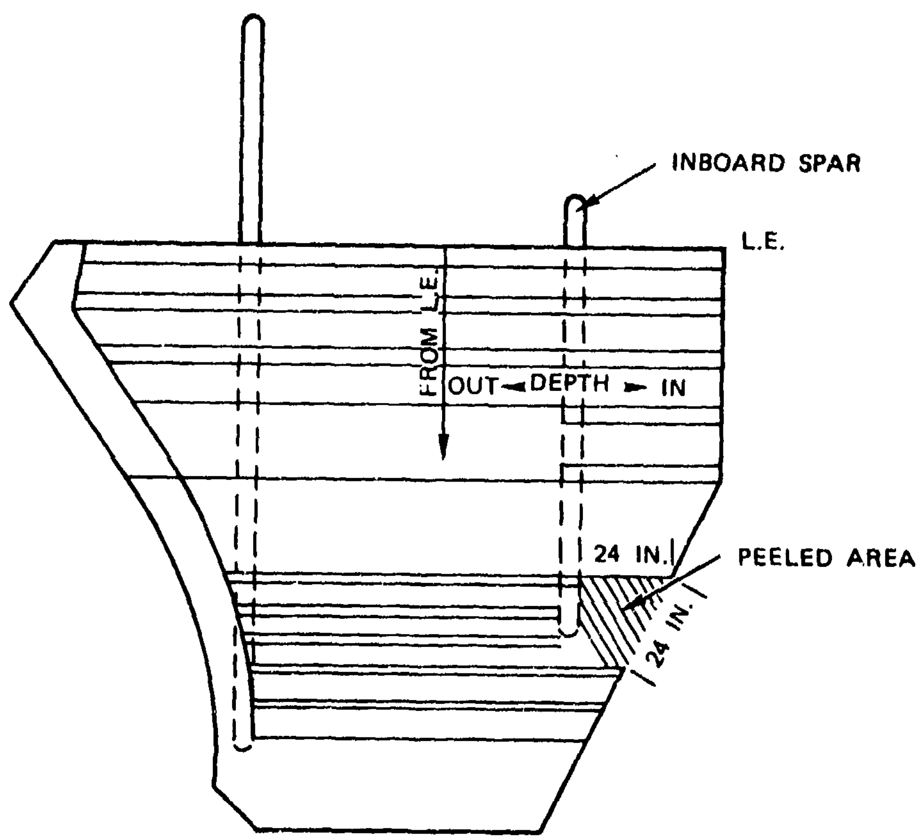


FIGURE 42. Silicone - Ablative Coated Fire Barrier After Fire Test No. 2.

leading edge of the overwing fairing was charred and the center section showed surface buckling and heating effects. Damage assessment measurements for the underside of the overwing fairing are given in Figure 43. The measurements shown are from the leading edge to the center of the damaged area.



DAMAGE ASSESSMENT MEASUREMENTS

ITEM	DISTANCE FROM L.E.	DEPTH		WIDTH
BURNED RIB	35 IN. (88.9 cm)	6 IN. (15.2 cm)	IN	13 IN. (33.1 cm)
BURNED RIB	42 IN. (106.7 cm)	4 IN. (10.2 cm)	IN	20 IN. (50.8 cm)
BURNED RIB	59 IN. (149.9 cm)	9 IN. (22.9 cm)	OUT	10 IN. (25.4 cm)
CRACKED RIB	50 IN. (127.0 cm)	11 IN. (27.9 cm)	–	–
BURNED RIB	66 IN. (167.7 cm)	9 IN. (22.9 cm)	OUT	6 IN. (15.2 cm)

FIGURE 43. Damage Measurements on Underside of Overwing Fairing.

The entire section from 50 inches (1.3 meter) to 66 inches (1.7 meter) and the inboard spar to the fairing edge had burned ribs, an area of missing inner skin, and the entire section peeled with skin separated along the seam at the 50 inch (1.3 meter) location. Rivets pulled through the outer skin edge. Several rectangular plastic pads at the 84 inch (2.1 meter) to 94 inch (2.4 meter) location were missing.

In the accessory bay the rigid foam between fuel cell No. 5 and the inlet duct showed minor surface damage. The lower fuel vent line had ablative missing in a 10-inch (254-cm)-long section. It appeared that the ablative loss may have been caused by the blown extinguisher bottle. Heat damage was not apparent in this section of line.

There were two partially burned plastic pads located next to a backup fire-fighting CO_2 nozzle at the outboard edge of the overwing fairing. In addition, pieces of aluminum structure which appear to have come from the inboard section of the fairing were noted. Infrared television cameras had revealed a small object leaving the right side of the test article at approximately T + 50 seconds. The only debris found were the above mentioned pads and an aluminum skin section. It is thought that a portion of the wing air seal burned and was ejected by the airflow.

Fire Test No. 3

This test series was to determine the extension in time-to-failure of the brackets holding the aircraft rudder and stabilator control rods after coating them with silicone ablative. These brackets are of aluminium and are located in the aft nacelle. Tests were conducted on the current unprotected design as installed in the left nacelle and on protected brackets in the right nacelle. A pool of JP-5 fuel with appropriate flame baffles and an air source to provide airflow over the test area were utilized in the testing (see Figure 44). Preloads on the control rods were used to provide simulated flight conditions for the brackets and time-to-failure was observed. Failure was defined as when one of the weights attached to the control rods deflected a significant amount.

Both left and right nacelle areas were burned to provide a comparison between protected and unprotected surfaces. The procedure for testing each nacelle was similar. A fan was placed in the forward part of the nacelle to simulate airflow around the engine. A semicircular baffle, representing the afterburner section, was installed forward of the stabilator control rod assembly (see Figure 44). A 24 x 42 inch (60.96 x 106.68 cm) pan with 1 inch (2.54 cm) of JP-5 fuel was hung 15 inches (38.10 cm) below the top of the nacelle centerline. The nacelles were burned until a significant deflection of a weight was observed. Control rods were loaded to a representative 10-pound (4.536-kg) force. Test time began when the entire pan of fuel was on fire. At test termination the fuel was dumped onto the test pad and extinguished.

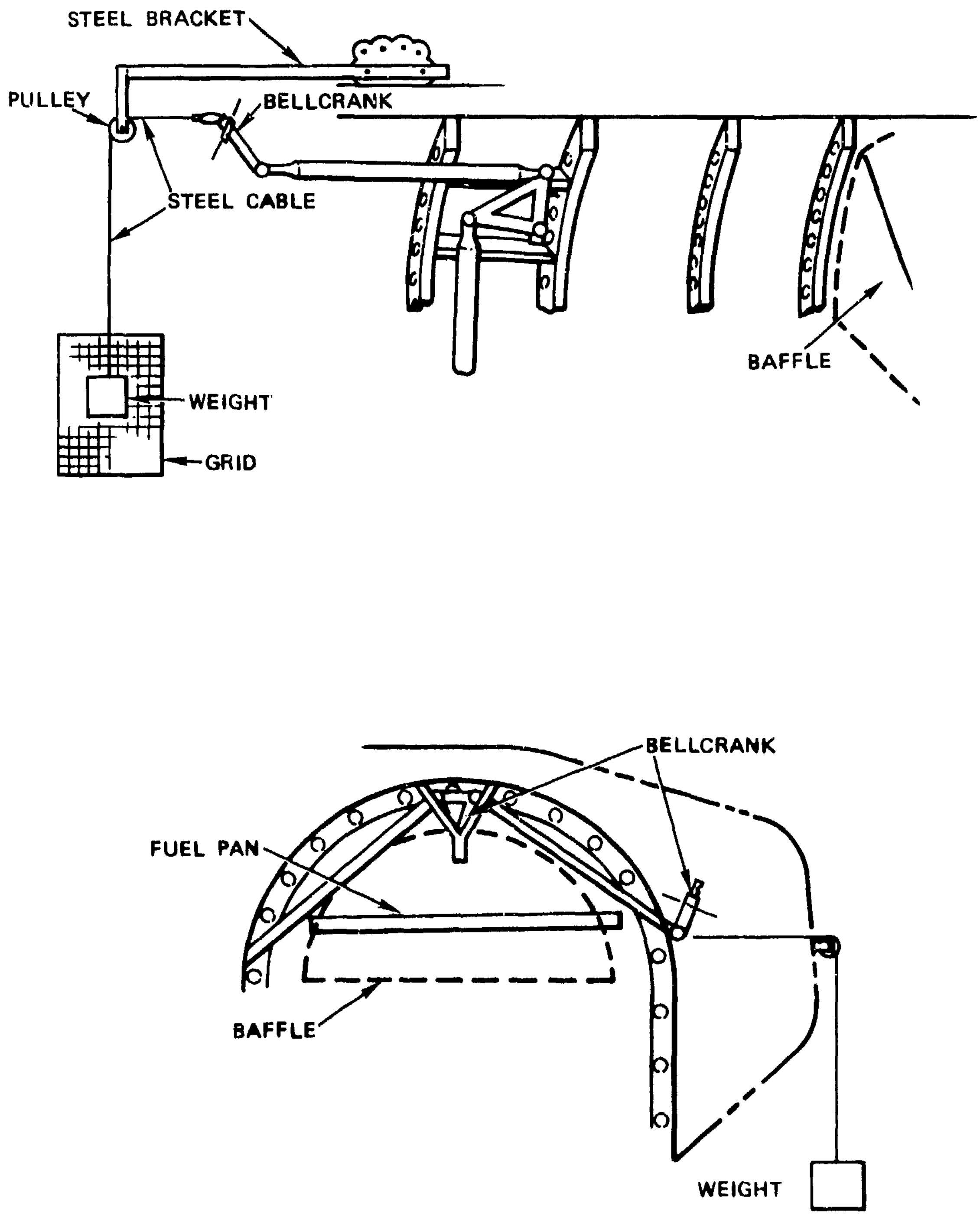


FIGURE 44. Test Configuration for Nacelle Fire Tests.

A post-test damage assessment of the unprotected left nacelle showed the aft assembly (rudder control) failed after 4 minutes of burn. The aluminum support brackets evidenced major damage, being cracked in four places. The skin and former ribs which support the aluminum brackets showed buckling and cracks which allowed the assembly to move inward and aft; these were essentially destroyed as shown in Figure 45. The forward assembly showed only a charring of the surfaces.

Deflections of the left nacelle test weights are plotted in Figure 46. This plot is derived from the video tape data. Final deflections after the structure cooled down were: fore, 0 inch; aft, - 2.0 inch (-50 mm). The aft weight was observed revolving from 40 to 90 seconds. It is suspected that the 1-inch (25.4-mm) weight drop during this time was due to heat expansion and unraveling of the cable.

The ablative-protected right nacelle was tested in the same manner as the left nacelle. After 2 minutes burn time, the rudder weight had dropped one inch and the test was terminated. Post-test damage assessment revealed that both left and right control rods on the forward bellcrank assembly evidenced paint damage and some surface beading. There was no apparent damage to the bellcrank support. The nacelle skin was buckled 12 inches (304.8 mm) left of the centerline.

