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ARTIFICIAL AND NATURAL ICING TESTS OF AH-64 (PHASE II)

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FINAL REPORT

US ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523 - 5000

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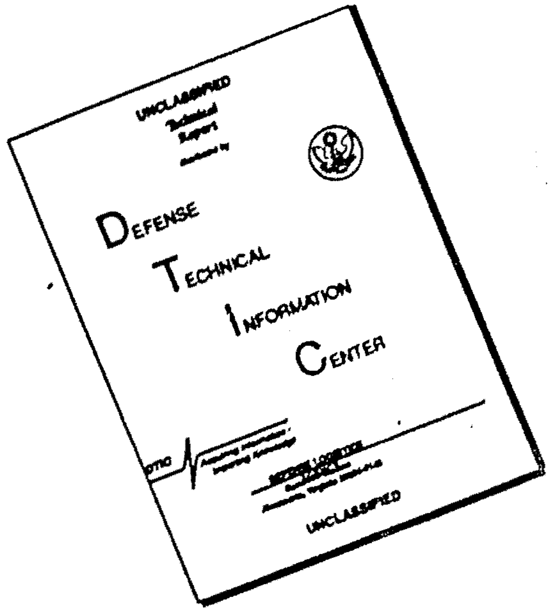

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Natural icing tests were conducted in November and December 1986 as a continuation of the icing evaluation required for the US Army Aviation Systems Command to establish a moderate icing envelope for the AH-64A Apache. The AH-64A ice protection and detection/rate measuring systems were found to be satisfactory and the aircraft demonstrated that it will have the capability to operate safely in icing conditions through the moderate level of intensity following correction of the one deficiency noted during testing. The deficiency was the high potential for engine foreign object damage due to shed ice from the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairings. Eleven shortcomings were identified. The most significant were: the susceptibility of the engines to damage due to ice shed from the external airframe components; unpredictable failure of the blade deice system due to the design/operation of the K-3 contactor relay; the potential failure of the deice controller caused by penetration of oil and water into the unit; the inadvertent unplugging of the helmet communication cord; and the poor location of the ice protection

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system advisory lights/anti-ice control panel. No degradation of aircraft handling qualities was noted as a result of aircraft ice accretion. Activation of the engine anti-ice system results in approximately 10% ~~percent~~ loss in power available and 8% ~~percent~~ decrease in maximum effective range. Keywords: Attack helicopters.

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INTRODUCTION

BACKGROUND

1. The US Army requires that the AH-64A helicopter operate safely in a moderate icing environment (ref 1, app A). The AH-64 icing program was divided into a development effort and a qualification effort. Both the development and qualification efforts used the artificial spray cloud produced by the Helicopter Icing Spray System and subsequent flight in natural icing conditions. Development and qualification tests on different individual aircraft systems were conducted simultaneously to prevent duplication of effort.

2. The AH-64 icing program began in St. Paul, Minnesota during the winter of 1982. Table 1 presents the tests required to complete development and qualification of the AH-64 anti-ice/deice systems.

Table 1. AH-64 Icing Tests

Date	Test	Results
Jan-Mar 1982	Artificial icing (development tests)	Ref 2, App A
Feb-Apr 1985	Artificial and limited natural icing (development and qualification tests)[1]	Appendix F
Nov-Dec 1985 Feb-Mar 1986	Natural icing (development and qualification)[2]	Appendix G
Nov-Dec 1986	Natural icing (qualification	This report

NOTES:

[1]Further development testing was required to integrate the LWC signal for automatic rotor blade deice system operation.
[2]Qualification not completed due to grounding of the AH-64 fleet.

The 1982 development phase of artificial icing was conducted by McDonnell Douglas Helicopter Co. (MDHC), formerly Hughes Helicopter Co., with the US Army Aviation Engineering Flight Activity (AEFA) in a supporting role (ref 2, app A). The development effort was not completed in 1982 because of problems with the automatic operation of the rotor deice system. In order to expedite the completion of icing development and qualification efforts, the US Army Aviation Systems Command (AVSCOM) tasked AEFA to conduct icing tests in conjunction with MDHC during the winter of 1985 (ref 3). The artificial icing portion of the development and qualification tests was completed in March 1985. Natural icing tests began during this icing season but could not be completed because of continued difficulties with the engine inlet liquid water content (LWC) sensor which prevented proper operation of the automatic blade deice system. Results of this test are included as appendix F.

3. AVSCOM again tasked AEFA to conduct natural icing tests with MDHC during the winter of 1985/86 (ref 4). During these tests, the engine inlet LWC sensor was replaced with a roof mounted aspirated LWC sensor. Additionally, electrical heater blankets were added to a portion of the aft nose gearbox/cross shaft fairings to prevent ice formation in the engine inlets. Grounding of the AH-64 fleet in March of 1986 prevented completion of the qualification effort. Test results for the 1985/86 icing season are presented in appendix G. Natural icing qualification tests resumed in November 1986 and were completed in December 1986.

TEST OBJECTIVES

4. The objectives of this test were to conduct natural icing flight tests of the AH-64A helicopter to:

a. Determine the effectiveness of the AH-64A ice protection and detection/rate measuring systems.

b. Determine the impact of ice accumulation on performance and handling qualities.

c. Determine the capability of the AH-64A to operate in moderate icing conditions.

DESCRIPTION

5. The AH-64A is a two-place, tandem seat, twin engine helicopter with four-bladed main and anti-torque rotors and conventional wheel landing gear. The helicopter is manufactured by MDHC and is powered by two General Electric T700-GE-701 turboshaft engines. The AH-64A has a movable horizontal stabilator. A 30mm gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has a wing with two pylons on each side for carrying

Hellfire missiles, 2.75-inch folding fin aerial rockets and/or external fuel tanks. Aircraft configurations tested were: Hellfire missiles outboard and the M-261 lightweight rocket launchers inboard; Hellfire missiles inboard and the M-261 lightweight rocket launchers outboard; and auxiliary fuel tanks inboard with the M-261 lightweight rocket launchers outboard. The test helicopter was USA serial number 84-24285 (production vehicle 145). Further description of the helicopter may be found in the system specification (ref 1), the operator's manual (ref 5), and appendix B.

6. The AH-64 anti-ice/deice systems are depicted in figure 2, appendix B. The deice system consisted of an outside air temperature (OAT) sensor, ice detector probe and signal processor unit, icing rate meter, blade deice control panel, slip rings for the main and tail rotor, and a deice controller. The main and tail rotor blades contained electrothermal resistive heating mats. Anti-ice systems were provided for sections of the windshield, pitot tubes, air data sensor, engines, engine inlets, nose gearbox/cross shaft fairings, and target acquisition designation sight/pilot night vision sensor (TADS/PNVS). The frangible HELLFIRE deice system was not installed on the test aircraft since satisfactory system performance was previously demonstrated (ref 2, app A).

TEST SCOPE

7. Previous portions of this program completed 8.3 hours of artificial and 22 hours of natural icing tests (apps F and G). The present evaluation included clear air and natural icing flight tests conducted in the vicinity of Duluth, Minnesota from 1 November through 18 December 1986. A joint MDHC/AEFA test team was used for these tests. A total of 10 flights were conducted resulting in 9.4 productive hours. Of these flights 2 were in clear air totaling 1.9 hours and 8 were in the natural icing environment totaling 7.5 hours of cloud immersion. Specific icing conditions are presented in table 1, appendix E. The aircraft configuration consisted of external fuel tanks on each inboard pylon and 2.75-inch rocket pods outboard. Anti-ice and deice systems were operated continuously while in the icing environment. Flight limitations contained in the operator's manual and the airworthiness release (ref 6, app A) were observed during testing.

TEST METHODOLOGY

8. Test methods used during artificial icing (1982 and 1985) are described in reference 2, appendix A and appendix F. Flight test data were obtained from calibrated test instrumentation and recorded on magnetic tape. Additional data were obtained using 16mm movie and video cameras on the test aircraft and both airborne and postflight still photos and video. A detailed listing of test instrumentation, test techniques and data analysis methods, and test results is contained in reference 2, appendix A and appendix F. Results of the artificial icing tests provide an essential supplement to the natural icing data in determining the capability of the AH-64A to operate in moderate icing conditions

9. Natural icing tests were conducted by flying in instrument meteorological conditions using instrument flight rules (IFR). The JU-21A scout aircraft configured with a cloud particle measuring system assisted in locating and documenting the icing conditions. Photos were taken from the JU-21A after the test aircraft exited the icing environment into visual meteorological conditions. Close coordination between air traffic control and the chase and test aircraft crews was required to find and stay in the icing environment and to implement in-flight aircraft rendezvous for photographic documentation. Procedures included radar vectoring, navigational aid holding, and block airspace assignment. Time in the clouds was limited by the availablility of the natural icing conditions and aircraft IFR fuel requirements. Video cameras were mounted on the test aircraft to document main rotor, tail rotor, engine inlet and nose gearbox/cross shaft fairing ice accretion and shedding characteristics both during and after cloud immersion.

10. Test data were recorded on magnetic tape in frequency modulation format. A detailed description of special equipment and instrumentation is provided in appendix C. Test techniques and data analysis methods are presented in appendix D. These include methods to determine cloud parameters and definitions of icing types and severities.

RESULTS AND DISCUSSION

GENERAL

11. Natural icing tests were conducted in November and December cf 1986 as a continuation of the icing evaluation required for AVSCOM to establish a moderate icing envelope, through 1.0 gm/m^3 LWC, for the AH-64A Apache helicopter. Results of the initial development testing performed in the winter of 1982 are presented in reference 2, appendix A. Results of additional artificial and limited natural icing tests conducted in February through April 1985 are presented in appendix F. Results of continued natural icing tests conducted in November/December 1985 and February through April 1986 are presented in appendix G. The qualification effort was completed in December 1986. The high potential for engine foreign object damage (FOD) due to shed ice from the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairing fairings was identified as a deficiency. Eleven shortcomings were identified. The most significant shortcomings are:

a. The susceptibility of the engines to damage due to ice shed from the external airframe components such as the canopy frames, windshield wipers, and fuselage handles/steps.

b. Unpredictable failure of the blade deice system due to the design/operation of the K-3 contactor relay.

c. The potential failure of the deice controller caused by penetration of oil and water into the unit.

d. The inadvertent unplugging of the helmet communication cord when the pilot or copilot/gunner (CPG) leans forward or turns his head to perform normal cockpit duties.

e. The poor location of the ice protection system advisory lights/anti-ice control panel.

ROTOR DEICE SYSTEM OPERATION

General

12. The AH-64A helicopter rotor deice system was previously evaluated for operational characteristics and effectiveness during 8.3 hours in the artificial icing environment and 22.0 hours in natural icing (apps F and G). Eight flights totaling 7.5 hours in natural icing were conducted during this evaluation at the conditions shown in figure 1, appendix E. The rotor deice system operational characteristics and effectiveness were satisfactory for the conditions tested.

Rotor Blade Heater-On Time

13. The AEG-Telefunken deice controller determined blade heater element on time as a function of OAT. Several modifications were made during previous testing to adjust the on time schedule and integrate the OAT signal for automatic system operation. The deice controller provided blade heater element on-times which were consistent with the design schedule and actual OAT (fig. 2, app E). No residual power required increases were observed following blade heating cycles and only a minimum increase in vibration levels occurred during rotor ice shedding. Rotor blade heater-on times are satisfactory.

Rotor Blade Heater Off-Time

14. The AEG-Telefunken deice controller determined blade heater element off time as a function of LWC. Heater-off time determines the amount of ice accreted on the rotor blades between heating cycles. This in turn affects increases in power required, vibration levels, and potential aircraft damage due to shed ice. The controller provided a blade heater-off time consistent with the design schedule in both automatic and manual modes of operation (fig. 3, app E). No unusual vibration levels occurred and power required increases of 6 to 8% indicated torque (collective fixed) were observed between heating cycles. One engine sustained FOD (para 31) due to ingestion of ice shed from an unknown source, possibly the main rotor blades. No other aircraft components were damaged during this test. The rotor blade heater off-times are satisfactory.

Ice Detection/Rate Measuring System

15. An aspirated Rosemount LWC probe (model 871ff2) was mounted on the aircraft upper fuselage (photo 1). This probe provided a LWC (icing rate) signal through a signal processor to the deice controller and to an icing rate meter on the pilot's instrument panel. Previous modifications to the ice detection/rate measuring system are described in appendix G. Further modifications prior to the beginning of this test include mounting the probe on a fiberglass ramp to reduce possible fuselage airflow interference and improving the pressurized air source to provide probe aspiration at approximately 23 pounds per square inch, gauge and 65 deg C. The JU-21A chase aircraft was used to document actual cloud conditions and verify ship system LWC indications. The ice detection/rate measuring system provided LWC readings that were consistent with the actual icing rate and is satisfactory.

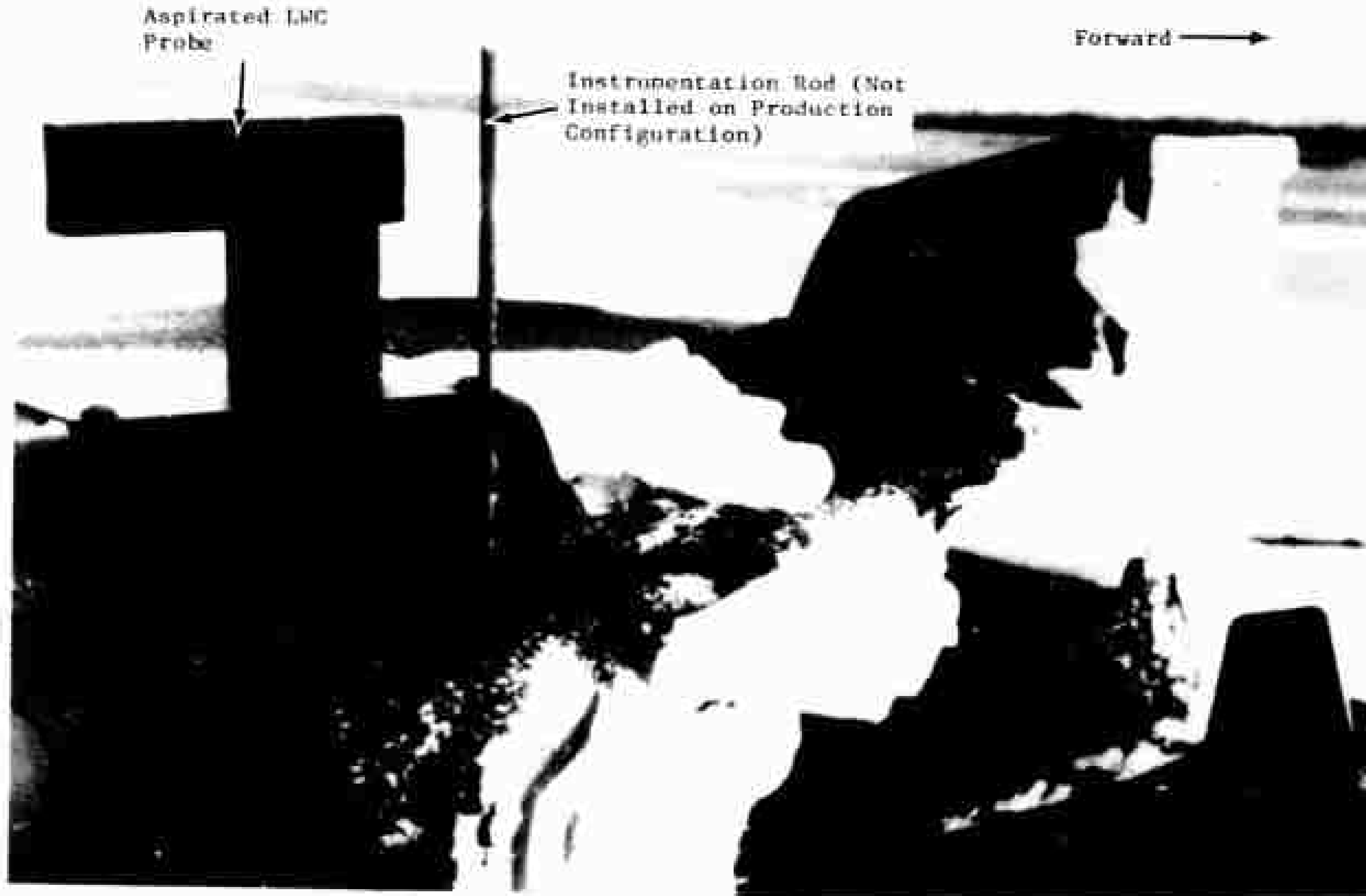


Photo 1. Liquid Water Content
Probe - Upper Right Fuselage

ANTI-ICE SYSTEMS OPERATION

General

16. The AH-64A anti-ice systems were previously evaluated in artificial and natural icing (apps F and G). Eight flights totaling 7.5 hours in natural icing were conducted during this evaluation. Specific test conditions for this evaluation are shown in table 1, appendix E. Anti-ice systems on the AH-64A helicopter protect the engines, engine inlet cowling rings, nose gearbox and cross shaft fairings, windshields, pitot-tubes, air data sensor (ADS), and TADS/PNVS. All anti-ice systems were activated prior to entering the icing environment and were operational for all icing flights. A detailed description of each system is presented in references 1 and 5, appendix A and appendix B.

Engine

17. Engine anti-icing was accomplished by a combination of hot air from the axial compressor and heat transfer from the air/oil cooler in the engine airframe. The system was controlled by a single engine anti-ice switch located on the pilot's lower left subpanel (fig. 1, app B). The engines were inspected daily and engine health indication tests were performed prior to each flight. No engine deterioration was observed and there were no indications of ice accumulation in the engine. The engine anti-ice system is satisfactory.

Engine Inlet Cowling Ring

18. Engine bleed air heated the engine inlet cowling ring to prevent ice accumulation. There were no accumulations of ice noted on the cowling ring. The engine inlet cowling ring anti-ice system demonstrated satisfactory operation.

Nose Gearbox/Cross Shaft Fairings

19. The forward nose gearbox/cross shaft fairings were anti-iced using electrical heater elements inside the fairings. These electric heaters were activated by the same two-position switch on the pilot's anti-ice panel (fig. 1, app B) that activated the engine anti-ice bleed air system. Heater blankets were added under a portion of the aft nose gearbox/cross shaft fairing when ice formation was discovered in the engine inlet following an icing encounter (para 3c(4), app G). Further modification of the aft fairing heater blankets was required during this test which eliminated small ice accretions in the engine inlets. The nose gearbox/cross shaft fairing anti-ice system with these modifications is satisfactory.

Heated Windshields

20. The two panels forming the pilot and CPG's windshield were anti-iced electrothermally. The windshield temperature was regulated between 65 and 85 deg F (18 to 29 deg C) which kept the heated windshields ice free. The windshield anti-ice system is satisfactory.

Pitot Tubes

21. The dual, wing mounted pitot tubes were anti-iced electrically and controlled by a single switch on the pilot's anti-ice control panel. No ice accretion was observed on the pitot tubes (photo 2). The pitot tube anti-ice system is satisfactory.

Air Data Sensor

22. The Air Data Sensor (ADS) was located above the main rotor and attached to a stand pipe through the main rotor mast. The rotating hub and arms of the ADS were electrically heated. The same switch that activated pitot heat activated the ADS anti-ice system. Ice did not accrete on the heated portion of the ADS (photo 3). On one occasion after departing icing conditions, an ADS failure occurred probably due to a weak heater. The ADS was replaced and no further incidences occurred. The ADS anti-ice system is satisfactory.

Target Acquisition Designation Sight/Pilot Night Vision Sensor

23. The TADS/PNVS anti-ice provisions included window, window frame, and selected turret surface panel heating. During natural icing tests the TADS and PNVS were coupled to the pilot's and CPG's helmets to provide turret movement which prevented loss in range of motion due to ice build-ups. The final TADS/PNVS anti-ice system configuration tested (described in ref 9, app A) is satisfactory.

UNPROTECTED AIRCRAFT COMPONENTS

General

24. Ice accretion and shedding characteristics of the unprotected aircraft components were previously evaluated throughout the tests reported in appendixes F and G and at the test conditions shown in table 1, appendix E. Ice formed on all unprotected stagnation areas and sharp protrusions from the airframe and main rotor. Large ice formations accreted on the canopy frames,

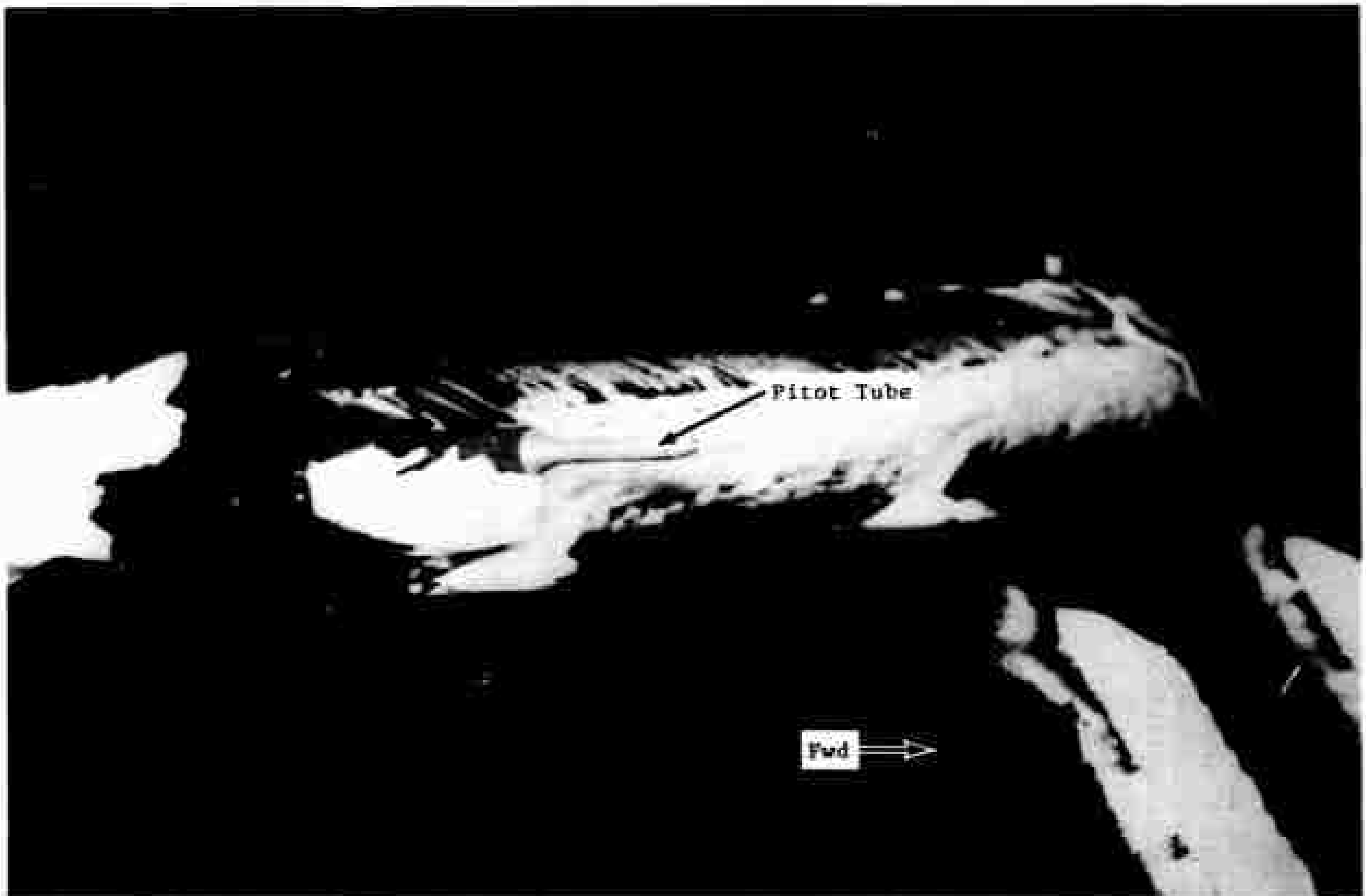


Photo 2. AH-64A - Left Wing
Conditions: Avg LWC = 0.43 gm/m^3
Avg OAT = $-11.0°C$
Time in Cloud = 70 minutes

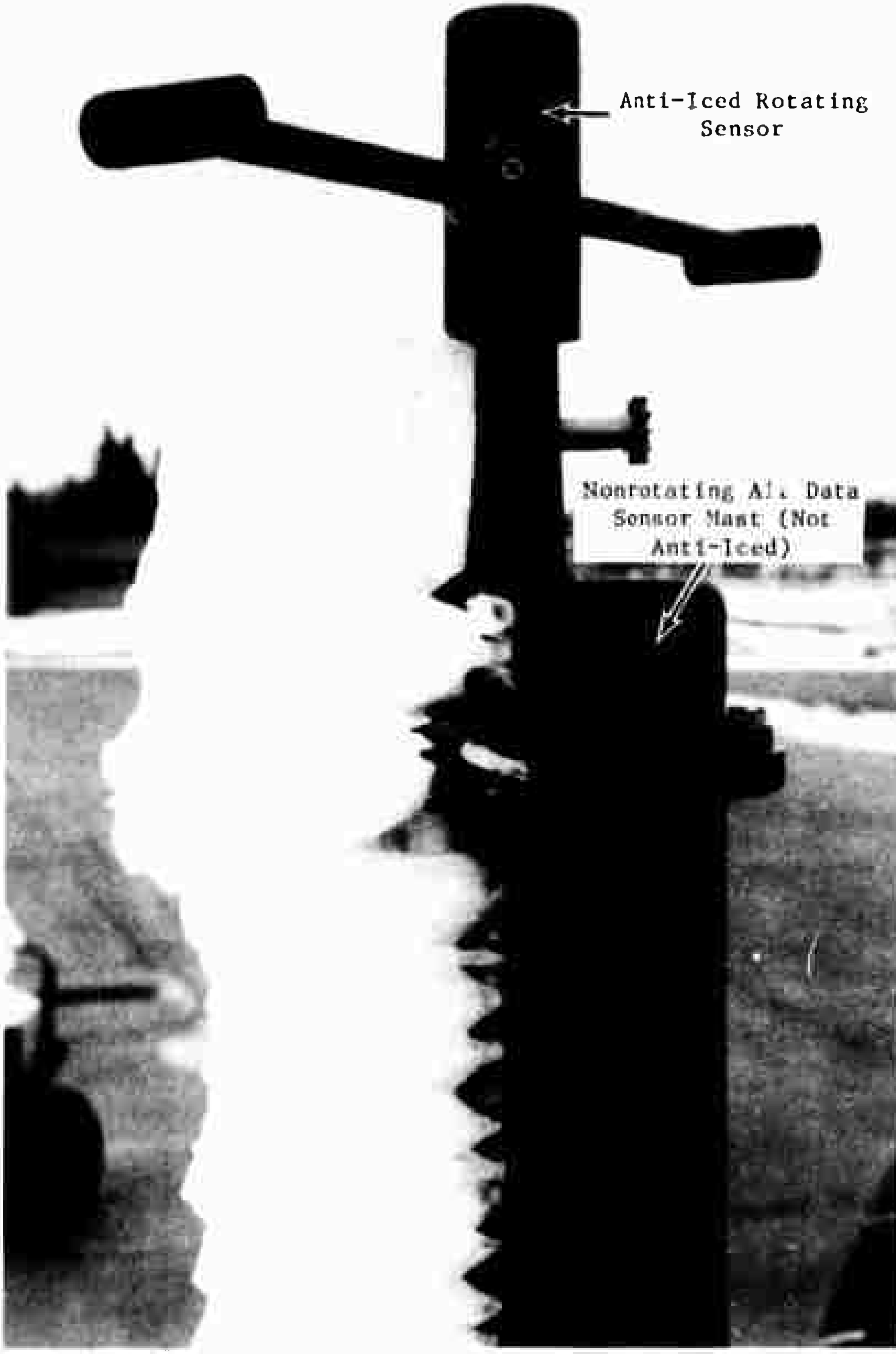


Photo 3. AH-64A - Air Data Sensor

Conditions: Avg LWC = 0.43 gm/m^3
Avg OAT = -8.5°C
Time in Cloud = 83 minutes

windshield wipers, fuselage handholds/steps and on the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairings which create the potential for engine FOD and/or aircraft damage due to shed ice. The high potential for engine FOD due to shed ice from the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairing is a deficiency.

Main Rotor

25. Minimal amounts of ice accumulated on the unprotected blade surfaces inboard of the heater mats and on the blade retention mechanisms. No restriction of any movable component on the main rotor head was noted. No aircraft damage resulted from ice shedding characteristics of the unprotected portion of the main rotor and rotating components. The ice accretion and shedding characteristics of the unprotected main rotor components are satisfactory.

Stabilator

26. Minor ice build-ups observed on the stabilator leading edge created no problems. Minor paint chips and dents occurred on the upper surface of the left stabilator due to tail rotor shed ice but were not considered a problem. The ice accretion and shedding characteristics of the stabilator are satisfactory.

Fuselage

27. Typical fuselage ice accretions are shown in photo 4, enclosure 6 to appendix F. Large accumulations of ice were noted on the wing leading edges, nose, handholds/steps, landing gear, landing gear struts, and surface irregularities on the fuselage but created no operating difficulties. All windows, doors, and access panels remained functional after icing encounters. Ice accretion and shedding from the right and left side fuselage handholds/steps (photo 4) create the potential for engine FOD (see para 31) but no damage was specifically documented as being due to ice shed from these areas. Future aircraft designs should incorporate recessed handholds/steps to minimize ice accretion and FOD potential.

Unprotected Windows

28. No ice protection was provided for any window panels except the two electrically heated forward facing windows dicussed in paragraph 20. Minimal ice accretions on the unprotected window areas created no obstruction to the field of view. The ice accretion and shedding characteristics of the unprotected windows are satisfactory.

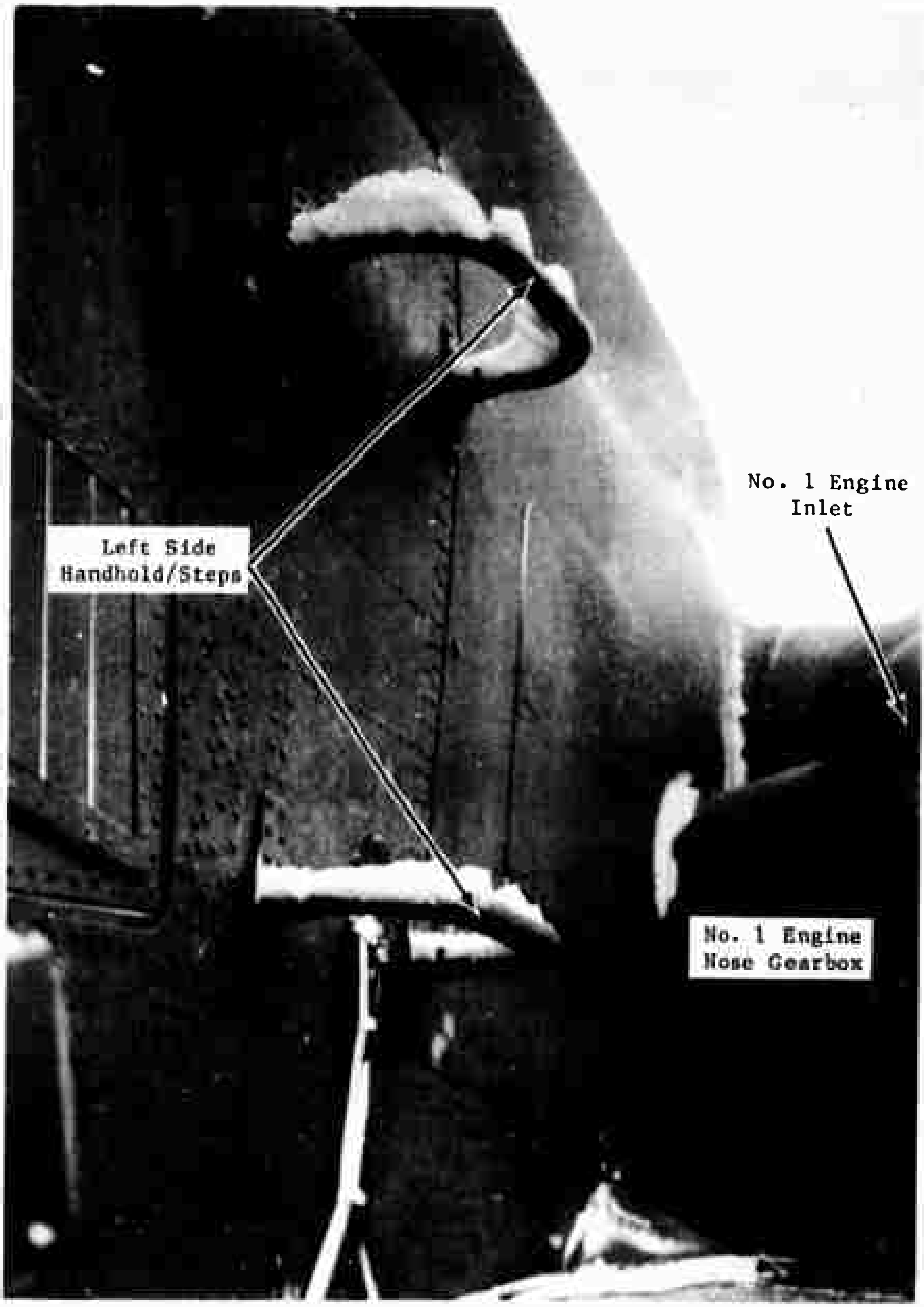


Photo 4. AH-64A - Left Side Fuselage

Conditions: Avg LWC = 0.28 gm/m^3
Avg OAT = -11.0°C
Time in Cloud = 51 minutes

Canopy Frames

29. The framework supporting the essentially flat plate canopy was not anti-iced. The forward canopy frames accreted large quantities of ice (photo 5) which could break away during flight and damage aircraft components or result in engine ice ingestion. No damage was specifically documented as being due to ice shed from the canopy frames. Future aircraft designs should incorporate canopy frames which minimize ice accretions.

Windshield Wipers

30. Large ice formations on both the pilot's and CPG windshield wipers (photo 5) could break away resulting in aircraft damage or engine ice ingestion. No damage was specifically documented due to ice shed from the wipers. The wipers were deliberately not activated in flight to prevent any damage. Future aircraft designs should incorporate windshield wipers which minimize ice accretions.

Engine Ice Ingestion

31. The No. 2 engine sustained FOD during one natural icing flight. No unusual cockpit indications were observed. Post-flight inspection revealed a 1/2 inch curl in a first stage compressor blade. A video camera mounted to observe one fourth of the No. 2 engine inlet did not reveal when the damage occurred. The most probable sources of the ingested ice are the handhold on the fuselage right side, windshield wipers, canopy frames, or the nonstandard camera mounted on the right forward avionics bay. The susceptibility of the engines to damage due to ice accretion and shedding from the external airframe components is a shortcoming.

32. Ice accretion on the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairings was previously documented during artificial icing tests (photo 6, enclosure 6 of appendix F). Ice accreted in this area is immediately in front of the engine inlet and presents a high probability of damage to the engine. No permanent modification to correct this problem has yet been implemented and the mismatch has been observed on most production aircraft. The fairing edges were smoothed during this test to eliminate the mismatch and prevent engine FOD. An example of the mismatch present with the most recent fairing configuration is shown in photo 6. The high potential for engine FOD due to shed ice from the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairing is a deficiency.

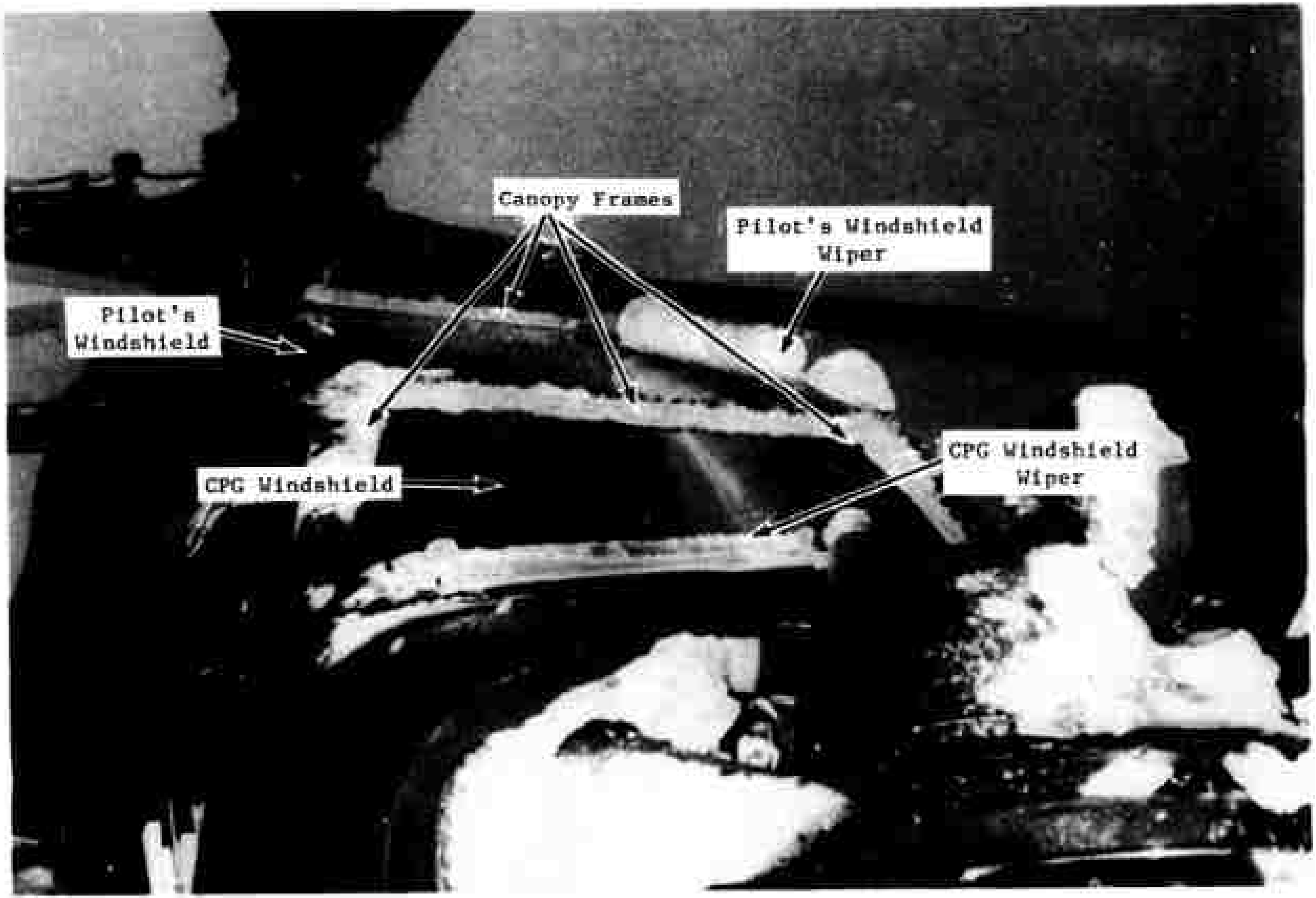


Photo 5. Canopy Frames and Windshield Wipers

Conditions: Avg LWC = 0.48 gm/m^3
Avg OAT = -9.0°C
Time in Cloud = 30 minutes

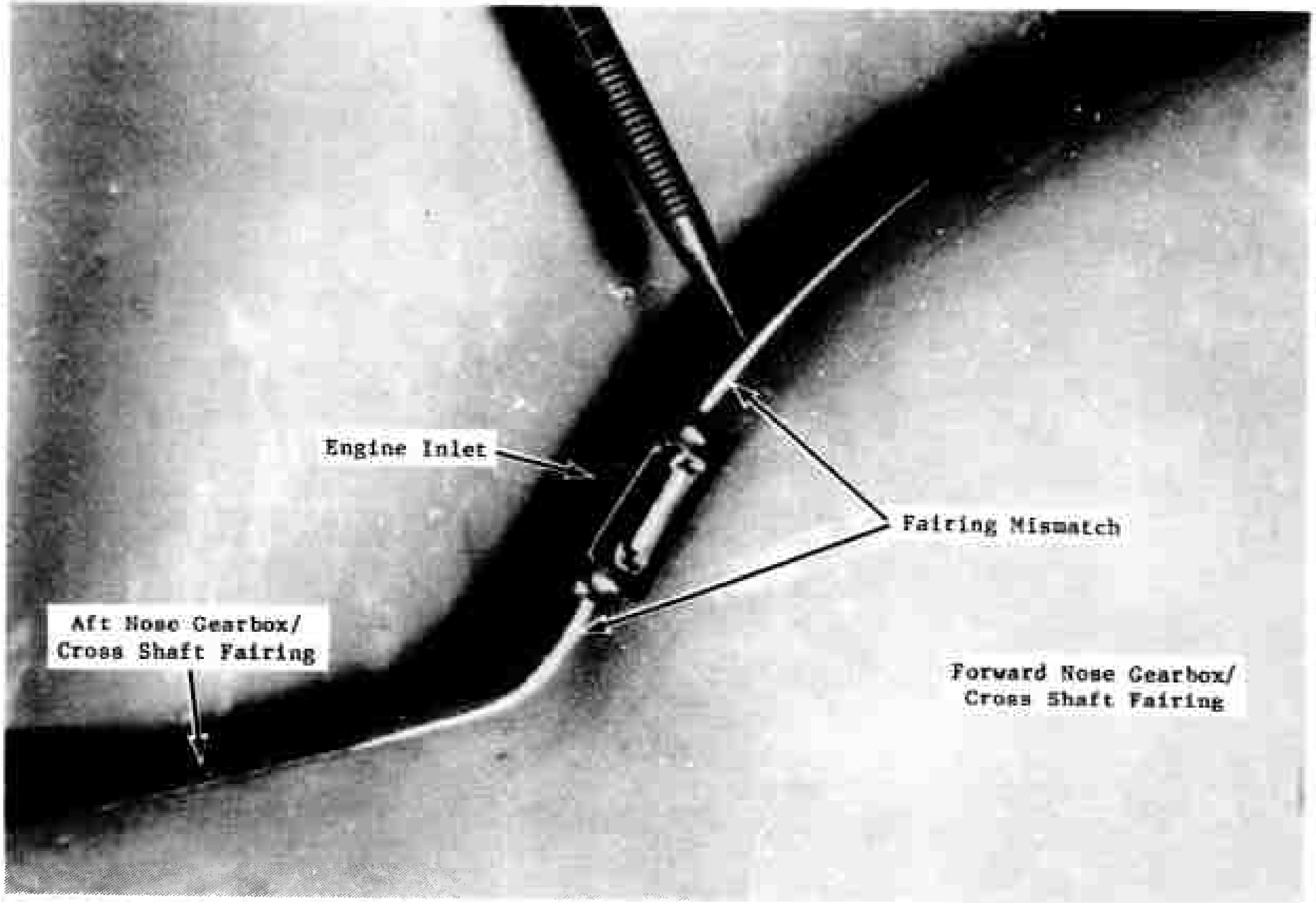


Photo 6. No. 1 Engine Inlet

Antennas

33. The ice accretion and shedding characteristics of the aircraft antennas were evaluated throughout these tests. Many AH-64A antennas are flush mounted and thus accrete little if any ice. Exceptions to this are the transponder antenna (cabin overhead), the radar jammer receiving antenna (upper fuselage), radar jammer transmitting antenna (TADS/PNVS assembly), and the two radar warning antennas located on the front of each forward avionics bay. No degradation in radio transmission or reception was noted in any aircraft systems, however, no specific tests were conducted to evaluate these characteristics. Ice shed from these antennas presented no operational difficulties. The ice accretion and shedding characteristics of the aircraft antennas are satisfactory. A study should be conducted to determine the possible degradation of transmission or reception of the radar jammer and radar warning antennas with ice accreted on their forward face.

M-130 Chaff Dispenser

34. The ice accretion and shedding characteristics of the M-130 chaff dispenser were evaluated throughout these tests. One dispenser was mounted on the left side of the tailboom. The M-130 system was not operated during this evaluation. No ice accretion or subsequent sheds were observed which would interfere with operation of the system during or after an icing encounter. The ice accretion and shedding characteristics of the M-130 chaff dispenser are satisfactory.

ALQ-144 Infrared Countermeasures Device

35. The dummy ALQ-144 installed on the test aircraft was identical to that used in previous tests and described in appendix F. The ice accretion and shedding characteristics of the ALQ-144 are satisfactory.

Hellfire Missile System

36. The frangible Hellfire deice system described in appendix B was not installed on the test aircraft since satisfactory system performance was previously demonstrated (ref 2, app A). The ice accretion and shedding characteristics of the Hellfire missile system are satisfactory.

Wing Pylon Articulation

37. Satisfactory performance of the wing pylon articulation system was previously demonstrated (app F). No restriction of pylon travel was noted during these tests.