The upper and lower control rods on the aft assembly evidenced paint damage and surface beading. The ablative-covered supports were not damaged with the exception of the forward rib. It had a 3-inch (76.2-mm) section of ablative missing and a few cracks 4 inches (102 mm) outboard of the assembly. The ribs show slight buckling and separation from the skin. The entire inboard side had buckled skin.

Weight deflections in the right nacelle are plotted in Figure 47. Final deflections after structure cool-down were: fore, 0 inch; aft, 0.56 inch (14.288 mm). The same cable was used for the aft weight in both the left and right nacelle tests. During this burn the cable did not unwind further.

The aft bellcrank assembly was not damaged. However, a deflection of 0.56 inch (14.3 mm) was achieved. This deflection is believed to be caused by the buckling of the aircraft skin and former ribs. The observed oscillation of the aft weight (see Figure 47) could also be caused by the skin buckling. The condition of the aft bellcrank is shown in Figure 48.

Since the first burn did not destroy either of the bellcrank assemblies, another burn test was conducted as the second segment of this test series. This burn had a duration of 4.24 minutes and was terminated due to fire spreading to the overwing fairing area. Observations made visually on the test pad did not indicate failure of either assembly during the burn.

FIGURE 45. Left Nacelle Aft Bellcrank Structure After Fire Test No. 3.

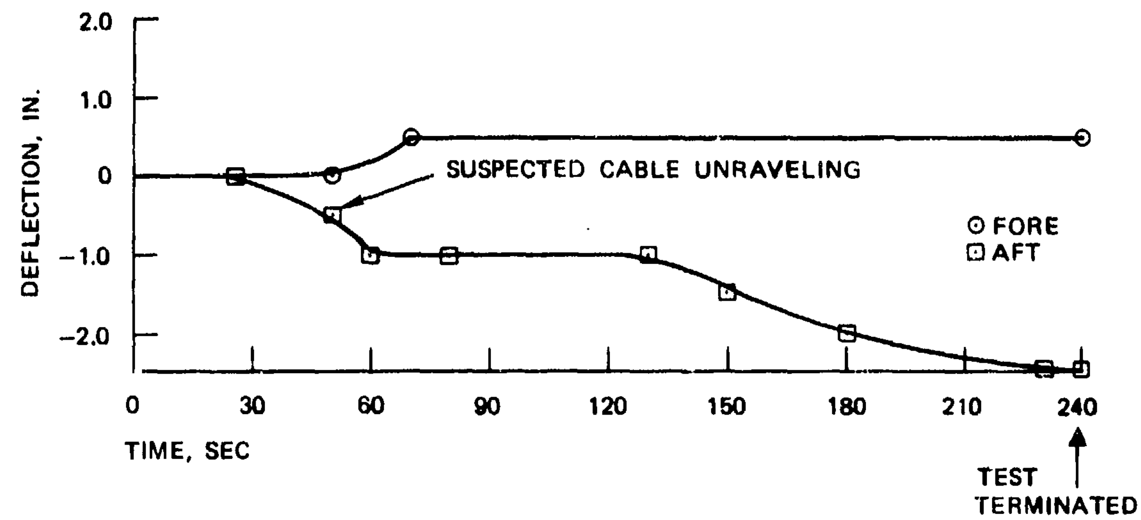


FIGURE 46. Weight Deflection Data Plot, Left Nacelle.

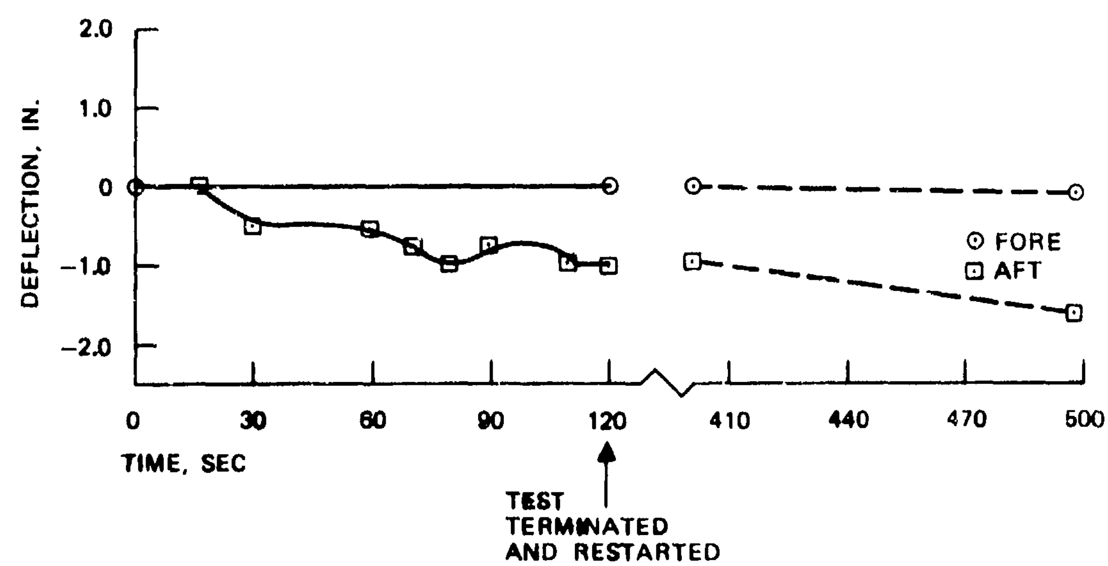


FIGURE 47. Weight Deflection Data Plot, Right Nacelle.

FIGURE 48. Right Nacelle Aft Bellcrank Structure After Fire Test No. 3.

Due to a crosswind blowing smoke in front of the video camera a plot of weight deflections was not possible. After the structure cooled there was no deflection in the forward weight and a 0.5-inch (12.7-mm) drop in the aft weight. The top of the nacelle on the outboard side burned completely through. The buckling of the skin was more severe and unprotected ribs were slightly twisted and cracked. The surfaces which had the ablative covering were only charred and both bellcrank assemblies appeared to be operable.

CONCLUSIONS

The F-14A fire protection testing program conducted by NWC entailed evaluating the materials and technology associated with a fire protection modification proposed by Grumman Aerospace Corporation for F-14A aircraft. Tests performed during the course of the program encompassed testing of candidate fire barrier materials and tests directed toward determination of the effectiveness of the proposed fire extinguishing system and fire containment methods proposed for the aircraft. Conclusions derived from the test results and recommendations regarding the proposed modification program are presented herein.

The tests performed during the course of this program fulfilled the program objective. Successful performance of all tests planned for the program was achieved, and the individual test goals were met. Conclusions associated with the major tests performed during the testing program are summarized in subsequent paragraphs.

CANDIDATE BARRIER MATERIALS

The initial foam material tests evidenced that foam specimen anomalies had a noticeable effect on test results. In casting relatively large blocks of fiber-reinforced polyurethane type foams, the mixed ingredients are injected into the bottom of a mold and then allowed to free-rise to the top. This process not only tends to orient the reinforcing fibers with the rise direction, but also creates homogeniety variations with foam density being greatest at the bottom of a casting block and gradually decreasing toward the top. Color variations in some of the specimens indicated that some of the isocyanate had not completely reacted. These process variations allowed for the presence of excessive voids in some of the samples, resulting in premature failures. No attempt was made to nondestructively measure foam homogeneity. The use of "soft" X-rays or acoustical techniques may provide this capability.

Of the polyurethane foam types tested, the 5F14RS foam was superior in terms of thermal resistance. It was capable of preventing burnthrough for periods in excess of 10 minutes with no supplemental intumescent coating protection. However, this was a special laboratory hand-mixed formulation with no production experience behind it. It was found that any of the other polyurethane foams tested could be made to resist burnthrough with the aid of proper intumescent coating. In applications where a predominant flow direction exists for the fire-exposed side of polyurethane foam type barriers, the foam located most upstream will receive the greatest heat load. This occurs because foam smoke outgassing, when exposed to fire, travels downstream forming a thickening boundary layer for protection.

Even with intumescent coating protection the polyurethane foams burn down to their basic carbonaceous char form within about 5 minutes after exposure to fires of the test intensity. Little actual test or measurement data of physical characteristics of these chars at temperature exist. Their ability to withstand the internal aircraft airflow generated and vibrational type loads is unknown and must be determined prior to any actual incorporation into aircraft. In the limited tests conducted with applied air pressure differential loads, the foam failed at relatively low levels; this suggests that metallic backside reinforcement would probably be required to meet environmental criteria during an actual aircraft inflight fire. Sizing for aircraft installation and methods of mechanical fastening of the foams was not investigated, but either of these factors could seriously affect the practicality of a given installation design. However, bonding of the foams to thin metal sheets, with all mechanical fastening being accomplished through the sheets, would appear to simplify this problem area.

Intumescent paint performance was also influenced by quality control of the particular coating and in its application process. Intumescent characteristics found desirable for thermal barrier application included formation of a mechanically strong insulating char (but of not especially large volume) of the exposed foam surfaces and a rapid initial rise followed by maximum paint adhesion in the gap filling application of heat. Improper adhesion was sometimes noted by detachment of sections of the coating upon initial application of heat. Residual traces of releasing agent on the virgin foam blocks, from the casting operation, prior to coating application was the suspected fault for this failure mechanism. In exposed foam surfaces application, 1000 modified and the 1200 flexible sheet were found to provide the best performance. In the gap filling application, M-30 followed by 1200 sheet were the most efficient. However, initial swelling action for both of these materials is relatively slow and the application of a thin outer coating of a fast rising material, such as 1000 modified, is recommended.

In the fire simulator tests, the test criteria for candidate barrier systems called for the barrier to withstand exposure to fire for a period of 15 minutes without failure, and the backface temperature (measured

6 inches (152 mm) from the barrier) not to exceed 400°F (205°C) for a period of 15 minutes. A TBS-758 silicone-coated stainless steel sheet specimen reached the test temperature criteria after 5 minutes duration when tested as a continuous sheet barrier. This type specimen was tested with lap and butt joint configurations. Both configurations failed after approximately 4 minutes test time. In a burn test conducted with the silicone insulator faced on both sides of the steel sheet and impregnated with fiberglass, rapid erosion of the fire-exposed fiber glass, with lap joint failure, occurred after approximately 60 seconds burn time.

In tests with the barrier penetrated by tubes and wire bundles, the weak link failure mode was evident. Once fire penetrated a weak spot, such as melting the tube, flame propagation from the inside rapidly melted out the rest of the tube thus defeating all other thermal protection. The tube then acted as a conduit for breaching the fire barrier. The wire bundle penetrations proved less vulnerable. Although the bundle melted where exposed directly to the fire stream, flame failed to penetrate the barrier during a 7-minute burn test.