Area Weapon System

38. The ice accretion and shedding characteristics of the 30mm chain gun area weapon system were evaluated in natural icing conditions. Weapons firing was not attempted following flight in icing conditions. Although live fire operations were not performed, the accreted ice did not hamper the traversing operation of the weapon. The gun was traversed in-flight through its full range in azimuth and elevation to verify unrestricted motion. A muzzle protector made of cloth prevented ice build-ups inside the barrel. An environmental cover (photo 7) was installed on the ammunition chute and turret assembly. This dual purpose cover prevented both ice accumulation inside the chute which would jam ammunition feed and cold air from entering the aft cockpit through the turret assembly. An additional seal was installed to improve the seal between the cover and the lower fuselage (photo 8). The ice accretion and shedding characteristics of the area weapon system are satisfactory. The production configuration of the area weapon environmental cover should incorporate a seal between the cover and fuselage similar to the one used on the test aircraft.

M-261 Lightweight Rocket Launcher

39. The ice accretion and shedding characteristics of the M-261 lightweight rocket launchers were evaluated during this test. Protection covers identical to those previously described and tested (app F) were installed on the launchers. Ice accreted on the protective covers as shown in photo 9. Rocket firing was not attempted following an icing encounter. The ice accretion and shedding characteristics of the M-261 lightweight rocket launcher are satisfactory. A test should be conducted to determine the consequences of the M-261 protective covers departing the launcher, with and without accumulated ice.

Proposed Wire Strike Protection System

40. Installation of the proposed Wire Strike Protection System (WSPS) would mount wire cutters on the front of the aircraft and the upper and lower fuselage. The proposed WSPS should be evaluated in natural icing conditions to determine possible impact on existing anti-ice/deice systems and the potential for aircraft damage (i.e., engine FOD) due to the shed ice.

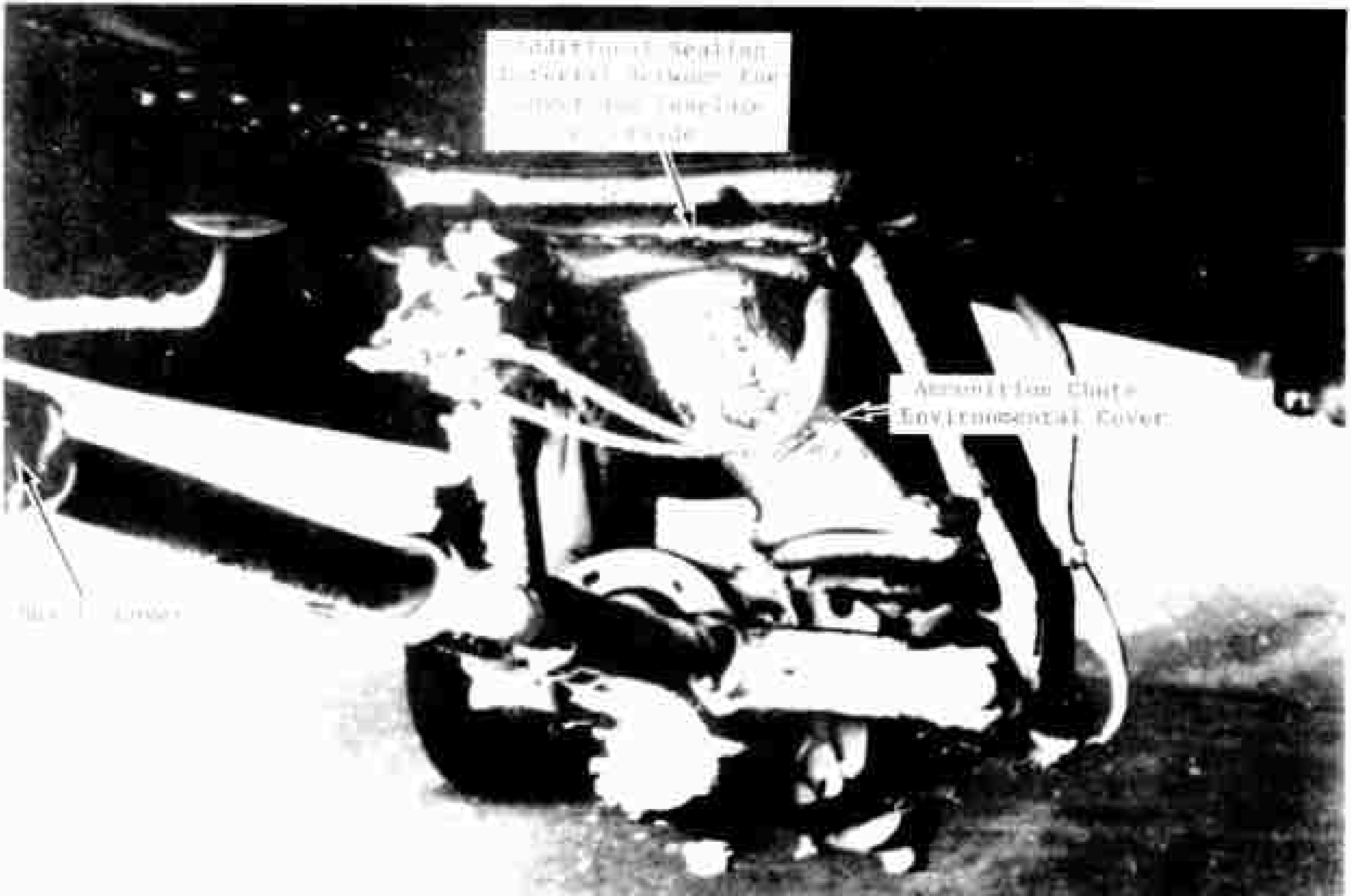


Photo 7. 30mm Area Weapon
Conditions: Avg LWC = 0.28 gm/m^3
Avg OAT = -11.0°C
Time in Cloud = 51 minutes

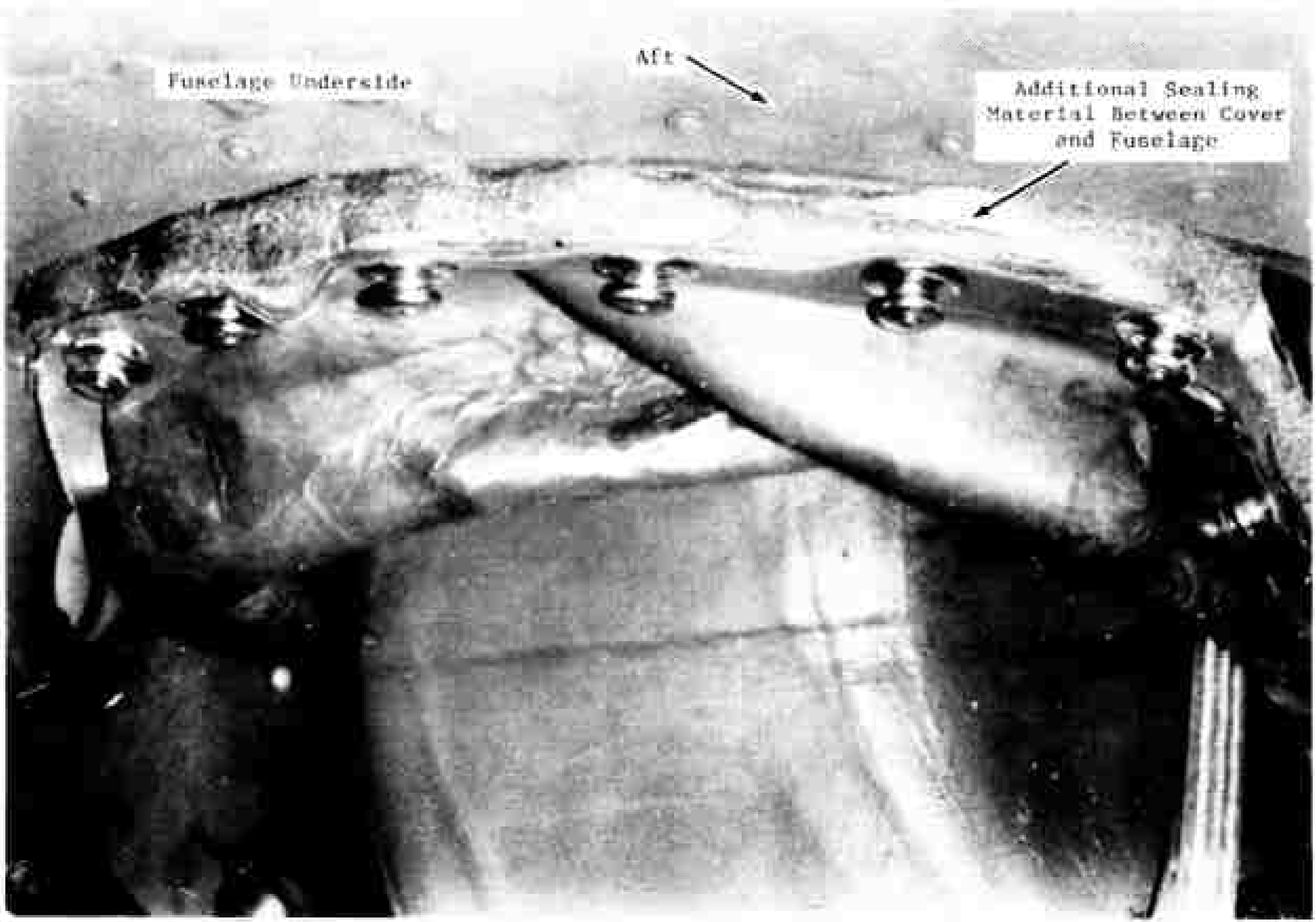


Photo 8. 30mm Ammunition Chute Cover - Upper Portion

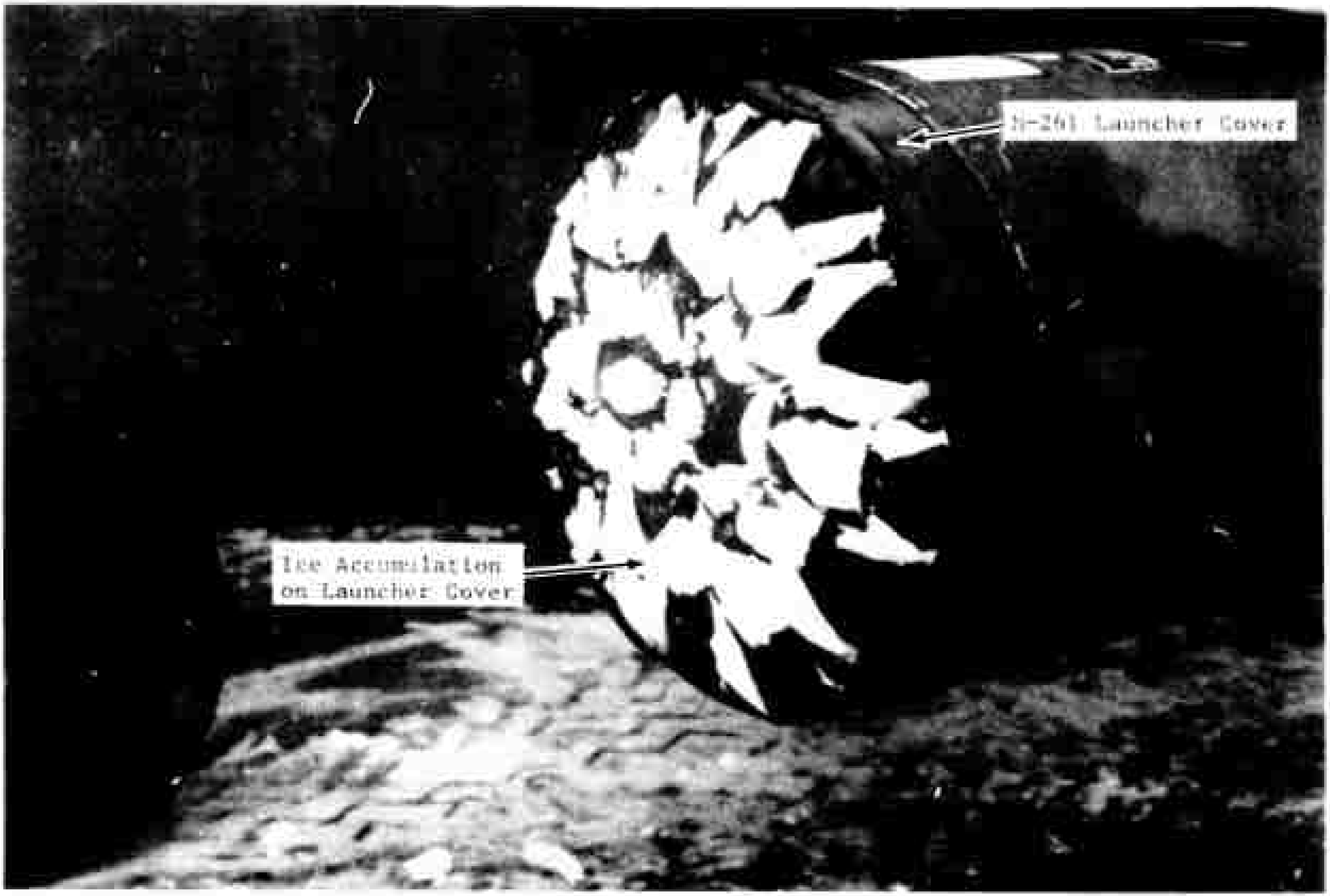


Photo 9. Left Wing Store - M-261 Lightweight Rocket Launcher

Conditions: Avg LWC = 0.33 gm/m^3
Avg OAT = -14.0°C
Time in Cloud = 60 minutes

PERFORMANCE

Level Flight Performance

41. Level flight performance characteristics of the AH-64A helicopter were evaluated at the specific natural icing test conditions listed in table 1, appendix E. Collective control position was fixed at pre-immersion trim position, altitude was maintained and airspeed was allowed to vary as necessary during the encounter. Indicated power required increases between blade deice cycles ranged from 6 to 8 percent torque. The level flight performance characteristics of the AH-64 during flight in icing conditions are satisfactory.

Power Loss with Operation of Anti-Ice Systems

42. The engine performance characteristics were recorded with the engine bleed air anti-ice system OFF and ON for comparison during previous testing (figs. 7 and 8, enclosure 4 of appendix F). An approximate 50 deg C increase in measured gas temperature was observed with anti-ice system use. The power available losses with activation of the anti-ice systems were significant. A note should be added to the operator's manual performance charts to indicate that an approximate 10 percent loss in power available and 8 percent decrease in maximum effective range occur when the engine anti-ice system is activated.

HANDLING QUALITIES

43. A qualitative handling qualities evaluation was performed during natural icing flights at the conditions listed in table 1, app E and during previous testing described in appendixes F and G. The evaluation was accomplished by performing typical instrument flight maneuvers with and without ice on the aircraft. No degradation of aircraft handling qualities was noted as a result of aircraft ice accretion.

44. General handling qualities were evaluated during the entire icing evaluation. With the attitude hold system engaged, the pilot was required to make continuous lateral and longitudinal control inputs during instrument flight to control pitch and roll attitudes. Additionally, uncommanded pitch, roll, and yaw attitude response occurred when the momentary force trim interrupt was engaged due to the operational characteristics of the digital augmentation system (previously reported in ref 10, app A). These aircraft characteristics increased the pilot workload. Future models of the AH-64 should incorporate an improved attitude hold system to reduce pilot workload during instrument flight.

VIBRATION

45. The aircraft vibration characteristics were monitored throughout these evaluations. Qualitative crew comments were compiled during the icing immersions and flights to home base with residual ice. No significant increase in vibration levels was noted. The airframe vibration characteristics resulting from flight in icing conditions are satisfactory.

RELIABILITY AND MAINTAINABILITY

Tail Rotor Elastomeric Bearings

46. The four-bladed, semi-rigid teetering tail rotor has four elastomeric bearings which connect each delta hinged hub assembly to a titanium fork and provide the teetering axis for each pair of blades. Postflight inspection revealed three elastomeric bearings debonded from the fork assembly following a natural icing flight at -11 deg C. No unusual in-flight aircraft response was noted as a result of elastomeric debonding. Debonding of the tail rotor elastomeric bearings in the cold weather environment (0 to -20 deg C) is a shortcoming.

Fault Detection/Location System

47. The Fault Detection/Location System (FD/LS) monitors the blade deice subsystems. These are the tail rotor heaters, main rotor heaters, distributor, and power controller. When AC electrical power is applied and the blade deice switch is in the OFF position, the FD/LS displays NO-GO (fail) messages for each subsystem, regardless of actual system status. These fail messages are continuously displayed when the blade deice control switch is in the OFF position and requires the crew to scroll through several pages of displayed NO-GO messages to identify the status of other systems. Additionally, display of NO-GO messages with the system turned OFF is contrary to the design philosophy of other anti-ice/deice systems. The continuous FD/LS should be modified to display NO-GO status messages only when the blade deice switch is in the ON position and a subsystem has failed. NO-GO status messages for the blade deice subsystems resulting from the blade deice switch being placed in the OFF position are a shortcoming.

Main and Tail Rotor Deice Controller

48. The main and tail rotor deice controller is mounted on the forward transmission deck (photo 10). Cooling vents and a fan

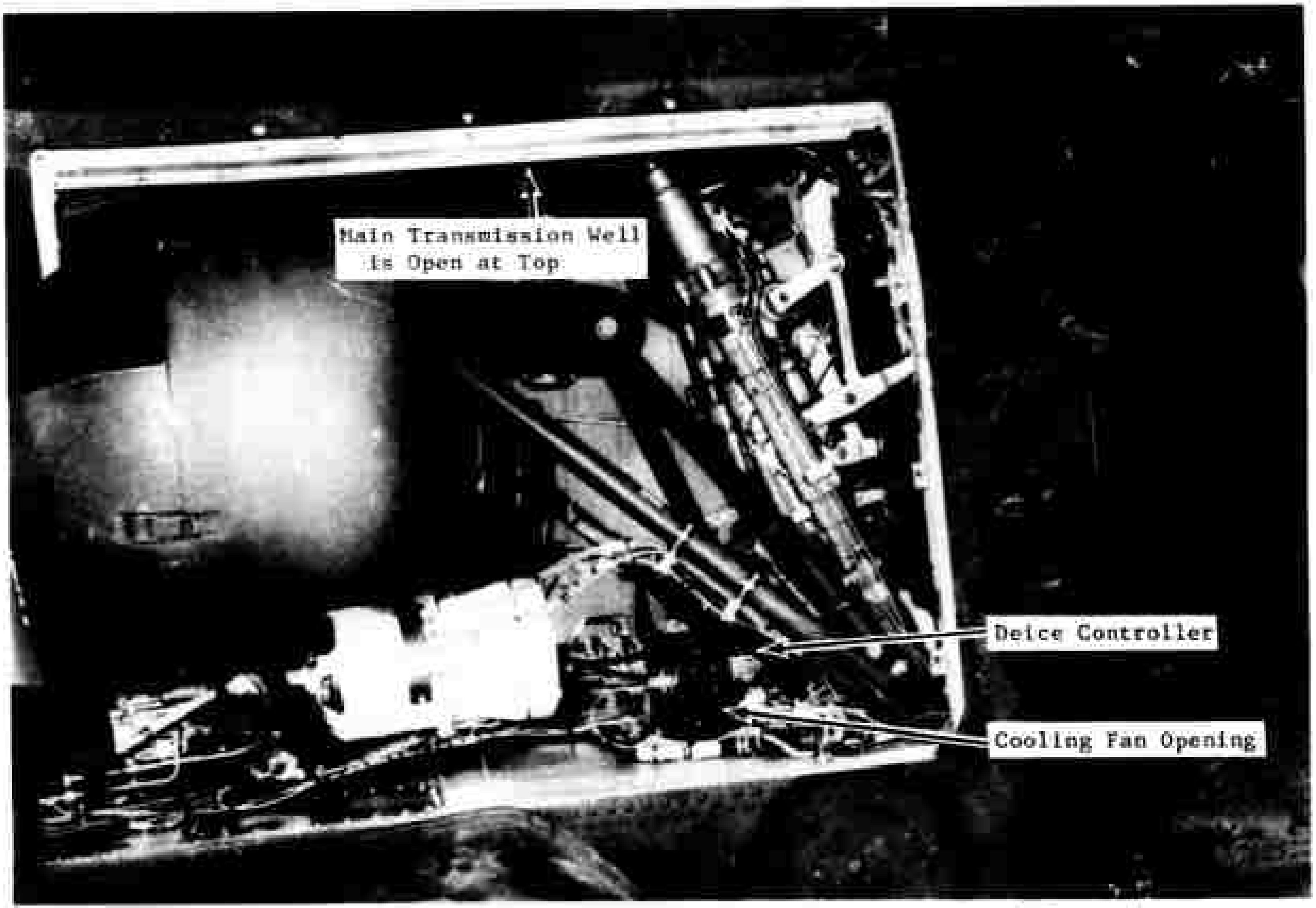


Photo 10. Fuselage Right Side - Main Transmission Access Panel Removed

inlet at the ends of the controller case (photo 11) allow oil, hydraulic fluid, and water to penetrate the unit. Following a flight in natural icing, the deice controller was removed from the aircraft and water was discovered inside the unit. The potential for failure of the deice controller caused by penetration of oil and water into the unit is a shortcoming.

49. The AEG-Telefunken controller is designed to prevent a blade heater element from exceeding 55 deg C. The controller measures blade element temperature as a function of electrical resistance during element heating and requires the controller to be "trimmed" for the electrical resistance characteristics of each blade/blade element. The controller potentiometers must be adjusted each time a main or tail rotor blade is changed (previously reported in app F). The required peculiar ground support equipment may not be available at unit level. Since main and tail rotor blade changes are a unit level maintenance function, this will cause a system maintainability problem. The requirement to manually adjust the deice controller to the rotor system electrical resistance characteristics following a main or tail rotor blade change remains a shortcoming.

K-3 Contactor Relay

50. The K-3 contactor relay (fig. 3, app B) monitors the blade deice system power source, normally the No. 2 generator, and is designed to switch to the No. 1 generator if an electrical current variation occurs. Switching power sources results in a momentary break in electrical power and a blade deice system failure. During several flights, the blade deice system failed when there was no apparent malfunction of the generators or deice controller. Normal system operation was regained by recycling the blade deice control switch. System analysis revealed the K-3 contactor relay was the most probable cause of failure. The unpredictable failure of the blade deice system due to the design/operation of the K-3 contactor relay is a shortcoming.

External Auxiliary Fuel Tanks

51. External auxiliary fuel tanks were installed on the test aircraft (photo 12). Fuel was transferred from the external tanks by means of the pressurized air system. When the aircraft is shutdown, the external tanks remain pressurized. When the aircraft was hangared for several days with the tanks pressurized, fuel began leaking from the tank access panels (photo 13) and fuel line fittings. The only way to vent pressure from the external tanks was to open the fuel filler caps following each flight. The tendency of the external auxiliary fuel tanks to

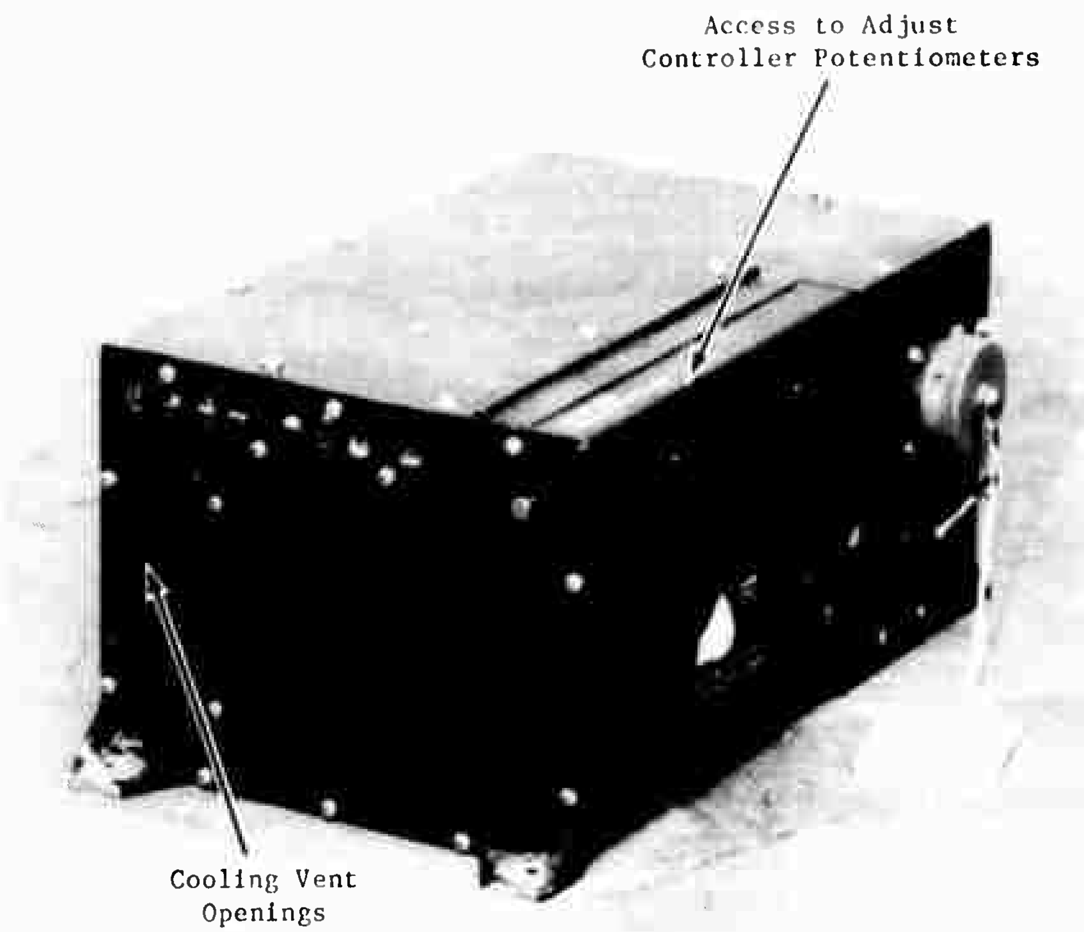


Photo 11. AH-64A Deice Controller

Photo 12. AH-64A - Front View

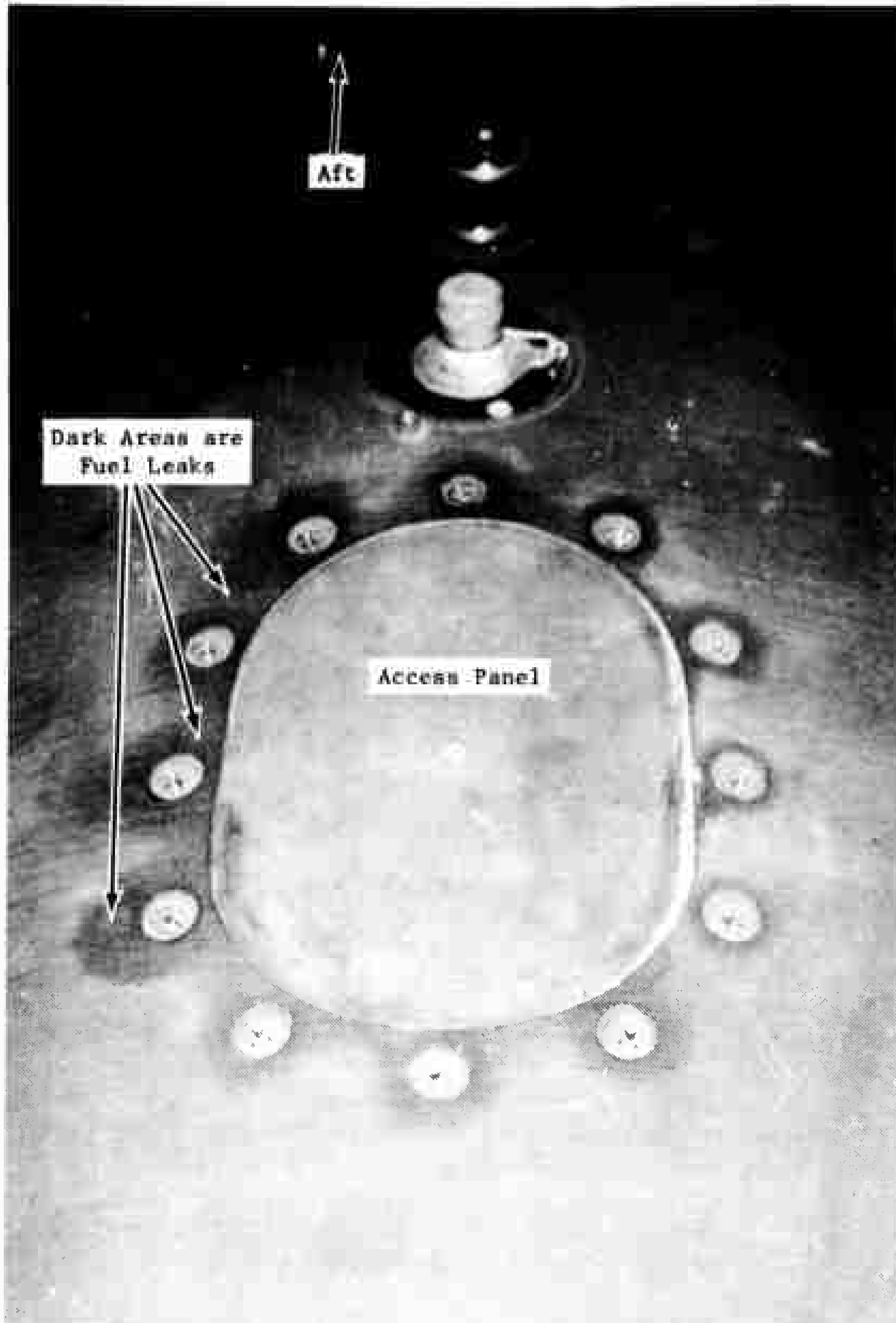


Photo 13. Auxiliary Fuel Tank - Top View

leak when the aircraft is stored after normal shutdown is a shortcoming.

Internal Laser Boresight

52. An internal laser boresight of the TADS assembly was attempted following a cold soak of the test aircraft at an average temperature of -14 deg F for 14 hours. This test was performed during February 1986 and the results were not included in appendix F. The test was performed by starting the auxiliary power unit and allowing the aircraft systems to run for 25 minutes. The first portion of the boresight was successfully completed using the medium field of view. When the final boresight sequence was attempted in the narrow field of view, there was no indication of the laser firing. Internal boresighting was attempted following two separate cold soak tests with the same results. The cause of the failure to internally boresight has not been identified. The failure of the TADS to successfully perform internal laser boresight following cold soak at -14 deg F is a shortcoming.

HUMAN FACTORS

General

53. Human factors were evaluated during this test and previous evaluations reported in appendixes F and G. The cold aft cockpit environment previously reported as a deficiency has been corrected by the installation of a production environmental ammunition chute and turret assembly cover (para 33). The poor location of the ice protection system advisory lights and control panel remains a shortcoming as prevously reported. Two additional shortcomings were identified.

Helmet Communication Cord

54. The production configuration position of the aircraft communication cord connector has been modified since the previous evaluation reported in appendix F. During flight, the pilot and CPG inadvertently unplugged their helmet communication cord from the aircraft connector by simply turning their heads to the right or leaning forward to perform normal cockpit duties (photo 14). The integrated helmet and display sight system (IHADSS) cord remained connected. Both the pilot and CPG experienced this loss of communication capability on numerous occasions during instrument flight. The inadvertent unplugging of the helmet communication cord is a shortcoming.

Photo 14. Pilot Station - Left Side View

Ice Protection Systems Advisory Lights/Control Panel Location

55. The pilot's anti-ice control panel (fig. 1, app B) was located at the aft end of the pilot's lower left console. Two advisory lights (deice system ON and engine inlet anti-ice system ON) were located on this panel and are not in the pilot's normal instrument scan pattern. Additionally, the pilot is required to memorize switch locations since it is difficult to simultaneously fly the aircraft and read the switch labels. The ice protection system advisory lights/control panel should be located in the pilot's normal instrument scan pattern. The poor location of the ice protection systems advisory lights and anti-ice control panel remains a shortcoming.

External Auxiliary Fuel Tanks

56. External auxiliary fuel tanks were installed on the test aircraft at the inboard wing stores location (photo 12). A fuel drain located on the bottom aft end of the tank is the standard type drain used on several different auxiliary fuel tanks in the Army inventory. Ground maintenance personnel are required to depress a Phillips head plunger in the center of the drain assembly to obtain a fuel sample. It is difficult for a crewmember to perform this task without getting fuel on his hands which creates the potential for cold weather injury. The lack of an adequate drain valve on the external auxiliary fuel tank creating the potential for cold weather injury is a shortcoming.

Operator's Manual

57. A significant number of modifications were made to the AH-64 helicopter during the icing tests. This information and recommended procedures will be forwarded to the Program Manager's Office under separate cover for review. The operator's manual should be updated to reflect the latest anti-ice/deice systems configuration, recommended procedures for instrument flight, and recommended procedures for flight in icing conditions.

CONCLUSIONS

GENERAL

58. The following general conclusions were reached as a result of the artificial and natural icing tests of the AH-64 helicopter.

a. One deficiency, previously reported, was still present.

b. Eleven shortcomings were found, three of which were previously reported.

c. Forty equipment performance reports/test incident reports were submitted (February 1985 - December 1986).

DEFICIENCY

59. The high potential for engine FOD due to shed ice from the mismatch at the joint between the fore and aft nose gearbox/cross shaft fairing is a deficiency (para 32). See appendix D for the definition of a deficiency.

SHORTCOMINGS

60. The following shortcomings have been identified and are listed in relative order of importance. See appendix D for the definition of a shortcoming.

a. The susceptibility of the engines to damage due to ice accretion and shedding from external airframe components (para 31).

b. The unpredictable failure of the blade deice system due to the design/operation of the K-3 contactor relay (para 50).

c. The potential failure of the deice controller caused by penetration of oil and water into the unit (para 48).

d. The inadvertent unplugging of the helmet communication cord (para 54).

e. The poor location of the ice protection systems advisory lights and anti-ice control panel (para 55).

f. NO-GO status fail messages for the blade deice subsystems resulting from the blade deice switch being placed in the OFF position (para 47).

g. The requirement to manually adjust the deice controller to the rotor system electrical resistance characteristics following a main or tail rotor blade change (para 49).

h. Debonding of the tail rotor elastomeric bearings in the cold weather environment (0 to -20 deg C) (para 46).

i. The lack of an adequate drain valve on the external auxiliary fuel tank creating the potential for cold weather injury (para 51).

j. The tendency of the external auxiliary fuel tanks to leak when the aircraft is stored after normal shutdown (para 51).

k. The failure of the TADS to successfully perform internal laser boresight following cold soak at -14 deg F (para 52).

RECOMMENDATIONS

60. The following recommendations were made:

a. Release the AH-64A helicopter for flight in the moderate icing environment after correction of the deficiency.

b. Correct the shortcomings as soon as practical.

c. Future aircraft designs should incorporate recessed handholds/steps to minimize ice accretion (para 27).

d. Future aircraft designs should incorporate canopy frames which minimize ice accretion (para 29).

e. A study should be conducted to determine the possible degradation of transmission or reception of the radar jammer and radar warning antennas with ice accreted on their forward faces (para 33).

f. The production configuration of the area weapon environmental cover should incorporate a seal between the cover and fuselage similar to one used on the test aircraft (para 38).

g. A test should be conducted to determine the consequences of the M-261 protective covers departing the launcher, with and without accumulated ice (para 39).

h. The proposed WSPS should be evaluated in natural icing conditions to determine possible impact on existing anti-ice/deice systems and the potential for aircraft damage (i.e., engine FOD) due to shed ice (para 40).

i. A note should be added to the operator's manual performance charts to indicate that an approximately 10 percent loss in power available and 8 percent decrease in maximum effective range occur when the engine anti-ice system is activated (para 42).

j. Future models of the AH-64 helicopter should incorporate an improved attitude hold system to reduce pilot workload during instrument flight (para 44).

k. The continuous FD/LS should be modified to display NO-GO status messages only when the blade deice switch is in the ON position and a subsystem has failed (para 47).

l. The operator's manual should be updated to reflect the latest anti-ice/deice systems configuration, recommended procedures for instrument flight, and recommended procedures for flight in icing conditions (para 57).

APPENDIX A. REFERENCES

1. Systems Specification, Hughes Helicopter, DRC-S-H10000B, 15 April 1982, with change 96, 4 June 1986.

2. Letter, USAAEFA, DAVTE-TA, 2 September 1982, subject: Letter of Effort, YAH-64 Icing Survey, USAAEFA Project No. 80-08.

3. Letter, AVSCOM, AMSAV-ED, 10 September 1984, subject: Artifical and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-33. (Test Request)

4. Letter, AVSCOM, AMSAV-ED, 17 July 1985, subject: Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23, (Phase II). (Test Request)

5. Technical Manual, TM 55-1520-238-10, *Operator's Manual for Army AH-64A Helicopter,* 28 June 1984, with change 1, 15 October 1984.

6. Letter, AVSCOM, AMSAV-ED, 23 October 1986, subject: Contractor Flight Release for AH-64A, S/N 84-24285,PV-145, for Artificial and Natural Icing Tests, Contract DAAJ09-86-C-A012.

7. Test Plan, USAAEFA Project No. 84-23, *Artificial and Natural Icing Tests of the AH-6A,* October 1984.

8. Test Plan, Hughes Helicopters, Report No. 77-FT-80005P-3, *Test Plan for Artificial and Natural Icing Tests of Production AH-64A Advanced Attack Helicopter,* January 1985.

9. Final Report, Martin Marietta Co, Test Report No. 00940775004, *TADS/PNVS on AH-64A, at Duluth, Minnesota, Nov/Dec 1986,* to be published.

10. Final Report, USAAEFA Project No. 84-10, *First Article Preproduction Tests of the AH-64A Helicopter,* November 1984.

APPENDIX B. DESCRIPTION

SYSTEMS DECRIPTION

General

1. The deice system consists of an outside air temperature (OAT) sensor ice detector/rate sensor and signal processor unit, ice rate meter, blade deice control switch, main and tail rotor slip rings, and a deice controller. The main and tail rotor blades contain electrothermal resistive mats. Anti-ice systems are provided for sections of the windshields, pitot-static tubes, air data sensor, engines, engine inlet cowling ring, nose gearbox/cross shaft fairings, target acquisition and designation system (TADS), and pilot night vision system (PNVS). The deice/anti-ice systems are activated by switches located on the pilot's anti-ice control panel (fig. 1). Deice/anti-ice systems on the test helicopter, serial number (S/N) 84-24285 (production vehicle 145), are shown in figure 2.

ELECTRICAL POWER SUPPLY

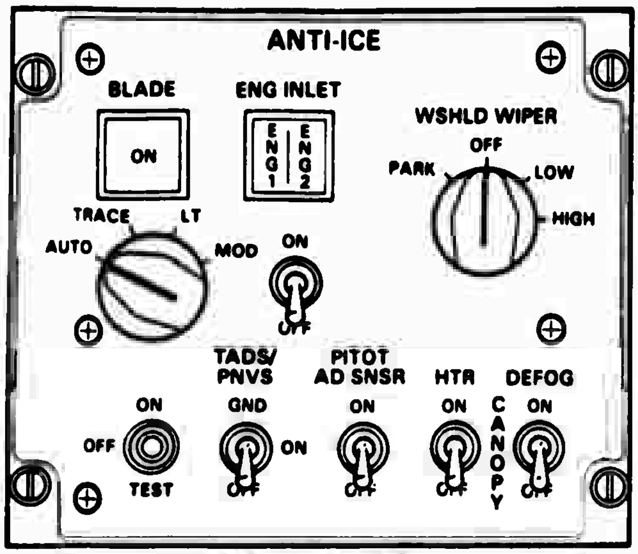
2. Power to the AH-64A electrical distribution system is supplied by 115/200 vac, 3-phase, 400 Hz, 35 Kva generators. The two generators are located on the transmission accessory gearbox. Normally each generator supplies its own ac essential bus. If one generator fails, the remaining generator takes over the bus of the failed generator.

3. Input power to the blade deice system is three-phase, 400 Hz, 115/200 vac which is fed to the blade deice controller through the blade deice system's three-phase double-throw contactor (K-3 contactor/relay). The blade deice system is normally powered by the No. 2 generator, however, the K-3 contactor/relay (fig. 3) is designed to sense a malfunction of the power source and transfer the blade deice system to the No. 1 generator. The 115/200 vac power to the blade deicing controller is converted to 268 vdc by a silicon control rectifier in the controller. The 268 vdc power is used to energize the blade deicing heaters.

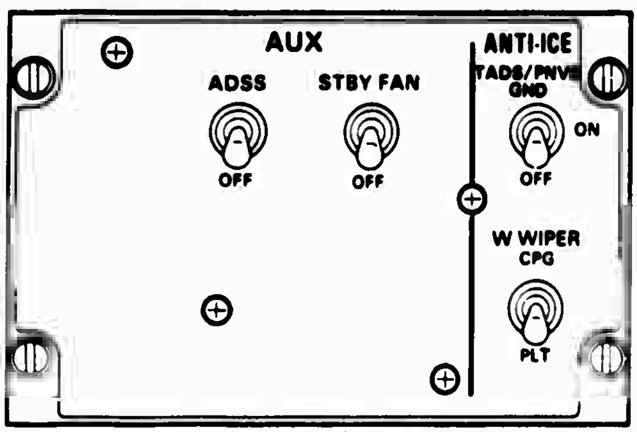
4. Two 250-ampere transformer/rectifiers powered from the ac buses provided the aircraft 28 vdc power supply.