Ceramic insulation material for aluminum tube protection proved to be the most effective with burnthrough time in excess of 8 minutes. Silicon material was marginal in this application although coatings of approximately 0.020 inch (0.51 mm) may prove adequate (this thickness coating was not tested). Backside auto-ignition of the various sealant materials proved a problem when the penetrating medium resisted burnthrough. Although sustained combustion of the silicon sealant occurred as early as 4.5 minutes burn test duration, actual burnthrough of the barrier at the penetrations did not occur until about 8 minutes burn time.

EXTINGUISHING AGENT CONCENTRATION

Measurement of agent concentrations in the engine nacelle volume showed adequate concentrations for periods of several seconds under the different airflow configurations employed in the tests. The loss of extinguishant from the nacelle under a fancase rupture failure mode, and the effect on the accessory bay volume, was of particular concern. However, in tests conducted under simulated failure mode conditions, the agent concentration proved to be more than adequate in the nacelle volume. The dispersion of extinguishant and concentrations in fire-critical areas was proved to meet, and in fact exceed, military specification requirements for an aircraft fire extinguisher system. As the result of these tests it was concluded that performance of the fire extinguisher system was adequate for the intended purpose. It represented a conservative system design for an aircraft of this type.

FULL-SCALE FIRE TESTS

During the first burn of the full-scale test article, the test data showed that the extinguishant put out the aircraft fire within 2 seconds after agent deployment. Had the aircraft been in flight, very little fire damage would have occurred. The titanium sheet that was added on the forward outboard section of the engine nacelle did keep the fire from burning out into the wheel well. The titanium sheet added on the nacelle top deck proved to provide adequate fire protection in that area. The torch effect resulting from fire entering the accessory bay through the fan blade penetration in the nacelle deck resulted in local melting of the overwing fairing itself. The presence of the fire barrier between the accessory bay and center trough inhibited the propagation of fire into the trough and prevented damage to the control rods.

The center trough fire barrier had some slight delamination of the silicone ablative from the stainless steel backing. It does not appear to be related to the heating by fire. The silicone applied over the top of fuel cell No. 5 in the overwing fairing accessory bay did not sustain any significant heat damage. This silicone also suffered extensive delamination prior to testing, probably due to inadequate cleaning of the metal prior to application of the silicone.

The endurance fire test subjected the test article to a worst-case condition. A fire relight occurred after extinguishment of the initial fire by a residual fire on the test pad which would not occur in flight. Fuel vapors inside the nacelle were forced through the engine bay door joints by the simulated in-flight ram air cooling and engine fancase rupture airflow. These vapors were ignited by the test pad fire*. The fire then re-entered the nacelle where it continued to burn in an intense fire in the forward part of the nacelle for approximately 20 minutes. The long period of this fire was due to a malfunction of the fuel shutoff valve. The valve actuated properly, however the valve butterfly went past the design stop allowing continued fuel leakage from the main engine fuel feed line.

The titanium sheet over the nacelle was intact. The overwing fairing compensator suffered heat failure to its aluminum shell and mounting structure. The fire path opened by this failure, in addition to the 12-square-inch (78.5-cm^2) hole deliberately cut in the barrier to represent a fan blade penetration, allowed fire to impinge on the top of fuel cell No. 6 and the diagonal longeron. The fuel cell was penetrated along with severe heat damage inflicted on the longeron. These effects would be of consequence only in the event the extinguisher failed to control an on-board fire. Extending the titanium over the small area of the affected fuel cell and applying insulation to the longeron could alleviate this potential problem.

* The test pad had been foamed to prevent fire relight. The foam reduced the size of the test pad fire but did not eliminate it.

The second burn test series in the overwing fairing accessory bay shows that the central trough fire barrier deflects the heat away from the trough. There was no failure of the fire barrier during the 1-minute burn test. On this test, the fire extinguishing bottle was intentionally not used; however, the nozzle failed during the fire, discharging the agent in an unscheduled manner. The particular bottle design on the test item was not the one proposed for production incorporation. It does serve as a warning that the production design should be failure tested.

As a result of the data obtained in the overwing fairing test it is apparent that the presence of the fire barrier is sufficient to defeat the propagation of fire into the trough. Special care is required to assure the bottom of the barrier is liquid-tight and that all gaps are closed off. Liquid fuel, if present, will run into the trough as that is the lowest point in the accessory bay. Wire bundles and tubing should be covered with some form of insulation so they do not allow easy penetration of the barrier by fire.

The aft nacelle fire tests showed that unprotected control surfaces in the left nacelle failed after approximately 2 minutes exposure to fire. The protected surfaces in the right nacelle endured a burn time of 6.5 minutes duration without sustaining any significant damage. Judging by the damage to unprotected surfaces and the good condition of the protected surfaces, it could be expected that the ablative-covered components would fail when surrounding unprotected structure sustained major fire damage.

RECOMMENDATIONS

Based on the results of this test program, the following recommendations are presented:

1. Titanium firewalls as installed in the test article are adequate. The Halon 1301 fire extinguishing system as finally configured has sufficient dispersion and concentrations to extinguish a fire in the protected areas. If a relight source were still present, the fire could reignite, placing added importance on the fire barriers. The firewalls and fire extinguisher should be considered for installation in production aircraft and for retrofit into early aircraft.

2. The overwing fairing compensator should be investigated to determine techniques to prevent burnthrough of the covers and support structures.

3. The titanium fire barrier should be extended over the top of fuel cell No. 6 under the overwing fairing area to prevent possible burn-through into a fuel supply.

4. Fire heating of the diagonal longeron should be investigated to determine if protective measures should be incorporated to prevent heat damage as seen in the fire endurance test.

5. The silicon-on-steel sheet is an adequate fire barrier technique for protecting the central trough. Further modifications and tests should be performed on the joints between barrier pieces to ensure the production parts will meet the fire criteria. Also, the sides and bottom edges of the barrier must be sealed to prevent liquid fuel from flowing into the trough. The tests showed that liquid fuel can be blown by ram air through the nacelle top deck into the accessory bay.

6. The firewall fuel shutoff valve should be modified to prevent the malfunction experienced in the first test series which allowed continued defueling of the aircraft.

7. The silicon ablative material applied to the aft nacelle control rod brackets significantly extended the time-to-failure and should be considered for incorporation in the aircraft modifications.

Appendix A

Development of Fuel Line Cutting Charge

DEVELOPMENT OF FUEL LINE CUTTING CHARGE

The NWC Explosives Technology Branch conducted a series of tests to develop the cutting charge used to simulate fuel line penetration by a liberated engine fan blade. The object of the tests was to demonstrate the cutting capabilities of a flexible linear shaped charge (FLSC) on a 3-inch (76-mm) fuel line for use in full-scale engine tests. A representative fuel line was obtained for the development and testing of the FLSC cutting assembly. The fuel line was aluminum with an outer 2.5-inch (64-mm) diameter instead of the 3-inch (76-mm) size orginally anticipated. The wall thickness was 0.050 inch (1.3 mm).

Two assemblies were designed and successfully tested on the sample fuel line. In the first design, a circle of FLSC 4 inches (100 mm) in diameter was taped in contact with a length of the fuel line. The FLSC was lead sheathed and contained 10 gr/ft (2 g/m) RDX explosive. The 4-inch (100-mm) circle contains a surface area of 12 square inches (8×10^3 mm^2). The length of fuel line was filled with water and pressurized to 30 psi (1.4 kpa). The cutting charge was successfully fired, giving a uniform cut throughout the circumference of the circle. The explosive cutting charge collapsed the straight length of fuel line since the ends were not held fixed.

The second design incorporated a section of aluminum tubing with a 2.625-inch (67-mm) inside diameter to provide a rigid support for the FLSC. A length of tube was cut in half, parallel to its axis, so it could be placed like a saddle on top of the fuel line. An elliptical hole was cut in the saddle to allow an easy exit of the plug cut from the fuel line and the fuel itself. The elliptical shape was used instead of a circle to ensure that no more than half the circumference of the fuel line would be severed. The FLSC was placed on the underside of the saddle along the elliptical contour and held in place with epoxy. Rubber gasket material was also glued to the inside of the saddle to provide a 0.040-inch standoff distance between the FLSC and the fuel line. The saddle was attached to the fuel line using 3-inch pipe clamps to ensure a firm method of attachment. Water was added to the fuel line and pressurized to 30 psi. This second design was successfully tested. The elliptical plug cut from the fuel line also had 12 square inches of surface area.

In both tests, an RP-80 Exploding Bridge Wire (EBW) detonator with a 0.25 x 0.25 inch (6.35 x 6.35 mm) booster pellet of PBXN-5 was used to initiate the FLSC. Special aluminum adaptors were required to hold the booster pellet and detonator in place for proper initiation of the FLSC. Two more units were then fabricated for the full-scale tests. Only the aluminum adapter was changed from the previous design to accept an SE-1 EBW detonator instead of the booster pellet and RP-80 detonator.

Appendix B

Test Facility Engineering

TEST FACILITY ENGINEERING

Pretest preparations included delineation of the test site preparation/modification tasks required for the full-scale tests. Test site modifications were necessary to accomodate an aircraft of the F-14A size. An overview of the NWC Aircraft Survivability Test Site is shown in Figure B-1. Facility engineering tasks consisted of construction of the aircraft test pad and mount; fabrication and test of the DASH (dynamic air source, high velocity) airflow nozzle; design and construction of the airflow mappi g system; preparation and installation of instrumentation and remote controls required for full-scale tests. Specific attention was focused on requirements for the aircraft test pad, airflow system, instrumentation and remote controlling, and safety considerations.

AIRCRAFT TEST PAD

Because of the nature of the tests to be conducted and the size of the F-14A test article, a new test pad was necessary to accommodate the proposed tests. Spacing of aircraft tie-down points and their capability to hold the aircraft with one engine operating were the basic criteria for determining test pad requirements. An initial data acquisition meeting was held with GAC personnel to ascertain aircraft mount design parameters.

The environmental effect of airflow mass flow generated during full-scale test performance was also considered. Without provision for control of the combined engine/test support airflows there was a possibility that test site visibility would be obscured by dust, with possible damage to the facility airflow (DASH system) turbines from ingested dust. Therefore construction of a blast deflector was included in site modification plans. Control of fuel spillage, pad washdown and external fire containment materials was also seen as necessary. Although large-scale fuel spillage was not anticipated during fire test performance, the possibility existed. To provide control of large volume quantities of liquids on the pad, construction of a drainage trough and catch basin were included in the site modification plans.

An 8-ton carbon dioxide system was available at the test site. Use of the system was planned as a backup for the aircraft on-board fire extinguishment system. Control of fires external to the test article would be provided by the NWC fire department with mobile fire fighting trucks.

FIGURE B-1. Overview of NWC Aircraft Survivability Test Facility.

AIRFLOW SYSTEM

Airflow system requirements were based upon the need to provide a 250-knot airflow over the aircraft overwing fairing structure, and the airflows required for inflight normal nacelle cooling and engine fancase rupture simulation. Because of the different quantities, pressures and velocities of airflow required for test conditions, several methods of supplying the required airflows were investigated.