ROTOR BLADE DEICE SYSTEM

5. The AH-64A rotor blade deice system utilizes the electrothermal cyclic deicing concept. A controlled amount of ice is allowed to build-up on the blade by varying heater off-time as a



PILOT ANTI-ICE PANEL



CPG AUX/ANTI-ICE PANEL

Figure 1. Pilot and Copilot/Gunner Anti-Ice Control Panels

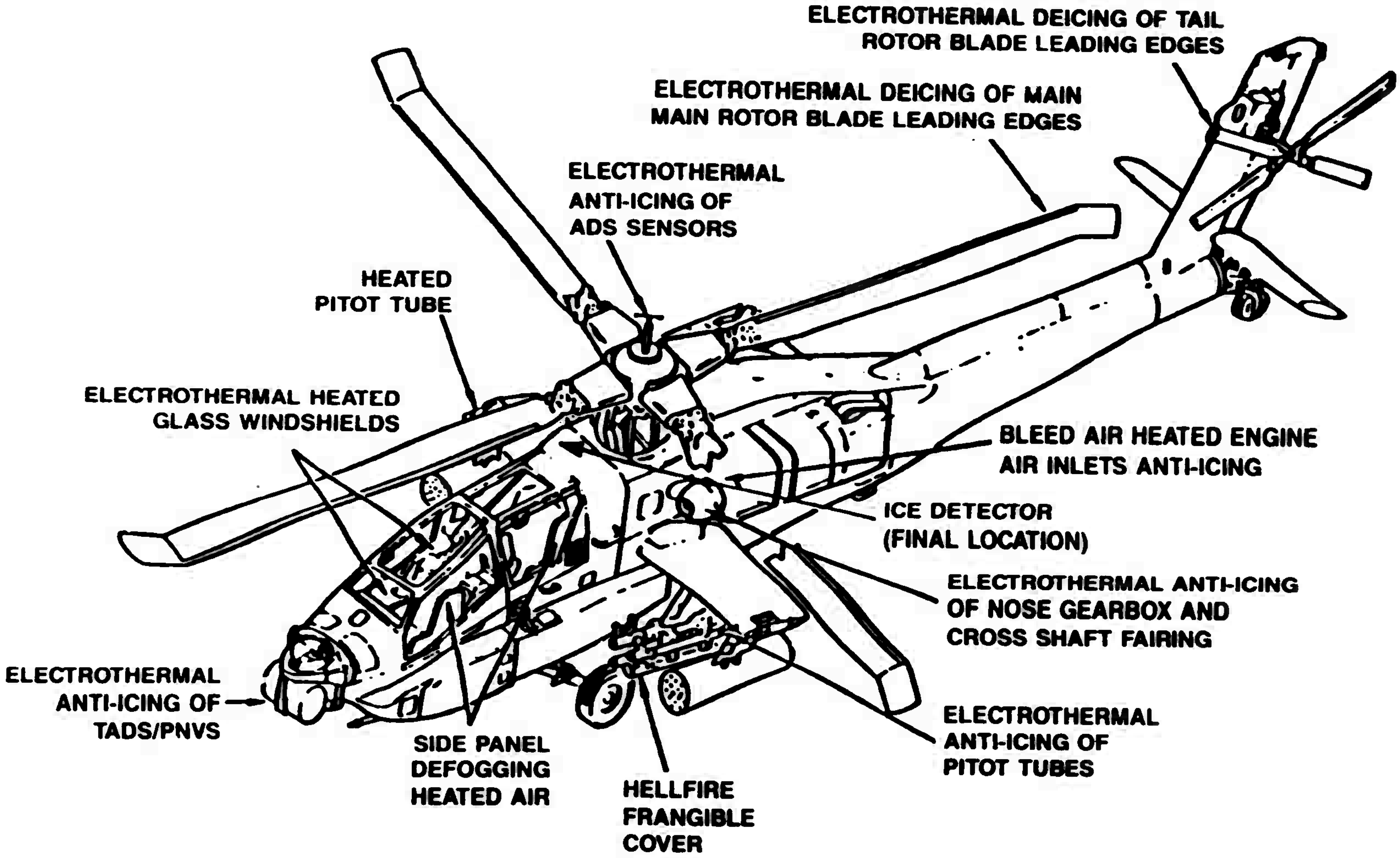


Figure 2. AH-64 Anti-Ice/Deice Systems

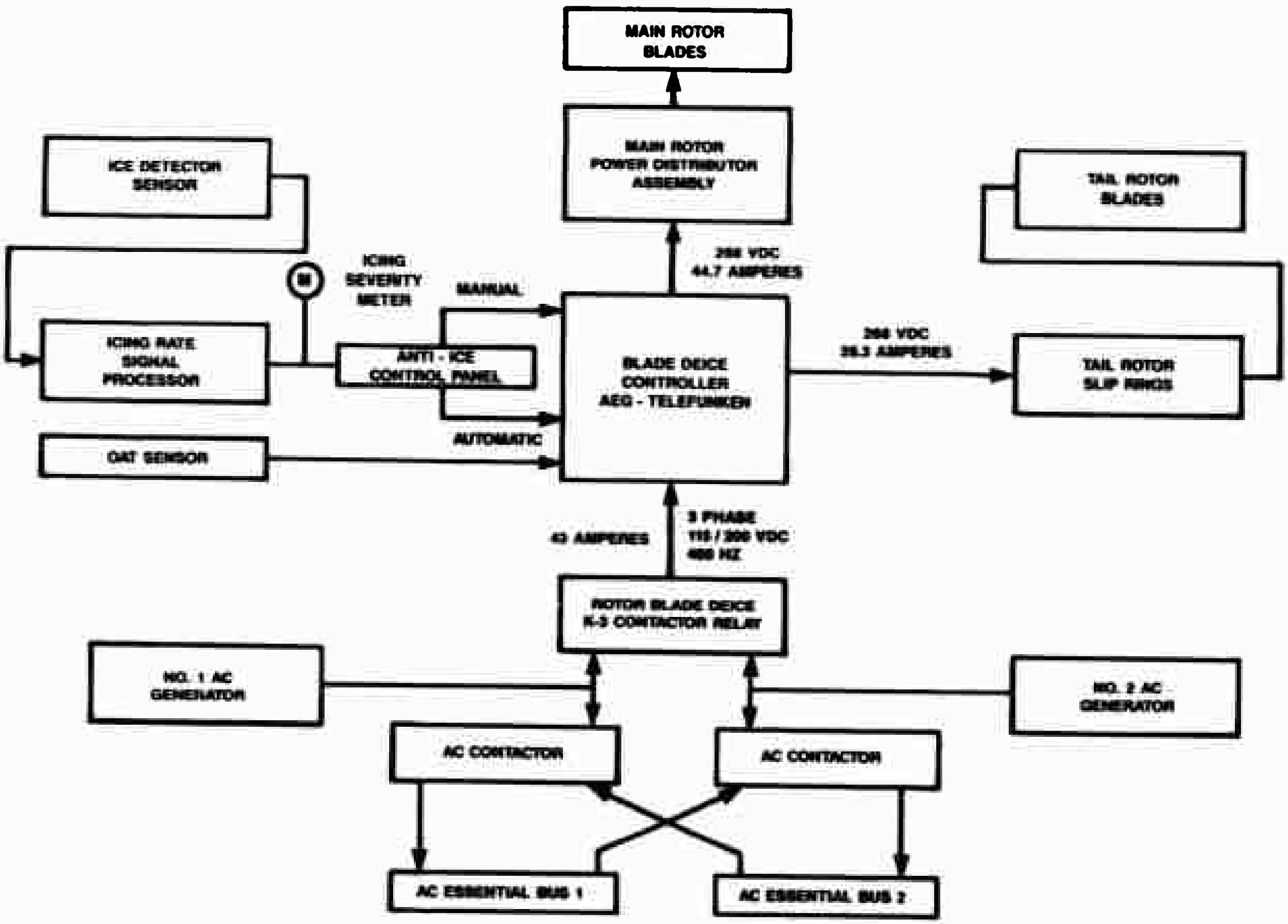


Figure 3. Blade Deice System Block Diagram

function of liquid water content (LWC) and removed periodically by energizing electrical heating elements to raise the blade surface temperature to break the ice bond and allow shedding.

6. The main blade heater consists of five spanwise heating zones running from root to tip. These zones are energized sequentially and produce chordwise ice shedding. The main blade heater elements are bonded to the inner surface of the blade structure and extend from 19 to 94 percent span.

7. The tail rotor blade heater is installed internally but consists of only one heater zone which covers 43 to 91 percent of the blade spanwise.

8. The amount of chordwise coverage of the heaters is essentially the same for the main and tail rotor blades, beginning from 10 percent aft of the leading edge on the upper surface to 26 percent (main) and 25 percent (tail), on the lower surface (fig. 4). The power density of the heaters (watts per square inch) is varied spanwise on the main rotor blade to take advantage of kinetic heating and reduce the electrical load requirements. The power density decreases in 14 steps from root to tip. The inboard and outboard steps are 36 and 30 inches in length, respectively. The remaining 12 steps each cover 12 inches of span. The tail rotor is uniformly heated at one power density over its span.

9. The blade deice system is shown schematically in figure 3. Three-phase, 115 vac, 400 Hz electrical power from the generator is rectified in the deicing system controller to 268 vdc for blade heating. The dc power from the controller is routed to the main and tail rotor through slip rings. The main rotor installation incorporates a distributor that sequentially delivers the electrical power to the five heating elements of each blade. The system deices all four of the tail rotor blades simultaneously and steps through the zones of pairs of opposing main rotor blades.

10. Figure 4 shows the blade and zone heating sequence that is used. After the blades have accreted the prescribed amount of ice, a pulse train is initiated to deice the tail and main rotor blades in a sequence that deices the tail rotor twice as often as the main rotor. The controller does not allow simultaneous heating of the tail rotor with the main rotor since the combined electrical load could exceed electrical system capacity.

11. The time interval between the applications of heat to any given zone is called heater-off time. For normal operation,

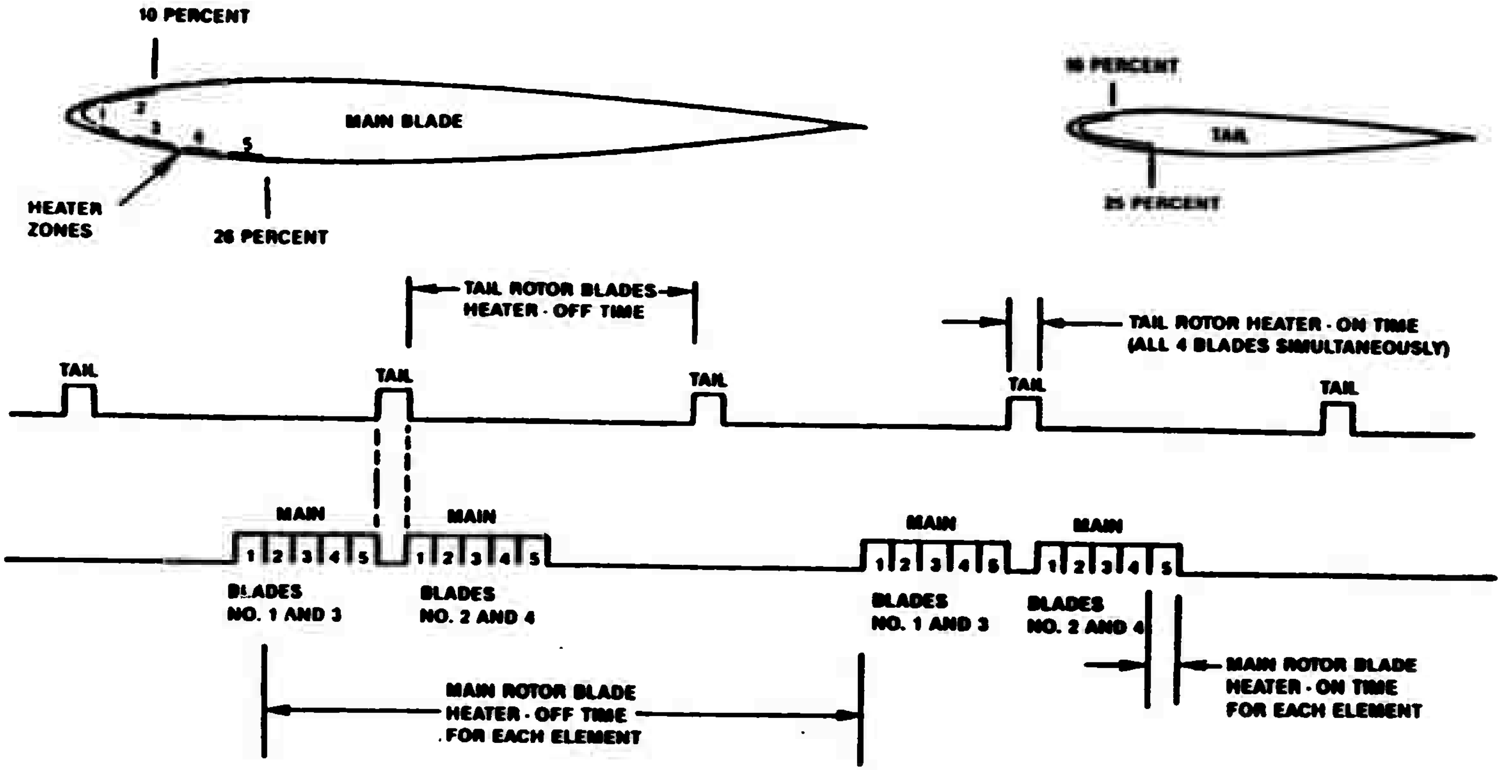


Figure 4. Blade Heater Elements and Distributor Pulse Sequence

blade heater-off time is controlled automatically by the blade deicing controller and is scheduled as a function of ice accretion rate signal multiplied by time. The controller integrates the icing rate signal (corresponding to LWC) from the ice detection/rate measuring system with respect to time and initiates a blade deicing signal each time the integral reaches a predetermined value. This predetermined value is selected to approximate a 0.25-inch thickness of ice accreted on the main rotor blade at the mid-span location. The tail rotor blade airfoil, which is approximately one-half the scale of the main rotor, is deiced at one-half the main rotor LWC-time integral (twice as often as the main rotor) to maintain the same proportion of ice accretion as for the main rotor blades.

12. The main rotor off time scheduled by the controller corresponds to the curve shown in figure 5. This curve is a constant LWC multiplied by the time value of 1.76 gm-minutes/m^3. The LWC signal from the ice detector processor is a dc voltage with a 0.2 gm/m^3 equivalent to 1 volt dc. The main rotor volt-time integral, therefore, is 8.8 volt-minutes. The tail rotor volt-time integral is approximately 4.4 volt-minutes.

13. Also shown in figure 5 are the manually selectable off times that are available in the event the system is not operated in automatic mode. The manual tail rotor off times are one half of the main rotor off times shown.

14. The duration of time that heat is applied to a given zone is called heater-on time and is controlled automatically by the blade deicing controller. The heater-on time varies as a function of OAT. The lower the air temperature, the longer the heaters are energized to raise blade surface temperature for deicing. The design heater-on time versus OAT schedule is shown in figure 6.

15. The design heater-on time is the same for each of the main blade heater elements and for the tail rotor. The deice controller must be manually adjusted for the electrical resistance characteristics of the main and tail rotor following a blade change to ensure proper system operation and over-temperature protection.

Ice Detection/Rate Measuring System

16. The ice detection/rate measuring system consisted of an Model 871FF2 aspirated Rosemount LWC probe and a Model 524Y4 signal processor unit. The LWC sensor is a vibrating probe which changes frequency as ice is accreted. The signal processor monitors the probe frequency shift and provides an analog output voltage signal to the deice controller which is proportional to

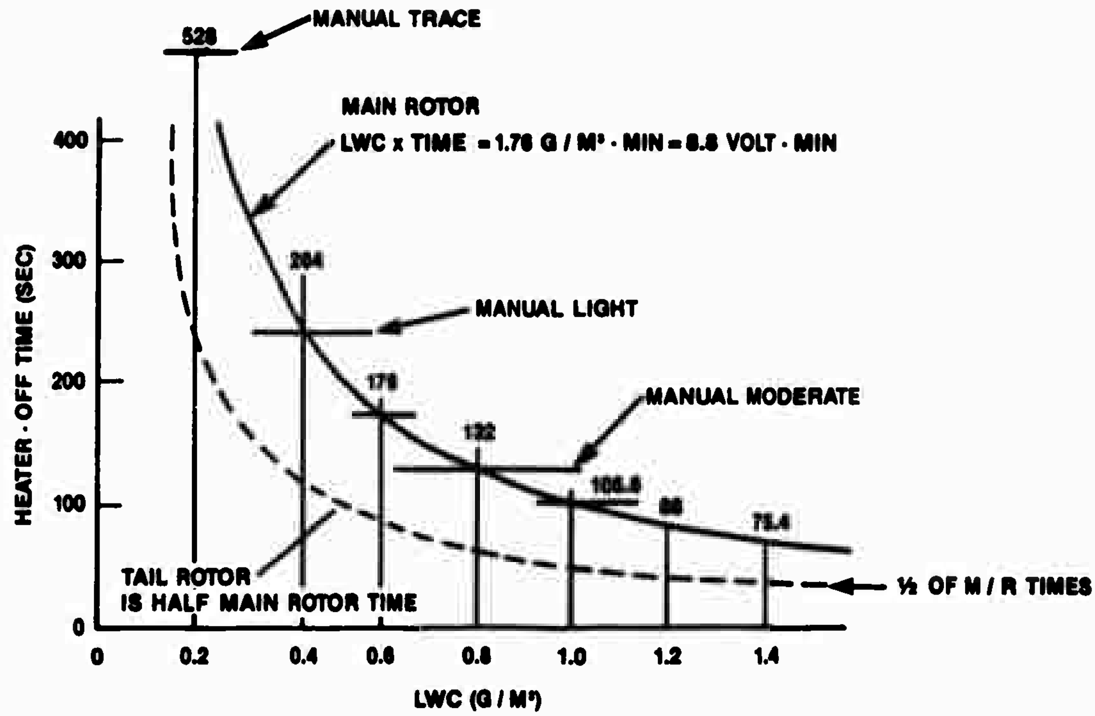


Figure 5. Blade Heater-Off Time Schedule

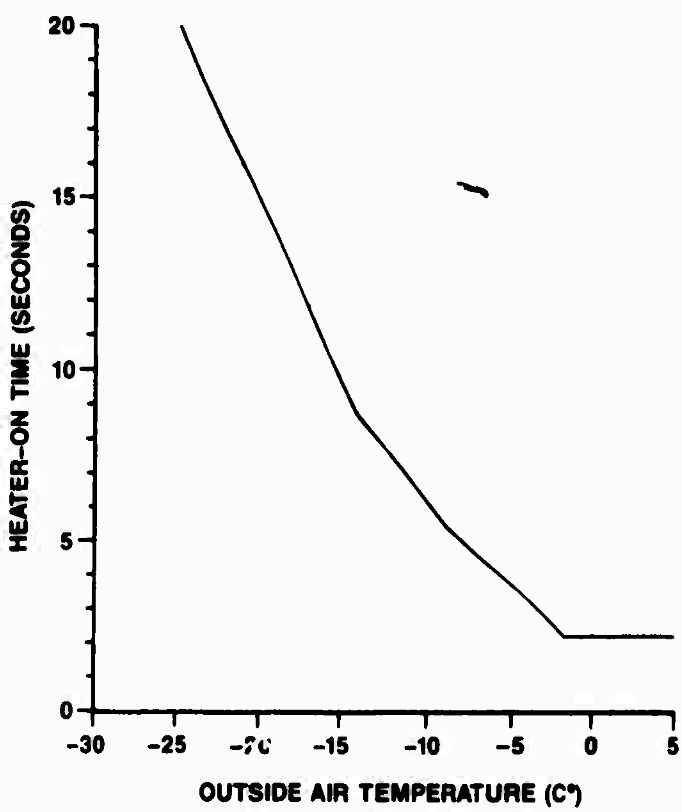


Figure 6. Blade Heater-On Time Schedule

sensed LWC. The probe was mounted on a fiberglass ramp on the upper right side of the fuselage. The ramp was installed to provide separation between the probe and fuselage. A metal rod (not part of the production configuration) was mounted adjacent to the probe (photo 1). Uniform ice accretion on the rod indicated that probe readings were not influenced by any water flowing from the heated windshields and the probe was operating in undisturbed free stream airflow.

17. The PITOT AD SENSOR switch on the anti-ice control panel (fig. 1) opens a valve which allows the aircraft pressurized air system to provide aspiration air to the probe. The pressurized air source was modified and the external air line was insulated to provided aspiration air to the probe at approximately 23 psig and 65° C. These modifications provided significant improvement in probe performance over that observed in previous testing (app F).

18. The ice detector/rate probe is powered by direct current and is on whenever ship's power is on. The icing rate signal is generated whenever ice build-up on the probe is between 0.015 inch and 0.060 inch thick. When the ice thickness reaches 0.060 inch, the probe is electrothermally deiced and the cycle repeated. The ENG ICE caution light on the pilot's caution/warning panel illuminates when the ice thickness reaches 0.015-inch and remains on for a minimum of 90 $\pm$10 seconds. If no further icing were encountered, the light would go out at the end of that time. If icing is continually encountered, the light will remain on.

19. During the time the sensing probe is deiced and recovering from its 6 second heater application, a hold circuit is employed that maintains the LWC signal at its last value. The hold circuit operates until the probe has thermally recovered and then releases when accreted ice thickness again reaches the 0.015-inch or 60 $\pm$10 seconds of time has elapsed. Thus, if the aircraft emerges from an icing environment prior to reaching 0.015-inch thickness the LWC signal would go to zero after this time interval to indicate a zero LWC condition. The probe recovery time (time the detector/rate sensor is off line) varies with LWC and OAT. Typical maximum recovery times for a probe velocity of 112 knots are 75 seconds at LWC = 0.12 g/m^3, 15 seconds at 0.5 g/m^3, and 10 seconds at 1.0 g/m^3.

20. The output of the ice detection/rate measuring system will vary with air temperature and local flow velocity over the probe at a fixed LWC. Additionally, accuracy will vary with LWC at a fixed temperature and flow velocity. The probe is calibrated in an icing tunnel to read LWC $\pm$10 percent for local flow velocity of 112 knots. At higher and lower local flow velocities, the

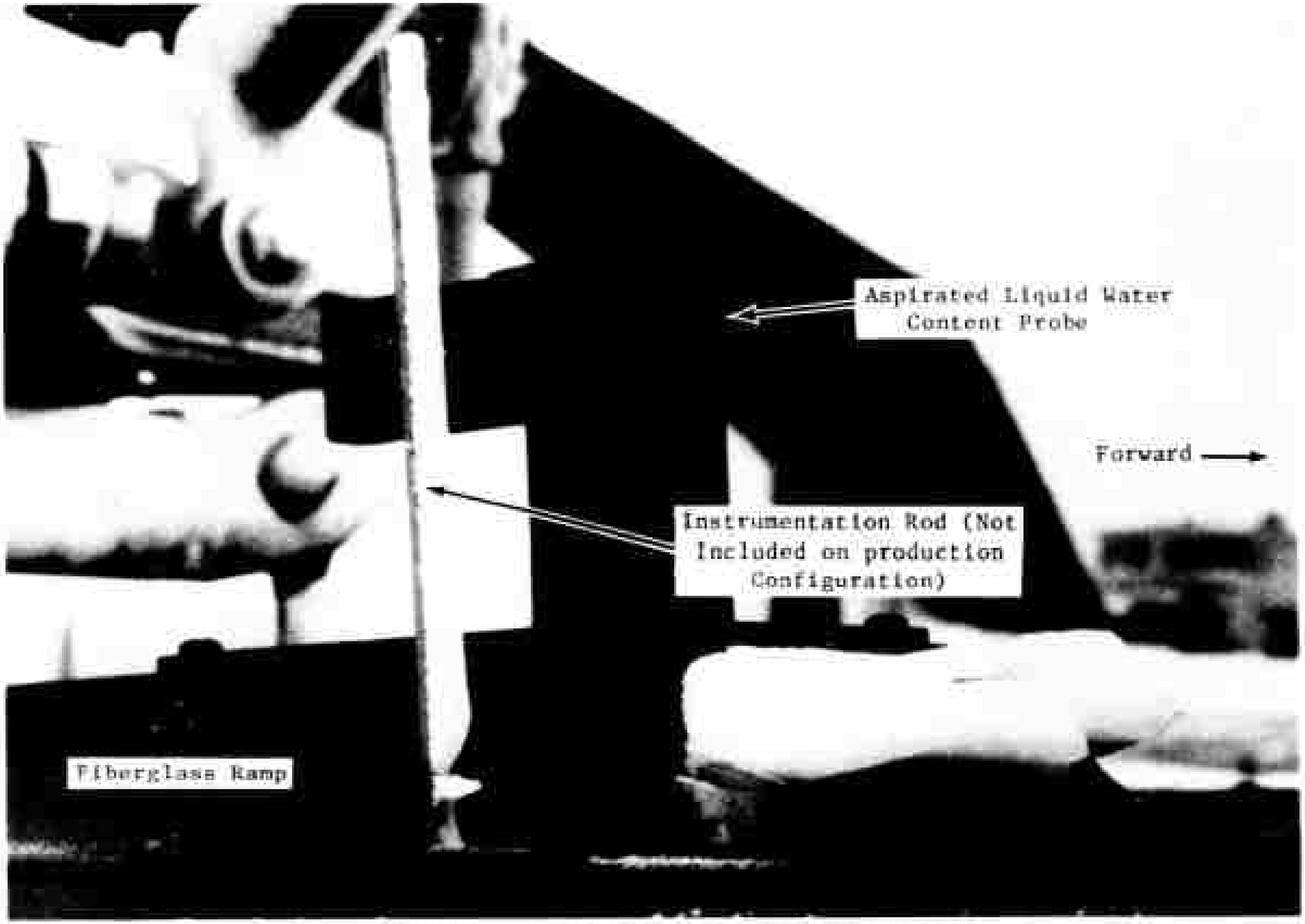


Photo 1. Upper Fuselage - Left Front View

system will indicate slightly higher or lower than actual LWC values.

21. The deice system included an icing rate meter which displayed LWC as measured by the ice detector/rate probe and the signal processor. The LWC meter (fig. 7) is mounted on the upper right side of the pilot's instrument panel.

Outside Air Temperature System

22. The OAT sensor is a flush-mounted resistance type temperature sensor installed on the right side of the fuselage between the engine nacelle and wing. This location was selected to shield the 1.25 inch diameter sensor from exposure to direct sunlight heating and also to be in an area that will not be subjected to ice impingement. The OAT sensor's resistance output is fed to the blade deicing controller for use in scheduling blade heater-on time. No cockpit indication of OAT sensor output is provided.

23. A second identical OAT sensor was installed immediately adjacent to the deicing controller sensor as test instrumentation to provide a signal for recording on the data acquisition system. This permitted a check on the OAT signal going to the blade deicing controller.

Blade Deice System Operation

24. Rotor blade deicing operation was initiated as follows. Prior to entering a potential icing environment, all ice protection system swiches were turned on. The blade deice system was set for automatic operation using the mode selector on the anti-ice control panel. When icing was encountered, the pilot was alerted by illumination of the ENG ICE segment light on the caution/warning panel (fig. 8). The normal procedure was to allow the collective to remain fixed and to maintain cruise altitude by allowing airspeed to decrease. As ice accreted on the rotor system, power required (indicated torque) increased 6 to 8 percent and airspeed dropped 5 to 10 knots. Blade deicing would commence after the scheduled ice accretion period elapsed and continued automatically. Power required and indicated airspeed returned to their original values following a blade shed cycle. Except for the indicated torque rises and indicated airspeed loss, operation of the blade deice system is essentially transparent to the crew.

25. Heater-off time shortens with increasing icing intensity (LWC) and heater-on time lengthens with decreasing OAT. At some high level of LWC and OAT, the system will saturate. On time

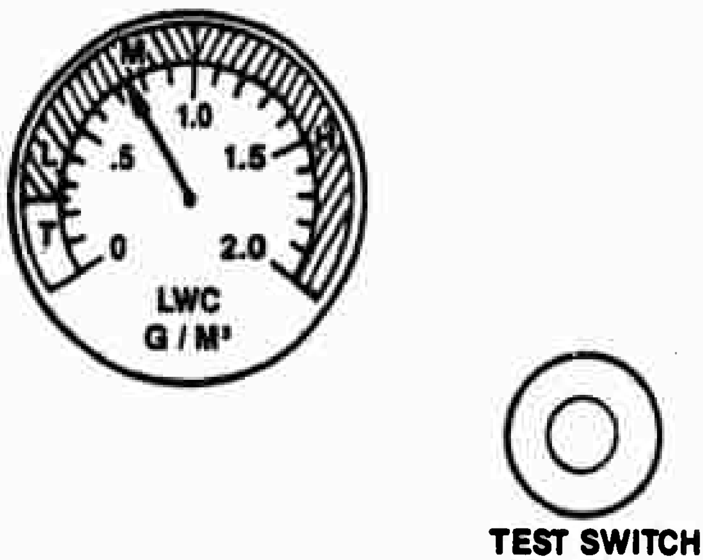
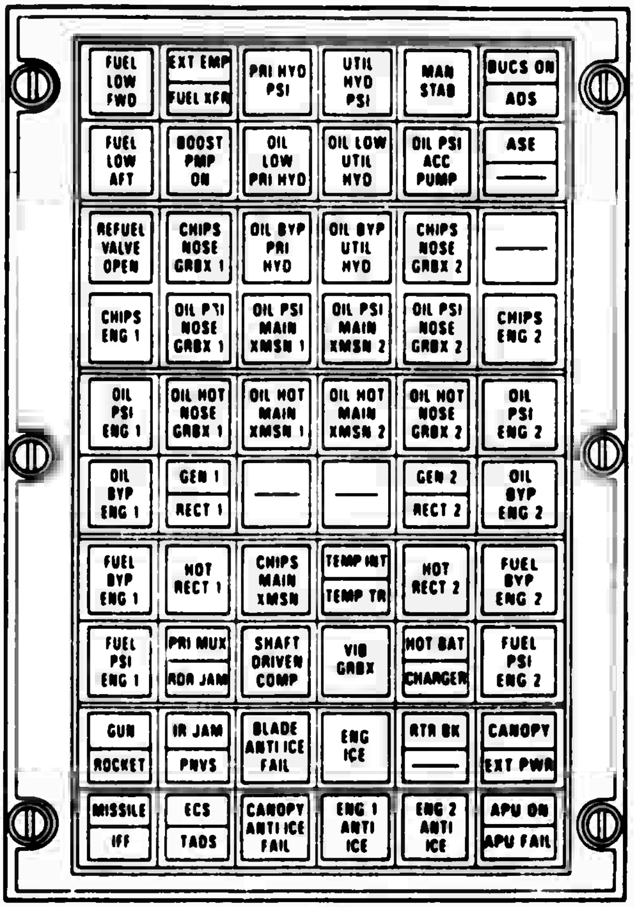
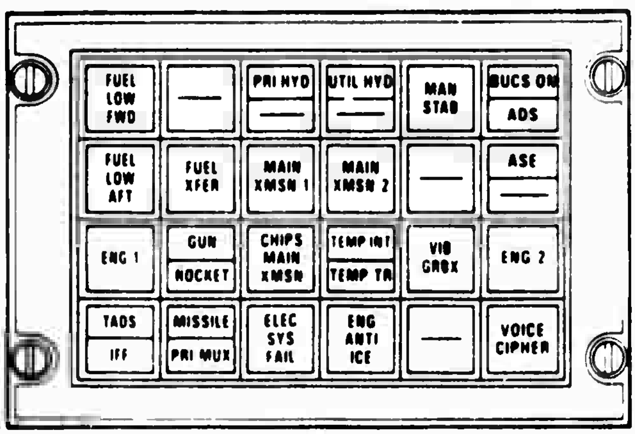


Figure 7. Icing Rate Indicator and Test Switch



PILOT



CPG

Figure 8. Pilot and CPG Caution/Warning Panels

takes precedence over off time and will prevent ice from building up to the desired thickness before deicing. In theory, this could have a deterimental effect on blade shedding characteristics. However, during the developmental phase of testing no adverse blade shedding characteristics resulted from short blade heater-off times. Saturation does not occur until -14.7 deg C at a LWC = 1.0 g/m^3.

26. The manual mode of blade deice system operation may be selected if the aircraft is operating in an icing environment and the pilot suspects the icing rate system is malfunctioning and not detecting the true icing intensity. In this mode, the blade heater on-time is one of three fixed values chosen with the mode selector on the pilot's anti-ice control panel: TRACE = 528 seconds, LIGHT = 264 seconds, and MODERATE = 132 seconds.

Rotor Blade Deice System Checkout

27. A deice system TEST switch and an icing rate indicator are provided to enable ground checkout of the system. The deice system test consists of a rapid heater on sweep or duty cycle (0.5 second on-time per element) and an advisory light to indicate heater operation. Any failures will activate the BLADE ANTI-ICE FAIL light on the caution/warning panel (fig. 8). The icing rate indicator test is activated by depressing the test button adjacent to the icing rate meter. This test conducts a check on the icing rate meter electronics only.

28. The Fault Detection/Location System (FD/LS) provides an "on command" check of four blade deice subsystems. The check is performed by selecting the FD/LS mode on the copilot gunner (CPG) data entry keyboard and entering the appropriate system code displayed on the menu. Required actions to be performed by the crew are displayed on the visual display. The four subsystems are:

a. main rotor heaters

b. tail rotor heaters

c. power controller

d. distributor

During normal aircraft operations, the FD/LS will display NO-GO (fail) messages for these subsystems when the blade deice switch is in the OFF position. When the switch is ON, NO-GO messages will be displayed only if a subsystem has failed.

Blade Deice System Protection Circuits

29. The blade deicing controller contains monitoring circuits to sense a system malfunction. The fault monitoring functions and protection features are:

a. Main or tail rotor heater undercurrent or overcurrent.

b. Main or tail rotor heater ground current (6.0 amp maximum).

c. Distributor mechanism does not advance.

d. Heater current on too long (+24 percent maximum).

e. Controller over temperature (>100 deg C).

f. Excessive input power (115 vac +10 percent).

g. System ON advisory light (caution while performing maintenance).

h. System disable above 4 deg C OAT to preclude blade over temperatures.

ENGINE ANTI-ICE SYSTEM

30. The engine anti-ice system incorporates two electrically operated valves, the engine anti-ice valve and the inlet anti-ice valve, to route hot engine air to the engine swirl frame, hollow inlet guide vanes and the bellmount shaped nacelle inlet cowling ring. Additionally, electrothermal anti-ice protection is provided on the forward nose gearbox/cross shaft fairing and a portion of the aft nose gearbox/cross shaft fairing forming the engine air inlet (fig. 9). The engine anti-ice system is controlled by a single two-position ON/OFF toggle switch located on the pilot's anti-ice control panel (fig. 1). In the ON position, green ENG 1 and ENG 2 advisory lights above the switch illuminate when the bleed valves are properly positioned and the heated nose gearbox/cross shaft fairings have reached a monitored temperature threshold. The advisory lights will extinguish and No. 1 and No. 2 ENG ANTI-ICE fail lights on the caution/warning panel (fig. 8) will illuminate if a bleed valve fails to open or if the heated fairings reach an over temperature or under-temperature condition.

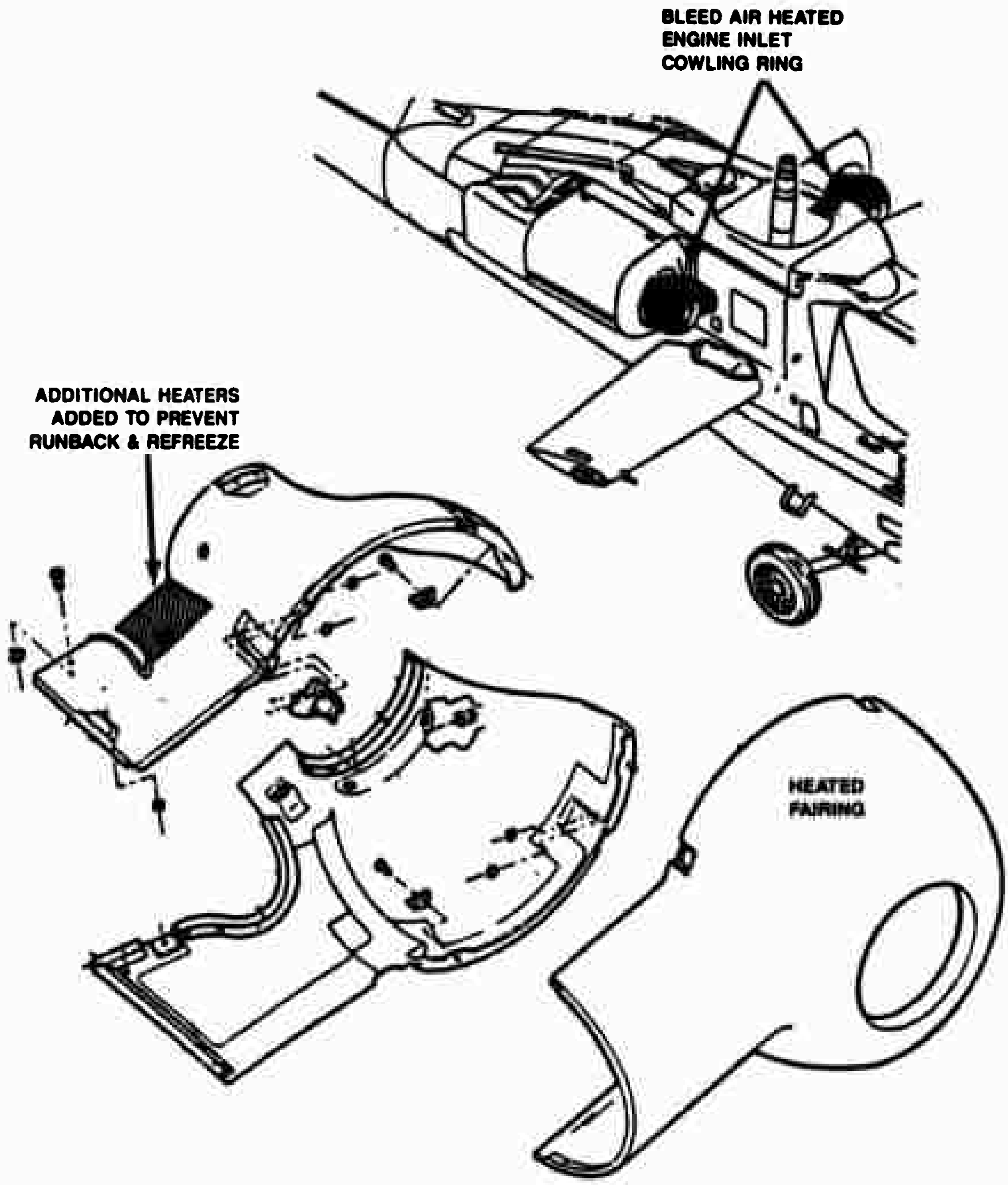


Figure 9. Nose Gearbox Fairing Installation

Engine Inlets

31. Hot engine air is routed to the bellmouth shaped nacelle inlet cowling ring (fig. 9) through the electrically operated two position inlet anti-ice valve. This valve is electrically (28 vdc) closed when the engine anti-ice switch is OFF and spring loaded open when electrical power is removed, engine anti-ice switch is ON or there is a complete electrical failure. The heated inlet cowling ring surface extends around the entire leading edge of the inlet to where it mates with the engine. Thus, the running wet surface extends into the engine during operation in icing conditions. The amount of engine bleed air flow was designed to anti-ice the inlet even under low power descent conditions.

Nose Gearbox/Cross Shaft Fairings

32. The forward nose gearbox/cross shaft fairings incorporate electrothermal heating blankets for anti-icing (fig. 9). Additional heater blankets were added to the aft nose gearbox/cross shaft fairing at the engine inlet to prevent runback and refreezing. The heater blankets are controlled automatically by a temperature controller which maintains the surface at a temperature sufficient to prevent moisture that impinges on the fairing from freezing.

33. The electrothermal heating blankets are turned on by the same switch that activates the engine air inlet anti-ice system (para 30). Overheat sensors are provided that activate the ENG ANTI-ICE fail lights on the pilot's caution/warning panel and shut the electric heaters off automatically. The controlling-type overheat sensor allows the system to operate at the higher temperature without additional overheat protection.

Canopy Anti-Ice System

34. Transparent electrothermal anti-ice elements (115 vac) are built into the two laminated glass panels located in front of and directly above the CPG. The pilot's windshield, which is the panel above the CPG, is heated in two zones with Phase A and B of the three-phase current. The CPG's panel is powered by Phase C. A single temperature sensor, located on the left side forward zone of the pilot's windshield, signals the controller to maintain the windshield's outside surface temperature between 65 and 85 deg F using a power density of 4.57 watts/in.^{2}. Canopy anti-ice is activated by the CANOPY HTR switch on the pilot's anti-ice control panel (fig. 1).

35. Failure of the canopy anti-ice system is indicated by the CANOPY ANTI-ICE FAIL light on the caution/warning panel (fig. 8)

and the continuous FD/LS visual display. An "on command" FD/LS check is provided through the CPG data entry keyboard. Unlike the blade deice system, a NO-GO (fail) message is displayed only when the CANOPY HTR switch is ON and the system has failed.

Defogging/Demisting

36. Cockpit defogging/demisting is accomplished by two sources. The two windshield sections are electrothermally heated as described in paragraph 34. This serves to defog as well as to anti-ice. Pressurized air from the shaft driven compressor (SDC) is mixed with ambient crew station air in diffuser outlets at the CPG and pilot stations to defog the canopy side panels. In case of SDC failure, No. 2 engine bleed air is used. Canopy side panel defogging is controlled by an ON/OFF switch on the pilot's anti-ice control panel (fig. 1).

Rain Removal

37. Two windshield wipers are mounted on the canopy frame. Both wipers are electrically driven and are normally controlled by a four-position rotary knob on the pilot's anti-ice panel (fig. 1). The wipers are powered from the 28 vdc emergency bus. Two speeds, HIGH or LOW, are selected by the pilot. The CPG has limited control over the windshield wipers. Normally, the W WIPER switch on the CPG's anti-ice panel (fig. 1) is left in the PILOT position. When the CPG position is selected the wiper on the front windshield moves at slow speed.

PITOT TUBES AND AIR DATA SENSOR

38. The pitot tubes installed on the outboard leading edge of both wings and the air data sensor (ADS) on top of the rotor mast are electrothermally anti-iced by 28 vdc power using internal heating elements. An ON/OFF toggle switch labeled PITOT AD SNSR is located on the pilot's anti-ice control panel. On the ADS, the rotating hub and arms are heated. The support pylon (mast) of the ADS is not anti-iced.

TADS/PNVS ANTI-ICE PROTECTION

39. Anti-icing for the TADS/PNVS consists of electrothermal heating of the three viewing windows and portions of their frames. In addition, thermostatically controlled heaters are located inside of the TADS shroud above the night side and dayside windows

to keep these limited shroud regions clear and ensure that adequate TADS elevation motion can be achieved. Unheated front ice shields protect the TADS azimuth gear and inboard sections (immediately adjacent to the centerpost) along the top two-thirds of the TADS shroud. A detailed description of the TADS/PNVS anti-ice systems is contained in reference 9, appendix A.

40. The TADS/PNVS anti-ice system is powered by 115 vac current and is normally controlled by a three-position switch on the pilot's anti-ice control panel (fig. 1). When the switch is in the ON position and the aircraft is airborne, electrical power is routed through the tail wheel squat switch relay to the TADS/PNVS heater switch. When the aircraft is on the ground and the squat switch is depressed, power to the TADS/PNVS heater switch s interrupted. On the ground, power may be provided to the TADS/PNVS heaters by moving the TADS/PNVS switch to the GND position.

HELLFIRE DEICE SYSTEM

41. The Hellfire deice system was evaluated during a previous test (ref 2, app A). This system utilizes a frangible glass dome attached to the missile seeker as shown in figure 10. A switch at the CPG station applies dc current to a squib enclosed in a plastic housing adjacent to the glass dome. Firing the squib extends a plunger which fractures the glass dome removing the accreted ice and exposing the missile seeker.

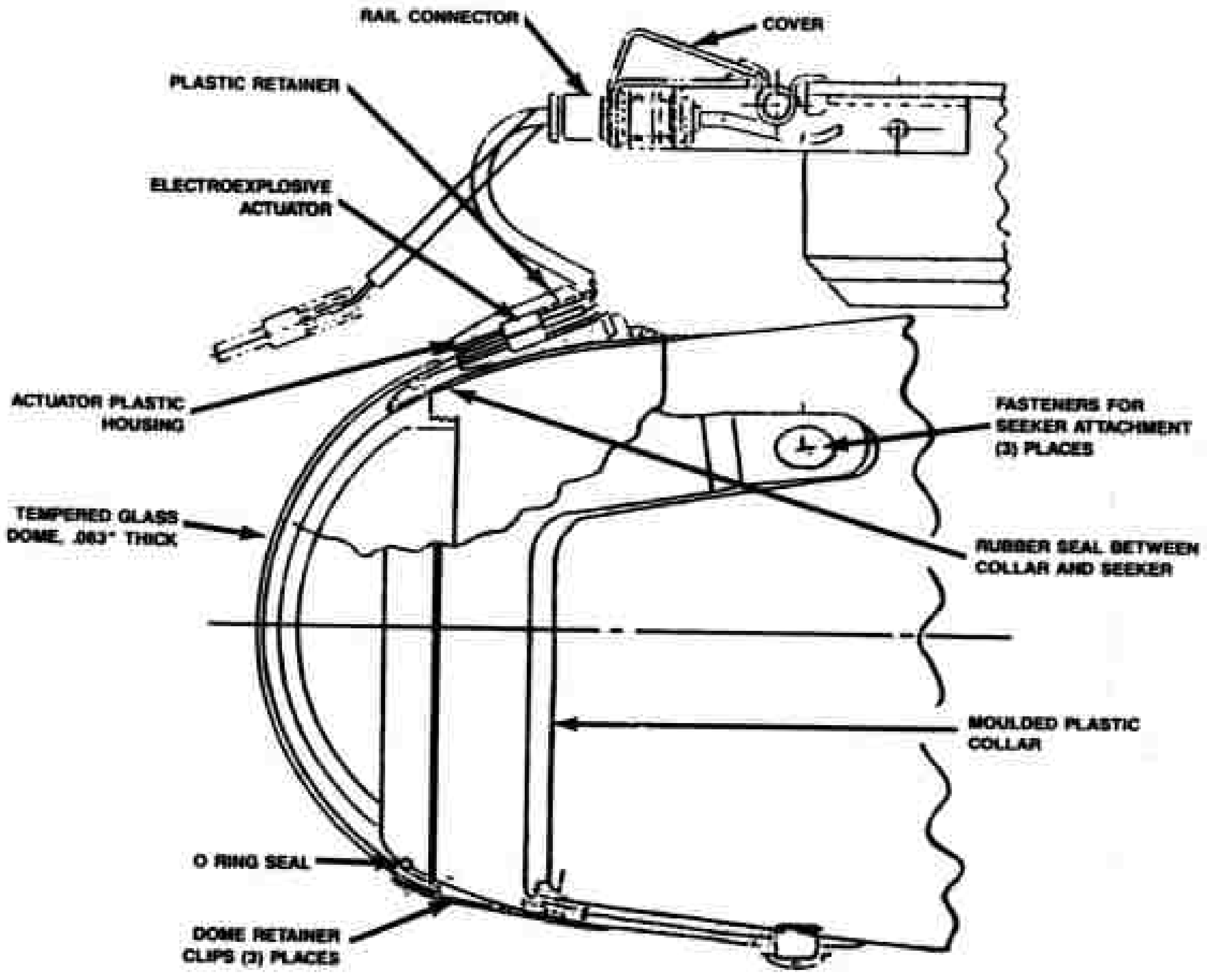


Figure 10. Frangible Dome Deicing Kit

APPENDIX C. INSTRUMENTATION AND SPECIAL EQUIPMENT

GENERAL

1. In addition to standard aircraft instruments, McDonnell Douglas Helicopter Company installed and maintained calibrated instrumentation. Data from the cockpit instrumentation was hand recorded on flight cards. The instrumentation system recorded frequency modulation (FM) data on magnetic tape. Additional equipment installed at the pilot's station included full instrument landing system instrumentation (localizer, glide slope, and marker beacon). Special equipment installed at the copilot/gunner station included an icing rate indicator for the aspirated system and a control panel for operation of the video camera and data system.