The full-scale testing program had been predicated upon use of the test facility DASH system as a primary airflow source. Although the DASH system produces over 500 lb/sec (225 kg/sec) of air, the operating pressures and temperatures would entail problems in controlling pressure/temperature factors for subsidary pressure flows in the test article nacelle. Therefore, the NR-2B and NR-10 airconditioning units were selected as sources for subsidary airflows.

The DASH system (see Figure B-2) is located at the NWC Aircraft Survivability Test Facility and is presently configured with two TF33-P5 turbofan jet engines. The system collects bypass airflow exiting from the fan stage of the engines. This airflow is then ducted into a 47-inch (1.2-meter)-diameter universal duct. Airflow diverters in the universal duct permit test personnel activity on a test specimen while the engines are at idle RPM. The exhaust from each engine is collected and directed vertically away from the test article.

Flow Distribution Test

One of the early requirements in the program effort was to map the airflow from the DASH system universal duct to determine the existing flow distribution. This data was necessary to determine the airflow parameters to design a new airflow distribution nozzle for the overwing fairing area of the test article. Tests were subsequently conducted to map the airflow discharge from the open end of the existing duct. In this task the airflow velocity was measured across the diameter of the duct using the test setup shown in Figure B-3.

Airflow velocity measurements were taken right at the end of the duct and at three other downstream Z-axis stations as shown in Figure B-4. Spacing of the Z-axis stations was equal to one, two and three duct diameters, respectively. As expected, the resulting data evidenced a pronounced variance in airflow velocity as measured across the duct diameter. Discharge velocity varied as much as 38 knots from one side to the other at the point of discharge from the duct. Velocities became somewhat more consistent in distribution further out from the discharge point. However, the map of Figure B-4 evidences that the air is discharged in a swirl-pattern which was unacceptable for the desired test purposes. Therefore it was evident that an airflow distribution nozzle was necessary to produce a more uniform airflow over the test article.

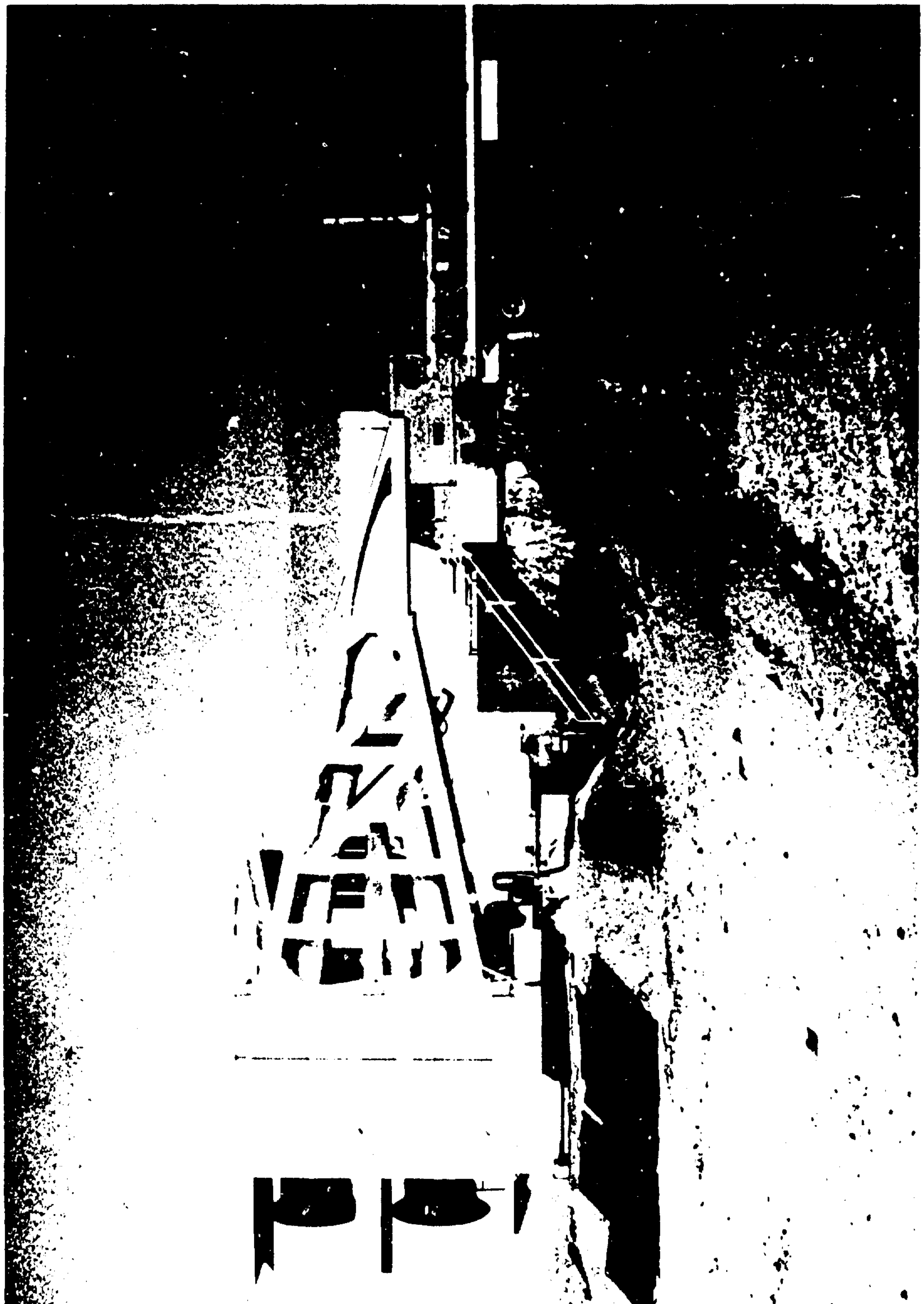


FIGURE B-2. Dynamic Air System, High Velocity (DASH) at NWC Aircraft Survivability Test Site.

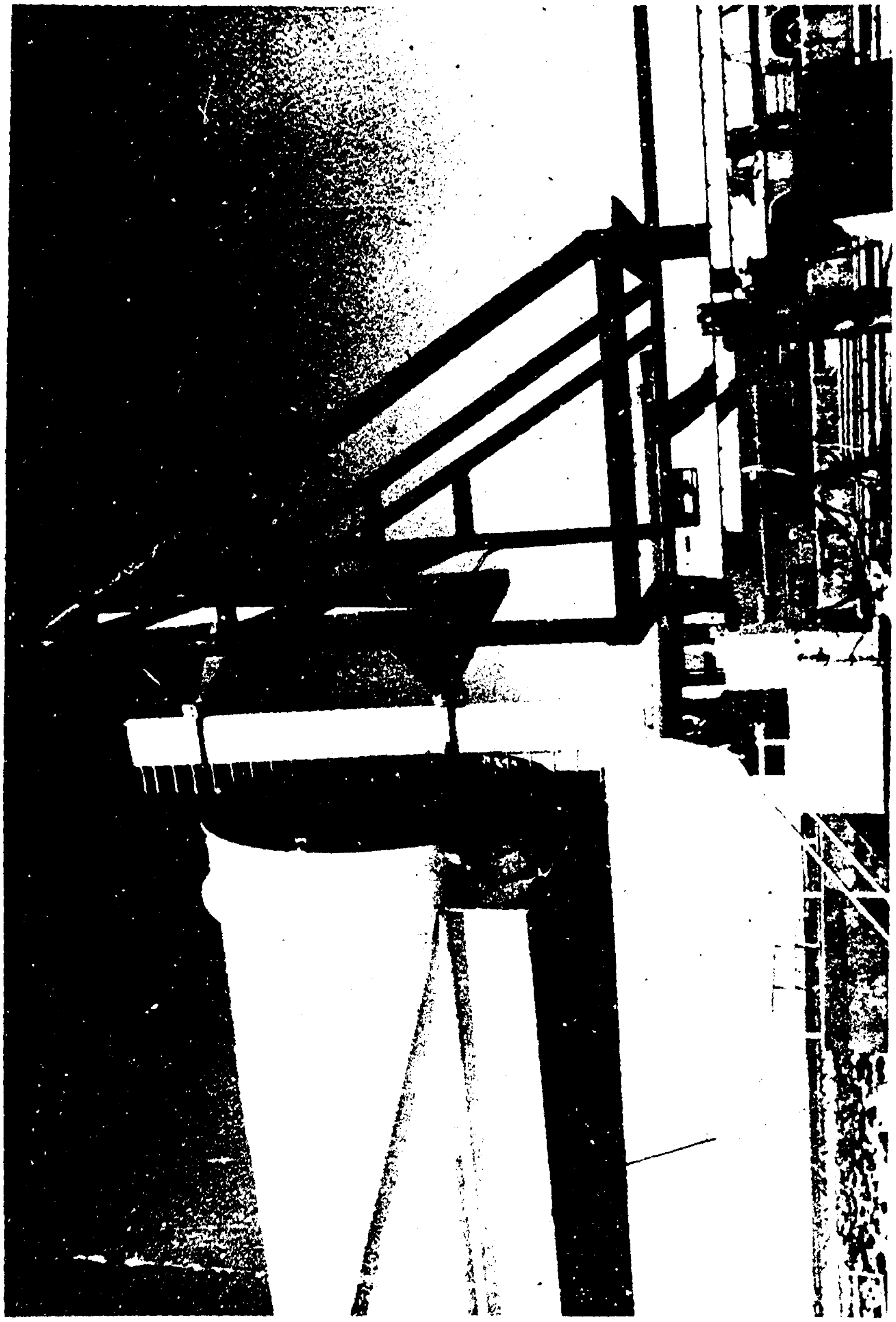


FIGURE B-3. Test Setup for Flow Distribution Test.