FM DATA PARAMETERS

2. The following data were recorded on magnetic tape in FM format.

Time code
Total air temperature
Observed air temperature (rotor deice system)
Blade deice current Phase A
Icing severity (aspirated system)
Target Acquisition Designation System temperature
    Top of dayside window
    Bottom of dayside window
    Dayside window frame
    Nightside window
    Nightside window frame
    Nightside shroud main heater
    Nightside shroud heater element
Pilot Night Vision System temperature
    Window
    Fairing

CAMERA SYSTEMS

3. Four video cameras were located on the test aircraft to document ice accretion characteristics. One camera was mounted on the right forward avionics bay to photograph the main rotor advancing blade leading edge and lower surface (photo 1). Another camera located on the top of the left wing tip photographed the tail rotor and horizontal stabilator (photo 2). The third camera photographed the left engine inlet (photo 3). A fourth camera was mounted on the transmission access panel to document ice

Photo 1. AH-64A - Left Side View

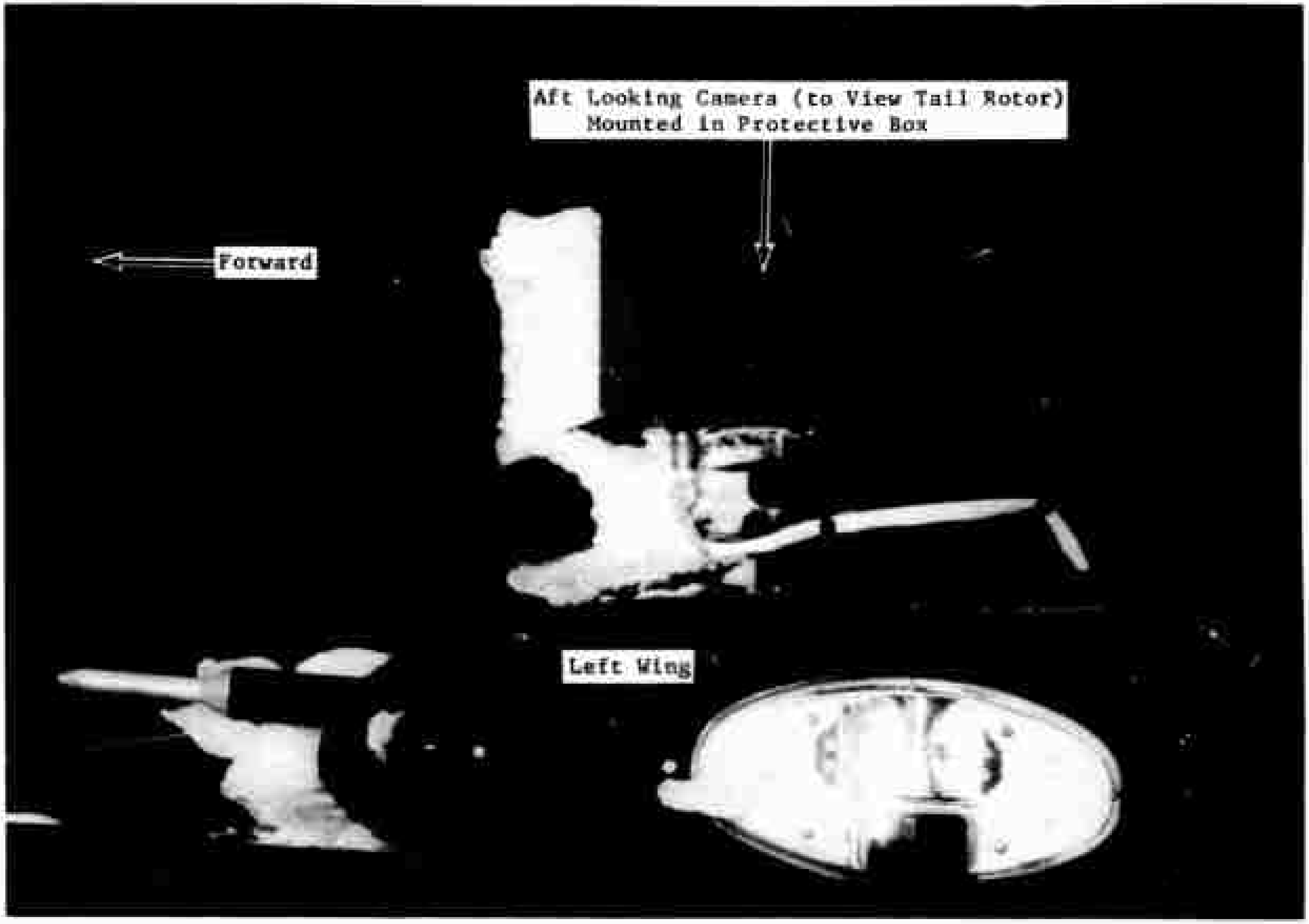


Photo 2. AH-64A - Left Side View

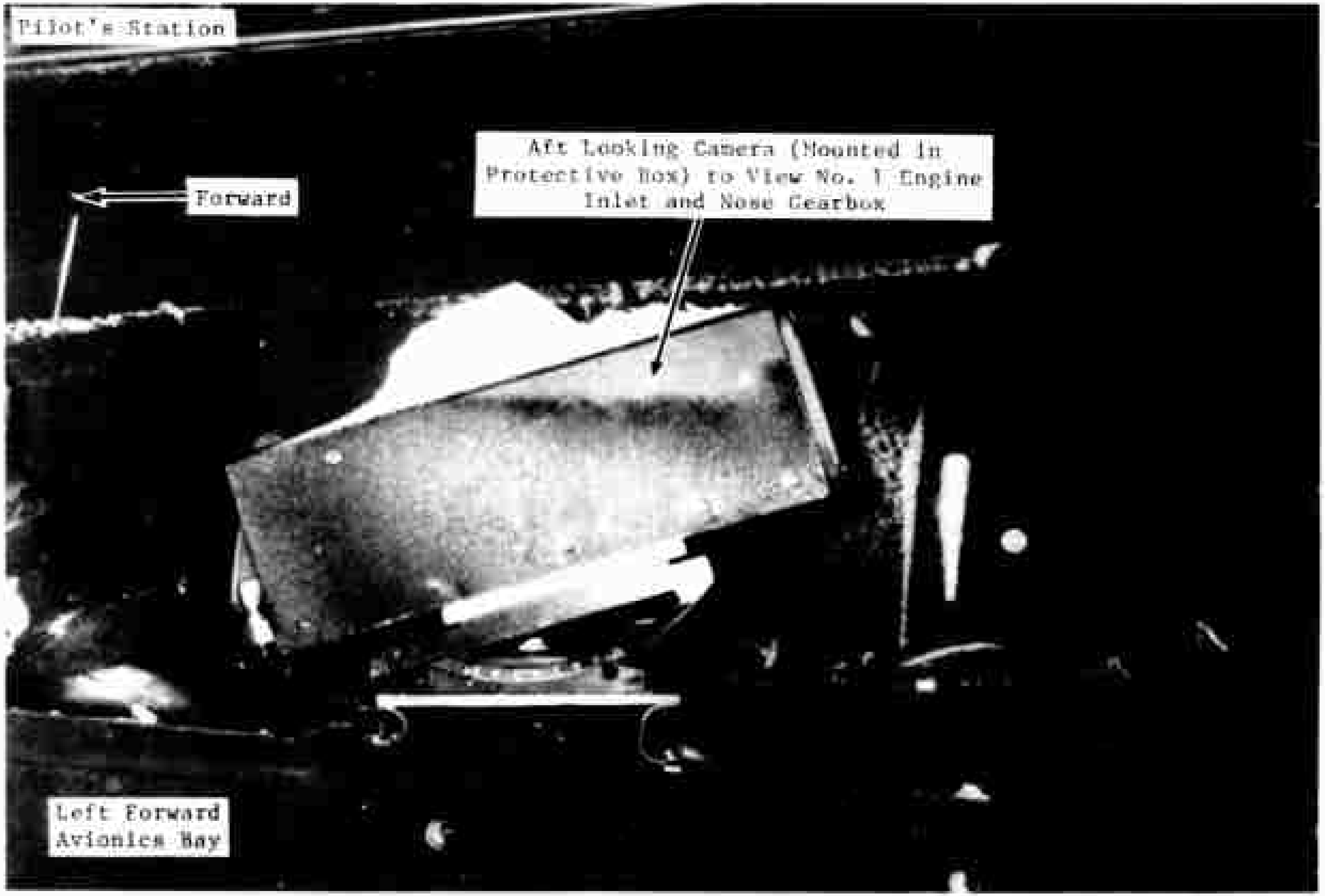


Photo 3. AH-64A - Left Side View

accretion on the right engine aft nose gearbox cross shaft fairing (photo 1).

4. An additional video camera was located on board the chase aircraft to document ice accretion on the test aircraft after exit from icing encounters. Single lens reflex 35mm cameras were used for still photo (color prints and slides) documentation both in the air and on the ground following icing flight.

CLOUD SAMPLING EQUIPMENT

5. For cloud measurements in the natural environment, the US Army Aviation Engineering Flight Activity employs a JU-21A fixed-wing aircraft, US Army S/N 66-18008, equipped with a cloud measurement package. The cloud measurement package consists of the following equipment: a Particle Measuring Systems, Inc. (PMS) Forward Scattering Spectrometer Probe (Model FSSP-100), a PMS optical array cloud droplet spectrometer probe (model OAP-200X), Rosemount total temperature sensor and display, Cambridge model 137 chilled mirror dew point hygrometer and display, Leigh Mk 12 ice detector unit, Cloud Technology Inc., model LWH-1 (Johnson Williams type) liquid water content (LWC) indicator system, Small Intelligent Icing Data System (SIIDS), and two visual accretion devices: Aeroplane & Armament Experimental Establishment's Vernier Accretion Meter (Harvey-Smith) and a Small Airfoil Section probe (OH-6 tail rotor section). Photos 4 and 5 show the exterior of the aircraft with the probes in place, while photo 6 shows the interior instrumentation rack with displays.

6. Each PMS probe projects a collimated helium-neon laser beam normal to the airflow across a small sample area. In forward flight, particles passing through the beam (sample area) are counted and measured into 15 size channels per probe, each probe operating over a different size range. While these probes are primarily intended as particle sizing devices, an LWC can be calculated from the drop size measurement and number count within the sample volume relative to airspeed.

7. The FSSP-100 determines particle size by measuring the amount of light scattered into the collecting optics aperture as the particles pass through the laser beam. A pulse height analyzer compares the maximum amplitude of the scattering signal pulses with a reference voltage derived from a separate measurement of the illuminating light signal. The pulse height analyzer output is encoded to give the particle size in binary code, and resolves particle sizes from 2 to 27 μm into 15 equally spaced increments 3 μm wide. It is capable of sizing particles having velocities

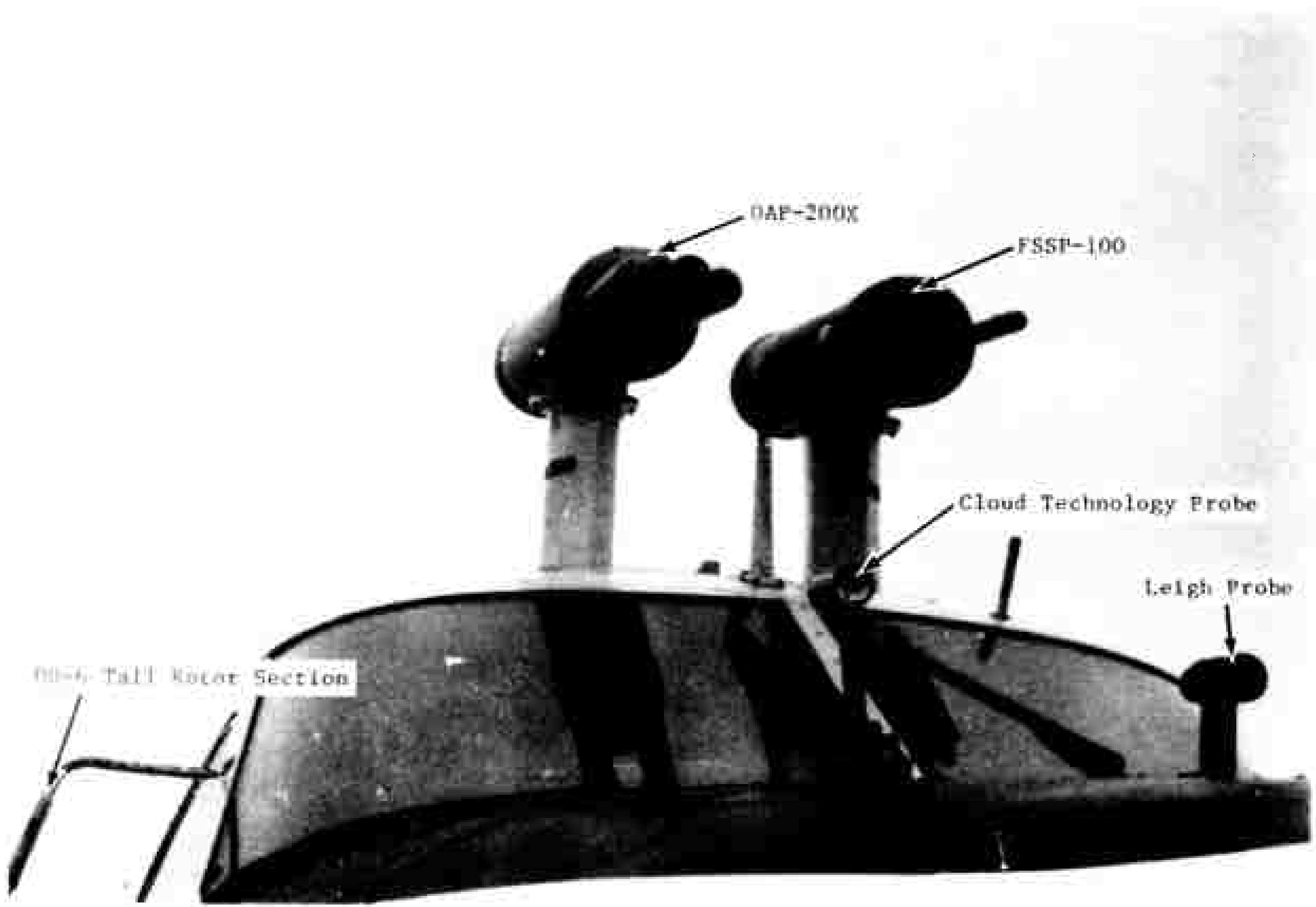


Photo 4. JU-21A Aircraft - Nose and Cabin Top View

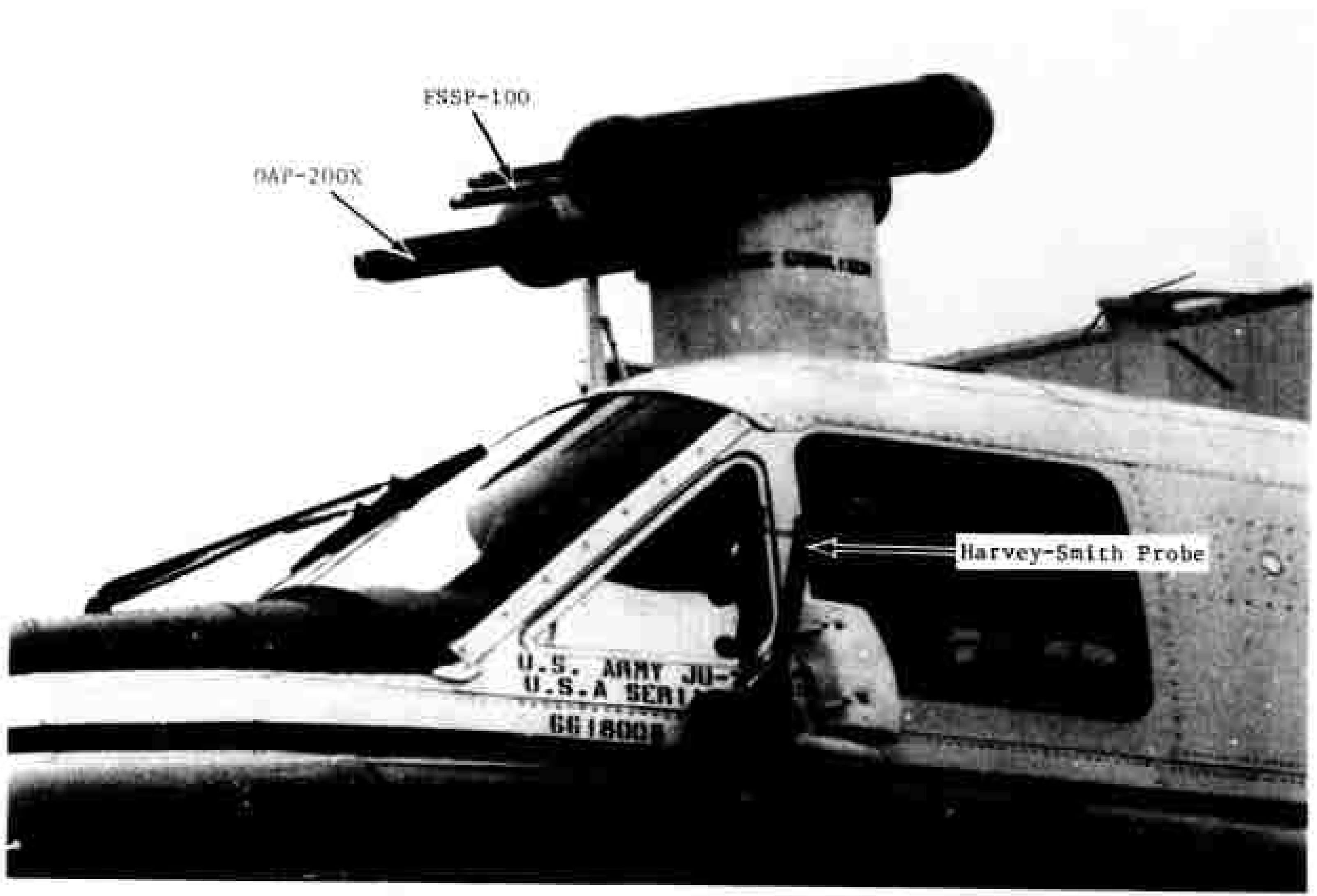


Photo 5. JU-21 Aircraft - Left Side View

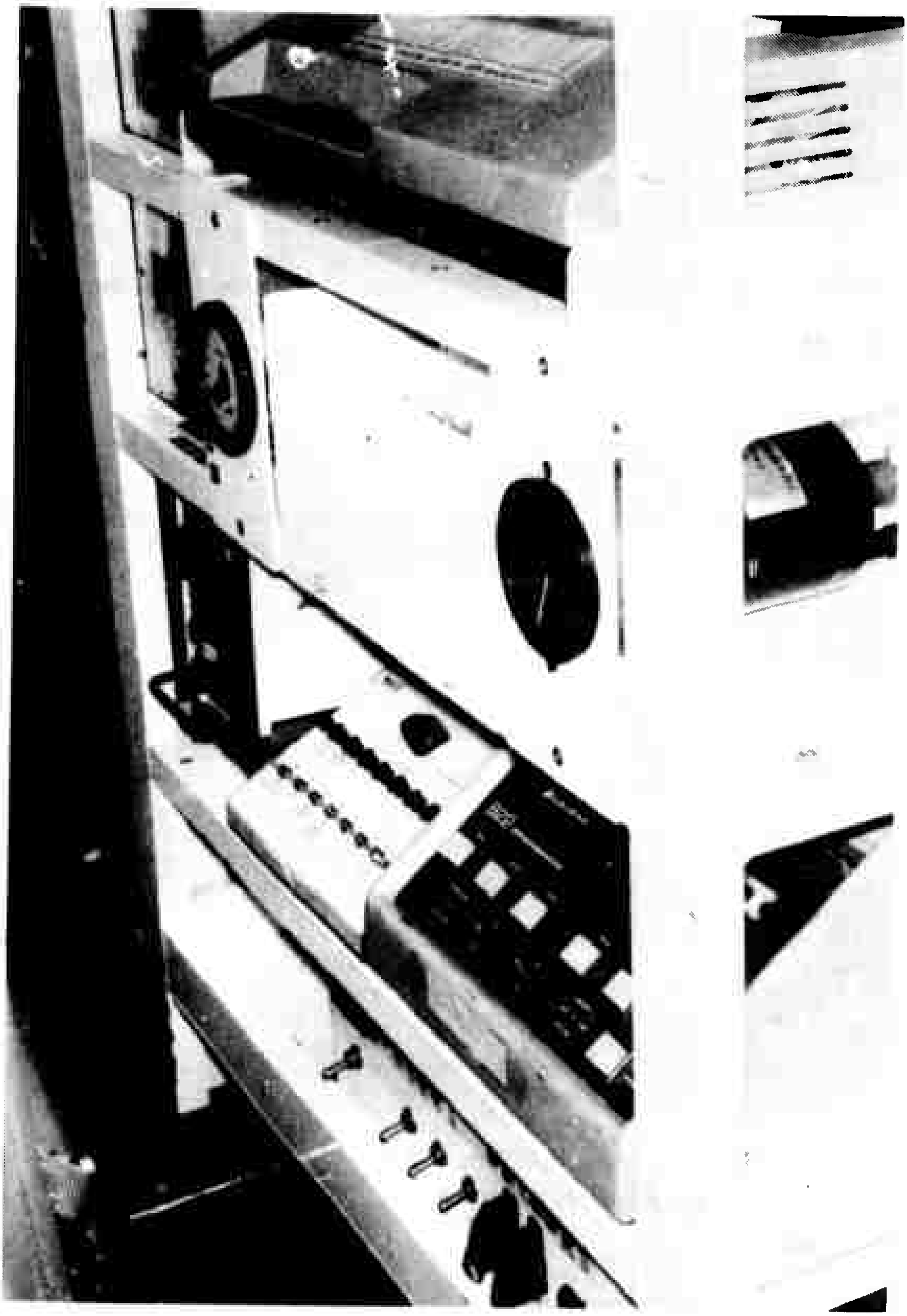


Photo 6. JU-21A Aircraft - Interior Instrumentation Rack

of 20 to 125 meters/sec (39 to 243 knots). A gate output signal provides a measure of particle transit time, and a velocity averaging counter and control system determines an average transit time. The system automatically rejects particles with transit times less than average since these are susceptible to edge effect errors and result from particles passing through regions of less than maximum intensity. A laser beam width of 0.186 mm and depth of field of 2.76 mm provides a total sample area of 0.513 mm^2 (before velocity reject).

8. The OAP-200X determines particle size using a linear array of photodiodes to sense the shadowing of array elements. Particles passing through the field of view illuminated by its laser area imaged as shadowgraphs on the array and a flip-flop memory element is set if the photodiode elements are darkened. Size is given by the number of elements set by a particle's passage, the size of each array element, and the optical magnification. Magnification is set for a size range of 20 to 300 μm, and 24 active photodiode elements divide particles into 15 size channels, each 20 μm wide. It is capable of sizing particles with velocities of 5 to 100 meters/sec (10 to 194 knots). Depth of field, effective array width, and sample area vary with sensed particle size to a maximum of 61mm, 0.44mm and 18.3mm^2, respectively.

9. The SIIDS was designed by Meteorological Research Inc. and is a data acquisition system programmed specifically for icing studies. A more complete description appears in the user's guide (ref 11, app A). It consists of four main components: a microprocessor, Techtran data cassette recorder, Axiom printer, and an operator control panel. The SIIDS has three operational modes: (1) data acquisition, in which averaged raw data are recorded on cassette tape and engineering units are displayed on the printer, (2) a playback mode in which raw averaged data read from the cassette are converted to engineering units displayed on the printer, and (3) a monitor mode used to set the calendar clock and alter programmed constants. During data acquisition, the operator may select an averaging period of 1/2, 1, 2, 5, or 10 seconds. The following parameters are displayed on the SIIDS printer in engineering units.

a. calendar: year, month, day, hour, minute and second

b. pressure altitude (feet)

c. airspeed (knots)

d. outside air temperature (°C)

e. dew point (°C)

f. total LWC observed by the FSSP (gm/m^3)

g. total LWC observed by both FSSP and OAP (gm/m^3)

h. median volumetric diameter (μm)

i. amount of LWC observed for each channel (total 30) of both probes (gm/m^3)

10. The Cloud Technology ice detector has a calibrated resistance wire which is mounted in the airstream and connected as one branch of a balanced bridge circuit. This wire is heated by an electric current. As the water droplets in the cloud strike the wire, they are evaporated, cooling the wire and decreasing its resistance. The change in resistance causes the bridge to become unbalanced. The degree of unbalance is a function of the LWC of the cloud. A second resistance wire, mounted with its axis parallel to the airstream direction and hence not subject to water-drop impingement, is connected as an adjacent branch of the bridge. This wire serves to compensate for variations in airspeed, altitude, and air temperature, so that the bridge becomes unbalanced only in the presence of water droplets. The output of the bridge is proportional to the rate of impingement of water on the sensing wire. This signal is converted to concentration of water per unit volume of air by means of an adjustment for true airspeed.

11. The Mark 12 ice detector draws in ambient air by means of an aspirator. During icing conditions, ice builds up on the sensor probe at a rate proportional to the LWC of the ambient air. This ice, accreted on the probe, occludes an infrared light beam crossing the central area of the probe at an oblique angle. The degree of occlusion is determined by the amount of ice on the probe. When the light occlusion reaches a predetermined level, representing maximum permitted ice thickness, a heating cycle is initiated to remove ice from the probe. The shedding of built-up ice restores the original light path conditions, the heating cycle is terminated and the accretion operation is resumed. The time taken to accrete ice between two predefined levels is used as a measure of the icing rate from which the LWC can be calculated.

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. The deice system on the AH-64A helicopter was functionally tested prior to each icing flight. All anti-ice systems (i.e., pitot heat, windshield anti-ice, and engine air induction system anti-ice) were activated prior to cloud immersion. The JU-21A icing scout aircraft located and documented the icing condition before the test aircraft entered the icing environment. The JU-21A then loitered in the area to facilitate a post-immersion rendezvous with the test aircraft for photographic documentation. The liquid water content (LWC), median volumetric diameter, and ambient temperature were documented by the JU-21A aircraft. The Rosemount icing rate meter in the test aircraft was also used to monitor LWC in natural clouds.

ICE ACCRETION AND SHEDDING

2. Ice accretions were documented using hand-held video from the chase aircraft and four video cameras mounted on the test aircraft. Post-flight photographs were made to document the ice remaining on the individual components of the airframe and rotors.

WEIGHT AND BALANCE

3. Prior to testing, the aircraft gross weight and longitudinal and lateral centers of gravity (cg) were determined by using calibarated scales. The aircraft was weighed with full instrumentation onboard, without fuel, and with auxiliary fuel tanks installed inboard and rocket pods outboard. The aircraft basic weight was 11,594 pounds with a longitudinal cg location at fuselage station 208.9 and a lateral cg location at buttline -0.2 left.

DEFINITIONS

4. Icing characteristics were described using the following definitions of icing severity. These definitions may be found in FM 1-230.

a. Trace Icing: Ice becomes perceptible. Rate of accumulation slightly greater than rate of sublimation. It is not hazardous even though deicing equipment is not used, unless encountered for an extended period of time (over 1 hour).

than this median diameter. The SIIDS data averaging intervals (sample accumulation rate) normally used was ten seconds for the natural icing environment.

7. To substantiate LWC measurements obtained from other devices, two visual accretion probes are mounted on the JU-21A. Mounted on the left side of the aircraft is the Aeroplane and Armament Experimental Establishment (A&AEE) Vernier Accretion Meter (Harvey-Smith). On the right side is the Small Airfoil Section probe constructed from an OH-6 tail rotor section. Readings of ice accretion on both probes were taken at timed intervals in order to determine the accretion rate. By visually aligning the datum lines of the "Harvey-Smith" (photo 1), the apex of the "V" was at zero on the calibrated scale (fig. 1). The pilot reads the ice depth on the scale at the point where the accreted ice on one leg of the "V" crossed the varnier scale on the other leg. A calibration chart (fig. 2) was used to determine LWC from the indicated icing rate (mm/minute) and airspeed. The Small Airfoil Section probe has a protruding 3/16-inch diameter steel rod painted with multi-colored 1/4-inch stripes to provide a reference for ice thickness estimation. Figure 3 was used to determine LWC for this probe's indicated icing rate (inches/minute).

8. Figure 2 was developed from the equation:

$$I = (V_t \times t \times 0.1852 \times LWC)/0.8 \qquad (1)$$

Where:

I = potential ice accretion (cm)
V_t = true airspeed (knots)
t = elapsed time in cloud (hours)
0.1852 = conversion factor ($m^3/nm\text{-}cm^2$)
LWC = liquid water content (gm/m^3)
0.8 = assumed density of ice (gm/cm^3)

The same equation is used to develop figure 3 by applying a conversion factor (2.54 cm/inch). Equation 1 assumes a catch efficiency of 100%. From tunnel tests conducted by A&AEE, catch efficiencies approaching 90% (drop size 15 μm at 120 knots) are obtainable for LWC at:

up to 1.0 g/m^2 at -20°C
up to 0.8 g/m^3 at -10°C
up to 0.5 g/m^3 at - 6°C
up to 0.2 g/m^3 at -3.5°C

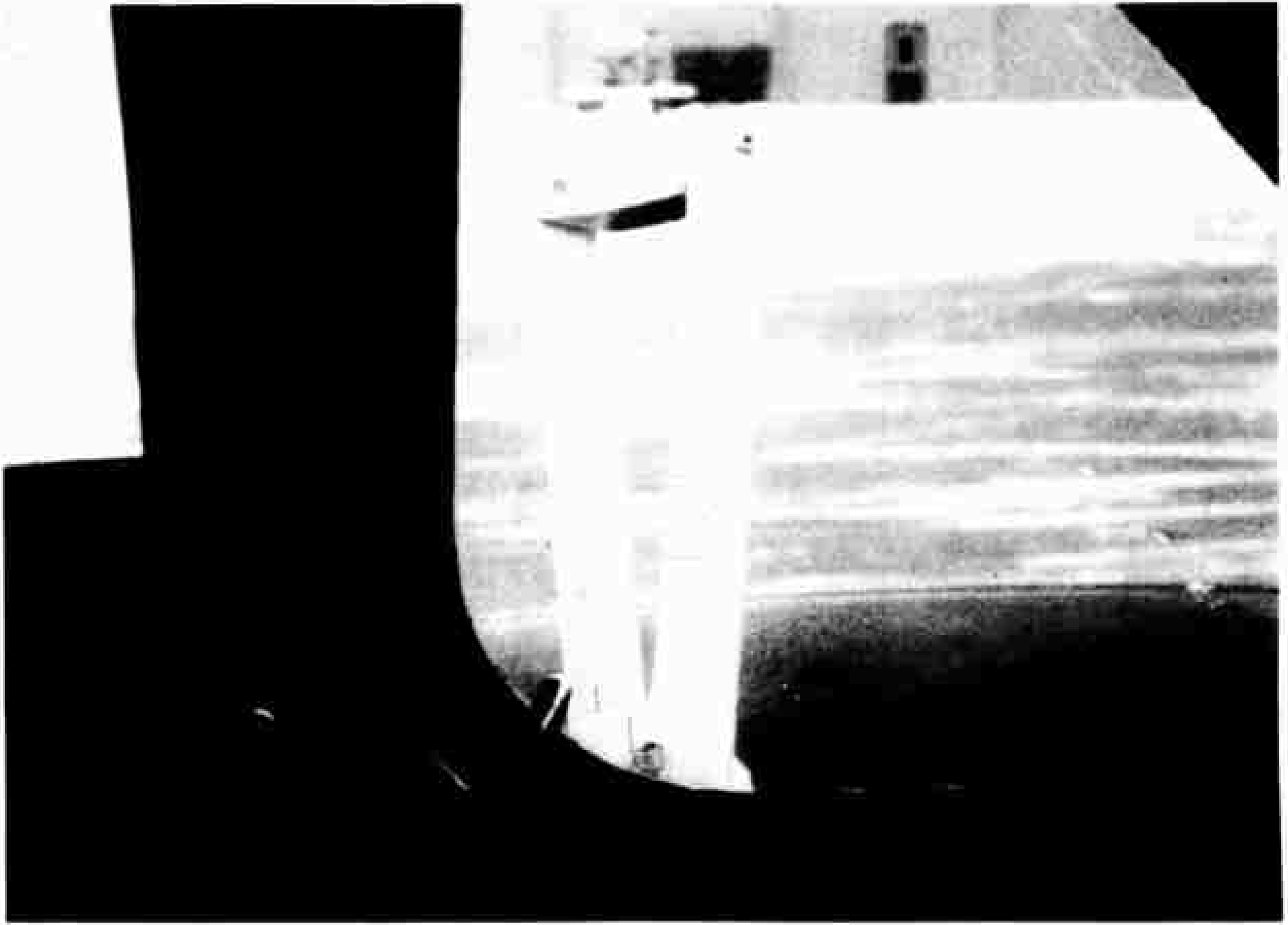


Photo 1. Harvey-Smith Meter - As Viewed by Pilot in the Left Seat

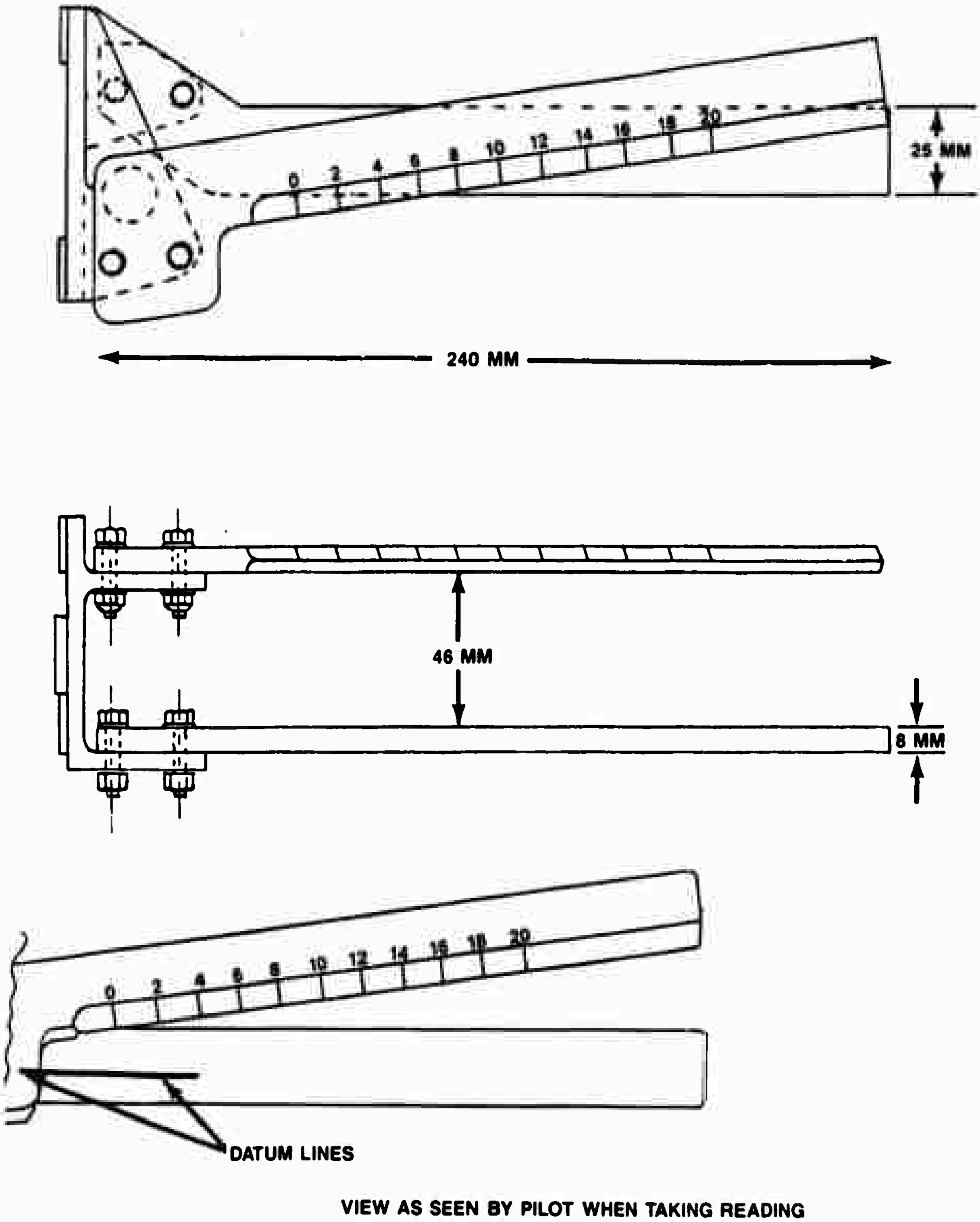


Figure 1. Vernier Accretion Meter (Harvey-Smith)

FIGURE 2
A&AEE VERNIER ACCRETION METER
"HARVEY-SMITH"
LIQUID WATER CONTENT CALIBRATION CHART

NOTES: 1. CALIBRATION BASED ON 100% CATCH EFFICIENCY
2. FREEZING FRACTION OF UNITY

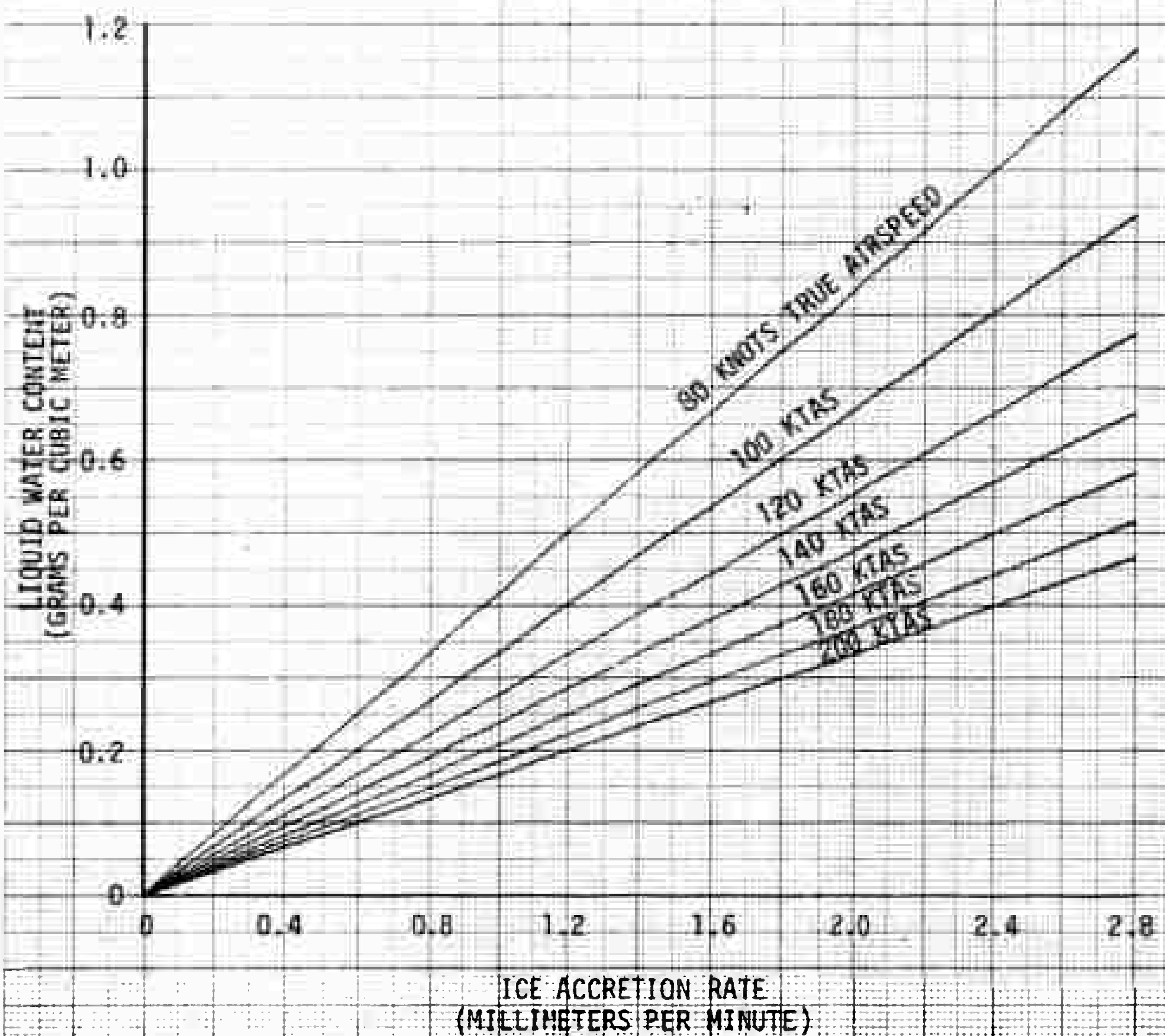
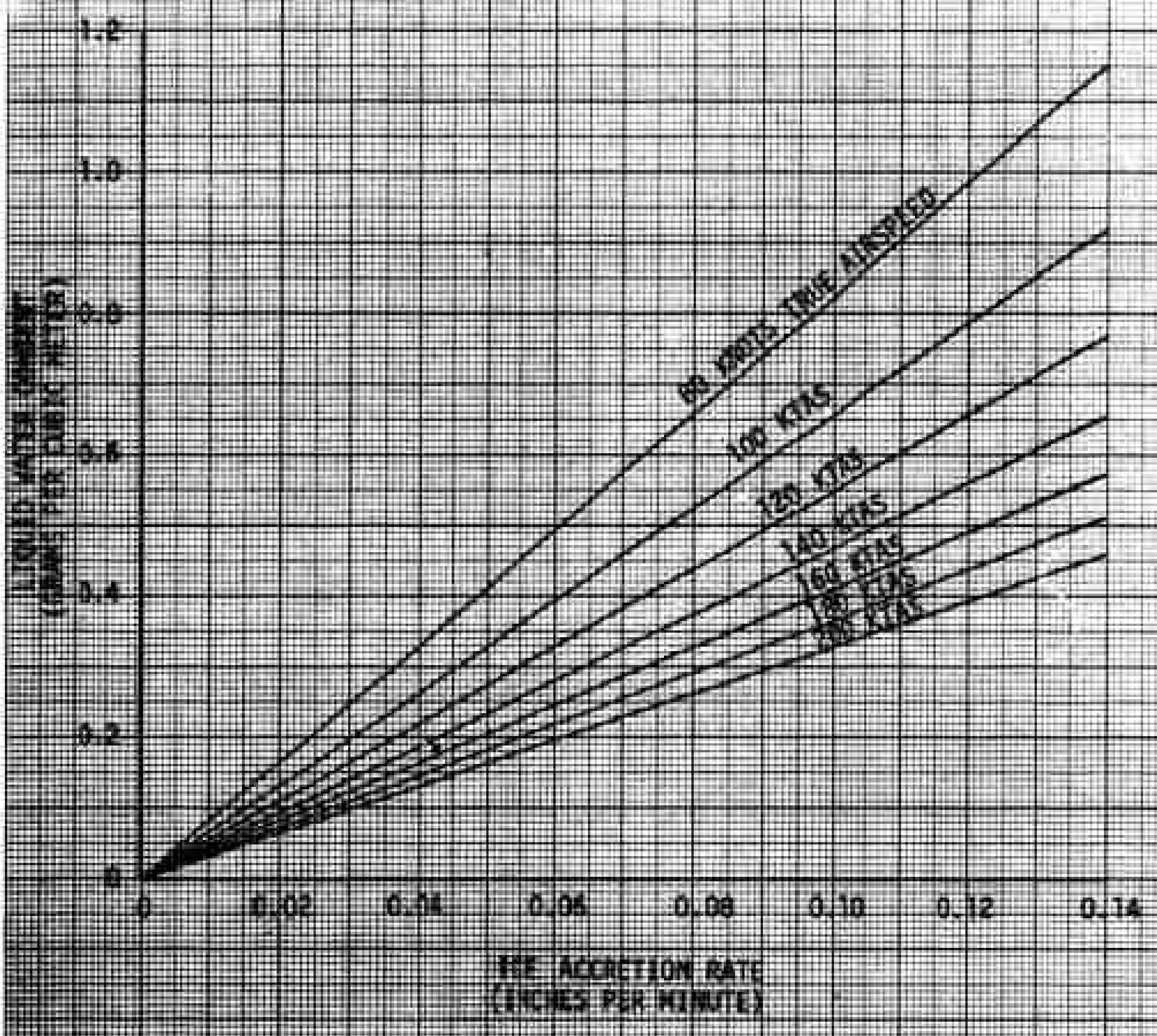


FIGURE 8
SMALL AIRFOIL SECTION PROBE
LIQUID WATER CONTENT CALIBRATION CHART

NOTES: 1. CALIBRATION BASED ON 100% CATCH EFFICIENCY
2. FREEZING FRACTION OF UNITY



APPENDIX E. TEST DATA

Table 1. Specific Test Conditions[1]

MDHC[2] Test Number	Date	Average Gross Weight (lb)	Average Longitudinal Center of Gravity Location (FS)	Average True Airspeed (knots)	Average Density Altitude (feet)	Average OAT (°C)	Time in Cloud (minutes)	Median Volumetric Diameter (microns)	Average Aspirated Rosemount LWC[3] (gm/m³)	Maximum Aspirated Rosemount LWC (gm/m³)	U-21 Instrumentation: Average FSSP[4] LWC (gm/m³)	U-21 Instrumentation: Average CT[5] LWC (gm/m³)	U-21 Instrumentation: Average Leigh Mk 12 LWC (gm/m³)	U-21 Instrumentation: Average Harvey-Smith LWC (gm/m³)
4	3 Nov 86	14,610	205.8	118	5250	-8.0	20	12	0.18	0.23	0.13	0.12	N/A	N/A
5	8 Nov 86	14,540	205.8	119	3600	-11.5	43	15	0.52	0.81	0.38	0.22	N/A	N/A
6	14 Nov 86	14,150	205.2	123	2140	-11.0	23	12	0.19	0.22	0.20	0.16	0.23	0.23
							51		0.28	0.30	0.24	0.23	0.28	0.23
7	20 Nov 86	14,210	205.4	124	5630	-14.0	60	15	0.33	0.40	0.39	0.20	0.30	0.33
8	30 Nov 86	14,700	205.1	126	960	-8.5	83	13	0.43	0.63	0.39	0.35	0.41	0.30
9	8 Dec 86	14,100	205.2	125	6060	-11.0	70	18	0.43	0.54	0.45	0.23	0.45	0.46
11	11 Dec 86	15,050	205.5	125	3300	-13.0	30	12	0.14	0.20	0.17	0.13	0.17	0.19
12	17 Dec 86	15,030	205.5	118	800	-8.0	50	8	0.10	0.10	0.09	0.14	0.11	0.14

NOTES:

[1] Rotor speed = 100%, Configuration: Auxiliary fuel tanks inboard, rocket pods outboard.
[2] McDonnell Douglas Helicopter Co.
[3] Liquid water content.
[4] Forward scattering spectrometer probe.
[5] Cloud Technology, Inc. (formerly Johnson-Williams).

FIGURE 1
AH-64 NATURAL ICING TEST CONDITIONS
AH-64A USA S/N 84-24288

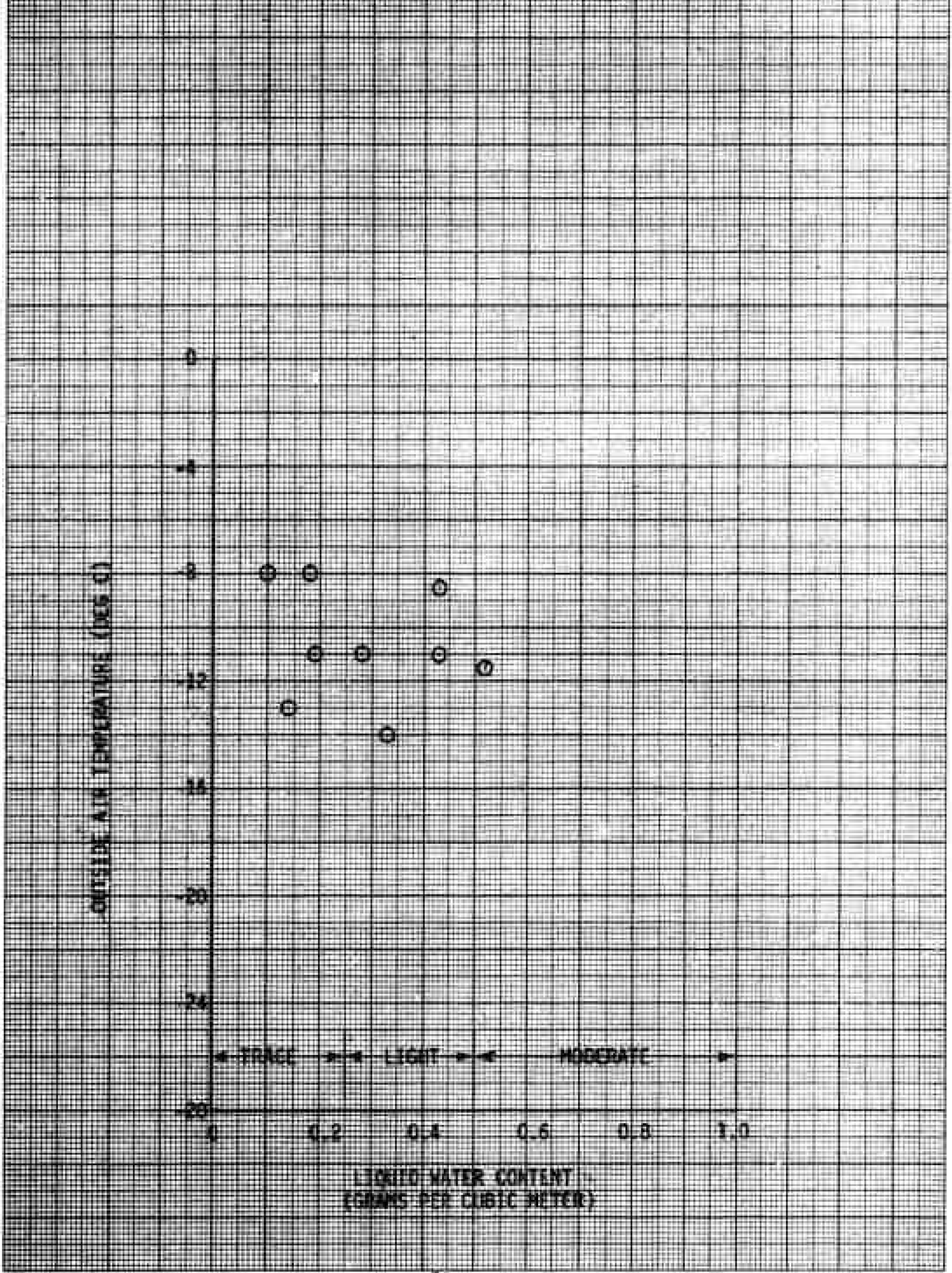


FIGURE 2
BLADE DEICE ON TIME
AH-64A USA S/N 84-24285

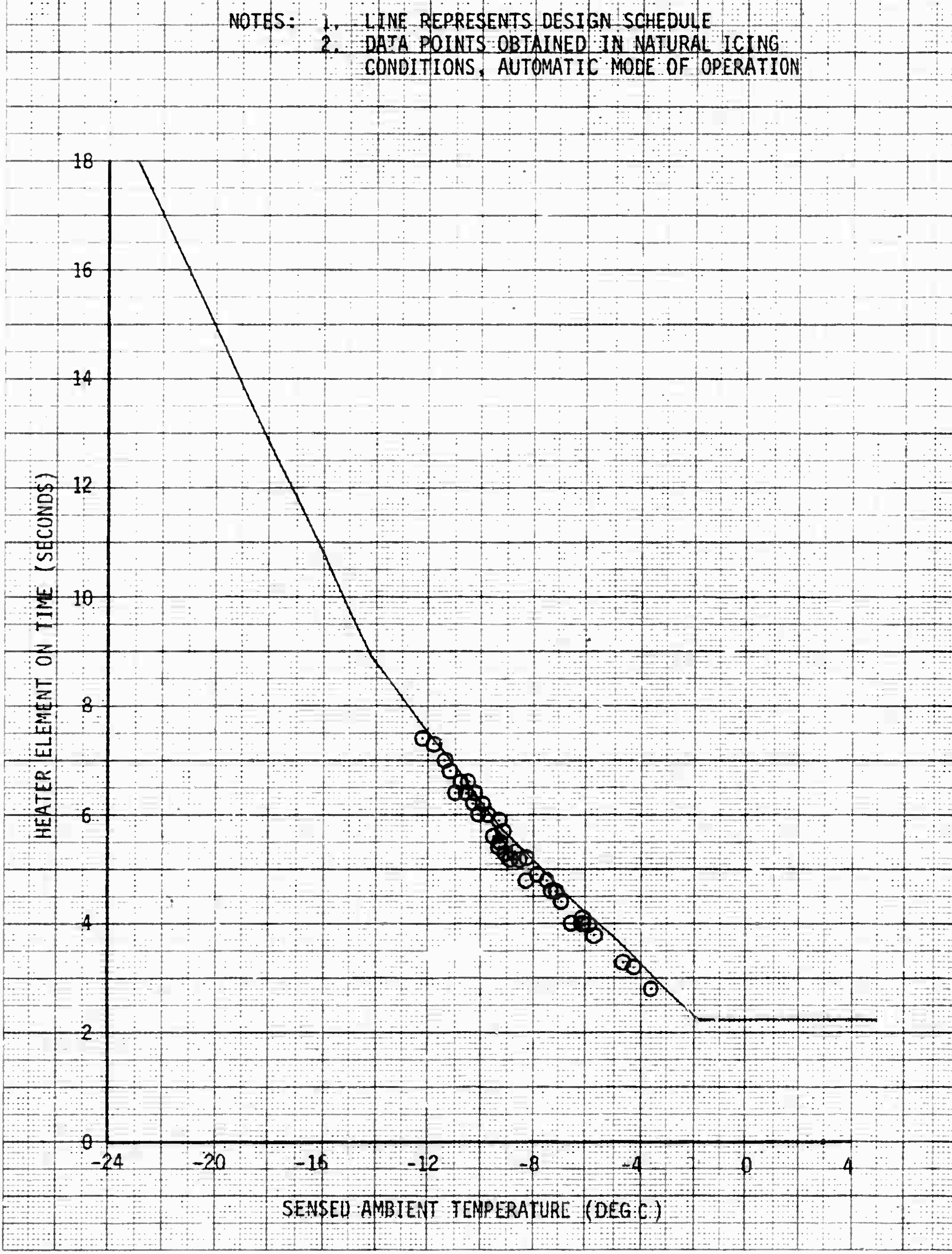
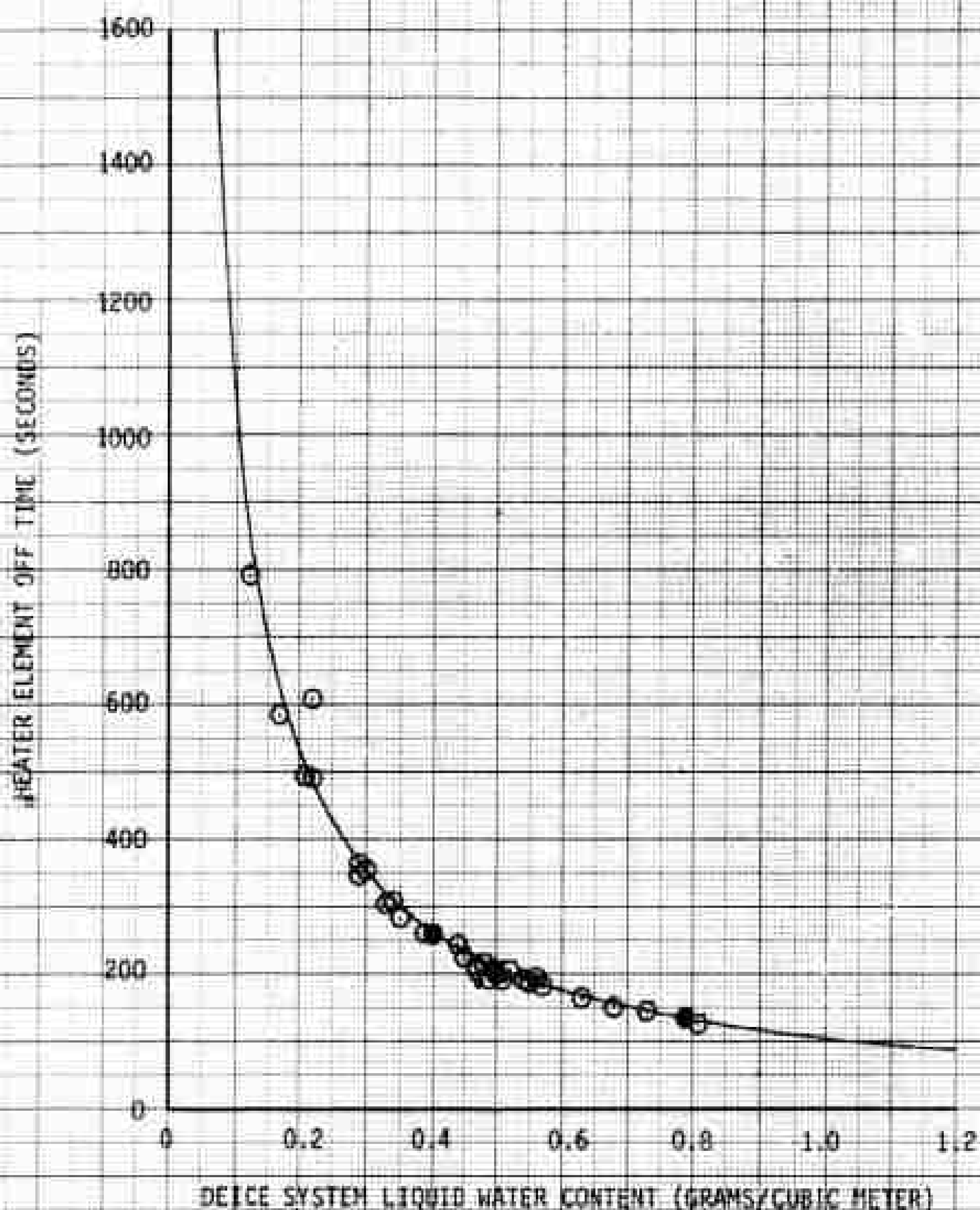


FIGURE 3
BLADE DEICE OFF TIME
AH-64A USA S/N 84-24285

NOTES: 1. LINE REPRESENTS DESIGN SCHEDULE
2. DATA POINTS OBTAINED IN NATURAL ICING CONDITIONS, AUTOMATIC MODE OF OPERATION
3. SHADED SYMBOLS DENOTE MANUAL MODE OF OPERATION



APPENDIX F. REPORT, *ARTIFICIAL AND NATURAL ICING TESTS OF THE AH-64A, USAAEFA PROJECT NO. 84-23*

DEPARTMENT OF THE ARMY
U.S. ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

REPLY TO
ATTENTION OF

SAVTE-TA

SEP 17 1985

SUBJECT: Report, Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23

Commander
US Army Aviation Systems
Command
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1. REFERENCES. a. Systems Specification, Hughes Helicopters, DRC-S-H10000B, 15 April 1982, with all current specification change notices.

b. Letter, USAAEFA, DAVTE-TA, 3 September 1982, subject: Letter of Effort, YAH-64 Icing Survey, USAAEFA Project No. 80-08.

c. Letter, AVSCOM, AMSAV-ED, 10 September 1984, subject: Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23. (Test Request)

d. Test Plan, USAAEFA Project No. 84-23, *Artificial and Natural Icing Tests of the AH-64A* October 1984.

e. Test Plan, Hughes Helicopters, Report No. 77-FT-8005P-3, *Test Plan for Artificial and Natural Icing Tests of Production AH-64A Advanced Attack Helicopter* January 1985.

f. Technical Manual, TM55-1520-238-10, *Operator's Manual for Army AH-64 Helicopter* 28 June 1984.

g. Letter, AVSCOM, AMSAV-E, 25 February 1985, subject: Airworthiness Release for Artificial and Natural Icing Test of the AH-64A Helicopter, USA S/N 82-23356.

2. INTRODUCTION. a. Background. (1) The US Army requires the AH-64 helicopter to operate safely in an icing environment up to and including the moderate level of intensity (ref 1a). The AH-64A incorporates a rotor blade deicing system, anti-icing systems, and ice detection systems. Qualification tests were required to substantiate airworthiness for operation in a moderate icing environment.

(2) A preliminary investigation of the AH-64 helicopter capability to operate in icing conditions was conducted during the 1981-82 icing season (ref 1b). The AH-64 main and tail rotor deicing systems operated satisfactorily when the controller was operated manually, however, the automatic functioning was inadequate. Since that evaluation, the AH-64A rotor deicing system has been redesigned to include a controller provided by AEG-Telefunken. The US Army Aviation Systems Command (AVSCOM) tasked the US Army Aviation Engineering Flight Activity (USAAEFA) to conduct artificial and natural icing tests of the AH-64A during the winter of 1984-1985 (ref 1c), in accordance with the approved test plans (refs 1d and 1e).

b. Test Objectives. The objectives of this test were to conduct artificial and natural icing flight tests of the AH-64A helicopter to:

(1) Determine the effectiveness of the AH-64A ice protection and detection systems.

(2) Determine the impact of ice accumulation on performance and handling qualities.

(3) Determine the capability of the AH-64A to operate in moderate icing conditions.

c. Description. (1) The AH-64A is a two-place, tandem seat, twin engine helicopter with four-bladed main and anti-torque rotors and conventional wheel landing gear. The helicopter is manufactured by Hughes Helicopter Incorporated (HHI) and is powered by two General Electric T700-GE-701 turboshaft engines. The AH-64A has a movable horizontal stabilator with three modes of operation: Manual, Automatic and Nap-of-the-earth/Approach. A 30mm gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has a wing with two pylons on each side for carrying Hellfire missiles, 2.75-inch folding fin aerial rockets and/or external fuel tanks.

(2) The deice system consisted of an outside air temperature sensor, ice detector, icing rate meter, blade deice control panel, slip rings for the main and tail rotor, and a deice controller. The main and tail rotor blades contained electrothermal resistive heating mats. Anti-ice systems were provided for sections of the windshield, pitot-static tubes, air data sensor, engines, engine inlets, nose gearbox and cross shaft fairings, target acquisition and designation system/pilot night vision system (TADS/PNVS). Deice capability was also provided for one of the Hellfire missiles. The test helicopter was USA serial number (S/N) 82-23356 (production vehicle 02). Further description of the helicopter may be found in the system specification (ref 1a), the operator's manual (ref 1f), and enclosure 1.

d. Test Scope. Clear air and artificial and natural icing flight tests were conducted in the vicinity of Duluth, Minnesota from 26 February through 11 April 1985. A joint contractor and USAAEFA test team was used for these tests. A total of 24 flights were conducted totaling 17.8 productive hours. Of these flights, 5 were in clear air totaling 3.0 hours, 13 were in the artificial icing environment totaling 8.3 hours of cloud immersion, and 6 were in natural icing for 6.5 hours of cloud immersion. The aircraft was flown with three different stores configurations: 8 Hellfire (4 Hellfire dummy missiles on each inboard pylon), 8 Hellfire inboard and 2.75-inch rocket pods outboard, and inboard Hellfire and outboard rocket pod on one side and outboard Hellfire and inboard rocket pod on the other side. A summary of test conditions is presented in table 1. Specific icing conditions are presented in table 1, enclosure 4. Anti-ice and deice systems were operated continuously while in the icing environment. During one flight in natural icing conditions, heater OFF times for the deice system were manually selected. The frangible Hellfire deice system was not activated since satisfactory system performance was previously demonstrated (ref 1b). Flight limitations contained in the operator's manual and the airworthiness release (ref 1g) were observed during testing.

e. Test Methodology. (1) Artificial icing was conducted by flying in a spray cloud generated from the Helicopter Icing Spray System (HISS). The JU-21A aircraft configured with the cloud particle measuring system was used to document the HISS cloud and provide photographic coverage while the test aircraft was in the artificial cloud. Ice accretion was also documented on the ground following icing encounters. A detailed discussion of the test sequence and procedures is contained in references 1d and 1e.

(2) Natural icing tests were conducted by flying in instrument meteorological conditions (IMC) using instrument flight rules (IFR). The JU-21A chase aircraft configured with the cloud particle measuring system assisted in locating and documenting the icing conditions. Photos were taken from the JU-21A after the test aircraft exited the icing environment into visual meteorological conditions. Close coordination between air traffic control and the chase and test aircraft crews was required to find and stay in the icing environment and to implement in-flight aircraft rendezvous for photographic documentation. Procedures used included radar vectoring, navigational aid holding, and block airspace assignment. Time in the clouds was limited by the availability of the natural icing conditions and aircraft IFR fuel requirements.

(3) Test data were recorded on magnetic tape in pulse code modulation format. A detailed description of special equipment and instrumentation is provided in enclosure 2.

(4) Test techniques and data analysis methods are presented in enclosure 3. These include methods used to determine cloud parameters and definitions of icing types and severities. Qualitative vibration assessment ratings were assigned in accordance with the Vibration Rating Scale (VRS).

Table 1. Test Conditions[1]

Icing Environment	Immersed Component	Number of Flights	Outside Air Temperature (deg C)	Liquid Water Content (gm/m^3)	Relative Humidity (%)	True Airspeed (kts)	Time in Cloud (hrs)
Artificial	Rotor Systems	9	-5.0 to -19.5	0.51 to 1.20	30 to 95	90, 95, 100	4.3
Artificial	Fuselage	4	-6.0 to -19.5	0.59 to 1.21	25 to 70	95	4
Natural	---	6	-7.0 to -18.0	0.10 to 0.45	N/A	45 to 119	6.5
Clear Air[2]	---	5	-3.0 to -14.0	N/A	N/A	90 to 130	---

NOTES:

[1]Average Gross Weight = 15,800 to 16,220 pounds.
Average Longitudinal CG Location = FS 204.8 to FS 205.9.
Main Rotor Speed = 289 rpm.
[2]3.0 hours productive flight time.

3. RESULTS AND DISCUSSION. a. General. Artificial and natural icing tests were conducted to establish a moderate icing envelope through 1.0 gm/m^3 liquid water content (LWC)) for the AH-64A Apache helicopter. A joint contractor (HHI) and USAAEFA test team performed the evaluation with HHI having primary responsibility for the development phase of testing and the government having primary responsibility during the second phase, or government evaluation. The government evaluation was not completed during these tests due to the ice detection system development requirements and unavailability of natural icing conditions at the conclusions of the test period. A summary of the specific test conditions for each flight is presented in table 1, enclosure 4. Additionally, the specific icing conditions in which the AH-64A was tested are presented in figure 1, enclosure 4 for the artificial environment and figure 2, enclosure 4 for the natural environment. The AH-64 rotor deice system operational characteristic should be further evaluated in the natural icing environment to verify calibration and modified damping characteristics of the ice detection subsystem. The cold aft cockpit environment during flight at approximately -20°C without the nonstandard boot installed on the 30mm gun turret and a properly sealed emergency air door was determined to be a deficiency. There were additionally eight shortcomings the most significant of which are:

(1) Debonding of the tail rotor elastomeric bearings in cold weather.

(2) Ice impact damage to the tail rotor blades.

(3) High potential for engine foreign object damage (FOD) due to shed ice from the fairing mismatch between th nose gearbox and cross shaft fairing.

b. Deice System Operation. (1) Rotor Deice System. The AH-64A helicopter deice system was evaluated for operational characteristics and effectiveness during 8.3 hours in the artificial icing environment and 6.5 hours in natural icing. During the artificial icing tests, the rotor deice system heater-on time was automatically controlled by the AEG Telefunken deice controller. Heater-off time was manually controlled using a test instrumentation panel with a signal equivalent to the HISS spray cloud measured LWC. Initially the heater-on time schedule corresponded to the dashed line shown in figure 3, enclosure 4. When problems were encountered with ice shedding characteristics at -5°C and below -15°C ambient temperature, the heater-on schedule was adjusted as shown in figure 3 for the remainder of the tests. The heater-off time conformed to the design line as shown in figure 4, enclosure 4. During one flight in natural icing conditions heater-off times were manually selected with the ship's control panel to verify system operation in the event that the automatic mode failed. No evidence of runback was observed during or following any of these tests. No residual ice accretions were observed on the tail rotor protected surfaces. Photographs showed some residual ice on the main rotor blade protected surfaces. The accreted ice on the main rotor blades did not appear to shed during every deice cycle especially between 30% to 70% of the main rotor span. Subsequent deice cycles were required to shed this residual ice accretion completely. No objectionable aircraft vibrations or problems were noted as a result of this residual ice. At the higher LWCs and the colder ambient temperatures apparent but not annoying vibration levels (VRS 5) were noted in the cockpit. The AH-64A rotor deice system operational characteristics should be further evaluated in natural icing conditions.

(2) Automatic Outside Air Temperature (OAT) Sensing Subsystem. During the initial clear air and artificial icing flights it was determined that the automatic OAT subsystem did not interface properly with the Telefunken deice system controller. A mismatch in amperage between the controller and the Lewis temperature probe caused the probe to self heat. The Telefunken controller was modified to compensate for this erroneous temperature signal. This modification allowed full integration and proper operation of the automatic OAT sensing subsystem. During subsequent artificial icing tests, the ice shedding characteristics of the main rotor blades at -5°C and below -15°C ambient temperature necessitated two changes to the Telefunken controller to modify the heater-on time schedule. The first change involved establishing 1.9 seconds as the minimum heater-on time at -5.2°C and all warmer ambient temperatures. The second change increased the rest of the schedule by 20%. This modified schedule is shown in figure 3, enclosure 4 and was used for the remainder of the testing.

(3) Ice Detection Subsystem. The ice detection subsystem consisted of a Rosemount ice detector in the left engine inlet (photo 1, encl 6) and an icing rate meter located on the pilot's instrument panel. The size and composition of the HISS cloud precluded correct functioning of the ice detector in the artificial icing cloud, and, therefore, automatic system operation was only evaluated in the natural environment. Initial natural testing in stratiform clouds indicated that the ice detector was operating in the saturated mode (LWC indications of approximately 2.0 gm/m^3) resulting in system OFF times much shorter than the design schedule for the actual LWC. The saturated condition also caused the icing severity meter on the pilot's instrument panel to indicate higher icing intensity than the actual conditions. Analysis of the data by Rosemount personnel revealed two problems: relay chattering caused by aircraft vibrations and a crystal failure. Once these were corrected, further natural icing tests in cumuliform clouds indicated a problem with the damping characteristics of the ice detector. Subsequent icing wind tunnel tests at the Rosemount facility in Burnsville, Minnesota revealed essentially no damping in the system. Changes in LWC resulted in large and persistent fluctuations in the undamped icing severity signal. Rosemount changed the damping characteristics of the ice detector, however, this modification was never evaluated in flight because of a lack of natural icing test conditions. A verification of the calibration and modified damping characteristics of the ice detection subsystem should be accomplished in the natural icing environment.

c. Anti-Ice Systems Operation. (1) General. The AH-64A anti-ice systems were evaluated for operational characteristics and effectiveness during rotor and fuselage artificial icing exposures of 4.3 and 4.0 productive flight hours respectively, as well as 6.5 hours of natural icing testing. Anti-ice systems on the AH-64A helicopter protect the engine, engine inlet cowling (bleed air), nose gearbox and cross shaft fairing, windshield, pitot-static tubes, air data sensor (ADS), and TADS/PNVS. All anti-ice systems were activated prior to entering the icing environment and were operational for all icing flights. A detailed description of each system is presented in enclosure 1.

(2) Engine. Engine anti-icing was accomplished by a combination of hot air from the axial compressor and heat transfer from the air/oil cooler in the engine frame. The system was controlled by one engine anti-ice switch located on the pilot's lower left subpanel (fig. 1). The engines were visually inspected (including borescope) daily and an engine health indication test was performed prior to every flight. No engine deterioration was noted during this program. There were no indications of ice accumulation in the engine. The engine anti-ice system demonstrated satisfactory operation in the icing environment.

(3) Engine Inlet Cowling Ring. Engine bleed air heated the engine inlet cowling ring to prevent ice accumulation. The cowl ring anti-ice surface temperature and anti-ice characteristics were evaluated throughout the artificial and natural icing tests. There were no accumulations of ice noted on the cowling ring throughout the tests. The engine inlet cowling ring anti-ice system demonstrated satisfactory operation.

(4) Nose Gearbox and Cross Shaft Fairings. The nose gearbox and cross shaft fairings were anti-iced electrically using embedded heater elements in the Kevlar material. These electric heaters were activated by the same single two position switch on the pilot's anti-ice panel (fig. 1) that activates the engine air inlet bleed air system. Surface temperatures were recorded throughout the icing tests. Some ice accumulation was observed on the front of the nose gearbox oil cooler but no adverse effects were noted. No ice accretion was noted on the heated surfaces, however, ice did accrete on the sharp edge of a poorly fitted fairing and may have caused engine ice ingestion damage which will be discussed later. The anti-ice provisions of the nose gearbox and cross shaft fairings are satisfactory.

(5) Heated Windshields. The two panels forming the pilot and copilot/gunner's (CPG) windshield were anti-iced electrothermally. The windshield temperature was regulated between 65° and 85°F (18° to 29°C) which kept the heated areas ice free throughout these tests. The field-of-view through the CPG windshield was occasionally obscured in the artificial icing environment by liquid water, requiring windshield wiper operation, (para 3e(5)). The heated windshield panels remained free of ice in the natural icing environment. Ice accretion characteristics of the AH-64A heated windshield are satisfactory.

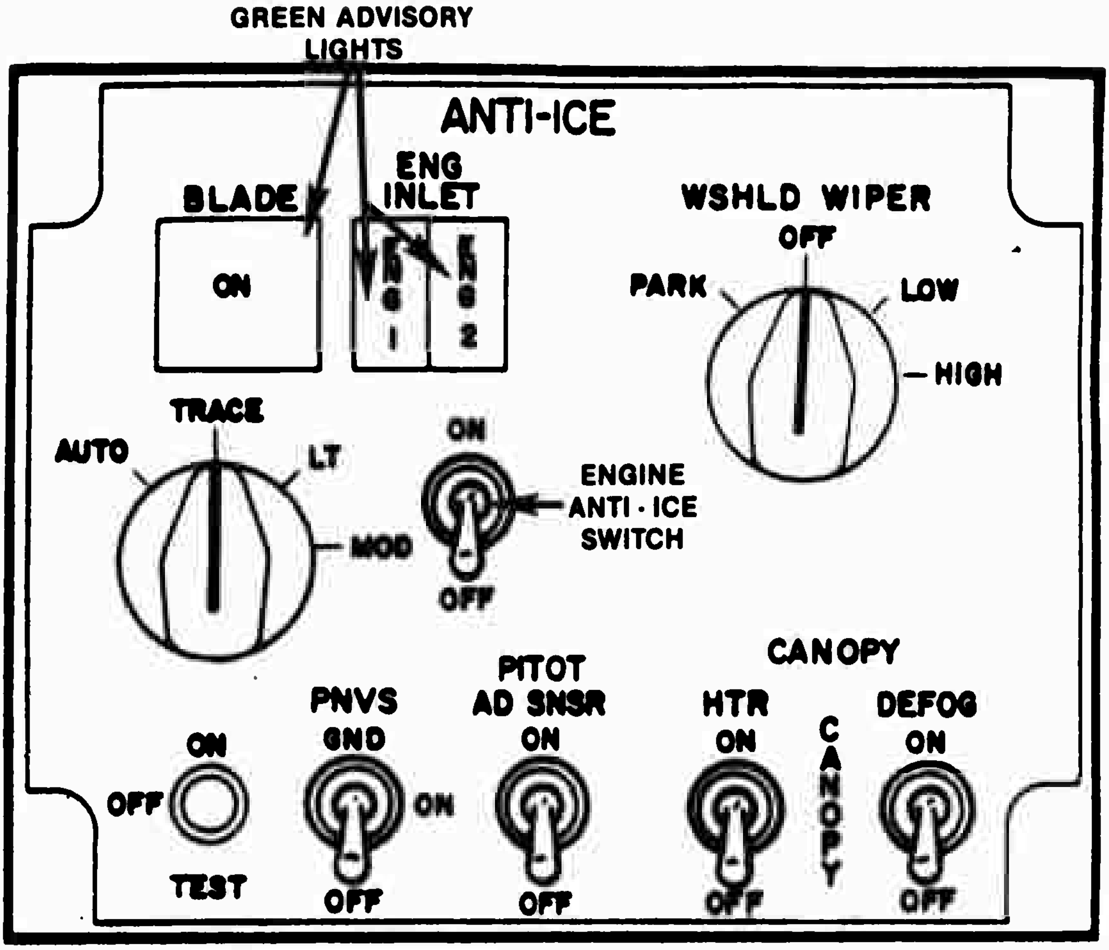

Figure 1. Pilot Anti-Ice Panel

(6) Pitot Tubes. The dual, wing mounted pitot tubes were anti-iced electrically and controlled by a single switch on the pilot's anti-ice control panel (fig. 1). The pitot heat was activated for all flights in icing conditions. No ice accretion was observed and the system operated without failure.

(7) Air Data Sensor (ADS). The ADS was located above the main rotor and attached to a standpipe through the main rotor mast. The rotating hub and arms of the ADS were heated. Control of the anti-ice function of the system was by the same switch as the pitot heat. Ice did not accrete on the heated portions of the ADS in either the natural or artificial icing environments.

(8) Target Acquisition and Designation/Pilot Night Vision System. The target acquisition and designation/pilot night vision system (TADS/PNVS) anti-ice provisions included window, window frame, and selected turret surface panel heating. During the artificial icing fuselage immersions, the PNVS was coupled to the pilot's helmet while the TADS turret was fixed forward but was slewed periodically to determine the effects of ice accretions on range of motion. Several changes were made to the TADS/PNVS anti-ice system during the evaluation in an attempt to improve performance. However, the final configuration tested failed to provide adequate anti-icing in artificial icing conditions at high LWC (1.0 gm/m^3). Ice buildups on the TADS/PNVS windows severely restricted the field-of-view and the PNVS horizontal range of motion after approximately 20 minutes in the cloud. The anti-ice system functioned satisfactorily in natural icing conditions at low LWC (0.3 to 0.4 gm/m^3) which was the most severe natural conditions available. The TADS/PNVS anti-ice system should be evaluated in moderate natural icing conditions (LWC > 0.5 gm/m^3).

d. Flight Control Surface Ice Accretion and Shedding Characteristics. (1) General. Flight control surface ice accretion and shedding characteristics were evaluated throughout these icing tests. Specific test conditions are presented in table 1 and figures 1 and 2, enclosure 4. No in-flight or post-flight difficulties associated with flight control surface ice accretion or shedding were identified.

(2) Main Rotor Blades. Ice that accreted on the heated surfaces of the main rotor blades was eventually shed though not always after each deice cycle. Minimal amounts of ice accumulated on unprotected surfaces and components of the main blades. Specifically, ice formed and was retained on the blades inboard of the heater mats and on the blade retention mechanisms. These accretions were small and should not present a significant problem either from accretion or subsequent shedding.

(3) Main Rotor Head. A typical ice accumulation on the unprotected main rotor head is shown in photo 2, enclosure 6. Ice as much as 3/4 inches thick accreted on sharp edged components during most icing exposures. Lesser thicknesses of ice were noted on flat surfaces (low catch efficiency) and areas

facing opposite the direction of rotation. No restriction of any movable component on the main rotor head was noted. It could not be determined if aircraft damage could be caused by shedding these ice accumulations. Future natural icing tests should attempt to document the ice shedding characteristics of the main rotor head unprotected rotating components.

(4) Tail Rotor. The tail rotor blades and hub area accreted ice in a manner similar to the main rotor system. High speed still and motion picture photography documented ice shedding from the protected portions of the tail rotor blades. Residual ice accumulations on unprotected portions of the tail rotor blades and hub area were noted but no difficulties were associated with these accretions. Ice shed from the tail rotor repeatedly struck the stabilator causing minor dents (photo 3, encl 6). Tail rotor blades were damaged during flights in both artificial and natural conditions. On two occasions in artificial conditions when the blade deice system was not functioning properly, a tail rotor blade was struck by shed ice causing debonding of the erosion shield and subsequent replacement. During a natural icing flight at -7.5°C and average LWC of 0.37 gm/m^3, two tail rotor blades were damaged in a similar manner. In none of the cases did the damage cause a noticeable increase in vibrations or change in aircraft handling characteristics. The Equipment Performance Reports submitted by USAAEFA on the incidents are presented in enclosure 5. The modified calibration and damping characteristics of the ice detection system (para 3b(3)), scheduled for evaluation in future icing tests, may alter the potential for tail rotor damage due to main rotor ice shedding. Ice impact damage to the tail rotor blades is a shortcoming.

(5) Stabilator. The ice accretion and shedding characteristics of the stabilator were evaluated throughout the program. Minor ice buildups were observed on the stabilator leading edge. No problems were identified due to these accretions.

e. Airframe Ice Accretion and Shedding Characteristics. (1) General. The airframe ice accretion and shedding characteristics were evaluated at the test conditions shown in table 1 and figures 1 and 2, enclosure 4. Ice formed on all stagnation areas and sharp protrusions from the airframe. Large ice formations accreted on the canopy frames, handholds, and poorly fitting panels on the cross shaft fairing. Also, the CPG windshield wiper system performed poorly in the artificial icing cloud.

(2) Fuselage. The fuselage ice accretion and shedding characteristics were evaluated during four artificial icing test flights and six natural icing test flights. Typical fuselage ice accretions are shown in photo 4, enclosure 6. Large accumulations of ice were noted on the wing leading edges, nose, steps, handholds, landing gear, landing gear struts, tail wheel, vertical stabilizer/ tail rotor pylon leading edge and many other surface irregularities on the fuselage. No operating difficulties were identified due to these ice

accumulations. All windows, doors and access panels remained functional after these icing encounters.

(3) Unprotected Windows. All window panels except the two electrically heated forward facing windows discussed in paragraph 3c(5) were unprotected from ice accumulation. During flight in the icing environment, minimal ice accreted on these unprotected window areas. Ice crystals were observed on an area of about 1/2 square foot directly above the pilot's head. This area is not used extensively for critical outside reference and these accumulations were not significant for normal flight, but could possibly reduce the pilot's field-of-view if the aircraft was required to perform an air-to-air combat mission immediately following flight in icing conditions.

(4) Canopy Frames. The framework supporting the essentially flat plate canopy was not anti-iced. The forward canopy frames accreted large quantities of ice (photo 4, encl 6). Supercooled water droplets which impinged upon the heated glass in some cases pooled on the window and the windshield wiper was used to aid removal. This water and the normally accreted ice combined (particularly around the CPG window) to form significant sized hard ice accumulations on the canopy frames and windshield wiper arms. During flights these ice formations became large enough to be broken away by free stream air and impinged on the nose gearbox or were ingested by the engine. The potential shedding problem created by these ice accretions should be further evaluated in the natural icing environment.

(5) Windshield Wipers. Frequent use of the windshield wiper system was required when flying behind the spray aircraft, particularly on the CPG windshield. The higher angle of incidence of this window relative to free stream air flow allowed for greater amounts of water impingement and pooling effects than the shallow angle on the pilot's window. The canopy frame ice (paragraph above) immobilized the windshield wipers by freezing them to the canopy frame on several occasions. This was not encountered in natural icing tests. The poor performance of the AH-64A CPG windshield wiper in the HISS cloud (artificial icing) should be further evaluated in natural icing conditions.

(6) Engine Ice Ingestion. In the artificial environment, several instances of ice particles entering the engine inlets were noted by the chase aircraft. No unusual cockpit engine indications were observed and borescope inspections failed to reveal any compressor damage. Following a natural icing encounter, ground personnel heard a high pitched whine emanating from the No. 2 engine which was not apparent to the flight crew. Borescope inspection revealed a 5/16 to 3/8-inch curl in one first stage compressor blade, requiring engine replacement. Ice could have shed from four possible areas of accretion (photo 4, encl 6) to cause the engine FOD: the handhold on the right side of the aircraft, the canopy frames and windshield wipers, the camera mount on the right forward avionics bay (FAB) (a nonstandard icing instrumentation

configuraticn), or the mismatched fairing fit between the nose gearbox and cross shaft fairings. The mismatched fairing (photo 5, encl 6) is the most probable area for ice buildup resulting in engine FOD because it is positioned directly in front of the engine inlet. Ice accretion on the mismatched fairings was observed on several flights in the artificial icing environemnt (photo 6, encl 6). The mismatched fairing fit between the nose gearbox and cross shaft fairings which creates a high potential for engine FOD due to shed ice is a shortcoming. The susceptibility of the engines to damage from shed ice should continue to be investigated in natural icing conditions.

(7) Antennas. The ice accretion and shedding characteristics of the aircraft antennas were evaluated throughout these tests. Many AH-64A antennas are flush mounted and thus accrete little if any ice. Exceptions to this are the transponder antenna located on the cabin overhead and the two forward facing radar warning antennas located on the front of each FAB. No degradation in aircraft radio transmission or reception was noted on any communications or navigation radios, although no specific tests were conducted to evaluate these characteristics. Ice shed from these antennas presented no operational difficulties. A study should be conducted to determine the possible degradation of reception with ice accumulation on the forward facing radar warning antennas.

(8) M-130 Chaff Dispenser. The ice accretion and shedding characteristics of the M-130 chaff dispenser were evaluated throughout these tests. Two dispensers were installed on the test aircraft, one on each side of the tailboom. The M-130 system was not operated during this evaluation. No ice accretions or subsequent sheds were observed which would interfere with the operation of the system during or after an icing encounter.

(9) Dummy ALQ-144 Infrared (IR) Countermeasures Device. The ice accretion and shedding characteristics of the dummy ALQ-144 IR Countermeasures device were evaluated during these tests. The dummy device was approximately the same shape as an ALQ-144 and was located just aft of the main rotor mast on the top of the fuselage. This location offered the advantage of masking some of the ice particles which would normally impinge and accumulate on the forward surfaces of the device. The resulting accumulations did not adversely effect the operations of the aircraft. Future natural icing tests should use an operational ALQ-144 to verify test results.

(10) Dummy Hellfire Missile System. Ice was accreted on the dummy Hellfire missiles and launcher racks at the specific flight conditions shown in table 1, enclosure 4. Typical accretions are shown in photo 7, enclosure 6. All sharp edges and stagnation points on the missiles and launcher racks accreted ice. Deice tests of the Hellfire missiles were not accomplished. Ice accretion and shedding characteristics of the Hellfire missile system remain the same as previously reported (ref 1b).

(11) Wing Pylon Articulation. Following artificial icing exposures of the fuselage (4 flights), the wing pylons which are articulated in pitch, were moved through the full range of travel. No restriction of pylon travel was detected. Natural icing tests verified these artificial icing test results.

(12) Area Weapon System. The ice accretion and shedding characteristics of the 30mm chain gun area weapon system were evaluated in artificial and natural icing conditions. Ice accreted on all stagnation points and sharp edges as shown in photo 8, enclosure 6. Weapons firing was not attempted following flight in icing conditions. Although live fire operations were not performed, the accreted ice did not appear to hamper the traversing operation of the weapon. The gun was traversed inflight through its full range in azimuth and elevation to verify unrestricted motion. For these flights, a temporary muzzle protector made of cloth tape accreted up to 1 1/2 inches of ice (photo 8, encl 6). Testing should be conducted to determine the effects of firing live ammunition through a 30mm gun muzzle protector with ice accretions. A nonproduction cover was also fitted over the ammunition chute. Without this cover ice would have been allowed to accumulate inside the chute and could prevent proper ammunition feed. Consideration should be given to incorporating the ammunition chute cover as a permanent installation and testing should be conducted to verify proper system operation during live fire exercises with the cover installed.

(13) Rocket Launcher, M200. Rocket launchers for the 2.75 inch folding fin aerial rockets were installed for three flights in artificial icing conditions. Forward facing protective covers were installed on the rocket pods. These covers were made of a thick, black plastic material and appeared to be very durable. Ice accreted on protective covers installed on the rocket launchers (photo 9, encl 6). Tests of these covers and their separation characteristics have not been accomplished. Further testing should be conducted on the 2.75 in. aerial rocket launcher system with the pod covers installed to determine the consequences of the covers departing the launcher, with and without accumulated ice.

f. Performance. (1) Level Flight Performance. Level flight performance characteristics of the AH-64A helicopter were evaluated at the specific natural icing test conditions listed in table 1, enclosure 4. Collective control position was fixed at pre-immersion trim position, altitude was maintained and airspeed was allowed to vary as necessary during the encounter. Figure 6, enclosure 4 is a typical time history of ice accretion over a full deice cycle in natural icing. Power required increases between deice cycles ranged from 3 to 7 percent for the majority of the natural icing conditions. These increases may not be representative since the icing severity indications from the ice detector were always much higher than the actual icing conditions. This erroneous signal resulted in OFF times that were much shorter than the design schedule for the actual LWC. An adjustable LWC signal multiplier was installed on the aircraft for the purpose of this test. When the multiplier was set at 0.4, to approximate actual icing severity, power required increases as high as 12% were recorded. Further tests should be conducted in the natural icing environment after the modified ice detection subsystem damping has been incorporated to document level flight performance degradation due to ice accretion.