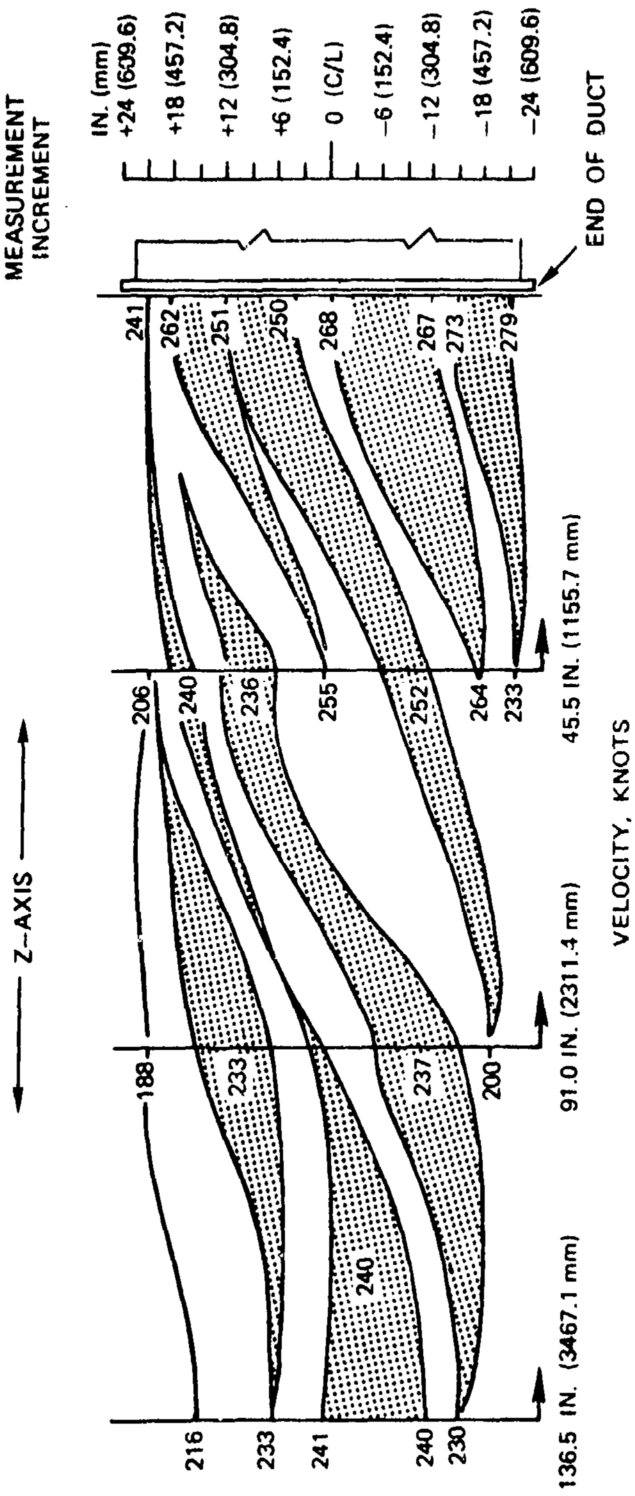


FIGURE B-4. Airflow Velocity, DASH System Discharge Mapped as a Free Jet.

An airflow distribution nozzle and support structure was designed and constructed for the test article. The support structure was designed to position the nozzle over the overwing structure of the test article when it was positioned and tied down on the test pad. This is shown installed in Figure B-5.

Upon establishing the airflow and measurement requirements, an air velocity sensor system was designed and built to facilitate mapping airflow over the F-14A test article. Airflow tests were subsequently conducted as described herein.

Air Velocity Sensor System (AVSS)

The AVSS shown in Figure B-6 was used to measure the airflow over the test article. The AVSS is comprised of a pitot tube rake assembly, a powered vertical and horizontal drive assembly with respective limit switches, a relay box which monitors the rake displacement, a scanivalve box which transforms pressures to electrical signals, and a hand-held controller box for manual control (if desired) of rake displacement. These components of the AVSS were mounted to two parallel I-beams and comprise a single unit which can be separated from the support structure. Readout instrumentation was contained in an instrumentation van which was connected to the AVSS by means of a 100-foot (30.48-meter) cable. Movement of the rake during mapping and data acquisition was performed automatically by utilizing a computer in the instrumentation van.

The pitot tube rake assembly consists of 8 stagnation pitot tubes and one combination pitot tube mounted to a 1.5-inch (38.1-mm)-diameter pipe as shown in Figure B-7. The stagnation pitot tubes are spaced 3 inches (76.2 mm) apart and protrude 4 inches (101.6 mm) into the airstream. The combination pitot static tube is mounted to the side of the row of stagnation tubes and has a similar construction except it protrudes 13 inches (320 mm) into the airstream. Vinyl tubes running internal to the 1.5-inch (38.1-mm)-diameter pipe connect the pitot tubes to the scanivalve. A thermistor mounted on the pipe perpendicular to the airflow measures stagnation temperature. Potentiometers provide positive feedback for controlling rake movement in the horizontal and vertical axes, and provide position data for indicating traversing movements. Limit switches at each end of the respective lead screws control maximum travel of the lead screws. Movement of the rake assembly in the Z-axis entails repositioning the support structure; this is a manual operation. A more detail description of construction and operation of the AVSS is provided in NWC Technical Memorandum 2897.[9]

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[9] Naval Weapons Center. *Remote Control Pitot Tube Rake Airflow Calibration System*, by D. L. Bishop, et al. China Lake, Calif., NWC, August 1976. (NWC TM 2897, publication UNCLASSIFIED.)

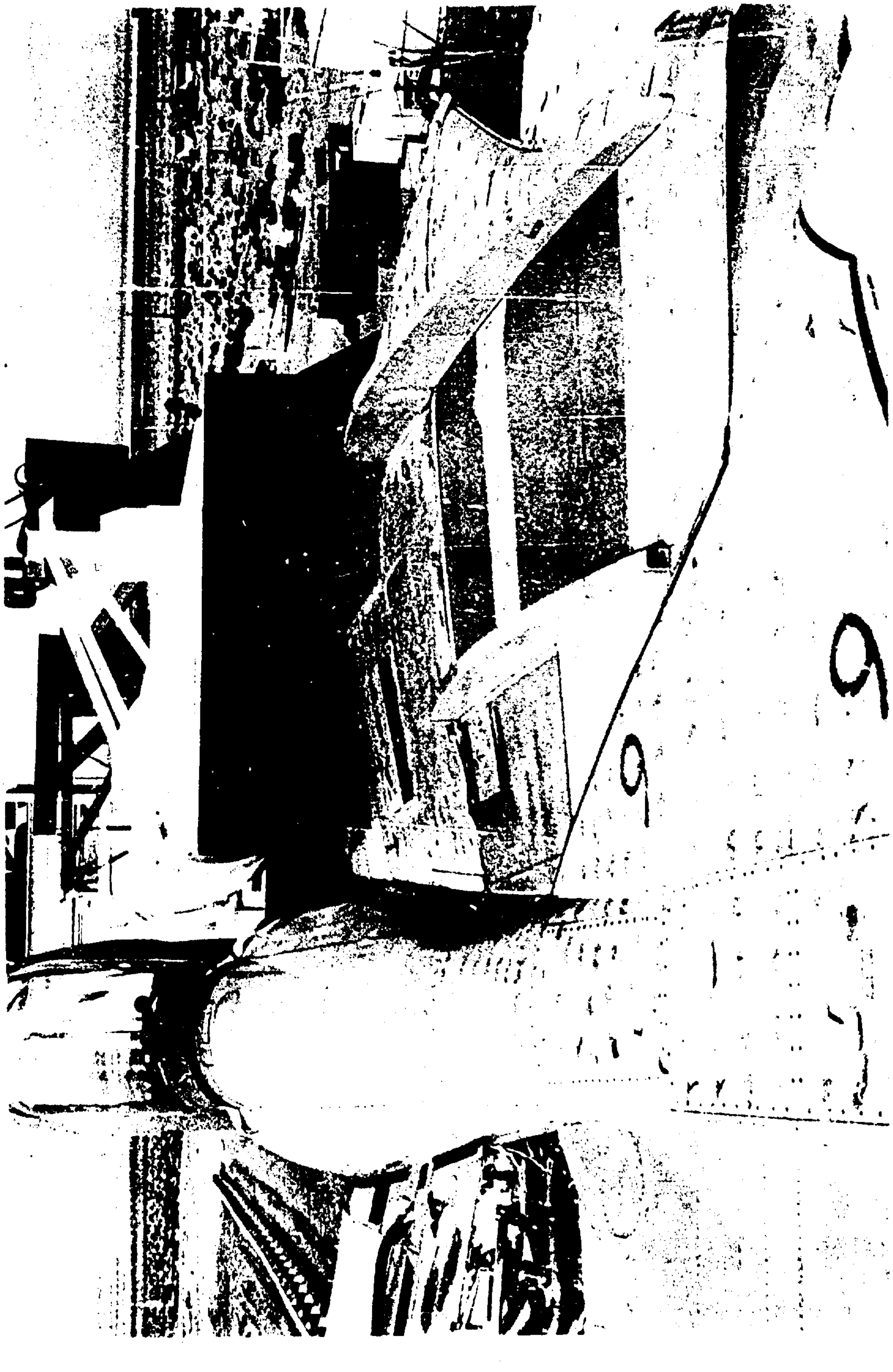

FIGURE B-5. DASH Nozzle Positioned Over Test Article.

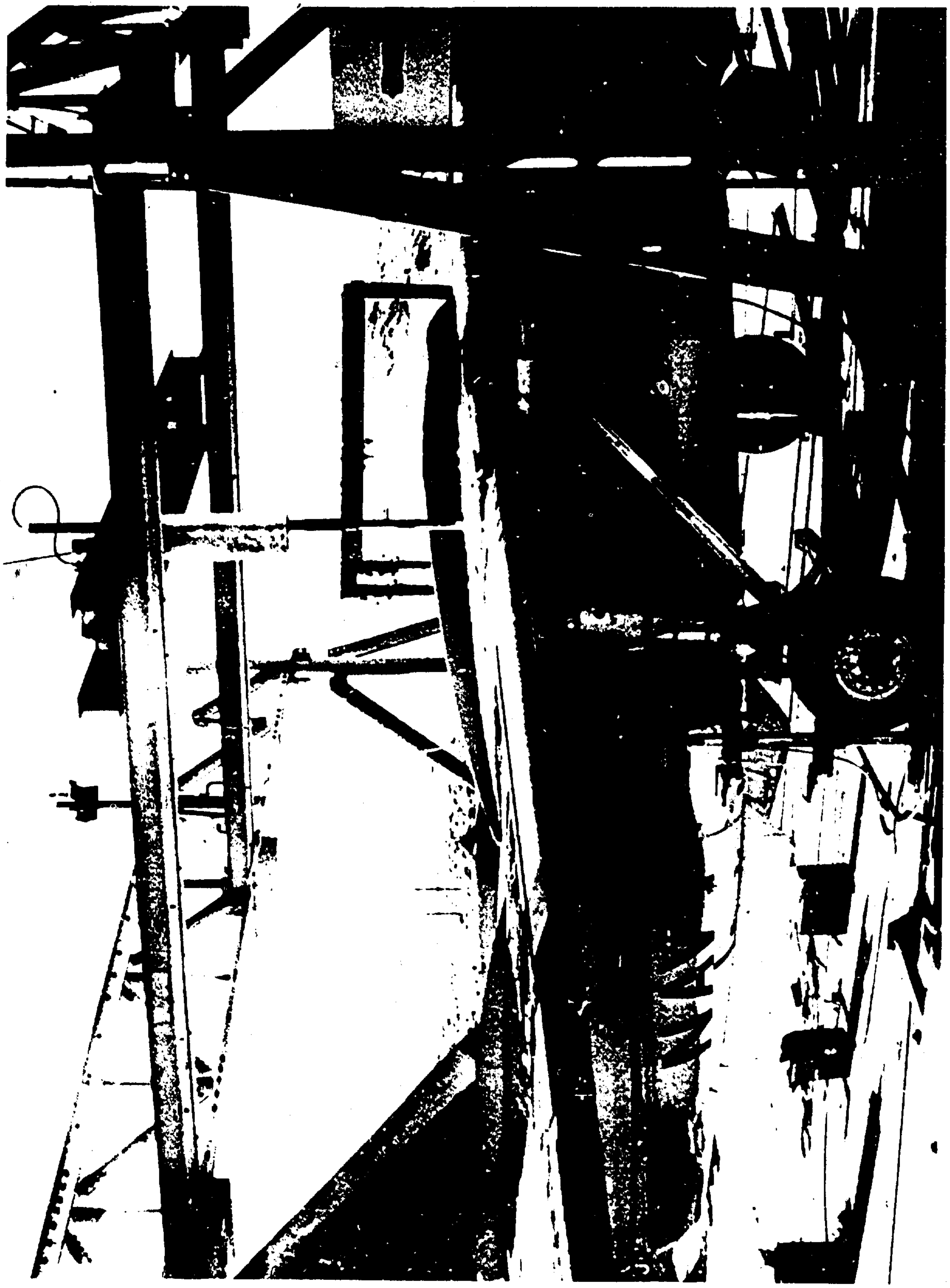

FIGURE B-6. Air Velocity Sensor System Used for Airflow Mapping Tests.

FIGURE B-7. Pitot Tube Array Used For Airflow Mapping.

Nozzle Airflow Measurement

Nozzle airflow measurements were made prior to installing the test article aircraft on the test pad, and after the test article was in position. For the first tests the nozzle airflow was mapped as a free jet. The purpose of the tests was threefold, i.e., (1) to verify that the nozzle/DASH system combination would produce a 250-knot airflow at a point 15.5 feet (5.7 meters) beyond the end of the nozzle (the desired location on the test article), (2) to map the airflow of the nozzle within the capability of the measurement system pitot tube rake system, and (3) to obtain airflow versus power setting calibration data for control of airflow velocity via DASH system power settings.

In mapping the DASH system nozzle as a free jet, the first test run was made with the pitot tube rake positioned 15.5 feet (5.7 meters) downstream from and aligned with the nozzle centerline. The test plan called for starting the test with a DASH system power setting of 70 percent and, after measuring airflow at this setting, increasing the power setting in 5 percent increments until the desired 250-knot airflow setting was bracketed. At this point the power setting was subsequently adjusted as necessary to provide the power setting which would produce an exact 250 knot airflow.

Data from both the DASH system control panel and from the pitot tube rake instrumentation were recorded at each power setting increment. Airflow velocity was calculated from the pitot tube data for correlation with DASH system power settings. These data provided a basis for DASH system operational parameters for subsequent tests.

After establishing the 250 knot airflow power setting, airflow measurements were made within the X - Y envelope traversed by the pitot tube rake. Airflow measurements were taken and recorded at each 6-inch (152-mm) increment during the traverse across the 123-inch (312-cm) width of the envelope. The envelope was traversed with the pitot tube rake in both the extreme up and down position.

Nozzle airflow measurements were then repeated at Z-axis positions 240, 120, 60 and 0 inches (609.6, 304.8, 152.4 and 0.0 cm) from the nozzle. Data recorded from these airflow measurements were plotted and significant boundaries of the airflow envelope established. The velocity and pressure/temperature measurements taken by the DASH engineer at each measurement increment were compared to the airflow measurements to verify DASH system power setting throughout the calibration runs. The mapping was repeated with the test article in position. A plot of the airflow profile over the test article is shown in Figures B-8 and B-9.

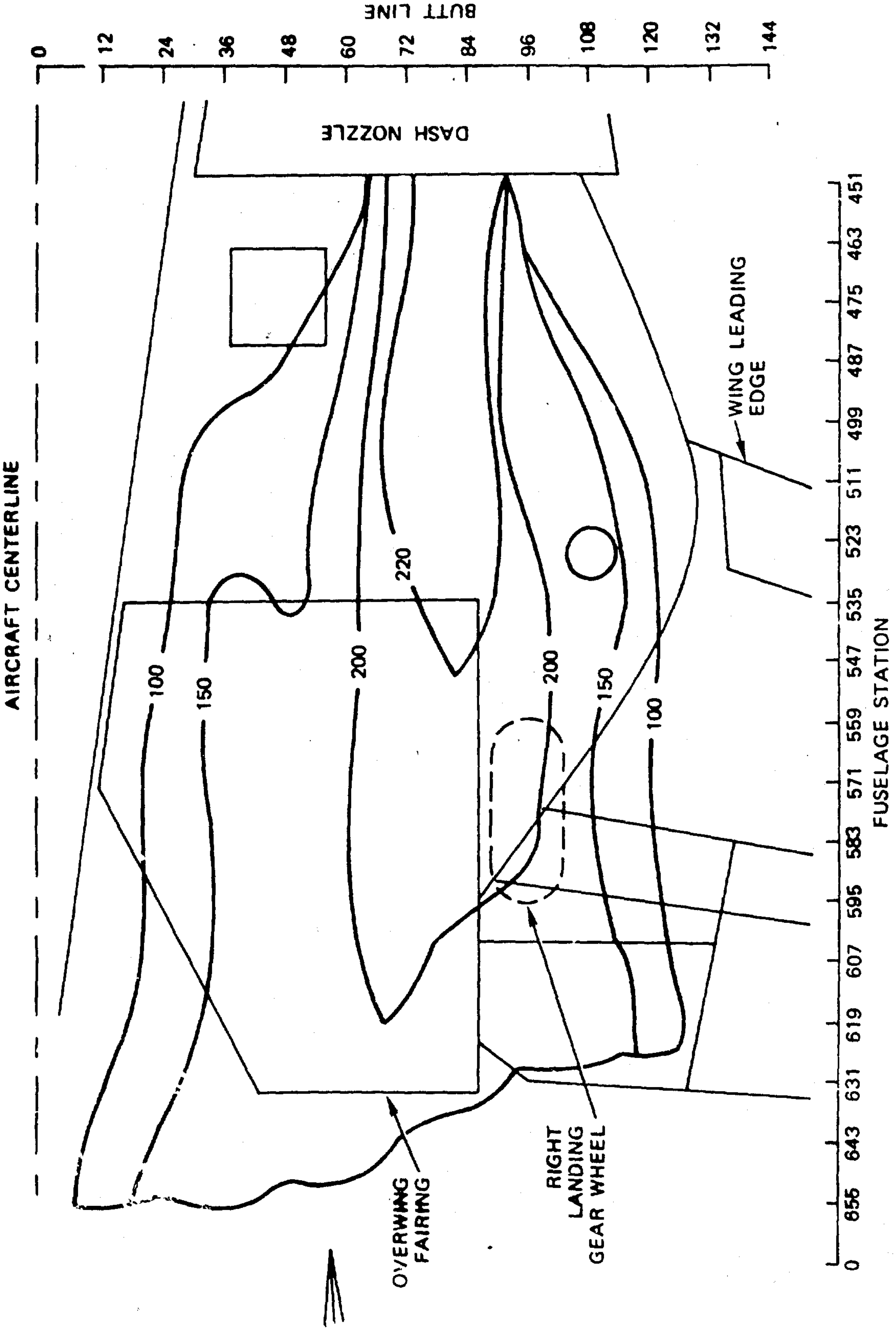

FIGURE B-8. Airflow Profile Measured 1 Inch Above Test Article Surface.

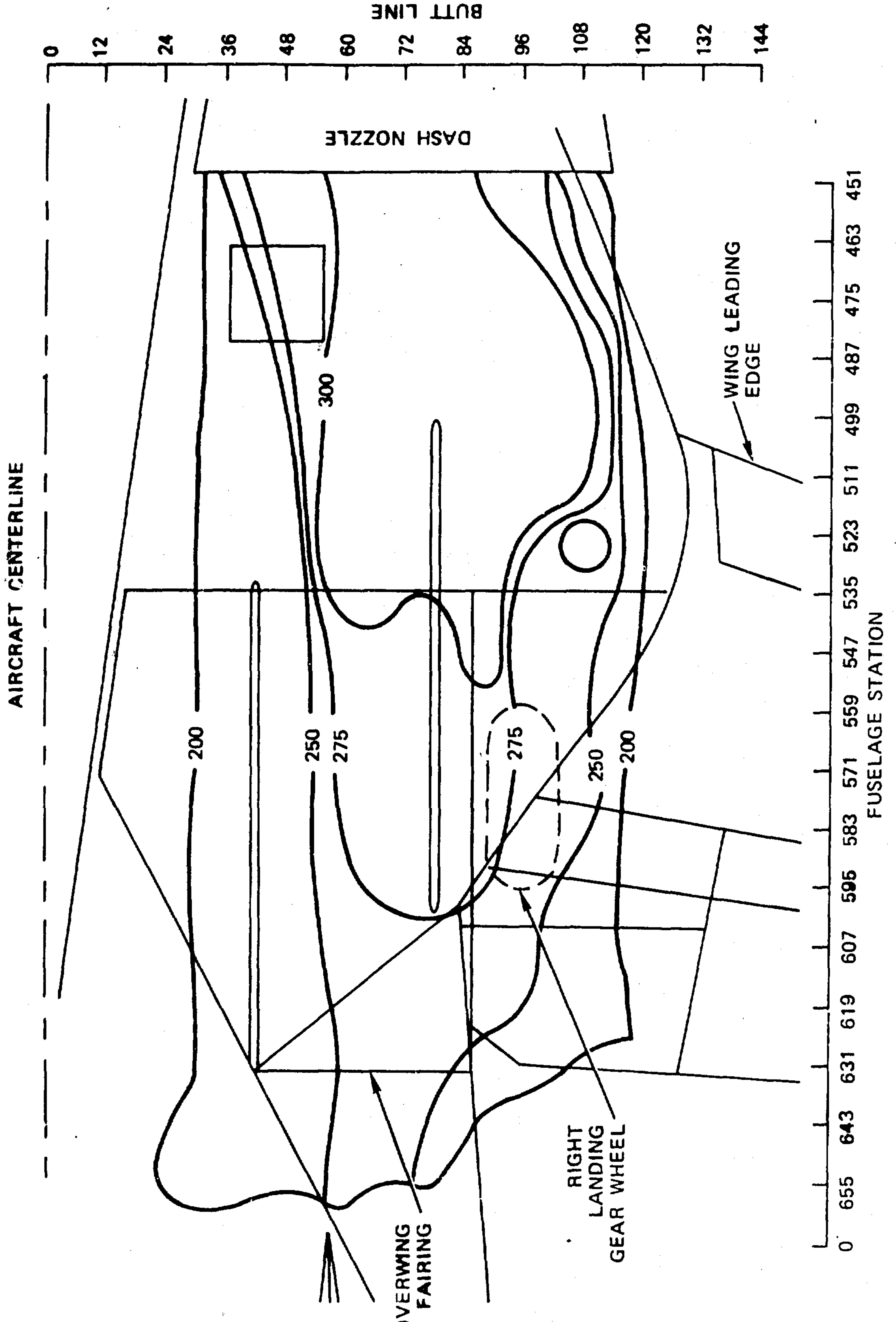

FIGURE B-9. Airflow Profile Measured 7 Inches Above Test Article Surface.