(2) Power Loss with Operation of Anti-Ice Systems. The engine performance characteristics were recorded with the bleed air anti-ice systems OFF and ON for comparison. The comparison is shown in figures 7 and 8, enclosure 4. An approximate 50°C increase in measured gas temperature was observed with anti-ice system use. The power available losses with activation of the anti-ice systems observed during these tests are significant. Performance information incorporating these losses should be published in the operator's manual for the pilot to use in preflight planning.

g. Handling Qualities. The effect of airframe and flight control surface ice accretion on the aircraft handling qualities was qualitatively evaluated throughout all the icing flights at the conditions listed in table 1, enclosure 4. The evaluation was accomplished by performing typical instrument flight maneuvers with and without ice on the aircraft. No degradation of aircraft handling qualities were noted as a result of aircraft ice accretion.

h. Vibration. The aircraft vibration characteristics were monitored throughout these evaluations. Qualitative crew comments were compiled during the artificial icing immersions and flights to home base with residual ice still accreted. At -15°C ambient temperature, a noticeable but not annoying increase in overall airframe vibration levels was observed (VRS 5). No significant increase in vibration levels was noted during flight in natural icing conditions. This may not be representative of a production aircraft since the final configuration of the ice detector system (para 3b(3)), which determines main rotor blade heater-off time, was not evaluated. Further evaluation of vibration characteristics in an aircraft using the modified ice detector system should be conducted under natural icing conditions.

i. Reliability and Maintainability. (1) Tail rotor elastomeric bearings. The four bladed semi-rigid teetering tail rotor is designed with four elastomeric bearings which connect each delta hinged hub assembly to a titanium fork and provide the teetering axis for each pair of blades. The elastomeric bearing is bonded to the fork assembly. On four separate occasions, postflight inspection revealed that an elastomeric bearing had debonded from the fork. An Equipment Performance Report (EPR) was submitted following each occurrance (EPR 82-23-4, 84-23-10, 84-23-12, 84-23-15, encl 5). A field repair procedure was used to rebond the bearing. Following the repair, approximately 20 hours was required for adhesive curing. A ground run and tail rotor balance was then performed. On one occasion, a postflight inspection immediately after shut down revealed no debonding. Approximately one hour later, after the aircraft was hangared, a debonded bearing was discovered (EPR 84-23-15, encl 5). No unusual inflight aircraft response was noted as a result of elastomeric debonding. Debonding of the tail rotor elastomeric bearing in the cold weather environment (0° to -20°C) is a shortcoming.

(2) Fault Detection Location System (FD/LS). The FD/LS incorporates continuous monitoring of selected subsystems to advise the pilot of system status during aircraft operation. The "on command" FD/LS program allows maintenance personnel and/or aircrew to perform system checks and troubleshoot failed or partially failed systems by entering a two digit code for that system on the data entry keyboard. A number of discrepancies were noted in the operation of the anti-ice/deice FD/LS (EPR 84-23-16).

(a) FD/LS Software Logic. When a fault is detected in a system containing two or more subsystems the FD/LS terminates its check at the first faulty subsystem and fails to check the remaining subsystems. It is not possible to interrogate the remaining subsystems until the first faulty subsystem is repaired. Failure of the FD/LS software logic to allow interrogation of all subsystems in a faulty system is a shortcoming.

(b) Tail Rotor Heater. A continuous tail rotor heater no-go status is displayed by the FD/LS. Investigation revealed that the tail rotor heater blankets were functioning correctly. This incorrect no-go status message is believed to be the result of an incorrect FD/LS software interface with the AEG-Telefunken Deice Controller. The tail rotor heater no-go status message when the system is functioning correctly is a shortcoming.

(c) Canopy Heater. When AC electrical power is applied and the canopy heater switch located on the pilot's anti-ice control panel is in the OFF position the FD/LS will display a canopy heater no-go message, regardless of system status. the keyboard mode swith must be rotated from "STBY" position to "FD/LS" and back to "STBY" to clear the incorrect status message. The software should be modified to give a canopy heater no-go message only when the canopy heater switch is in the ON position and the system has failed.

(d) Engine Anti-Ice System. When the ENG anti-ice switch is placed in the ON position, the following occurs:

(1) Anti-ice bleed valve opens allowing bleed air to route to the forward ENG inlet cowling.

(2) The ENG anti-ice thermal switch starts heating.

(3) The nosegear box electrothermal heaters begin heating.

The engine anti-ice caution light on the caution panel and the master caution light illuminate until the nosegear box fairing temperature reaches 205° F to 235° F and the thermal switch reaches 155°F which closes the switch contacts and energizes the anti-ice fail relays. At this time, the caution lights go out and the green engine inlet lights on the pilot's anti-ice panel illuminate. If for any reason the engine anti-ice thermal switch did not close or the bleed valve did not open, the engine anti-ice caution lights will remain illuminated. When the aircraft enters icing conditions, the engine anti-ice caution lights and the master caution light cycle on and off continuously. It is believed that this condition is the result of the nose gearbox fairing temperature dropping below 205°F. The caution light remains on for 3 to 5 minutes and then goes out when the nose gearbox temperature again goes above 205°F. This intermittent illumination of the engine anti-ice caution lights provides the pilot with a fail indication when in fact the system is functioning. Furthermore, the nose gearbox fairing heaters are designed to operate at 30 AMPS. Circuit protection is provided through 30 AMP circuit breaker which may result in the circuit breaker opening during normal system operation. This circuit breaker has not opened on any flights to date. Consideration should be given to performing an electrical load survey of this system. It should also be noted that there is no provision to interrogate the engine anti-ice subsystems through the on call or continuous FD/LS system. Consideration should be given to the addition of FD/LS software to check the engine anti-ice bleed valve, anti-ice thermal switch and nose gearbox fairing temperature. This will aid in the troubleshooting of a system since the failed component can't be identified with normal maintenance procedures. The engine anti-ice system already incorporates the necessary electrical signals to provide FD/LS interrogation of engine anti-ice subsystems. The periodic illumination of the engine anti-ice fail/master caution lights during normal system operation and the failure of the FD/LS to provide the capability to interrogate the engine anti-ice subsystems is a shortcoming.

(3) Main and Tail Rotor Deice Controller. The requirement exists to trim the deice controller to the electrical resistance characteristics of each main and tail rotor blade. This procedure is required each time a main or tail rotor blade is changed. The reason for this is that the AEG-Telefunken controller has the inherent capability of preventing a blade or blade element from overheating. The overheat temperature limit is 55° C. The controller measures blade element temperature by the change in electrical resistance as the element is heated. Hence the requirement for the controller to be "trimmed" for the variation in resistance in each blade/blade element. The specific procedure and equipment required is not contained in the -23 series TMs. The procedure and the peculiar ground support equipment required is only documented in the rotor blade deice HHI production test procedure (PTP) PTP 77-TP-7541-3, Rev A, Main and Tail Rotor Blade Deice system. Furthermore, charts are required which are contained only in the AEG-Telefunken manual. The required equipment and charts are only available at HHI in Mesa, Arizona. Since main and tail

rotor blade changes are an AVUM function, this will cause a system maintainability problem at unit level. The requirement to manually adjust the deice controller to the electrical resistance characteristics to each main blade and the tail rotor following a blade change is a shortcoming.

j. Human Factors. (1) Ice Protection Systems Advisory Light Location. The pilot's anti-ice control panel was located on the pilot's lower left console at the aft end. Two advisory lights (deice system ON and inlet anti-ice system ON) were located on this panel and are not in the pilot's normal instrument scan pattern. The ice protection system control panel and/or the advisory lights should be located in the pilot's normal instrument scan pattern. The poor location of the ice protection systems advisory lights is a shortcoming. Consideration should be given to moving the advisory lights and/or the pilot's anti-ice control panel to a more suitable location.

(2) Cabin Environment. The cabin environment at both crew stations was monitored throughout these tests and qualitative pilot and CPG comments were recorded. To resolve a previously reported problem (ref 1b), the emergency air door in the pilot's compartment was sealed for these tests and a boot was installed around the ammuntion chute on the 30mm gun turret (photo 10, encl 6). With these seals installed, the cabin environment was satisfactory at all ambient temperature conditions. On one flight at -20°C the boot was removed from the 30mm gun turret. After 30 minutes in this environment, the pilot experienced extreme discomfort in the legs and numbness of the feet. The nonstandard boot installed for this test on the turret of the 30mm gun should be incorporated as a permanent installation in the AH-64 and the emergency air door should be modified to provide a better seal. The cold aft cockpit environment during flight at approximately -20°C without the nonstandard boot installed on the 30mm gun turret and a properly sealed emergency airdoor constitutes a safety of flight hazard and is a deficiency.

4. CONCLUSIONS. a. General. The following general conclusions were reached as a result of the partially completed artificial and natural icing tests of the AH-64A helicopter. Potential problem areas requiring further evaluation are described under recommendations (para 5).

(1) One deficiency was found.

(2) Eight shortcomings were found.

(3) Seventeen Equipment Performance Reports were submitted.

b. Deficiency. The cold aft cockpit environment during flight at approximately -20°C without the nonstandard boot installed on the 30mm gun turret and a properly sealed emergency air door constitutes a safety of flight hazard (para 3j(2)).

c. Shortcoming. The following shortcomings (listed in the relative order of importance) have been identified.

(1) Debonding of the tail rotor elastomeric bearings in the cold weather environment (0° to -20°C) (para 3i(1)).

(2) Ice impact damage to the tail rotor blades (para 3d(4)).

(3) The high potential for engine FOD due to shed ice from the fairings mismatch between the nose gearbox and cross shaft fairing (para 3e(6)).

(4) The requirement to manually adjust the deice controller to the electrical resistance characteristics of each main rotor blade and the tail rotor following a blade change (para 3i(3)).

(5) Failure of the FD/LS software logic to allow interrogation of all subsystems in a faulty system (para 3i(2)(a)).

(6) The periodic illumination of the engine anti-ice fail/master caution lights during normal operation and the failure of the FD/LS to provide the capability to interrogate the engine anti-ice subsystems (para 3i(2)(d)).

(7) The poor location of the pilot's anti-ice control panel and/or advisory lights (para 3j(1)).

(8) The tail rotor heater no-go message when the system is functioning correctly (para 3i(2)(b)).

5. RECOMMENDATIONS. The following recommendations are made:

(1) The AH-64 rotor deice system operational characteristics should be further evaluated in natural icing conditions (para 3b(1)).

(2) A verification of the calibration and modified damping characteristics of the ice detection subsystem should be accomplished in the natural icing environment (para 3b(3)).

(3) The TADS/PNVS anti-ice system should be evaluated in moderate natural icing conditions (LWC > 0.5 gm/m^3) (para 3c(8)).

(4) Further natural icing tests should attempt to document the ice shedding characteristics of the main rotor head unprotected rotating components (para 3d(3)).

(5) The potential shedding problem resulting from icé accretions on the canopy frames and windshield wiper arms should be further evaluated in the natural icing environment (para 3e(4)).

(6) The poor performance of the AH-64A CPG windshield wiper in the HISS cloud (artificial icing) should be further evaluated in natural icing conditions (para 3e(5)).

(7) The susceptibility of the engines to damage from shed ice should continue to be investigated in natural icing conditions (para 3e(6)).

(8) A study should be conducted to determine the possible degradation of reception with ice accumulation on the forward facing radar warning antennas (para 3e(7)).

(9) Future natural icing tests should use an operational ALQ-144 to verify test results (para 3e(9)).

(10) Further testing should be conducted to determine the effects of firing live ammunition through a 30mm gun nozzle protector with typical ice accretions (3e(12).

(11) Consideration should be given to incorporating the nonstandard ammunition chute cover used during these tests as a permanent installation and testing should be conducted to verify proper system operation during live fire exercises with the chute cover installed (para 3e(12)).

(12) Further testing should be conducted on the 2.75 inch aerial rocket launcher system with the pod covers used during this test installed to determine the consequences of the covers departing the launcher, with and without accumulated ice (para 3e(13)).

(13) Further tests should be conducted in the natural icing environment after the ice detection subsystem damping has been incorporated to document level flight performance degradation due to ice accretion (para 3f(1)).

(14) Performance information incorporating power loss due to operation of the bleed air anti-ice system should be published in the operator's manual for pilot use in preflight planning (para 3f(2)).

(15) Further evaluation of vibration characteristics in an aircraft using the modified ice detector system should be conducted under natural icing conditions (para 3h).

(16) The FD/LS software should be modified to give a canopy heater no-go status message only when the canopy heater switch is in the ON position and the system has failed (para 3i(2)(c)).

(17) Consideration should be given to performing an electrical load survey of the nose gearbox fairing heater system (para 3i(2)(d)).

(18) Consideration should be given to the addition of FD/LS software to check the engine anti-ice bleed valve, anti-ice thermal switch, and nose gearbox fairing temperature (para 3i(2)(d)).

(19) Consideration should be given to moving the advisory lights and/or the ice protection system control panel to a more suitable location (para 3j(1)).

(20) The nonstandard boot installed on the gun turret of the 30mm gun for this test should be incorporated as a permanent installation in the AH-64 and the emergency air door modified to provide a better seal (para 3j(2)).

6. AUTHORS. This report was prepared by CW4 James M. Adkins, Project Officer, Autovon 350-4986, Commercial (805) 277-4986, and Mr. Ralph Woratschek, Project Engineer, Autovon 350-4023, Commercial (805) 277-4023.

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SYSTEMS DESCRIPTION

GENERAL

1. The deice system consists of an outside air temperature sensor, ice detector, icing rate meter, blade deice control panel, slip rings for the main and tail rotor and a deice controller. The main and tail rotor blades contained electrothermal resistive heating mats. Anti-ice systems were provided for sections of the windscreens, pitot-static tubes, air data sensor, engines, engine inlets, nose gearbox and cross shaft fairings, target acquisition and designation system, and the pilot night vision system. Deice capability was also provided for one of the Hellfire missiles. The test helicopter was serial number (S/N) 82-23356 (production vehicle 02).

ELECTRICAL POWER SUPPLY

2. Power to the AH-64A electrical distribution system is supplied by two 115/200 VAC, 3-phase, 400 Hz, 35 kVA generators. The two generators are located on the transmission accessory gearbox. Normally each generator supplies its own ac essential bus. If one generator fails, the remaining generator takes over the bus of the failed generator.

3. Input power to the blade deice system is 3-phase, 400 Hz, 115/200 vac which is fed to the blade deice controller through the blade deice system's 3-phase double-throw contactor. The blade deice system is normally connected to the No. 2 generator, however, when only the No. 1 generator is operating the contactor transfers to the No. 1 generator. The 115/200 vac power to the blade deicing controller is converted to 268 vdc by a silicon control rectifier in the controller. The 268 vdc power is used to energize the blade deicing heaters.

4. The aircraft's 28 vdc power supply comes from two 250-ampere transformer/rectifiers powered from the ac buses.

ROTOR BLADE DEICE SYSTEM

5. The AH-64A rotor blade deice system utilizes the electrothermal cyclic deicing concept. A controlled amount of ice is allowed to build up on the blade and then removed periodically by energizing electrical heating elements to raise the blade surface temperature to break the ice bond and allow shedding.

6. The main blade heater consists of five spanwise heating zones running from root to tip. These zones are energized sequentially and produce chordwise ice shedding. The main blade heater elements are bonded to the inner surface of the blade structure and extend from 19 to 94 percent spanwise.

7. The tail rotor blade heater is installed internally but consists of only one heater zone which covers 43 to 91 percent of the blade spanwise.

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8. The amount of chordwise coverage of the heaters is essentially the same for the main and tail rotor blades, i.e., to 10 percent aft of the leading edge on the upper surface and 26 and 25 percent, respectively, on the lower surface. The power density of the heaters (watts per square inch) is varied spanwise on the main rotor blade to take advantage of kinetic heating and thus reduces the electrical load requirements. The power density decreases in 14 steps from root to tip. The inboard and outboard steps are 36 and 30 inches in length, respectively. The remaining 12 steps each cover 12 inches of span. The tail rotor is uniformly heated at one power density over its span.

9. The blade deice system is shown schematically in figure 1. Three-phase, 115 vac, 400 Hz electrical power from the generator is rectified in the deicing system controller to 268 vdc for blade heating. The dc power from the controller is routed to the main and tail rotor through slip rings. The main rotor installation incorporates a distributor that sequentially delivers the electrical power to the five heating elements of each blade. The system deices all four of the tail rotor blades simultaneously and steps through the zones of pairs of opposing main rotor blades.

10. Figure 2 shows the blade and zone heating sequence that is used. After the blades have accreted the prescribed amount of ice, a pulse train is initiated to deice the tail and main rotor blades in a sequence that deices the tail rotor twice as often as the main rotor. The controller does not allow simultaneous heating of the tail rotor with the main rotor since the combined electrical load would exceed electrical system capacity.

11. The time interval between the applications of heat to any given zone is called heater-off time. For normal operation, blade heater-off time is controlled automatically by the blade deicing controller and is scheduled as a function of ice accretion rate signal multiplied by time. The controller integrates the icing rate signal (corresponding to liquid water content (LWC)) from the ice detection system with respect to time and initiates a blade deicing signal each time the integral reaches a predetermined value. This predetermined value is selected to approximate to a 0.25-inch thickness of ice accreted on the main rotor blade at the mid-span location. The tail rotor blade airfoil, which is approximately one-half the scale of the main rotor, is deiced at one-half the main rotor LWC-time integral (twice as often as the main rotor) to maintain the same proportion of ice accretion as for the main rotor blades.

12. The main rotor off-time scheduled by the controller corresponds to the curve shown in figure 3. This curve is a constant LWC multiplied by time value of $1.76 g/m^3$ - minutes. The LWC signal from the ice detector processor is a dc voltage with $0.2\ g/m^3$ equivalent to 1 volt dc. The main rotor volt-time integral, therefore, is 8.8 volt-minutes. The tail rotor volt-time integral is approximately 4.4 volt-minutes. Tail rotor off-time is half of the main rotor off-time.

13. Also shown in figure 3 are the manually selectable off-times that are available in the event the ice rate detection system is not used. The manual tail rotor off-times are one-half of the main rotor off-times shown.

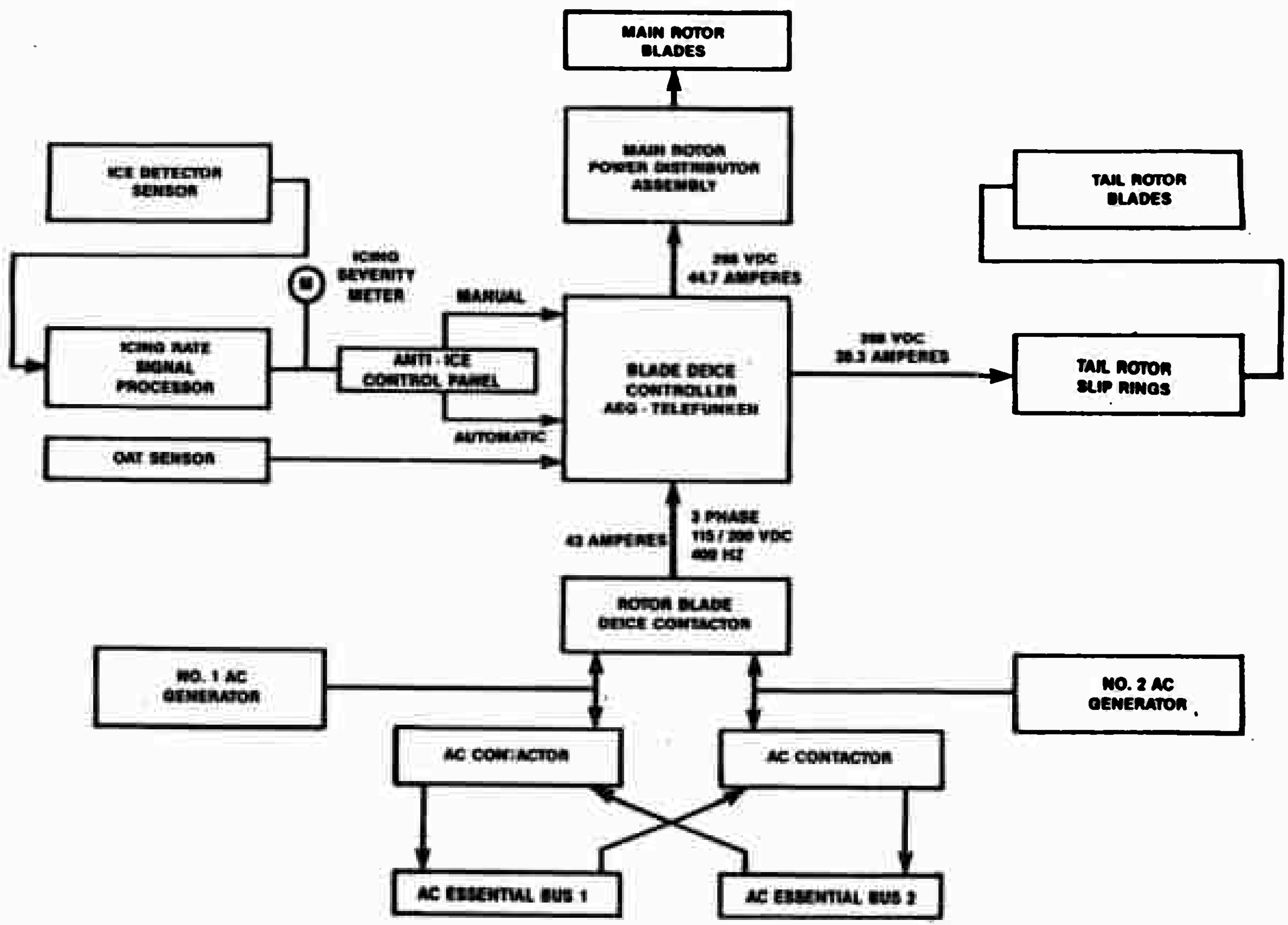


Figure 1. Blade Deice System Block Diagram

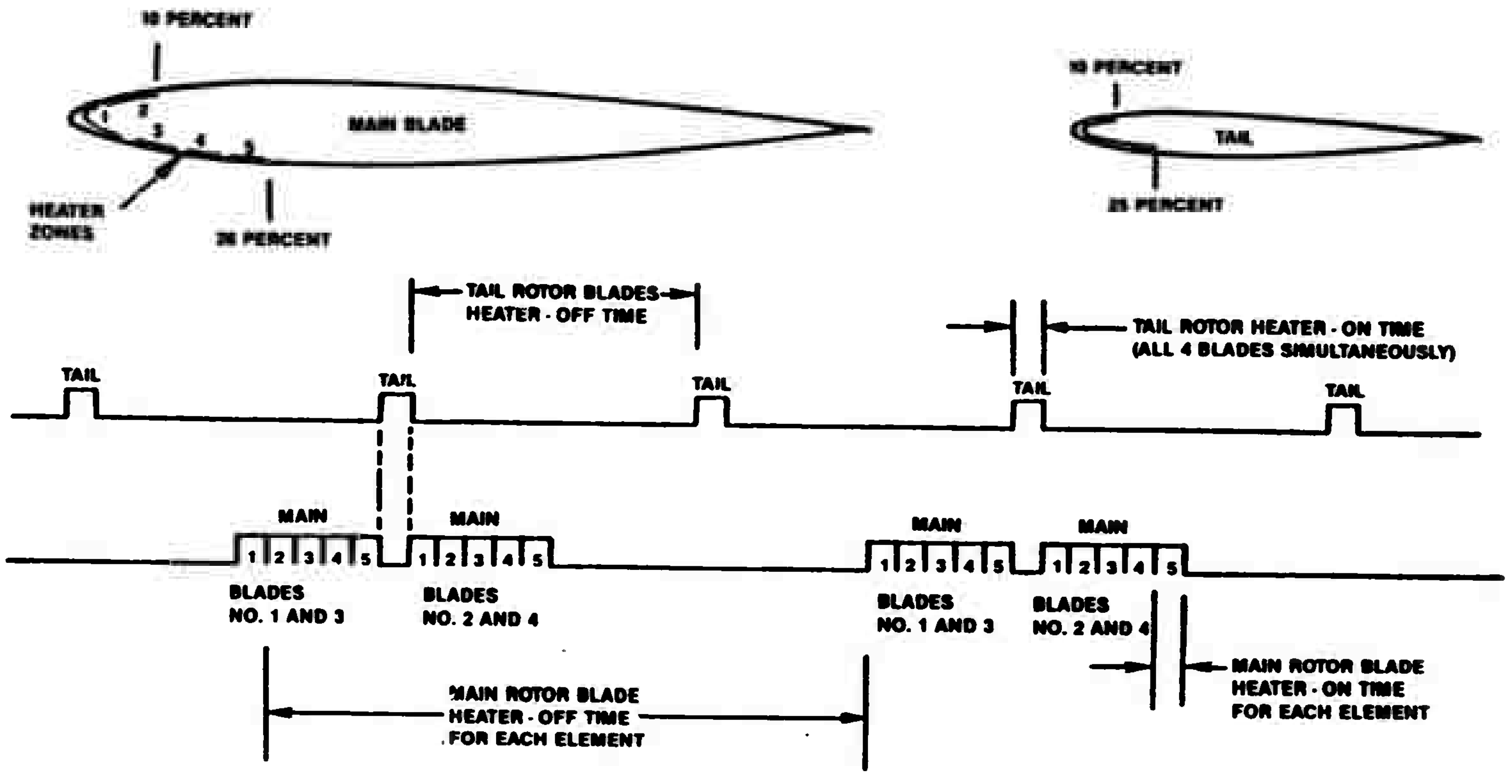


Figure 2. Blade Heater Elements and Distributor Pulse Sequence

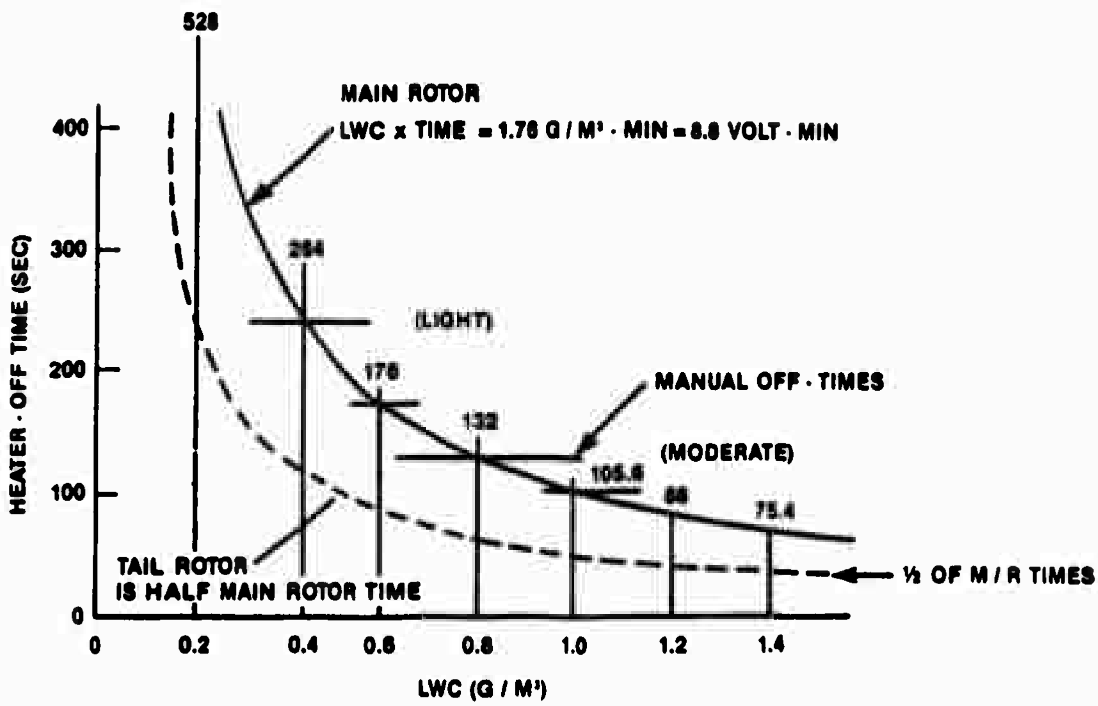


Figure 3. Blade Heater-Off Time Schedule

14. For the artificial icing program, the aircraft was equipped with additional special test controls for use in scheduling heater-off time.

15. The duration of time that heat is applied to a given zone is called heater-on time. Heater zone on-time is also controlled automatically by the blade deicing controller. The heater-on time varies as a function of outside air temperature (OAT). The lower the air temperature, the longer the heaters are energized to raise blade surface temperature for deicing. The design heater-on time versus OAT schedule is shown in figure 4, enclosure 4. A ± 25 percent gain adjustment capability was incorporated into the test aircraft controller to allow optimization of the heater-on time schedule.

16. The design heater-on time is the same for each of the main blade heater elements and for the tail rotor. The gain of the heater-on time can be adjusted on the ground independently for the main rotor and the tail rotor but not for each main rotor heater element.

Ice Detection System

17. The ice detection system consists of a Rosemount Inc., Model 871FG1, ice detector and a Model 524Y4 signal processor. The ice detector is of the ice accretion type that employs a vibrating sensing probe which changes frequency as ice is accreted. The probe is located inside the left-hand engine air inlet duct on the outboard side. Engine airflow provides a relatively high and fairly constant velocity over the probe both in hover and forward flight. The system includes a signal processor to output an analog voltage signal proportional to the LWC.

18. The ice detector is powered by dc current and is on whenever ship's power is on. The icing rate signal is generated whenever ice buildup on the probe is between 0.015 inch and 0.060 inch thick. At the 0.060-inch thickness point, the probe is electrothermally deiced and the cycle repeated. The ENG ICE caution light comes on at the 0.015 inch point and remains on for a minimum of 90 ± 10 seconds. If no further icing were encountered, the light would go out at the end of that time. If icing is continually encountered, the light will remain on.

19. During the time the sensing probe is deiced and recovering from its 6 second heater application, a hold circuit is employed that maintains the LWC signal at its last value. The hold circuit operates until the probe has thermally recovered and releases when accreted ice thickness again reaches 0.015-inch or 60 ± 10 seconds of time has elapsed. Thus, if the aircraft emerges from an icing environment prior to reaching the 0.015-inch thickness point, the LWC signal would go to zero after this time interval and indicate a zero LWC condition. The probe recovery time (time the detector is off line) varies with LWC and OAT. Typical maximum recovery times for a probe velocity of 112 knots are 75 seconds at LWC = 0.12 g/m^3, 15 seconds at 0.5 g/m^3, and 10 seconds at 1.0 g/m^3.

20. The output of the ice detector system will vary with air temperature and local flow velocity over the probe at a fixed LWC. In addition, the accuracy will vary with LWC at a fixed temperature and flow velocity. The ice detector system is calibrated in an icing tunnel to read LWC ± 10 percent for a local flow velocity of 112 knots. At higher and lower local flow velocities, the system will indicate slightly higher or lower than actual LWC values. Since the probe is mounted in the engine inlet, flow velocity variations result from variations in engine power. Therefore, the accuracy of the ice detector system ove. the most likely range of air temperature, velocities, and LWC values to be encountered in actual use has not been determined.

21. The deice system included an icing rate meter which displayed LWC as measured by the ice detector and a signal processor system. The LWC meter display is mounted on the pilot's instrument panel.

Outside Air Temperature System

22. The OAT sensor is a flush-mounted resistance-type temperature sensor installed on the right-hand side of the fuselage between the forward portion of the engine nacelle and the wing. This location was selected to shield the 1.25-inch diameter sensor from exposure to direct sunlight heating and also to be in a surface area that will not be subjected to ice impingement. The OAT sensor's resistance output is fed to the blade deicing controller for use in scheduling blade heater-on time. A cockpit indication is not provided.

23. A second identical OAT sensor was installed immediately adjacent to the deicing controller sensor as test instrumentation to provide a signal for recording on the data acquisition system. This permitted a check on the OAT signal going to the blade deicing controller.

Blade Deice System Operation

24. Rotor blade deicing operation is initiated as follows. When entering a potential icing environment, all ice protection system switches are turned on including the blade deice switch. When actual icing is encountered, the pilot will be alerted by illumination of the ENG ICE segment on the caution and warning panel. Blade deicing would commence after the scheduled ice accretion period has elapsed and continue automatically as long as icing is present. The off-time is automatically regulated based on integration of the icing rate signal from the ice detection system.

25. An alternate mode of operation would be to turn the ice protection system on when the ENG ICE segment is illuminated which indicates icing is present. For the proper deicing scheduling it is important to turn the blade deice system on immediately so integration of the icing rate can commence as soon as ice starts to accrete.

26. Heater-off time shortens with increasing icing intensity (LWC) and heater-on time lengthens with decreasing OAT. At some high level of LWC and low OAT, the system will saturate. When saturated, the time interval between the single

tail rotor pulse and the main rotor/tail rotor/main rotor train pulses will be zero. Since the on time takes precedence over off time, the system pulses continuously and the ice on the blades is not allowed to build up to the desired thickness before deicing. In theory, this could have a detrimental effect on ice shedding, however, saturation does not occur until -19.3°C at an LWC = 1.0 g/m^3.

27. In the event the aircraft is operating in an icing environment and the pilot suspects the icing rate system is malfunctioning and not detecting the true icing intensity, the pilot may select a manual mode of operation. In this mode, the blade deicing heater-on time is still regulated according to OAT but the blade heater-off time is one of three fixed off-times that the pilot can select based on his assessment of icing intensity. These selections are shown on the anti-ice control panel of figure 4 as TRACE, LT, and MOD representing trace, light, or moderate icing intensities. The pilot would have to base his assessment of the icing intensity from indications other than the ice detector. These indications might be observation of ice accretion rate on visible portions of the aircraft, premature and asymmetric blade ice shedding resulting in increased vibration levels, or lack of change in vibration levels when blade deicing heater cycles occur.

Rotor Blade Deice System Checkout

28. A deice system TEST switch and an icing rate indicator are provided to enable ground checkout of the system. The deice system test consists of a rapid heater-on sweep or duty cycle (0.5 second on-time per element) and a heater-on light to indicate heater operation. Any failures will activate the BLADE ANTI-ICE FAIL caution segment on the caution and warning panel. The icing rate indicator test is activated by depressing the test button adjacent to the LWC meter. This test conducts a check on the icing rate meter electronics only.

Blade Deice System Protection Circuits

29. The blade deicing controller contains monitoring circuits to sense a system malfunction. The fault monitoring functions and protection features are:

a. main or tail rotor heater undercurrent or overcurrent

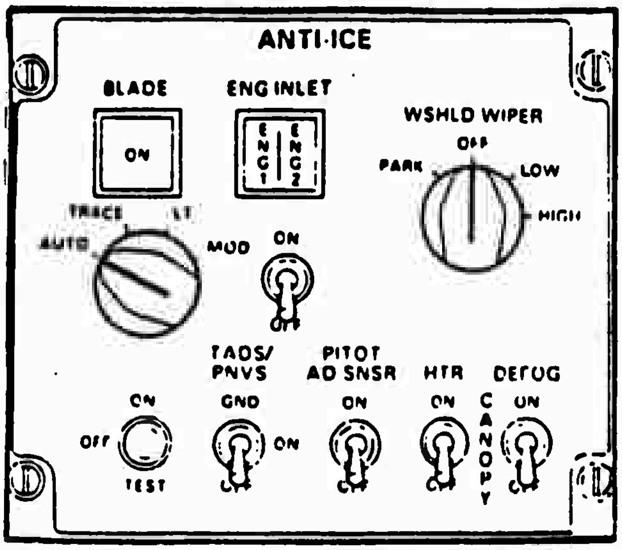
b. main or tail rotor heater ground current (6.0A maximum)

c. distributor mechanism does not advance

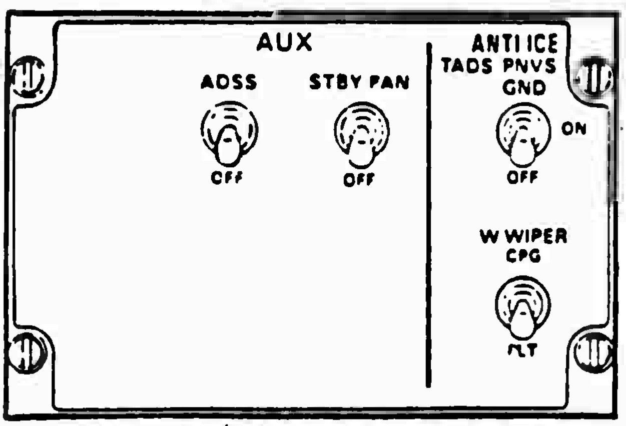
d. heater-on current on too long (+25 percent maximum)

e. controller overtemperature (>100°C)

f. input power must be 115 vac ±10 percent



Pilot Anti-Ice Panel



CPG Anti-Ice Panel

Figure 4. Pilot and CPG Anti-Ice Control Panels

g. system ON advisory light (caution while performing maintenance)

h. system disable above 4°C OAT to preclude blade over temperatures.

30. A built-in test equipment (BITE) test is provided to verify that the monitoring circuitry in the controller is functioning properly. Fault isolation may be performed with the controller installed by use of an extension harness to provide access to the BITE control and indicator. The test aircraft will also display to the pilot and copilot/gunner (CPG) the fault detection/ location system outputs from the deice controller on the normally installed high action displays.

Special Test Operating Provisions

31. Because of the limited vertical thickness and larger droplet diameters of the helicopter icing spray system (HISS) cloud, the test program had to provide for blade deicing system operation with the ice detector not immersed in the cloud. Automatic operation of the system in the HISS cloud is not feasible. To tailor the heater-off times for proper blade deicing, special test adjustment of LWC input parameter is necessary. For these purposes, provisions were made in the test aircraft to permit:

a. Manual deice capability for main and tail rotor blades independent of LWC. This permitted single cycle evaluation.

b. Manual insertion of an LWC signal equivalent to the HISS setting. This permitted simulated automatic operation while operating in the HISS spray cloud even though the ice detector was not immersed.

c. Manual adjustment of the volt-time integral which controls heater-off time. This permitted fully automatic operation in natural icing after the volt-time integral was optimized.

32. A special blade deicing test control panel as shown in figure 5 was installed in the CPG flight station. The false OAT adjustment was not operative.

ENGINE AIR INLETS

33. The engine air inlets use engine bleed air to heat the bellmouth-shaped nacelle inlet ring. The heated surface area extends around the entire leading edge of the inlet to where it mates with the engine. Thus, the running wet surface continues into the engine during operation in icing conditions. The amount of engine bleed air flow is designed to anti-ice the inlet even under low power descent conditions.

34. The bleed air system is activated by an electrically operated, two-position valve which fails open in the event of complete electrical failure. The system is operated by a single, two-position on/off toggle switch on the pilot's anti-ice panel (fig. 4). In the ON position, advisory lights above the

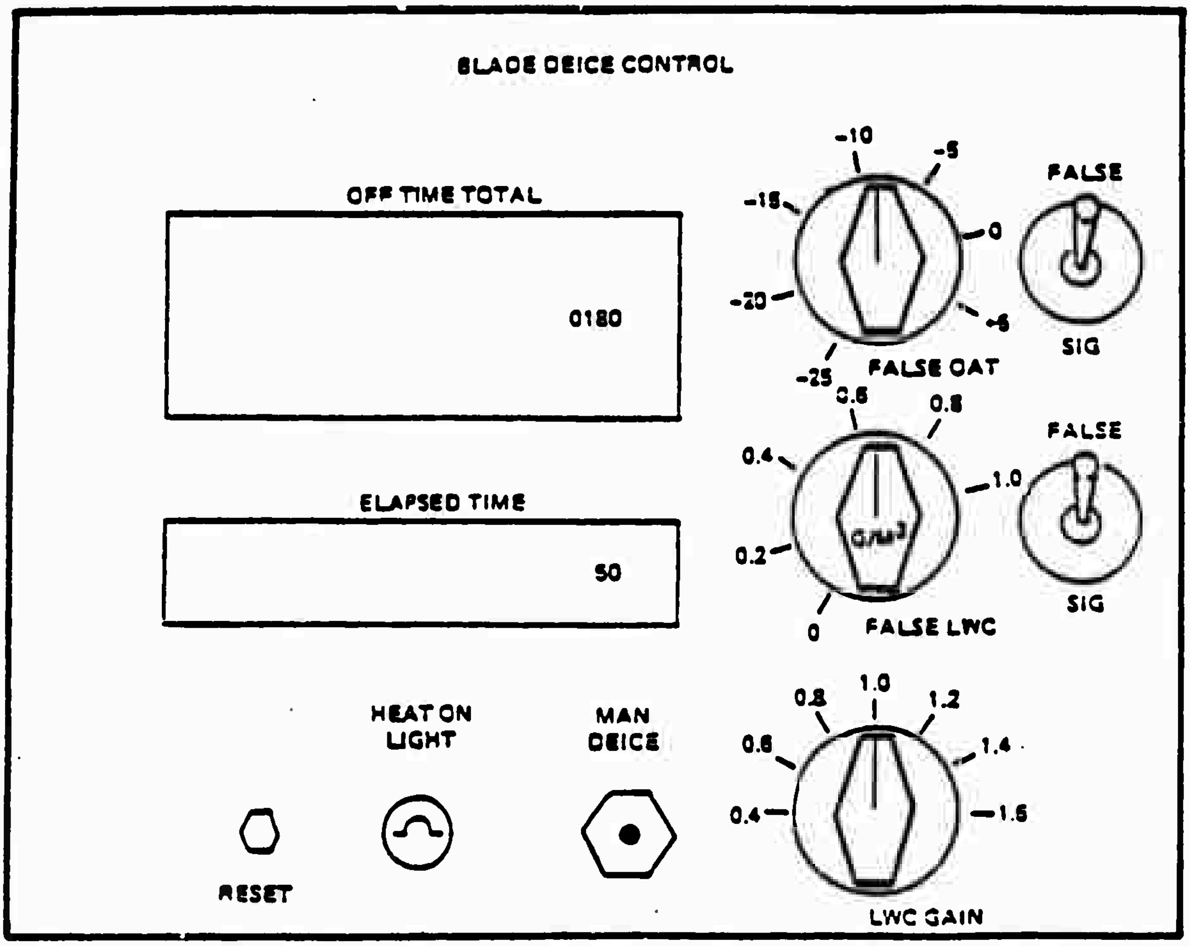


Figure 5. Blade Deice Test Control Panel

switch for ENG 1 and ENG 2 will illuminate. This switch also operates the electrothermal heater blankets on the nose gearbox and cross shaft fairings.

NOSE GEARBOX AND CROSS SHAFT FAIRING

35. The nose gearboxes and cross shafts from the engine to the transmission are covered with a fairing that incorporates electro-thermal heating blankets for anti-icing. The heater blankets are controlled automatically by a temperature controller which maintains the surface at a temperature sufficient to evaporate the moisture that impinges on the fairing. This evaporation precludes runback and subsequent refreezing downstream of the heater area.

36. These electric heaters are turned on by the same single two-position switch on the pilot's anti-ice panel that activates the engine air inlet bleed air system. Overheat sensors are provided that activate the ENG ANTI-ICE segment on the pilot's and the CPG's caution and warning panels and shut the electric heaters off automatically. The overheat sensor is a controlling-type sensor which allows the system to continue operation at the higher temperature without further overheat protection.

WINDSHIELDS

37. The two windshields, located in front of and directly above the CPG, have transparent electrothermal anti-ice elements (115 vac) built in the laminated glass panels designed to prevent ice accumulation. The pilot's windshield, which is the panel above the CPG, is heated in two zones with phase A and B of the 3-phase current. The CPG's panel is powered by phase C. A single temperature sensor is located on the left-hand side of the forward zone of the pilot's windshield. This sensor signals the controller to maintain the windshield's outside surface temperature between 65° and 85°F. An overheat caution segment labeled CANOPY ANTI ICE FAIL is provided to indicate failure of the automatic control and the windshield heater is automatically shut off. The windshield is heated at a power density of 4.57 watts/in^2.

Defogging/Demisting

38. Cockpit defogging/demisting is accomplished from two sources. The two windshield sections are electro-thermally heated as described previously. This serves to defog as well as to anti-ice. Pressurized air from the shaft driven compressor (SDC) mixes with ambient crew station air in diffuser outlets at the CPG and pilot stations to defog the canopy side panels. In case of SDC failure, engine bleed air from the left engine is used. Canopy side panel defogging is controlled by an on/off switch on the pilot's anti-ice control panel.

Rain Removal

39. Two windshield wipers are mounted on the canopy frame to service the two windshields. Both wipers are electrically driven and are normally controlled

by a four-position rotary knob on the pilot's anti-ice panel (fig. 4). They are powered from the 28 vdc emergency bus. Two speeds, HIGH or LOW, may be selected by the pilot. The CPG has limited control over the windshield wipers. Normally the WSHLD WIPER switch on the CPG's anti-ice control panel (fig. 4) is left in the PILOT position. By selecting CPG, the CPG's wiper blade moves at a LOW speed.

PITOT TUBES AND AIR DATA SENSOR

40. The pitot tubes installed on the outboard leading edge of both wings and the air data sensor (ADS) on top of the rotor mast are electrothermally anti-iced by internal heating elements. Both are anti-iced with 28 vdc power. An on/off toggle switch labeled PITOT AD SNSR is located on the pilot's anti-ice control panel. On the ADS, the rotating hub and arms are heated. The support pylon (mast) of the ADS is not ice protected.