INSTRUMENTATION

Instrumentation tasks involved delineating instrumentation requirements, procurement, setup, installation, and calibration. Instrumentation for the test program included an infrared thermal visualization camera system along with temperature, pressure and flow measurement capability. Data recording and processing were integral parts of the test program.

The ability to real-time visually monitor events occurring in the engine nacelle and overwing fairing during the fire tests was extremely important. As video cameras could not be used in the fire environment, a system of lights tied into the instrumentation monitoring circuits was used. To achieve this end, a forward oblique view of the aircraft showing internal structure was generated and the drawing mounted on a board. Small light emitting diodes (LEDs) were placed at points corresponding to selected temperature sensors located in the test article. This assembly comprised the display board which was mounted in the test control room.

The display board is shown in Figure B-10. In addition to the lights on the aircraft drawing, the lights in the three categories, namely (1) Status, (2) Fire Caution, and (3) Terminate Test, correspond to the sensors installed in the test article. The Status lights indicate normal test sequence and expected fire progression within the engine nacelle; the Fire Caution lights call attention to excessive temperatures in adjoining areas but, at this point, do not indicate need for termination of the test. Illumination of any light in the Terminate Test group calls for immediate termination of the test and immediate fire control action due to fire breaching the fire barriers and reaching critical test control locations.

The individual LEDs were driven by an electronics system that sensed the parameter and then illuminated the diode when the parameter changed through a selected value. The trip points selected for thermocouple data included (1) when skin temperatures of critical components exceeded 400°F (204°C) or (2) when air temperatures in a test volume exceeded 700°F (370°C). A selection of events (simple on/off events) were also displayed.

Detecting and monitoring the spread of heat caused by the fire was done by an infrared thermal visualization camera and related recording system. The resulting pictures, with isotherms, provided a means of analyzing hot spots and heat flow paths in real-time as well as providing a hard-copy record. In this manner test personnel could visually monitor fire progression in the test article during test performance.

A block diagram outlining the instrumentation and control of the test article is presented in Figure B-11. Three buildings, a concrete test pad and the DASH air facility comprise the physical facilities requirements for the test series. Power and control lines are routed separately from instrumentation lines so as to reduce the potential of

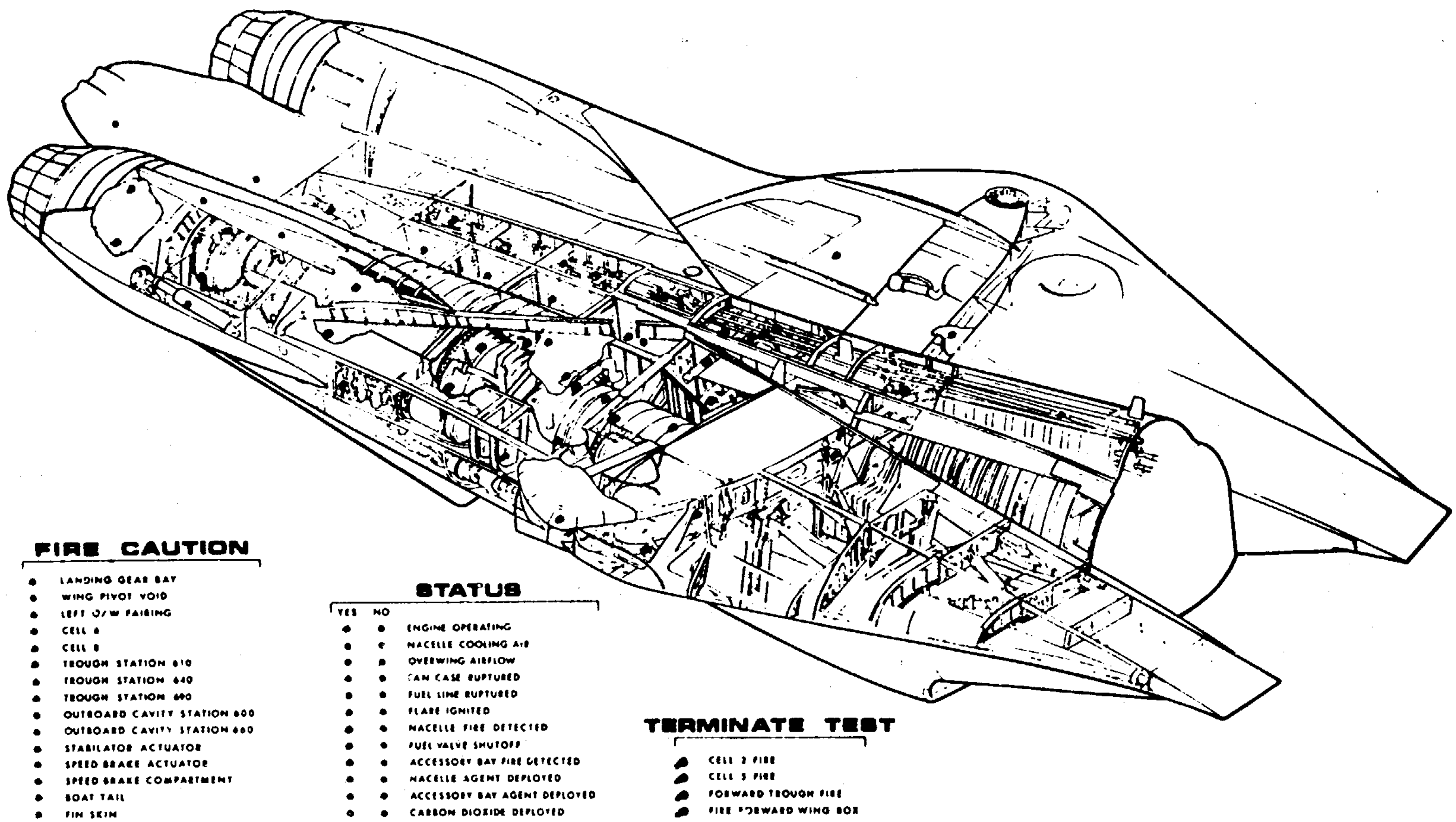

FIGURE B-10. Display Board Used to Monitor Fire Tests.

noise on recorded data. Tables B-1 through B-3 list the instrumentation and control parameters for measurement, operation and control of the facility and test article. Placement of temperature sensors was chosen so as to identify the fire propagation in the test volumes.

Chromel-alumel thermocouples were used throughout the test article with each couple terminating in a 150°F (65°C) reference oven. The signal from the oven went directly to a 90-channel amplitude multiplexer before being sent over land lines to the facility block house. All pressure sensors were of the strain gauge bridge type. Pressure sensor lines went through the instrumentation bunker to the facility block house where bridge balance units were mounted. The data system is shown in block diagram form in Figures B-12 and B-13.

Photographic and television coverage were also included to record visual events during the test. Film coverage at both normal speed (24 frames/second (fps)), and high speed (400 fps) were utilized. Test operators had a selection of monitoring views from both a standard black-and-white and color video camera, and an infrared camera displayed in color and black and white. These test monitors were in addition to the sensor-activated display board.

Engine operation parameters, Channels 74 through 78, utilized standard aircraft sensor systems; only the engine power-level angle (throttle position) was recorded. Necessary engine readouts were connected to the engine sensor systems to assure proper operation and control during the test. Critical test article events, Channels 79 through 84, were used for data feedback and were recorded to allow precise measurement of event times.

Photography and video cameras were used throughout the test program to document test progress and test performance. Motion picture documentation and still photography were used to record test item status before, during and after testing. This record, in conjunction with recorded data from other instrumentation, provided a basis for detailed evaluation of the fire protection system.

As with any hazardous testing it was necessary to remote the critical aircraft control functions. These included engine operation/shutdown, fuel and ignition for the fire test, operation of the fire warning system and fire extinguisher. Operational controls incorporating appropriate servo actuators were installed and their functional operation verified. Remoting of instruments and indicators necessary to monitor system performance was accomplished prior to beginning testing operations.

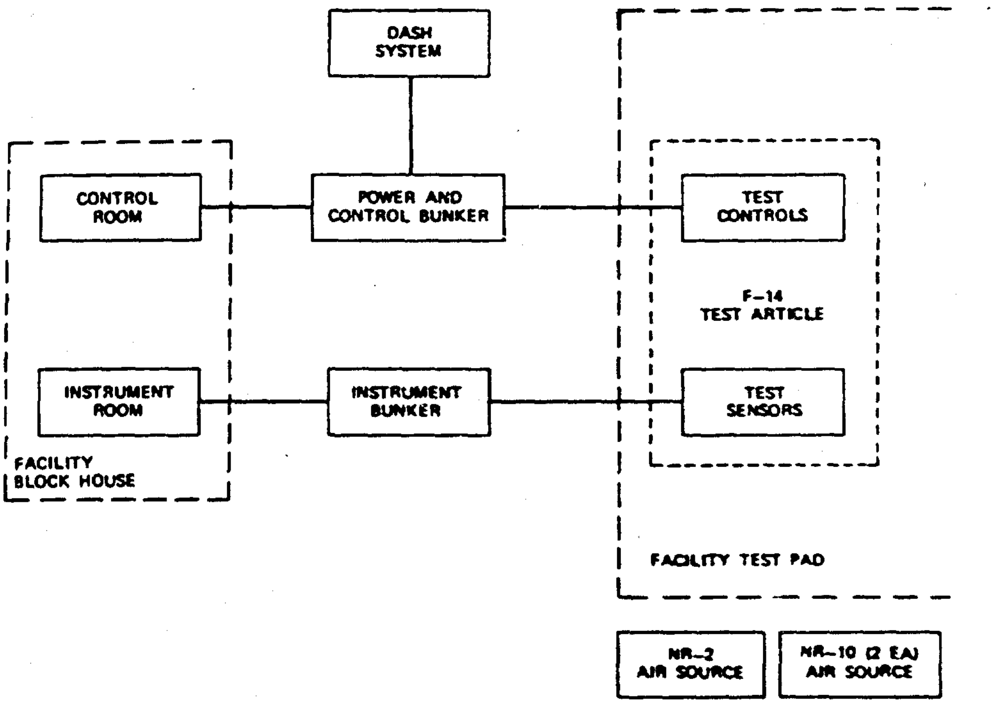

FIGURE B-11. Test Facility Instrumentation and Control, Simplified Block Diagram.