TARGET ACQUISITION DESIGNATION SYSTEM (TADS)/PILOT NIGHT VISION SYSTEM (PNVS) ANTI-ICE PROTECTION

41. Anti-icing for the TADS/PNVS consists of electrothermal heating of the three viewing windows and portions of their frames. In addition, continuous heaters are located inside the TADS shroud above the night side and the dayside windows to keep these limited shroud regions clear and ensure that adequate TADS elevation motion can be achieved. Unheated front ice shields protect the TADS azimuth gear and inboard sections (immediately adjacent to the centerpost) along the top two-thirds of the TADS shrouds.

42. The power density of the windows is between 3.2 and 4.0 watts/in^2. The PNVS and TADS window heaters are ac powered, whereas the boresight windows are dc powered.

43. TADS/PNVS anti-ice is controlled by a switch located on the pilot's anti-ice control panel or by a switch on the CPG's antiice control panel.

GENERAL

1. In addition to standard aircraft instruments, calibrated instrumentation was installed and maintained by the Hughes Helicopter Incorporated. Data from the cockpit instrumentation was recorded on flight cards, and the specially installed instrumentation system recorded pulse code modulated (PCM) and multiplex bus (MUX) data on magnetic tape.

Pilot Panel

2. The pilot's panel instrumentation and test instrumentation are listed below:

- Airspeed (ship's system)
- Altitude (ship's system)
- Altitude radar (ship's system)
- Rate of climb (ship's system)
- Free air temperature (ship's system)
- Rotor speed (ship's system and sensitive)
- Engine torque (both engines ship's system)
- Engine turbine gas temperature (both engines ship's system)
- Engine gas generator speed (both engines ship's system)
- Engine power turbine speed (both engines ship's system)
- Stabilator position (ship's system)
- Icing rate (ship's system left engine inlet)
- Full instrument landing system instrumentation
    - Localizer
    - Glide slope
    - Marker beacon
    - DME
- Control panel for operation of data system to include event marker
- Time code readout.

Copilot/Gunner Panel

3. The copilot/gunner panel instrumentation and test instrumentation are listed below.

- Airspeed (ship's system)
- Altitude (ship's system)
- Engine torque (both engines ship's system)
- Rotor speed (ship's system)
- Engine power turbine speed (both engines ship's system)
- Total air temperature (Rosemount system with deice capability)

Icing rate indicator (aspirated independent system mounted on aft cockpit overhead)
Manual control panel for deice system (with LWC active only)
Control panel for operation of data system to include event marker
Time code readout
Camera controls.

PCM Data Parameters

4. The following data were recorded on magnetic tape in PCM format.

Airspeed
Altitude
Observed air temperature (Rosemount sensor)
Observed air temperature (Rotor deice system)
Event markers (pilot and copilot)
Engine torque (2)
Engine measured gas temperature (2)
Engine gas generator speed (2)
Main rotor speed
Control positions
    Longitudinal
    Lateral
    Directional
    Collective
Vibrations
    3 axis at pilot seat
    3 axis at copilot seat
    2 axis at tail rotor 90 degree gear box
    Tail rotor mast structural attachment vertical
Stabilator incidence angle
Icing severity (Deice system sensor)
Icing severity (Aspirated independent system)
Voltage
    Ship's system Bus 2
    Battery
Rotor blade heater ON/OFF status
Current
    Generators Phase A (2)
    Windshield Phase A
    Blade Deice Phase A
    Nose Gearbox Heat Phase A
    Battery
Engine inlet surface temperature (left)
Time code
Pilot's station temperature
    Head level
    Waist level
    Foot level
Copilot's station temperature

Head level
Waist level
Foot level
Cockpit supply duct air temperature
FAB air inlet temperature (left and right)
FAB average exhaust temperature (left and right)
Cockpit air temperature at exhaust to TADS/PNVS turret
Pressurized air system manifold pressure

MUX Parameters

5. The following MUX parameters were recorded on magnetic tape:

Heading and Attitude Reference System (HARS) attitudes
Pitch
Roll
HARS magnetic heading
HARS rates
Pitch
Roll
Yaw
Sideslip angle (pacer system)
True airspeed longitudinal (pacer system)
True airspeed lateral (pacer system)
Observed air temperature (pacer system)

Onboard Video

6. The data recording system was configured to record the video imagery of the copilot/gunner's TADS optical relay tube display. Time code was inserted on the video frame to allow coordination with other data.

CAMERA SYSTEMS

7. A video camera and a 16mm motion picture camera were located onboard the chase and Helicopter Icing Spray System aircraft and used to document ice accretion on the test aircraft both in the spray cloud and after exit from icing encounters. Single lens reflex 35mm cameras were used for still photo (color prints and slides) documentation both in the air and on the ground following icing flights.

8. Initially three high speed 16mm cameras were mounted on the test aircraft to document these icing tests. One camera was mounted on the right forward avionics bay (FAB) to photograph the main rotor advancing blade leading edge and lower surface. Another camera located on the top of the left wing tip photographed the tail rotor and horizontal stabilator. The third camera photographed the left engine inlet. Mechanical failure of the tail rotor camera shortly after the beginning of testing only permitted the monitoring of the main rotor blades and left engine inlet during the artificial and left engine

inlet during the artificial icing tests. A second camera failure restricted the natural icing monitoring to the left engine inlet.

HELICOPTER ICING SPRAY SYSTEM (HISS)

9. The HISS is installed in a modified CH-47C helicopter and consists of an internally mounted 1800 gallon water tank and a hydraulically rotated external spray boom assembly suspended 19 feet beneath the aircraft from a cross-tube through the cargo compartment. The spray boom consists of two 27-foot center sections, vertically separated by 5 feet and two 17.6-foot outriggers. The spray cloud is generated by pumping water at known flow rates from the tank to 97 Sonicore model 125-HB nozzles installed on the boom center sections and using bleed air from the aircraft engines and an auxiliary power unit to atomize the water. A radar altimeter with aft-facing antenna, wired to red and yellow station-keeping lights on the underside of the CH-47, is used to position the test aircraft at a known standoff distance. At the 150 foot standoff distance used for icing tests, the size of the visible spray cloud is approximately 8 feet high by 36 feet wide.

CLOUD SAMPLING EQUIPMENT

10. For cloud measurements in both the natural and artificial environments, USAAEFA employs a JU-21A fixed-wing aircraft, US Army S/N 66-18008, equipped with a cloud measurement package. This package consists of the following equipment: a Particle Measuring system (PMS), forward scattering spectrometer probe (model FSSP-100), a PMS optical array cloud droplet spectrometer probe (model OAP-200X), Rosemount outside air temperature sensor and display, Cambridge model 137 chilled mirror dew point hygrometer and display, Cloud Technology ice detector unit, and a Small Intelligent Icing Data System (SIIDS).

TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. All anti-ice systems (i.e., pitot heat, windshield anti-ice, and engine air induction system anti-ice) were activated while enroute to the test area. For artificial icing, the test aircraft then entered the artificial spray cloud from a position below and approximately 150 feet behind the spray aircraft. Test and spray aircraft separation distance was maintained during the icing flight by observing yellow (greater than 160 feet) and red (closer than 140 feet) lights mounted on the bottom of the spray aircraft. The visual indications were supplemented as required by information relayed from the spray and chase aircraft. Airspeed and outside air temperature (OAT) were established with the calibrated instrumentation system of the spray aircraft. All artificial flights were flown with a predetermined liquid water content (LWC) and OAT. Flight continued in the cloud condition until the spray aircraft water limit was reached. For natural icing the JU-21A would locate and document the icing condition before the test aircraft entered the icing environment. The JU-21A would then loiter in the area to facilitate a post-immersion rendezvous with the test aircraft for photographic documentation. The LWC, OAT, and relative humidity were documented by the JU-21A chase/scout aircraft configured with the particle measuring system instrumentation. The two Rosemount icing rate meters in the test aircraft were also used to monitor LWC in natural clouds.

ICE ACCRETION AND SHEDDING

2. Ice accretion on the test aircraft was documented using hand-held video and high-speed motion picture cameras photographing from both the chase aircraft and spray aircraft. Post-flight photographs were made to document the ice remaining on the individual components of the airframe and rotors.

3. Ice shedding characteristics were qualitatively assessed by crew members in the test, spray, and chase aircraft.

ENGINE PERFORMANCE

4. Engine and engine inlet anti-ice system performance were monitored during the icing tests. Data were obtained with all anti-ice systems ON and then compared to baseline data with the anti-ice systems OFF.

5. The engine power required to operate the anti-ice/deice sytems was determined by measuring engine performance at various test conditions. Shaft horsepower was calculated using equation 1.

Encl 3

$$SHP = \frac{N \times Q \times K}{5252.113} \qquad (1)$$

where:

SHP = Calculated shaft horsepower (shp)
N = Main rotor rotational speed (rev/min)
Q = Engine output shaft torque (ft-lb)
K = Gearing constant between engine and main rotor (72.4243)

Data on SHP, turbine gas temperature (TGT), and gas generator speed (N_g) were referred as follows:

a. Referred SHP (RSHP):

$$RSHP = \frac{SHP}{\delta\sqrt{\theta}} \qquad (SHP) \qquad (2)$$

b. Referred gas temperature (RTGT):

$$RTGT = \frac{TGT + 273.15}{\theta^{0.96}} - 273.15 \qquad (°C) \qquad (3)$$

c. Referred gas generator speed (RN_g):

$$RN_g = \frac{N_g}{\theta^{0.5}} \qquad (\%) \qquad (4)$$

WEIGHT AND BALANCE

6. Prior to testing, the aircraft gross weight, longitudinal and lateral centers of gravity (cg) were determined by using calibrated scales. The aircraft was weighed with full instrumentation on board, without fuel, and in the clean configuration. The aircraft weight was 12,318 pounds with a longitudinal cg location at fuselage station 211.1 and a lateral cg location at buttline -0.8 (left).

DEFINITIONS

7. Icing characteristics were described using the following definitions of icing severity. These definitions may be found in FM 1-30.

a. Trace icing: Ice becomes possible. Rate of accumulation slightly greater than rate of sublimation. It is not hazardous even though deicing equipment is not used, unless encountered for an extended period of time (over 1 hour).

b. Light icing: The rate of accumulation may create a problem if flight is prolonged in this environment (over 1 hour). Occasional use of deicing/anti-icing equipment removes/prevents accumulation. It does not present a problem if the deicing/anti-icing equipment is used.

c. Moderate icing: The rate of accumulation is such that even short encounters become potentially hazardous and use of deicing/anti-icing equipment or diversion is necessary.

8. Results were categorized as deficiencies or shortcomings in accordance with the following definitions.

Deficiency: A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued or indicates improper design or other cause of an item or part, which seriously impairs the equipments operational capability.

Shortcoming: An imperfection or malfunction occurring during the life cycle of equipment, which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the usability of the material or end product.

Table 1. Specific Test Conditions[1]

Flight Number	Configuration[2]	Icing Environment	Immersed Component	Average Gross Weight (lbs)	Average Longitudinal Center of Gravity (fs)	Average Density Altitude (ft)	Average OAT (deg C)	Average Relative Humidity (%)	Average[3] LWC (gm/m^3)	Average True Airspeed (kts)	Total Time in Cloud (min)
1	Hellfire & Rockets	Artificial	Rotor	15,920	204.8	5390	-5.5	60	0.98	120/112	30
2	Hellfire & Rockets	Artificial	Rotor	16,160	205.9	5910	-9.0	53	1.19	106/100	30
3	Hellfire & Rockets	Artificial	Rotor	16,130	205.7	2970	-5.0	30	1.01	95	22
4	Hellfire & Rockets	Artificial	Fuselage	16,100	205.2	5480	-9.5	26	1.21	96	57 1/2
5	Hellfire & Rockets	Artificial	Rotor	15,890	204.8	9280	-19.5	84	0.51	90	30
6	Hellfire & Rockets	Artificial	Fuselage	15,990	205.2	10,010	-19.5	69	0.59	96	60
7	Hellfire & Rockets	Artificial	Rotor	16,260	204.9	7600	-18.5	95	0.56	95	30
8	Hellfire & Rockets	Artificial	Rotor	16,190	205.2	3240	-5.0	32	1.20	96	31 1/2
9	Hellfire & Rockets	Artificial	Rotor	16,120	205.5	7910	-16.0	43	1.03	95	21
10	Hellfire & Rockets	Artificial	Rotor	16,220	205.1	6800	-15.5	57	1.02	95	30
11	Asymmetric Stores	Artificial	Fuselage	15,900	205.3	9850	-6.0	43	1.05	94	60
12	Asymmetric Stores	Artificial	Rotor	15,830	205.3	10,240	-10.5	47	1.05	94	33
13	8-Hellfire	Artificial	Fuselage	15,850	205.4	10,610	-14.5	56	1.02	96	60
14	8-Hellfire	Natural	---	15,820	205.5	1060	-7.0	--	0.32	100	67
15	8-Hellfire	Natural	---	15,840	205.4	1050	-7.5	--	0.32	108	76 1/2
16	8-Hellfire	Natural	---	15,920	205.4	2570	-7.5	--	0.22	102	81
17	8-Hellfire	Natural	---	15,870	205.4	1780	-7.5	--	0.26	45/103	80
18	8-Hellfire	Natural	---	15,960	205.3	7940	-12.0	--	0.15 to 0.45	119	—
19	8-Hellfire	Natural	---	15,800	204.9	7220–9470	-17.0 to -18.0	--	0.10 to 0.40	100	60

NOTES:

[1]Rotor speed = 100%

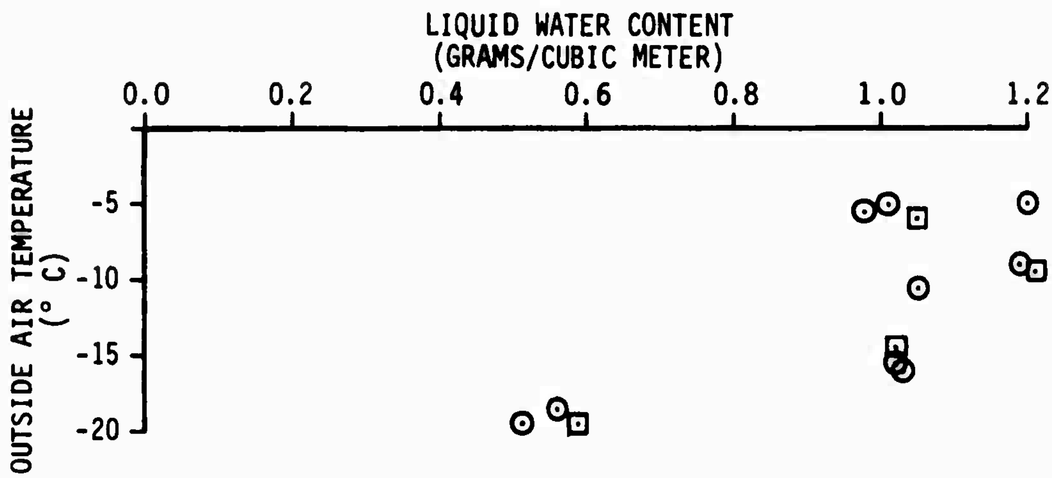
[2]Configurations: Hellfire & Rockets = 8-Hellfire Inboard, Rocket Pods Outboard
Asymmetric Stores = Inboard Hellfires & Outboard Rocket Pod on Left side, Outboard Hellfire & Inboard Rocket Pod on Right Side
8-Hellfire = 8-Hellfire Inboard

[3]Artificial LWC calculated from HISS water flow rate

FIGURE 1

ARTIFICIAL ICING TEST CONDITIONS

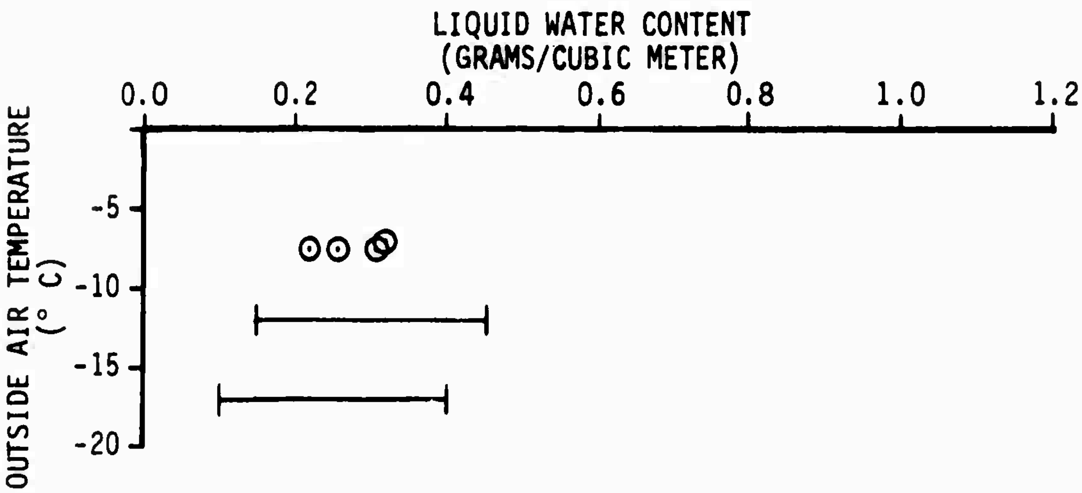
AH-64 USA S/N 82-23356



NOTES: 1. ○ - ROTOR SYSTEM
□ - FUSELAGE
2. ROTOR SPEED = 100 PERCENT
3. AVG GROSS WEIGHT = 16,040 POUNDS
4. AVG LONG. C.G. = FS 205.2

FIGURE 2
NATURAL ICING TEST CONDITIONS

AH-64A USA S/N 82-23356



NOTES:1. ROTOR SPEED = 100 PERCENT
2. AVG GROSS WEIGHT = 15,870 POUNDS
3. AIRSPEED = 100 - 119 KTAS
4. AVG LONG. C.G. = FS 205.3
5. ⊢⊣ DENOTES RANGE OF LWC VARIATION DURING FLIGHT IN CUMULIFORM CLOUDS
6. ⊙ — DENOTE STRATIFORM CLOUDS

FIGURE 3
BLADE DEICE ON TIME
AH-64A USA S/N 82-23356

NOTES: 1. ⊡ — — ORIGINAL SCHEDULE
⊙ — — MODIFIED SCHEDULE
2. OPEN SYMBOLS DENOTE ARTIFICIAL ICING CONDITIONS
3. SHADED SYMBOLS DENOTE NATURAL ICING CONDITIONS

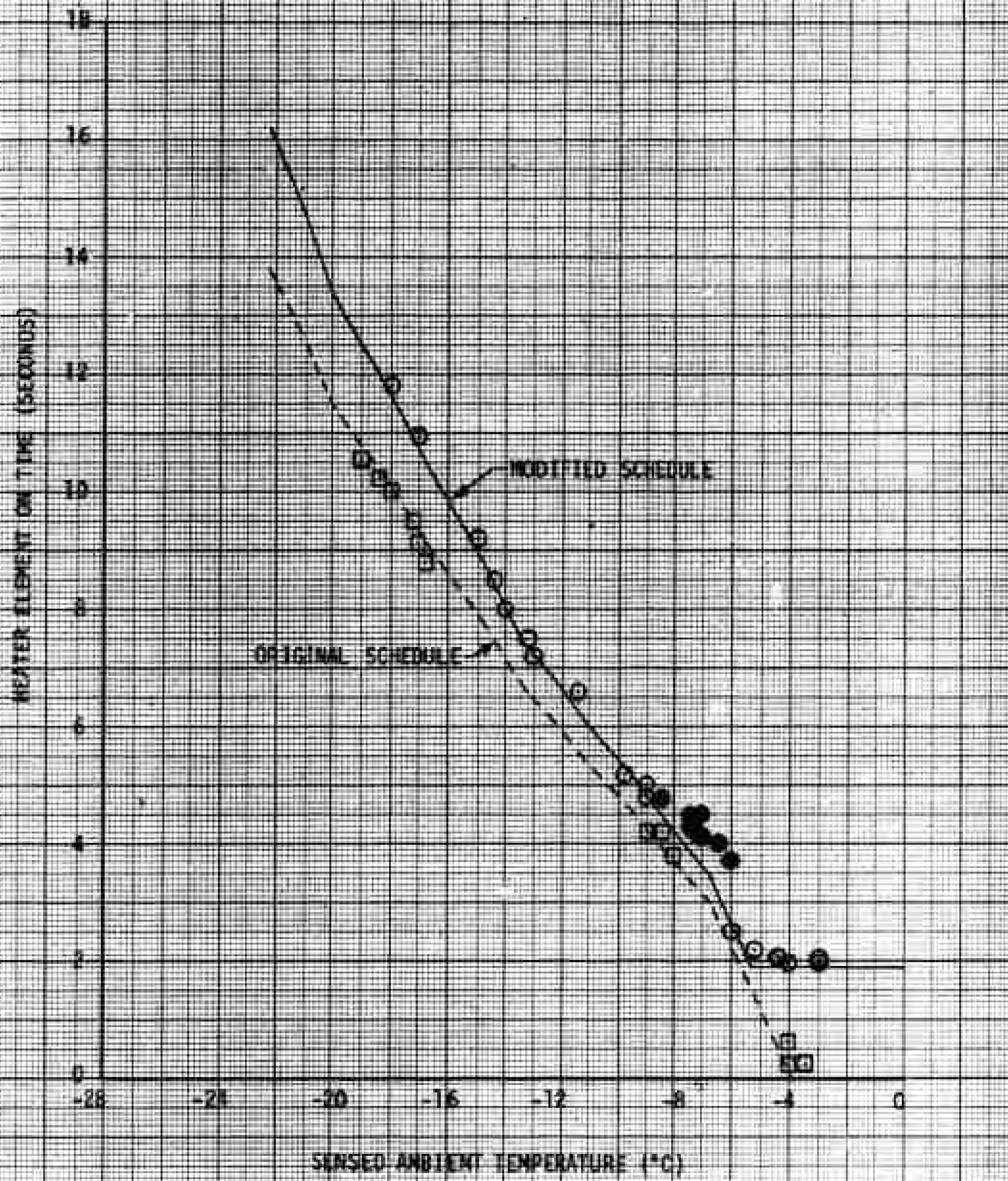


FIGURE 4
BLADE DEICE OFF TIME
AH-64A USA S/N 82-23356

NOTES:
1. SOLID LINE OBTAINED FROM CONTRACTOR FOR IDEAL SCHEDULE.
2. DATA POINTS OBTAINED IN NATURAL ICING CONDITIONS, AUTOMATIC MODE OF OPERATION
3. SHADED SYMBOLS DENOTE LWC SIGNAL MULTIPLIER SET AT 0.4, OPEN SYMBOLS DENOTE MULTIPLIER SET AT 1.0

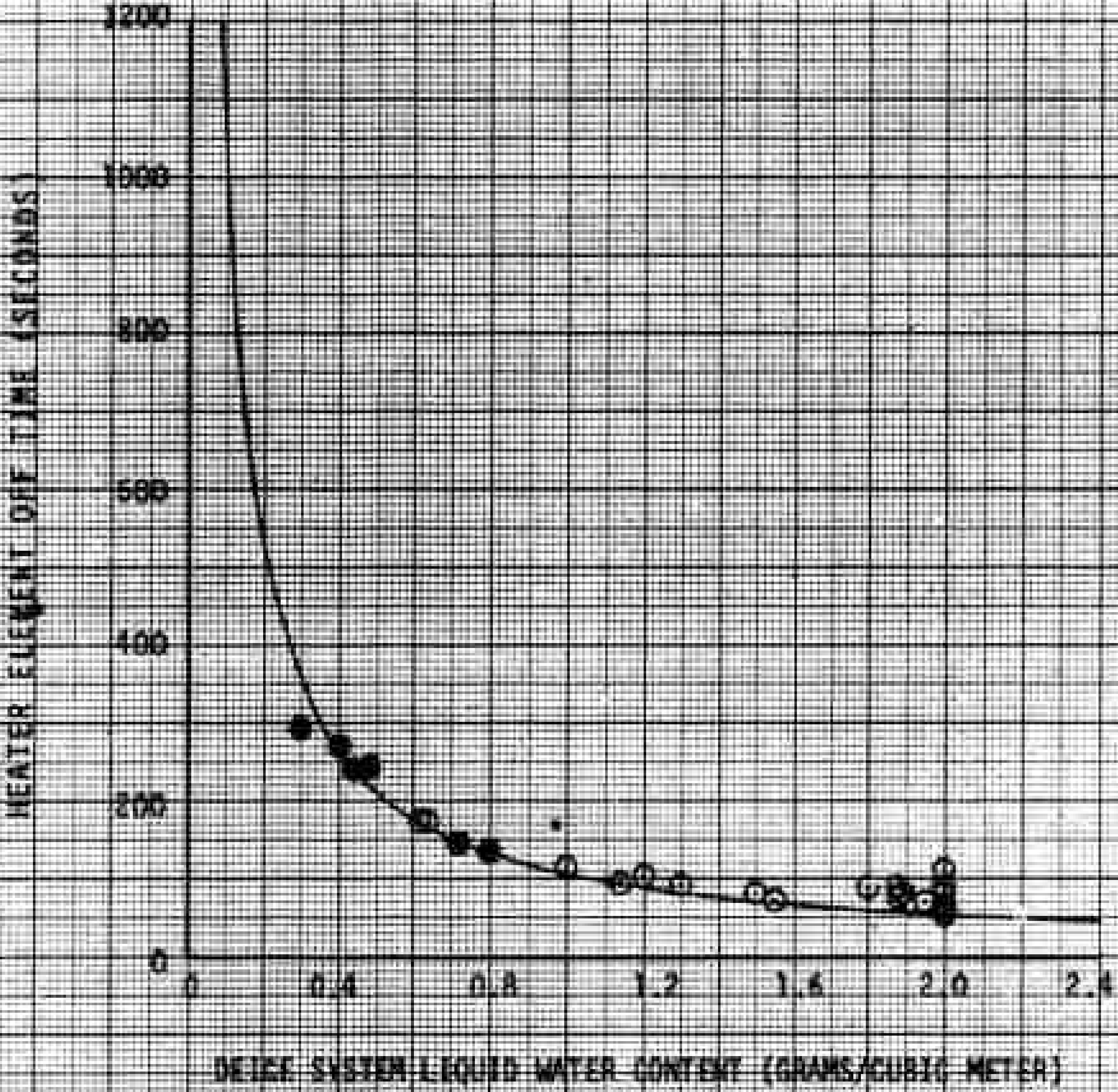


FIGURE 5
LIQUID WATER CONTENT INDICATIONS COMPARISON
AH-64A USA S/N 82-23356

NOTES: 1. DATA POINTS OBTAINED IN NATURAL ICING CONDITIONS, AUTOMATIC MODE OF OPERATION
2. SHADED SYMBOLS DENOTE DEICE SYSTEM LWC SIGNAL MULTIPLIER SET AT 0.4, OPEN SYMBOLS DENOTE MULTIPLIER SET AT 1.0

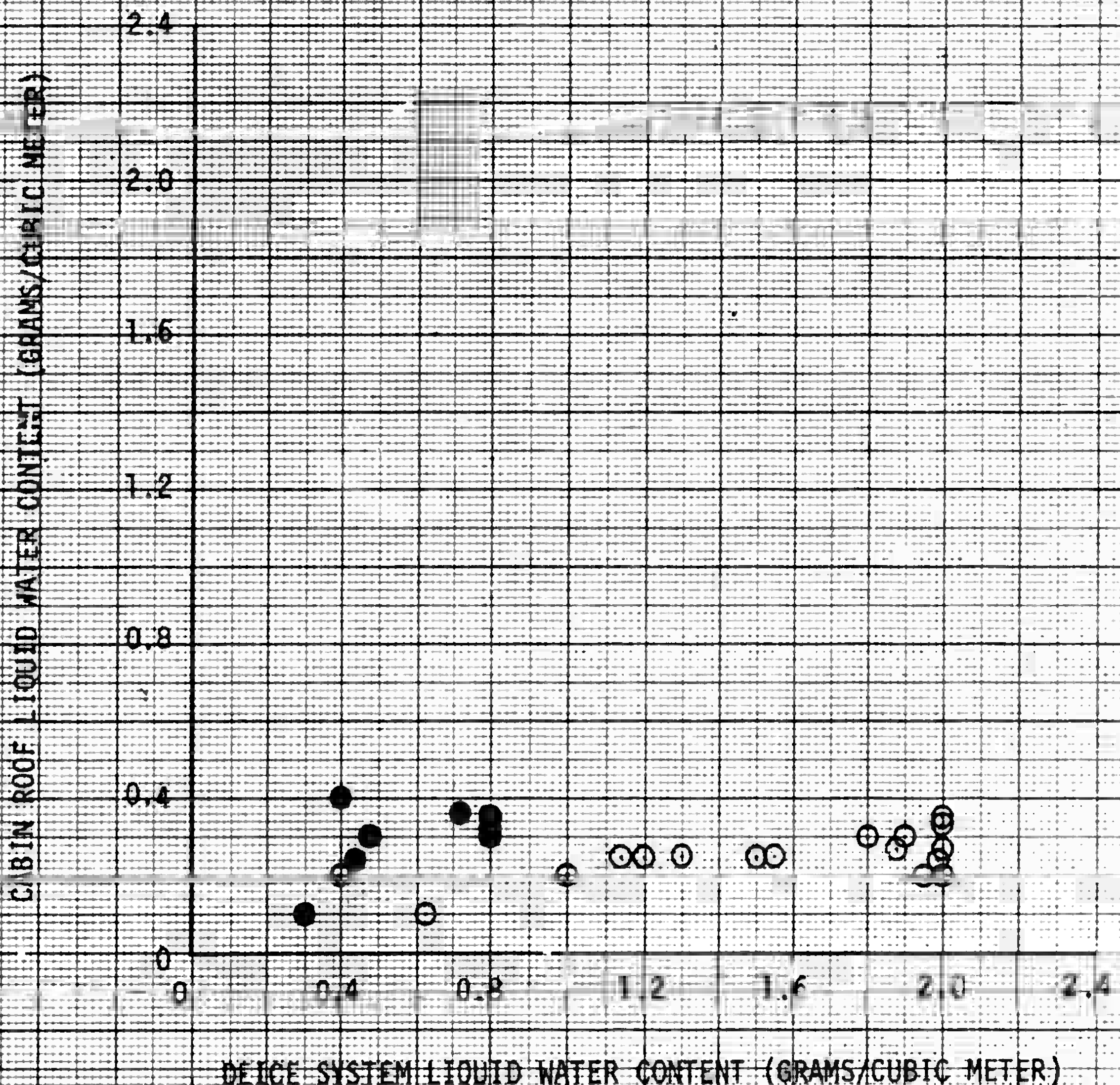
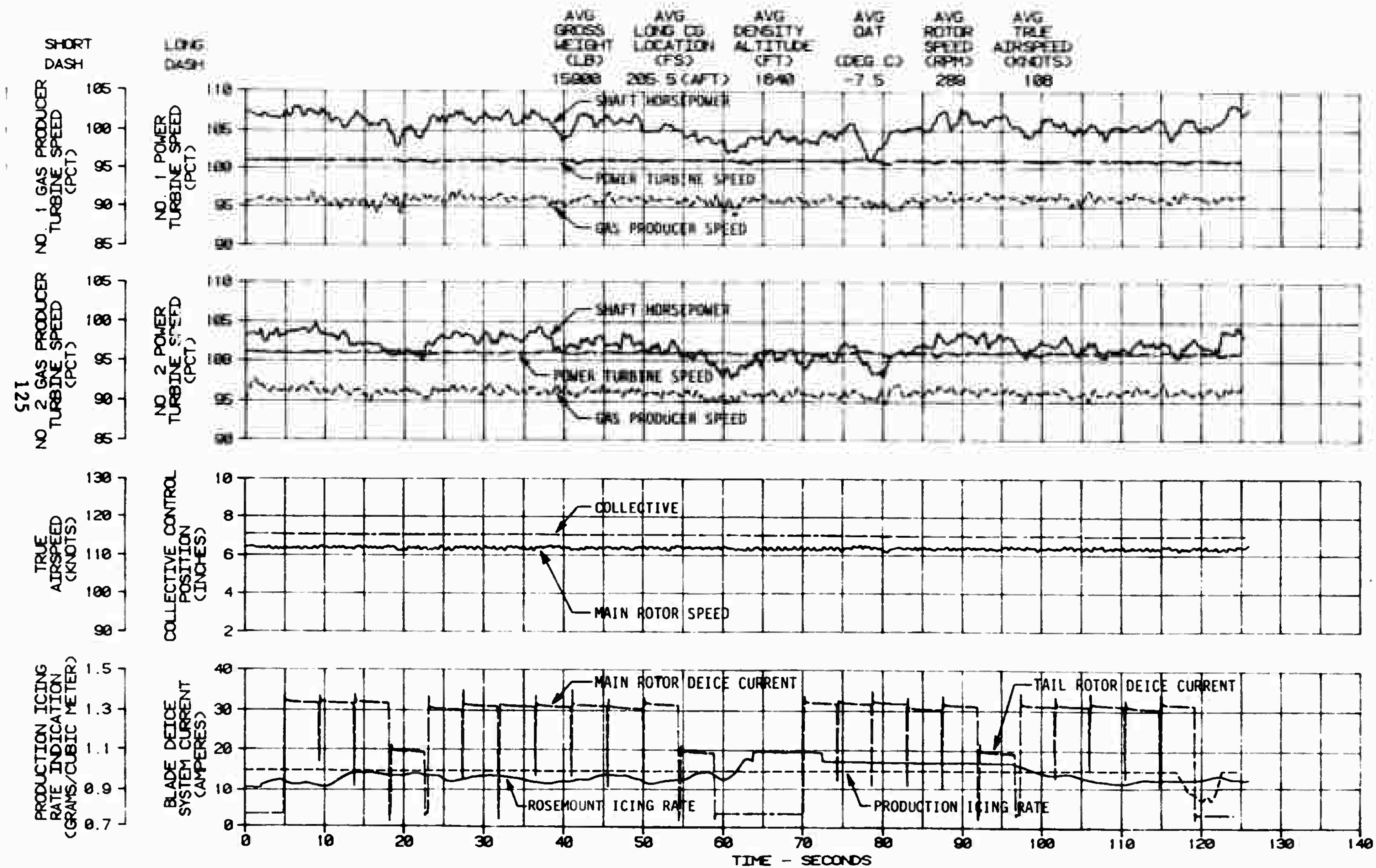


FIGURE 6
BLADE DEICE CYCLE
AH-64A USA S/N 82-23356



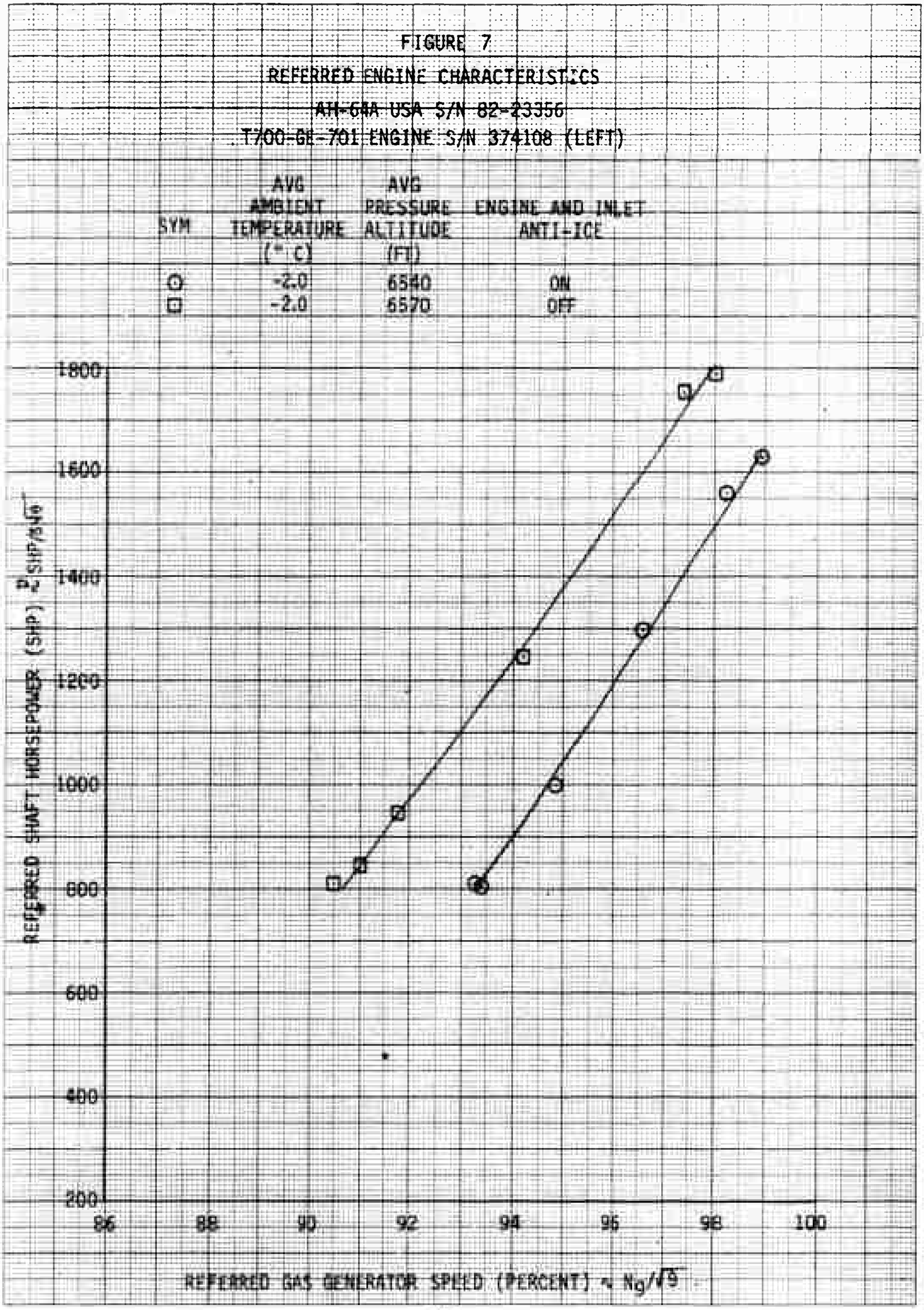


FIGURE 7
REFERRED ENGINE CHARACTERISTICS
AH-64A USA S/N 82-23356
T700-GE-701 ENGINE S/N 374108 (LEFT)
SYM
AVG AMBIENT TEMPERATURE (°C)
AVG PRESSURE ALTITUDE (FT)
ENGINE AND INLET ANTI-ICE
-2.0
6540
ON
-2.0
6570
OFF
REFERRED SHAFT HORSEPOWER (SHP) ~ SHP/δ√θ
1800
1600
1400
1200
1000
800
600
400
200
86
88
90
92
94
96
98
100
REFERRED GAS GENERATOR SPEED (PERCENT) ~ Ng/√θ

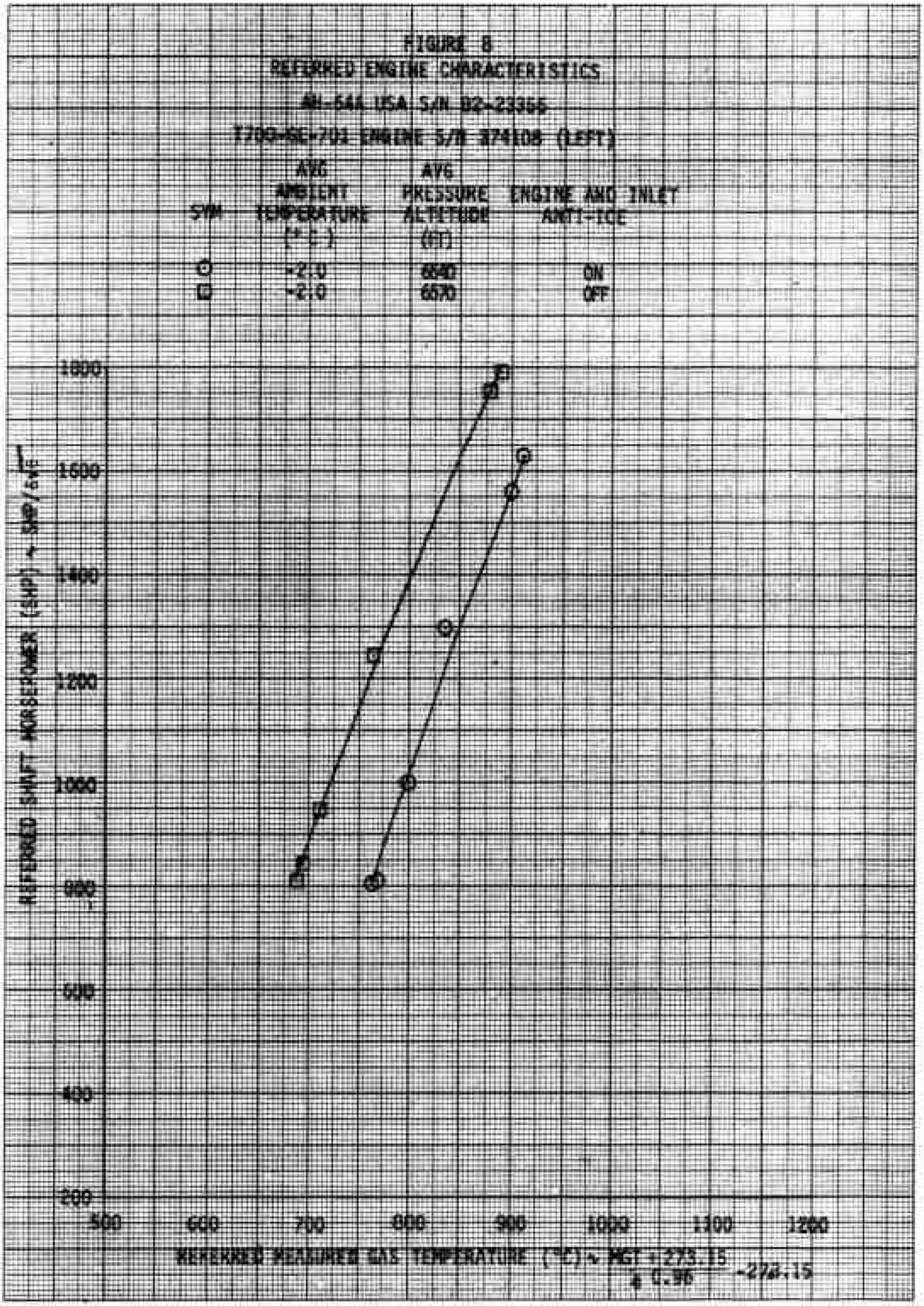


FIGURE 8
REFERRED ENGINE CHARACTERISTICS
AH-64A USA S/N 82-23356
T700-GE-701 ENGINE S/N 374108 (LEFT)
SYM
AVG AMBIENT TEMPERATURE (°C)
AVG PRESSURE ALTITUDE (FT)
ENGINE AND INLET ANTI-ICE
-2.0
6840
ON
-2.0
6570
OFF
REFERRED SHAFT HORSEPOWER (SHP) ~ SHP/δ√θ
1800
1600
1400
1200
1000
800
600
400
200
500
600
700
800
900
1000
1100
1200
REFERRED MEASURED GAS TEMPERATURE (°C) ~ MGT + 273.15 / θ 0.96 -273.15

EQUIPMENT PERFORMANCE REPORTS

The following Equipment Performance Reports (EPR's), DARCOM Form 2134, 1 September 1976, were submitted by USAAEFA during this evaluation.

EPR Number	Subject
84-23-01	Wiring harness assembly: TADS system
84-23-02	Wiring harness assembly: TADS system
84-23-03	Vibration gearbox caution/warning system
84-23-04	Tail rotor elastomeric bearing
84-23-05	APU control panel
84-23-06	Main/tail rotor blade deice controller
84-23-07	Visual MUX assembly (TADS FLIR)
84-23-08	Tail rotor blade damage
84-23-09	Holder assembly
84-23-10	Tail rotor elastomeric bearing
84-23-11	Engine foreign object damage
84-23-12	Tail rotor elastomeric bearing
84-23-13	Power supply: strobe lights
84-23-14	Nose gearbox wiring harness
84-23-15	Tail rotor elastomeric bearing
84-23-16	Fault detection/location system
84-23-17	Engine anti-ice system

Encl 5

EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 22 March 1985 / OFFICE SYMBOL: SAVTE-TA
TO: Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	FROM: Commander US Army Aviation Engr Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-1	USAAEFA Proj. No. 84-23	AH-64 Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Wiring Harness Assembly: TADS System

11. FSN:	12. MFR PART NO.: 13076054
13. DRAWING NO.:	14. MFR: Martin Marietta
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 26 February 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
			X	a. INCIDENT		a. REPLACED
				b. INFORMATION	X	b. REPAIRED
X	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
	c. INSPECTION		X	b. MAJOR		e. REMOVED
	d. OTHER			c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During cross country flight, faults appeared in TADS system. Faults were evident throughout two days of cross country upon arrival for icing testing. Faults were (1) TADS stuck in NFOV (2) TADS cools down and has no display, (3) TADS computer doesn't power up (4) PNVS operates intermittently, uncommanded. Martin Marietta troubleshot TADS approximately 16 hours plus and found burnt wire. 1A1W2J4-42 to 1A1W2TB2-8A located in wiring harness assembly P/N: 13076054 harness assembly located at fuselage station 35.50 behind PNVS rotating mount. The wiring burnt had been nicked or cut on terminal block frame, apparent sharp edge on frame where wires are hand tied in groups of 3 or 4 wires. Cut resulted in short of +28V wire to OPT. ADJ. motor for field of view. Traced wire back to within 2 inches of connector which would have been unrepairable at connector. Assembly P/N: 13076054 would have been replaced. Repair time was 8 hours plus. Martin Marietta manual reference: DEP 9-1270-476-23P. See attached drawing for location of repair and harness assembly.