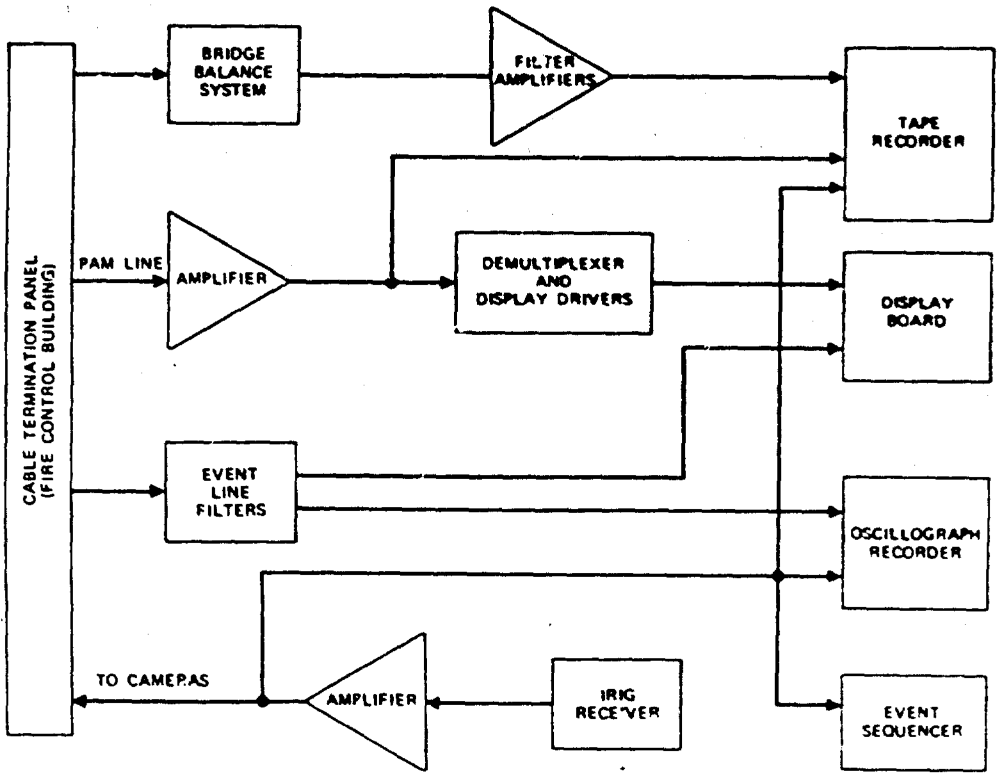

FIGURE B-12. Data System, Fire Control Building, Simplified Block Diagram.

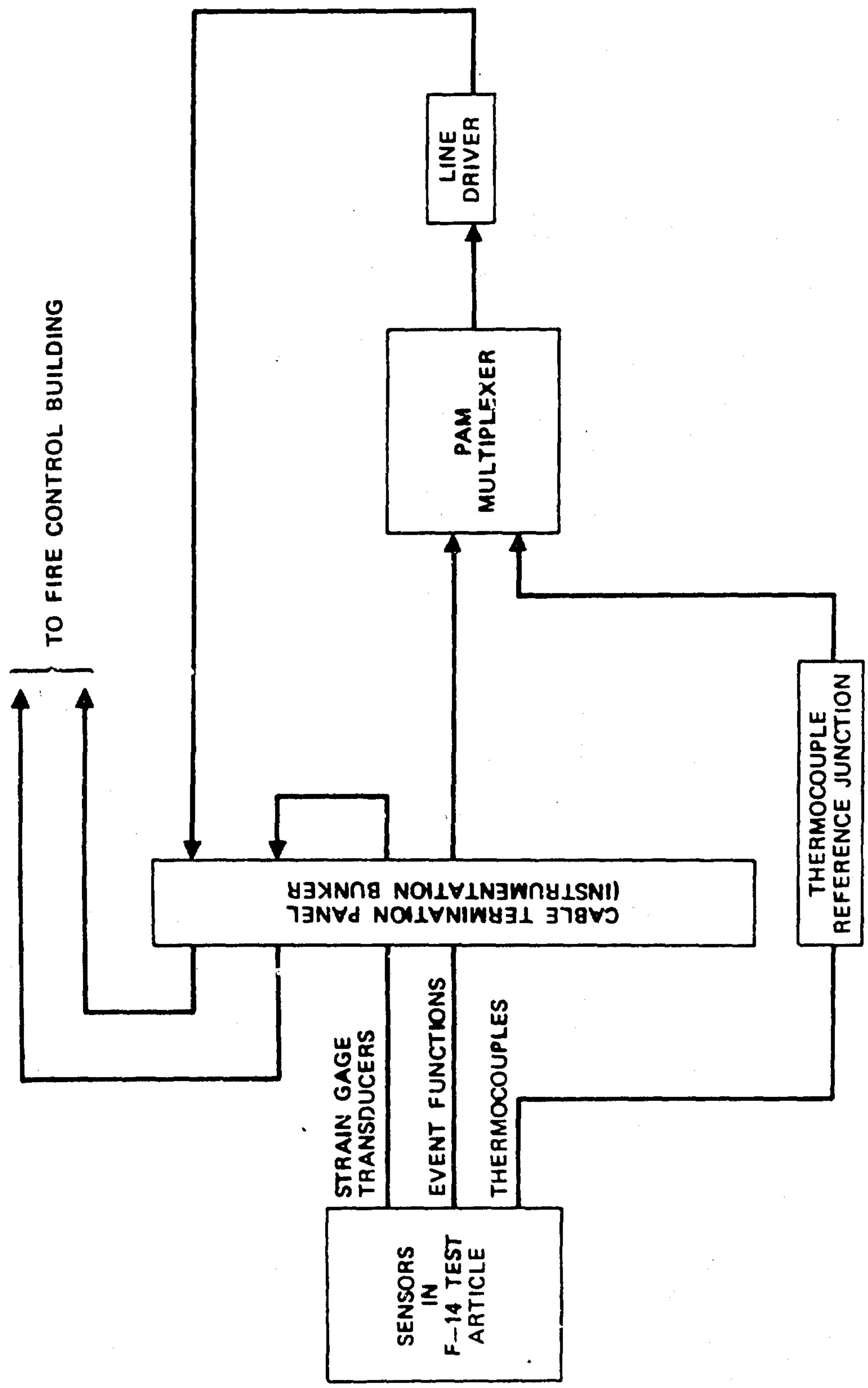

FIGURE B-13. Data System, Instrumentation Bunker. Simplified Block Diagram.

TABLE B-1. Thermocouple Listing

Channel number	Location	Function
1	Flare igniter nozzle	Data feedback
2-6	Inboard side of nacelle on 3-ft. (0.9-meter) centers starting at station (St.) 570	Fire information
7-11	Top side of nacelle on 3-ft. (0.9-meter) centers starting at St. 570	Fire Information
12-16	Outboard side of nacelle on 3-ft. (0.9-meter) centers starting at St. 570	Fire information
17	Inside nacelle cooling airflow exhaust vent	Fire information
18,19,20	Inside fuel cell 6	Fire caution
21	Inside fuel cell 8	Fire caution
22	Inside speedbrake actuator compartment	Fire caution
23	Inside speedbrake compartment	Fire caution
24	Inside boat tail	Fire caution
25	Inside stabilizer actuator compartment	Fire caution
26	Inside outboard cavity (approximately St. 660)	Fire caution
27	Inside outboard cavity (approximately St. 600)	Fire caution
28	Inside landing gear bay near inboard trunion	Fire caution
29	Inside wing sweep fairing, under side, outboard of wing pivot	Fire caution
30	Inside fuel cell 5	Test termination
31-33	Top of wing fairing by finger: aft of wing pivot; aft end of outboard over-wing fairing strake; over aft portion of overwing fairing inflation seal	Fire information
34	Top side of inlet duct (approximately St. 536)	Fire information

TABLE B-1. (Contd.)

Channel number	Location	Function
35	Top side of inlet duct (approximately St. 565)	Fire information
36	Top side of simulated nacelle wound	Fire information
37	Aft of overwing fairing hold down actuator	Fire information
38	Wing fuel vent line 6 in. (152 mm) outboard of trough fire barrier	Fire information
39	Wing fuel vent line 3 in. (76 mm) inboard of trough fire barrier	Fire information
40	Top of foam between cell 5 and inlet duct St. 536	Fire information
41	Top of foam between cell 5 and inlet duct St. 555	Fire information
42-45	Top of cell 5 approx. 6 in. (152 mm) from each corner	Fire information
46	Tube above cell 5 outboard of trough fire barrier	Fire information
47	Right control rod, center of fire barrier	Fire information
48	Center control rod, center of fire barrier	Fire information
49	Left control rod, center of fire barrier	Fire information
50	Center of upper trough, forward of wing box	Test termination
51	Center of upper trough, forward of wing box	Test termination
52	Void space right-hand side, forward of wing box	Test termination
53-55	Center of upper trough, equally spaced from approximately St. 575 to speed brake actuator compartment	Fire caution
56-57	Outboard fin skin forward and aft of nacelle cooling airflow exhaust vent	Fire information

TABLE B-1. (Contd.)

Channel number	Location	Function
58	Wing sweep bar	Fire information
59	Inside fuel cell 2	Test termination

NOTE: All thermocouples are to measure air temperature except channels 38, 39, 47-49 which measure skin surface temperature.

TABLE B-2. Instrumentation List

Channel number	Parameter	Mag tape record	F-14A display panel	Control panel display
1-59	Test volume air and surface temperatures	X	47 channels X	- -
60	Airflow static pressure, forward of wing box	X	-	-
61	Pressure differential, nacelle to accessory bay	X	-	-
62	Pressure differential, accessory bay to trough	X	-	-
63	Total fuel flow Δp	X	-	-
64	Fuel flow, after failure location, Δp	X	X	-
65	DASH airflow velocity, Δp	-	X	X
66	DASH airflow velocity, p_s	-	-	X
67	DASH airflow velocity, t	-	-	X
68	Nacelle cooling airflow Δp	X	X	-
69	Nacelle cooling airflow p_s	X	-	-
70	Nacelle cooling airflow, t	X	-	-
71	Simulated fan failure airflow, Δp	X	X	-
72	Simulated fan failure airflow, p_s	X	-	-
73	Simulated fan failure airflow, t	X	-	-

TABLE B-2. (Contd.)

Channel number	Parameter	Mag tape record	F-14A display panel	Control panel display
74	TF-30-P1 engine PLA	X	X	X
75	TF-30-P1 engine TIT	-	-	X
76	TF-30-P1 engine oil temperature	-	-	X
77	TF-30-P1 engine oil pressure	-	-	X
78	TF-30-P1 engine fuel flow	-	-	X
79	Nacelle fire detection	X	X	X
80	Accessory bay bleed air leak detector	X	X	X
81	Firewall fuel shutoff valve	X	X	X
82	Nacelle fire extinguisher (on-board)	X	X	-
83	Accessory bay fire extinguisher (on-board)	X	X	-
84	Test facility CO_2	X	X	-

TABLE B-3. Control Function List

Channel number	Parameter	Mag tape record	F-14A display panel	Control panel display
-	TF-30-P1 Throttle	-	-	X
81	Firewall fuel shutoff valve	X	X	X
68	Nacelle cooling airflow	X	X	-
71	Simulated fan failure airflow	X	X	-
-	External fuel supply	-	-	X
82	Nacelle fire extinguisher (on-board)	X	X	X
83	Accessory bay fire extinguisher (on-board)	X	X	-
1	Flare ignition	X	X	-
64	Fuel line rupture	X	X	-
84	Facility CO_2	X	X	-

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