* Vendor's P/N under Martin Marietta 13075408 for harness assembly.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
MICHAEL L. WALES ACFT MECH (805)277-2522; AV: 350-	MICHAEL A. GULICK CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

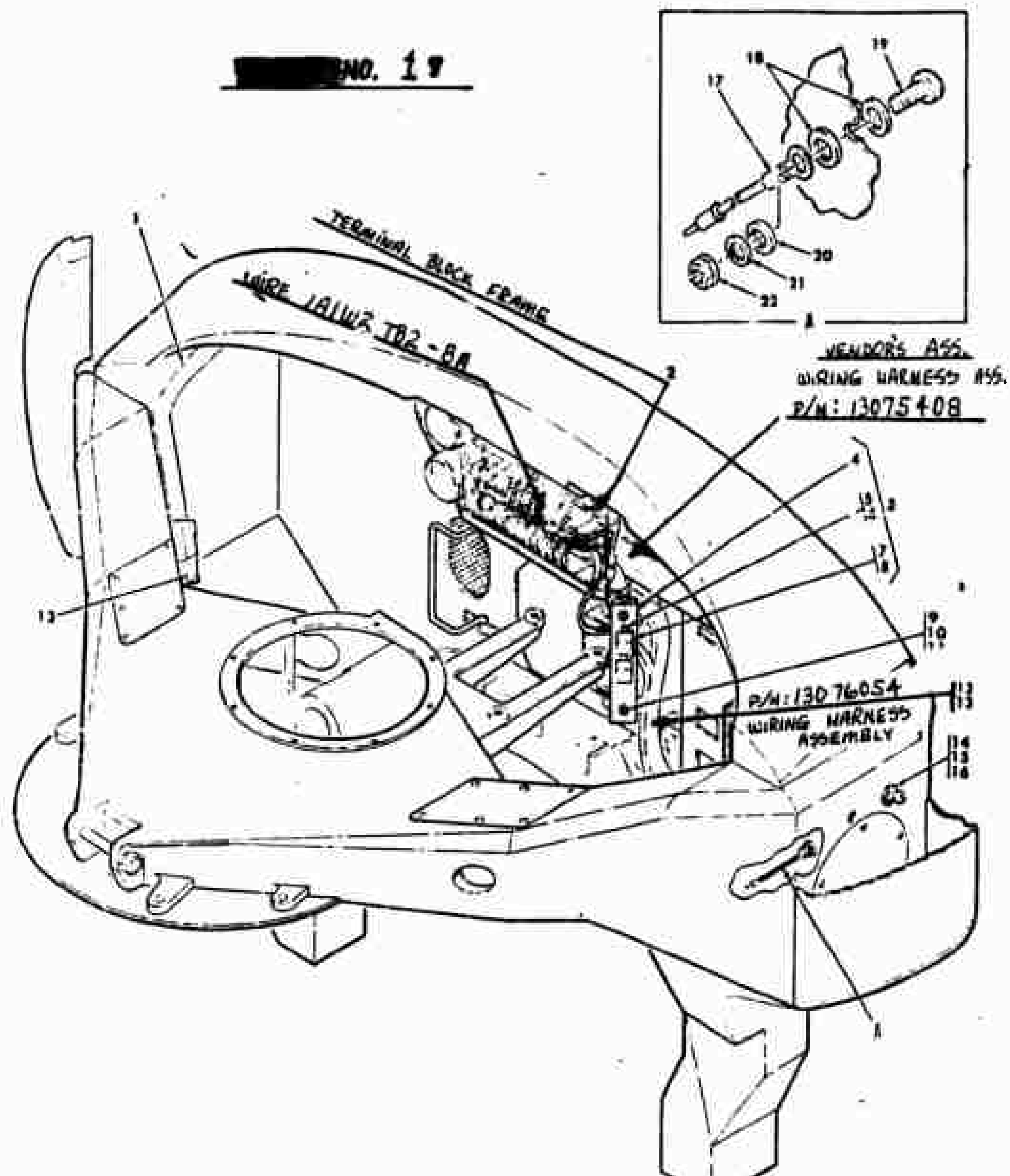


Figure 11. Aircraft interface assembly (13076250) (Group 0110) (Sheet 1 of 2).

Attachment 1 of EPR No. 84-23-1

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 22 March 1985

OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Engineering Flt Actv ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-2	USAAEFA Proj. No. 84-23	AH-64 Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64 A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Wiring Harness Assembly: TADS System

11. FSN: ------	12. MFR PART NO.: 130761665
13. DRAWING NO.: ---------	14. MFR: Martin Marietta
15. QUANTITY: 1	16. NEXT ASSEMBLY: -----------
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE: ------------

III INCIDENT DATA

19. DATE OF OCCURRENCE: 26 February 1985

22. MAINT SPT, ELM, CODE:

20. TYPE OF REPORT:		21. ACTION TAKEN:	
X	a. INCIDENT		a. REPLACED
	b. INFORMATION	X	b. REPAIRED

23. OBSERVED DURING		24. TEST ENVIRONMENT.
X	a. OPERATION	
	b. MAINTENANCE	
	c. INSPECTION	
	d. OTHER	

25. INCIDENT CLASSIFICATION:			
			c. ADJUSTED
	a. CRITICAL		d. DISCONNECTED
X	b. MAJOR		e. REMOVED
	c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During cross country flights, faults were noted in TADS System. Upon arrival for icing testing, TADS system was troubleshot by Martin Marietta and fault was discovered to be stripped wires causing short of +28V wire to OPT. ADJ. FOV, specifically wire 1A4W1P1-2 to 1A4W1J3-40, dayside. Three other wires were damaged in adjacent area to funny harness. Damage appeared to be from rough and jagged edges in casting with TADS/PNVS systems operated, harness moves in acft. Interface assembly P/N: 13076250 (Vendor # 13076075) during left/right swiveling motion. Access to this harness was through covers P/N: 13075830-1 and P/N: 13074621. Fairing on left side has to open P/N: 13075114 for access to cover P/N: 13074621. Funny harness involved in repair was P/N: 13076165. The short was from wire 1A4W1P1-2 to 1A4W1J3-40 dayside. This short also caused short and damage to wire 1A1W2TB2-C TO side. This short also caused short and damage to wire 1A1W2TB2-C TO 1A1W2J10-2.

See attached drawing and refer to EPR 84-23-1

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	
MICHAEL L. WAITE ACFT MECH (805)277-2522; AV: 350-	MICHAEL A. GULICK CPT, AV, C, Plans and Programs

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[redacted] 2 9

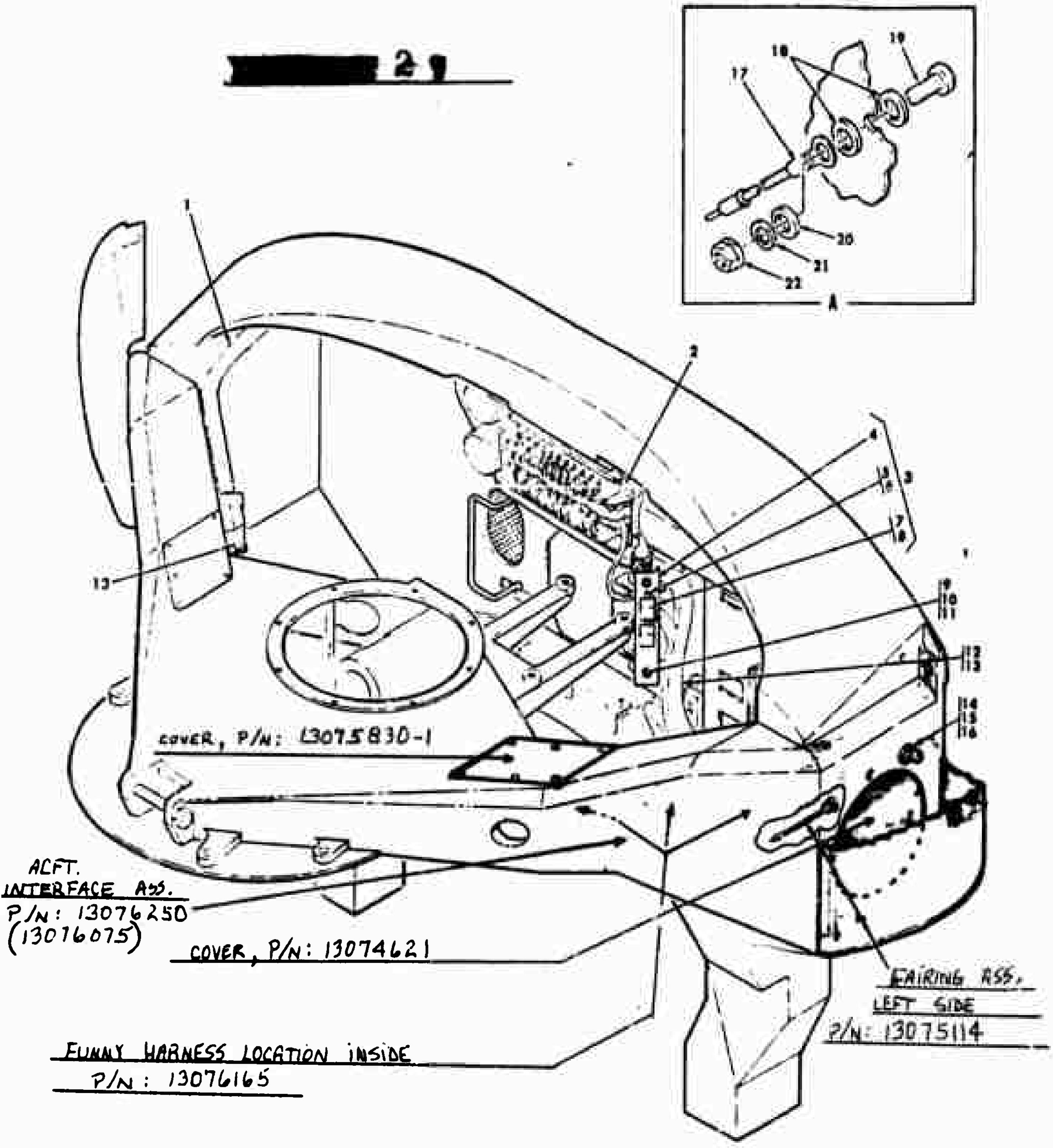

Figure 11. Aircraft interface assembly (13076250) (Group 0110) (Sheet 1 of 2).

Attachment 1 of EPR No. 84-23-2

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 22 March 1985

OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Engineering Flt Actv ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-3	USAAEFA Proj. No. 84-23	AH-64 Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL YAH-64A	5. SERIAL NO. 83-23356
6. QUANTITY 1	7. LIFE PERIOD:
8. MFR Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Vibration Gearbox Caution/Warning Light

11. FSN:	12. MFR PART NO.:
13. DRAWING NO.:	14. MFR:
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 5 March 1985

22. MAINT SPT. ELM. CODE:

23. OBSERVED DURING	24. TEST ENVIRONMENT:	20. TYPE OF REPORT:	21. ACTION TAKEN:
X a. OPERATION		X a. INCIDENT	a. REPLACED
b. MAINTENANCE		b. INFORMATION	b. REPAIRED
c. INSPECTION		25. INCIDENT CLASSIFICATION:	c. ADJUSTED
d. OTHER		a. CRITICAL	d. DISCONNECTED
		b. MAJOR	e. REMOVED
		X c. MINOR	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

On Flight #190, 5 March: Light for gearbox vibration illuminated frequently and at random through duration of flight. Electricians troubleshot wiring for any obvious faults from gearbox monitors (accelerometers) to signal processor then to annunicator panel. No apparent wiring damage found. Signal processor ordered, P/N: 7-211E10004-3, S/N: 1019, installed on acft. Signal processor S/N: 1074 removed. Aircraft Flight # 191 6 March, fault repeated as same as Flight # 190. Electricians ordered accelerometer, P/N: 7-311E10007. Removed and replaced intermediate gearbox accelerometer located near fuselage station, 525.0. Next flight, fault reoccurred. Electricians removed accelerometer that was installed new at intermediate gearbox and installed in 90^{o} gearbox, and removed old one. Next flight, fault reoccurred. Fault is thought to be in -3 signal processor.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29.
MICHAEL L. WAITS ACFT MECH (805)277-2522, AV: 350-	MICHAEL A. GULICK CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134 — Previous edition may be used until exhausted.

EQUIPMENT PERFORMANCE REPORT
(DARCOM/AMCR 700-38)

DATE: 22 March 1985

OFFICE SYMBOL: SAVTE-TA

TO: Commander
US Army Aviation Systems Command
ATTN: AMCPM-AAH-SE
4300 Goodfellow Blvd, St. Louis, MO 63120

FROM: Commander
US Army Avn Engineering Flight Activity
ATTN: SAVTE-TA
Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-4	USAAEFA Proj. No. 84-23	AH-64 Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Elastromeric bearing (Mount. Resilient, Tail Rotor)

11. FSN:	12. MFR PART NO.: 7-211421025
13. DRAWING NO.:	14. MFR: LORD
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 12 March 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
	a. OPERATION		X	a. INCIDENT		a. REPLACED
	b. MAINTENANCE			b. INFORMATION	X	b. REPAIRED
X	c. INSPECTION			25. INCIDENT CLASSIFICATION:		c. ADJUSTED
	d. OTHER			a. CRITICAL		d. DISCONNECTED
			X	b. MAJOR		e. REMOVED
				c. MINOR		NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During ground inspection, after No. 2 flight, resilient mount was found to have debonded from forked housing on tail rotor yoke housing. Further inspection revealed only one of the four mounts had debonded. Maintenance crews removed the outboard T/R to remove debonded mount. Clean up on fork assembly and old mount was required before rebonding and cure time. Proceeded through bonding procedures titled: Bonding Procedures for the 7-211421008 Tail Rotor Fork Assembly. Time cured before testing bond on ground was approximately 20 hours. Ground run was accomplished approximately 22 hours after bonding. Ground inspection after run revealed no debonding. Balancing of T/R was required after rebonding of mount.

Reference: TM 55-1520-238-23 and 23P; Fig. 127, Item 20

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:
MICHAEL L. WAITS
ACFT MECH
(805) 277-2622; AV: 350-

MICHAEL A. GULICK
CPT, AV, C, Plans and Programs

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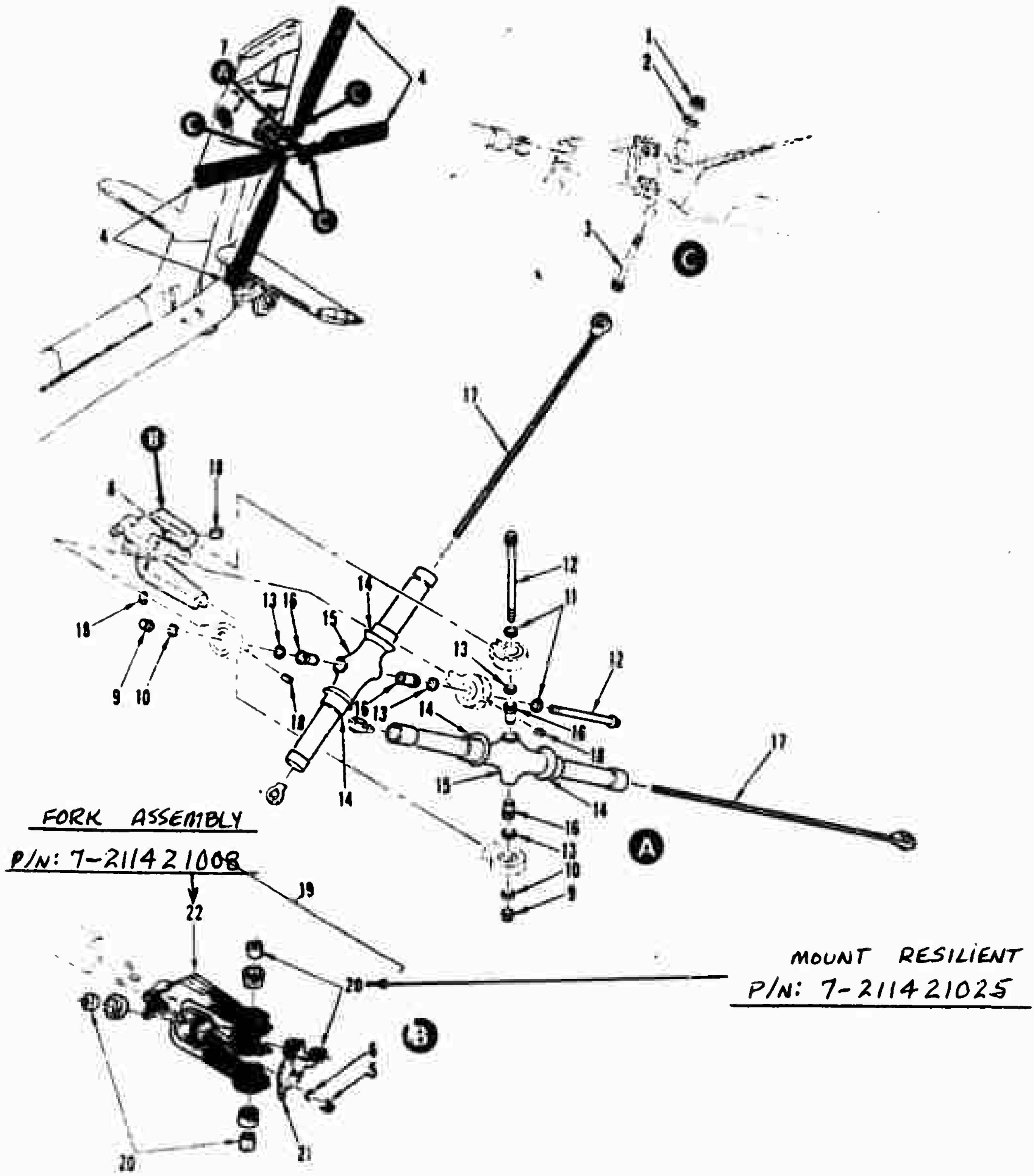


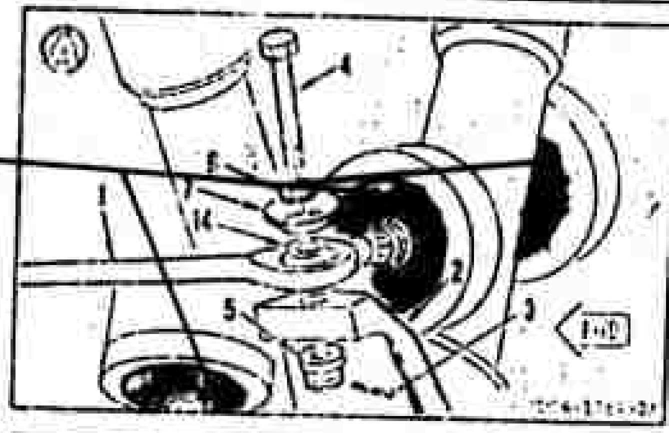
FIGURE 127. GROUP 05 TAIL ROTOR INSTALLATION
(HEAD BLADES) 7-311420002

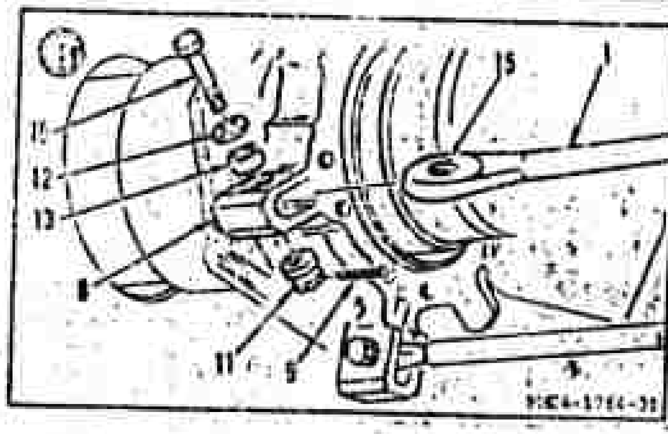
Attachment 1 of EPR No. 84-23-4

4

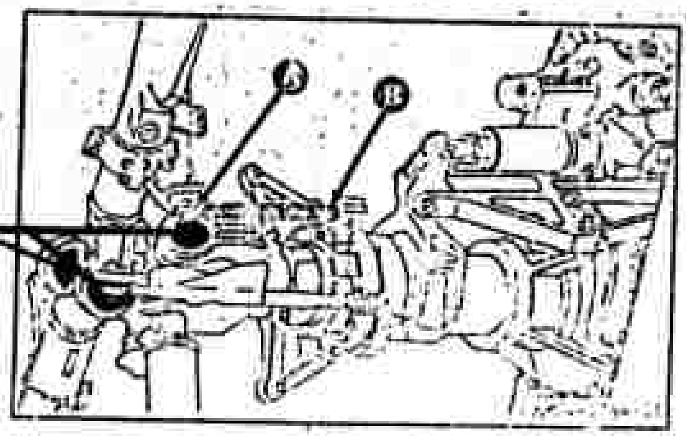
11-5-11. TAIL ROTOR PITCH LINK ASSEMBLY REMOVAL/INSTALLATION (cont) 11-5-11

RESILIENT MOUNT, T/R
P/N: 7-211421025





RESILIENT MOUNT T/R
P/N: 7-211421025



TO NEXT PAGE

Attachment 2 of EPR No. 84-23-4

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 22 March 1985

OFFICE SYMBOL: SAVTE-TA

TO: Commander
US Army Aviation Systems Command
ATTN: AMCPM-AAH-SE
4300 Goodfellow Blvd, St. Louis, MO 63120

FROM: Commander
US Army Avn Engineering Flt Activity
ATTN: SAVTE-TA
Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-5	USAAEFA Proj. No. 84-23	AH-64A Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: APU Control Panel, Start/Run

11. FSN:	12. MFR PART NO.: 7-311650003
13. DRAWING NO.:	14. MFR:
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 14 March 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
			X	a. INCIDENT	X	a. REPLACED
				b. INFORMATION		b. REPAIRED
X	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
	c. INSPECTION		X	b. MAJOR		e. REMOVED
	d. OTHER			c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

On APU start, pilot observed traces of smoke from his right console. Pilot terminated APU run. Electricians removed APU panel from right console and discovered an apparent burnt diode. Damaged panel was removed and a new APU panel was installed. Further trouble-shooting of damaged panel reaffirmed the one apparently burned diode, and another one was damaged due to the one that burned. No apparent cause for failure was discovered.

Reference: TM 55-1520-238-23

See attached drawing for diodes location and APU panel.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:
MICHAEL L. WAITS
ACFT MECH
(805)277-2522; AV: 350-

29. APPROVING COMMANDER:
MICHAEL A. GULICK
CPT, AV, C, Plans and Programs

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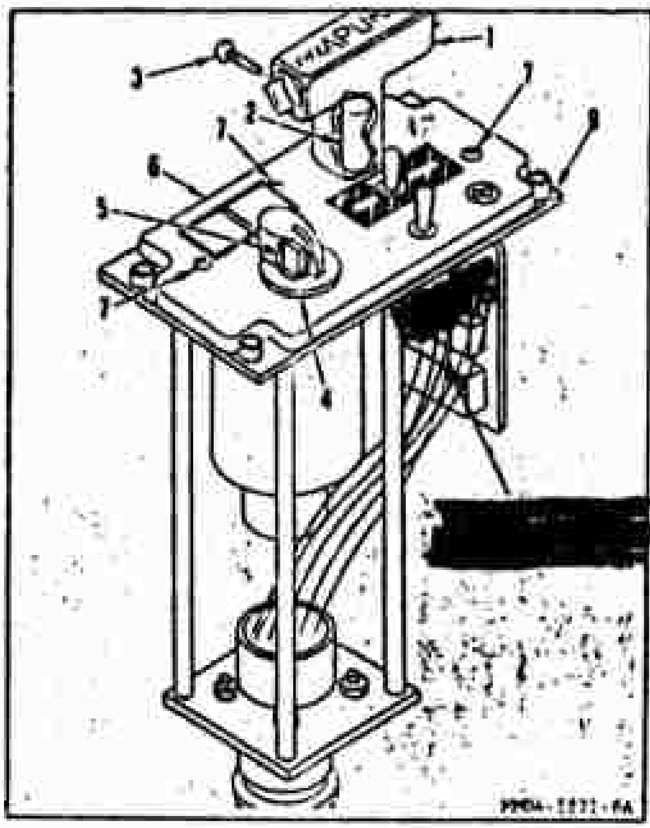
15-2-24. APU CONTROL PANEL START/RUN SWITCH REPLACEMENT (AVIM) (cont) 15-2-24

LOCATION OF APU PANEL
ON PILOT'S RT. CONSOLE



[redacted]

Attachment 1 of EPR No. 84-23-5

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 22 March 1985

OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Avn Engineering Flight Actv ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-6	USAAEFA Proj. No. 84-23	AH-64A Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO.
6. QUANTITY:	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Main and Tail Rotor Blade De-ice Controller

11. FSN:	12. MFR PART NO.: 7-311A10023-5 (S/N: 0052)
13. DRAWING NO.:	14. MFR: AEG - Telefunken
15. QUANTITY:	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE:

22. MAINT SPT, ELM, CODE:

20. TYPE OF REPORT:
- a. INCIDENT
- X b. INFORMATION

21. ACTION TAKEN:
- a. REPLACED
- b. REPAIRED
- c. ADJUSTED
- d. DISCONNECTED
- e. REMOVED
- f. NONE

23. OBSERVED DURING
- a. OPERATION
- X b. MAINTENANCE
- c. INSPECTION
- d. OTHER

24. TEST ENVIRONMENT.

25. INCIDENT CLASSIFICATION:
- a. CRITICAL
- X b. MAJOR
- c. MINOR

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

The requirement exists to trim the De-ice controller to the electrical resistance characteristics of each main and tail rotor blade. This procedure is required each time a main or tail rotor blade is changed. The reason for this is that the AEG-Telefunken controller has the inherent capability of preventing a blade or blade element from overheating. The overheat temp. limit is 55° Celcius. The controller measures blade element temperature by the change in electrical resistance as the element is heated. Hence the requirement for the controller to be "trimmed" for the variation in resistance in each blade/blade element. The procedure is only documented and the peculiar ground support equipment described in the rotor blade de-ice production test procedure (PTP) PTP 77-TP-7541-3, Rev A, Main and Tail Rotor Blade De-ice system. Furthermore charts are required which are contained only in the AEG-Telefunken manual. The specific procedure and equipment required is not contained in the -23 series TMs. The equipment is only available at HHI in Mesa, AZ. Since main and tail rotor blade changes are an AVUM function this will cause a maintainability problem at unit level.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	
MICHAEL L. WAITS [illegible]T MECH (850)277-2252; AV: 350-	MICHAEL A. GULICK CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

EQUIPMENT PERFORMANCE REPORT (DARCOM AMCR 700-38)	DATE: 22 March 1985
	OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd. St. Louis, MO 63120	Commander US Army Avn Engineering Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-7	USAAEFA Proj. No. 84-23	AH-64A Artificial and Natural Icing Tests

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: VISUAL MUX Assembly (TADS FLIR)

11. FSN:	12. MFR PART NO.: 13075911
13. DRAWING NO.:	14. MFR: Martin Marietta
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 19 March 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:	24. TEST ENVIRONMENT:	20. TYPE OF REPORT:	21. ACTION TAKEN:
X a. OPERATION		a. INCIDENT	X a. REPLACED
b. MAINTENANCE		X b. INFORMATION	b. REPAIRED
c. INSPECTION		25. INCIDENT CLASSIFICATION:	c. ADJUSTED
d. OTHER		a. CRITICAL	d. DISCONNECTED
		X b. MAJOR	e. REMOVED
		c. MINOR	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

Target acquisition designation system (TADS) FLIR image modulates inflight making the system marginally useable. This image modulation is not the result of any anti-ice systems being activated. The electro-optical multiplex unit has been replaced. System does not appear to have been functioning properly since the beginning of the icing tests. A representative FLIR image is required to complete the TADS portion of the icing test since one of the test objectives requires comparision of FLIR imagery with anti-ice systems on and off.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	FOR THE COMMANDER:
MICHAEL L. WALTS ACFT MECH (805)277-2522; AV: 350-	MICHAEL A. GULICK CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

3/8 PO

EQUIPMENT PERFORMANCE REPORT
(DARCOM/AMCR 700-38)

DATE: 29 March 1985

OFFICE SYMBOL: SAVTE-TA

TO: Commander
US Army Aviation Systems Command
ATTN: AMCPM-AAH-SE
4300 Goodfellow Blvd, St. Louis, MO 63120

FROM: Commander
US Army Aviation Flight Activity
ATTN: SAVTE-TA
Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-8	USAAEFA Project #84-23	AH-64A ICING TESTING

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: TAIL ROTOR BLADE

11. FSN:	12. MFR PART NO.: 7-311422050
13. DRAWING NO.:	14. MFR: COMPOSITE DYNAMICS
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 24 MARCH 1985

22. MAINT SPT, ELM, CODE:

20. TYPE OF REPORT:
- XXX a. INCIDENT
- b. INFORMATION

21. ACTION TAKEN:
- XXX a. REPLACED
- b. REPAIRED
- c. ADJUSTED
- d. DISCONNECTED
- e. REMOVED
- f. NONE

23. OBSERVED DURING:
- XXX a. OPERATION
- b. MAINTENANCE
- c. INSPECTION
- d. OTHER

24. TEST ENVIRONMENT:

25. INCIDENT CLASSIFICATION:
- a. CRITICAL
- XXX b. MAJOR
- c. MINOR

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

Post flight inspection revealed damage to the #3 and #4 tail rotor blades. This flight was a 1 hour 16 minute immersion in natural icing conditions at -5° C to -8° C and with a liquid water content (LWC) of 0.3 to 0.5 gm/m. Damage to the #4 blade (an inboard blade) was on the inboard side and is described in detail in the attached HHI Field Data Report. Damage to the #3 blade was a two inch deformation of the leading edge, approximately 1.5 inches from blade tip. The #3 blade also sustained a 1" spanwise crack, approximately 1.5 inches from the trailing edge on the outboard side of the blade and one inch inboard of the blade tip. The #3 blade is an outboard tail rotor blade. Tail rotor blade damage is attributed to ice shedding. However, it has not been determined whether the shed ice came from the main rotor or fuselage.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:
for JAMES M. ADKINS
CW4, USA
(805) 277-4986
Gary L Bender

29. FOR THE COMMANDER:
MICHAEL A. GULICK
CPT, AV, C, Plans and Programs
(805) 277-4643

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

PO

EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 29 March 1985 OFFICE SYMBOL: SAVTE-TA
TO: Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	FROM: Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-9	USAAEFA Project #84-23	AH-64 Artificial & Natural Icing Test

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-233356
6. QUANTITY: 1	7. LIFE PERIOD:
8 MFR: Hughes	9 USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Holder (TADS)	
11. FSN:	12. MFR PART NO.: 13075990
13. DRAWING NO.:	14. MFR: MARTIN MARIETTA
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 24 March 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING	24. TEST ENVIRONMENT:	20. TYPE OF REPORT:	21. ACTION TAKEN:
X a. OPERATION		X a. INCIDENT	a. REPLACED
b. MAINTENANCE		b. INFORMATION	b. REPAIRED
c. INSPECTION		25. INCIDENT CLASSIFICATION:	c. ADJUSTED
d. OTHER		a. CRITICAL	d. DISCONNECTED
		X b. MAJOR	e. REMOVED
		c MINOR	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During a 1.2 hour flight in natural icing conditions it was noted by the CPG that the day TV image was inoperative. Post flight inspection revealed a broken internal part described in the enclosed pages. This part cannot be changed without performing critical adjustments to the optical system. Therefore, it has been proposed that the entire dayside assembly be replaced. Action to be taken is uncertain at this time.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
for JAMES M. ADKINS CW4, US Army (805) 277-4986 *Gary L Bender*	MICHAEL A GULICK CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134 — *Previous edition may be used until exhausted.*

AH

PO

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 29 March 1985

OFFICE SYMBOL: SAVTE-TA

TO: Commander
US Army Aviation Systems Command
ATTN: AMCPM-AAH-SE
4300 Goodfellow Blvd, St. Louis, MO 63120

FROM: Commander
US Army Aviation Flight Activity
ATTN: SAVTE-TA
Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-10	USAAEFA Project #84-23	AH-64 Artificial & Natural Icing Test

I MAJOR ITEM DATA

4. MODEL: AH-64A	5. SERIAL NO.: 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: Hughes	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: Elastromeric Bearing (Mount Resilient Tail Rotor)

11. FSN:	12. MFR PART NO.: 7-211421025
13. DRAWING NO.:	14. MFR: LORD
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 25 March 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
			X	a. INCIDENT		a. REPLACED
				b. INFORMATION	X	b. REPAIRED
	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
X	c. INSPECTION		X	b. MAJOR		e. REMOVED
	d. OTHER			c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During ground inspection following a ground run for tail rotor change due to T/R damage on FLT 15, resilient mount was found to have debonded from forked housing on tail rotor yoke hosuing. Further inspection revealed only one of the four mounts had debonded. Maintenance crews removed the outboard T/R to remove debonded mount. Clean up on fork assembly and old mount was required before rebonding and cure time. Proceeded through bonding procedures titled: Bonding Procedures for the 7-211421008 Tail Rotor Fork Assembly. Time cured before testing bond on ground was approximately 20 hours. Ground inspection after run revealed no debonding. Balancing of T/R was required after rebonding of mount. This was not the same elastrometric bearing that was debonded on FLT #2 (see EPR #004).

REFERENCE: TM 55-1520-238-23 and 23P; FIG. 127 ITEM 20.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:
MICHAEL L. WAITS
ACFT MECH
(805) 277-2522; AV: 350-

29. FOR THE COMMANDER:
MICHAEL A. GULICK
CPT, AV, C, Plans and Programs

DARCOM FORM 1 SEP 76 2134

Previous edition may be used until exhausted.

EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 30 March 1985 OFFICE SYMBOL: SAVTE-TA
TO: Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	FROM: Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-11	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL: AH-64A	5. SERIAL NO.: 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: ENGINE

11. FSN: 2840-01-114-2111	12. MFR PART NO.: 6044T06G01 (S/N: 374133)
13. DRAWING NO.:	14. MFR: GENERAL ELECTRIC
15. QUANTITY: 1 EA.	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 28 MARCH 1985	20. TYPE OF REPORT:	21. ACTION TAKEN:
22. MAINT SPT, ELM, CODE:	X a. INCIDENT	X a. REPLACED
23. OBSERVED DURING: / 24. TEST ENVIRONMENT:	b. INFORMATION	b. REPAIRED
a. OPERATION	25. INCIDENT CLASSIFICATION:	c. ADJUSTED
b. MAINTENANCE	a. CRITICAL	d. DISCONNECTED
X c. INSPECTION	X b. MAJOR	e. REMOVED
d. OTHER	c. MINOR	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

Following a 1 hr. and 20 minute flight in natural icing conditions (total 2.0 hr. flight time), the ground crew reported an unusual engine noise during landing and shutdown. The flight crew was not aware of any unusual engine noise. Engines were bore scoped by General Electric representative. Number two (2) engine was found to have a 5/16 to 3/8 inch curl in the tip of one blade in the first stage compressor. General Electric recommended that engine was flyable for a maximum of 10 hours and engine should be changed prior to departing Duluth. AVSCOM Engineering gave VOCO release on 29 March 1985 for five (5) hours flight time to complete icing tests. Engine will be replaced prior to departing from Duluth.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER	29. FOR THE COMMANDER:
JAMES M. ADKINS CW4, USA (805) 277-4986	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 277-4643

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EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 29 March 1985 / OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-12	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: ELASTROMETRIC BEARING (MOUNT RESILENT TAILROTOR)

11. FSN:	12. MFR PART NO.: 7-211421025
13. DRAWING NO.:	14. MFR: LORD
15. QUANTITY:	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 28 MARCH 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
			X	a. INCIDENT		a. REPLACED
				b. INFORMATION	X	b. REPAIRED
	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
X	c. INSPECTION		X	b. MAJOR		e. REMOVED
	d. OTHER			c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During ground inspection following Flight #17 the resilent mount was found to have debonded from forked housing on tailrotor yoke housing. Maintenance crew removed the outboard T/R to remove debonded mount. Repair was made as before on mount I.A.W. rebonding procedures (See EPR #004). Mount cured for the 24 hours and then was run-up for T/R balancing. This was the same mount that debonded after Flight #15 (See EPR #010). Tailrotor balancing was good and ground inspection after run found no discrepancies with bonding of mount.

REFERENCE: TM55-1520-238-23 AND 23P; FIG.127, ITEM 20.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
JAMES M. ADKINS CW4, USA (805) 277-4986 SIGNED	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 27704643 SIGNED

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

EQUIPMENT PERFORMANCE REPORT
(DARCOM AMCR 700-38)

DATE: 4 APRIL 1985	OFFICE SYMBOL: SAVTE-TA

TO: Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	FROM: Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.: 84-23-13	2. TECOM/AVSCOM PROJ NO.: USAAEFA Project #84-23	3. TEST TITLE AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: POWER SUPPLY (STROBE LIGHTS)	
11. FSN:	12. MFR PART NO.: 7-311B12022-3
13. DRAWING NO.:	14. MFR:
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 3 APRIL 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:
- [ ] a. OPERATION
- [ ] b. MAINTENANCE
- [x] c. INSPECTION
- [ ] d. OTHER

24. TEST ENVIRONMENT:

20. TYPE OF REPORT:
- [x] a. INCIDENT
- [ ] b. INFORMATION

25. INCIDENT CLASSIFICATION:
- [ ] a. CRITICAL
- [ ] b. MAJOR
- [x] c. MINOR

21. ACTION TAKEN:
- [ ] a. REPLACED
- [ ] b. REPAIRED
- [ ] c. ADJUSTED
- [ ] d. DISCONNECTED
- [ ] e. REMOVED
- [ ] f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

During pre-flight inspection of cockpit area, strobe lights were not working both sides. Electrician troubleshot power supply for strobe lights P/N: 7-311B2022-3 located, left XSMN panel, access, next to left transformer. Previous months had shown two other power supply failures. Power supply had all voltages going to it as required. Condensation was only probable cause noted after troubleshooting. Cleaned/dried connectors and reapplied PWR. Strobes worked.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER: MICAHEL WAITS ACFT MECH (805) 277-2522	29. FOR THE COMMANDER MICAHEL A. GULICK CPT, AV, C, PLANS & PROGRAMS (805) 277-4643

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 4 APRIL 1985
	OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-14	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO.: 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: NOSE GEARBOX FAIRING, LOWER HALF	
11. FSN:	12. MFR PART NO.: 7-311140174-2
13. DRAWING NO.:	14. MFR: HUGHES
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 3 APRIL 1985		20. TYPE OF REPORT:	21. ACTION TAKEN:
22. MAINT SPT, ELM, CODE:		a. INCIDENT	a. REPLACED
23. OBSERVED DURING:	24. TEST ENVIRONMENT:	X b. INFORMATION	b. REPAIRED
a. OPERATION		25. INCIDENT CLASSIFICATION:	c. ADJUSTED
X b. MAINTENANCE		a. CRITICAL	d. DISCONNECTED
c. INSPECTION		b. MAJOR	e. REMOVED
d. OTHER		X c. MINOR	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

When installing nose gearbox fairing, lower half, P/N: 7-311140174-2, the wire bundle to the nose gearbox was found to be pinched between the fairing and fan (D/S Covering) P/N: 7-31132001. There was no definite indication to the desired routing or anywhere to secure the wires from being left to dangle. Further inspection revealed the wires had been chaffing in that area due to no way to secure them to fairing. Velcro had been tried but it appeared that it doesn't hold. An alternate routing was tried behind a fairing support (see * on drawing), but connector at J48/P48 location was short from routing and won't connect.

See Attached Drawings.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
MICHAEL WAITS ACFT. MECHANIC (805) 277-2522	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 277-4643 SIGNED

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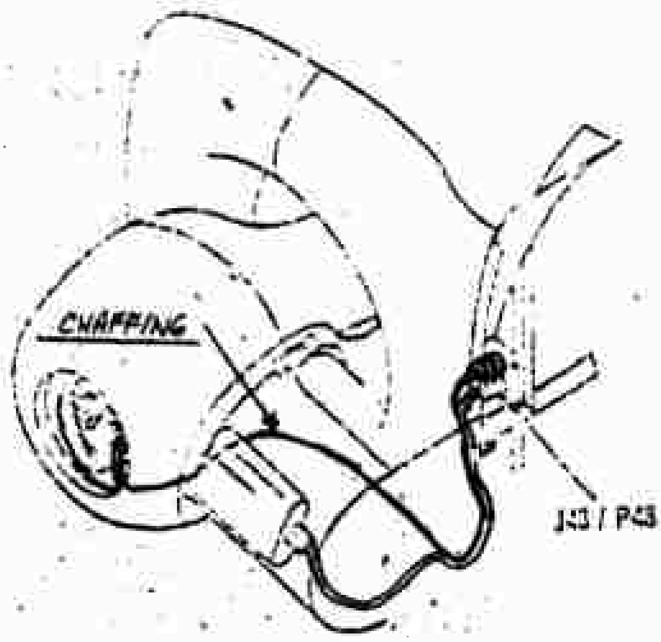

ENGINE 2 NOSE GEARBOX FAIRING

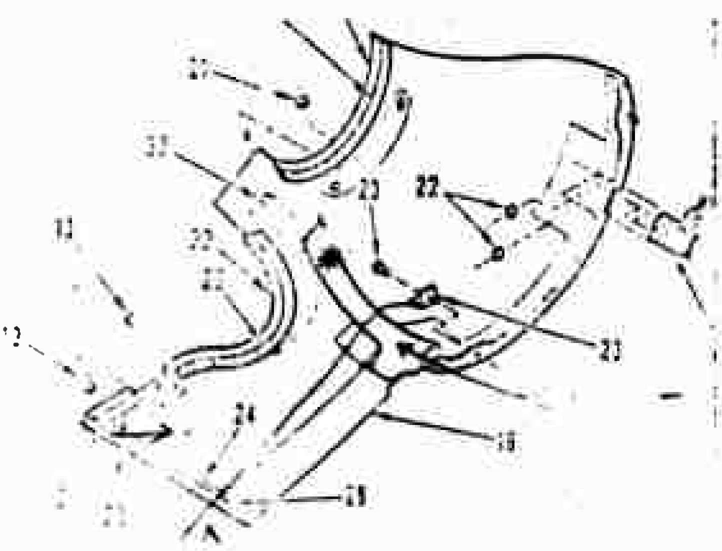

FIGURE 69. GROUP 02 NOSE GEARBOX
7-311140170-1.

EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 5 APRIL 1985
	OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-15	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO.: 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: ELASTROMERIC BEARING (MOUNT, RESILIENT TAILROTOR)	
11. FSN:	12. MFR PART NO.: 7-211421025
13. DRAWING NO.:	14. MFR: LORD
15. QUANTITY: 1	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 5 APRIL 1985			20. TYPE OF REPORT:		21. ACTION TAKEN:	
22. MAINT SPT, ELM, CODE:			X	a. INCIDENT		a. REPLACED
23. OBSERVED DURING:		24. TEST ENVIRONMENT:		b. INFORMATION	X	b. REPAIRED
	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
X	c. INSPECTION		X	b. MAJOR		e. REMOVED
	d. OTHER			c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

After first flight, a post flight inspection revealed no debonding of the T/R resilient mounts. Approximately 1 hour later, after acft. had been relocated inside the hangar and allowed temperature change, an inspection of the mounts revealed that the #1 T/R mount had debonded fo the third time (see EPR's 010, 012). The first debonding when adhesive was mixed, it was heated prior to application so as to speed up cure. This was #3 T/R mount, (see EPR #004), and has not since debonded. Other factors involved besides coldness could be the use of tap water instead of distilled or any materials in the bonding procedures used that might leave a filmy substance on fork assembly. P/N: 7-311421039 on the inside of the race area, contributing to debonding.

See EPR's oo4, 010, 012.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
MICHAEL WAITS ACFT. MECHANIC (805) 27702522	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 277-4643

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EQUIPMENT PERFORMANCE REPORT (DARCOM/AMCR 700-38)	DATE: 1 APRIL 1985
	OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-16	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO. 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: FAULT DETECTION/LOCATION SYSTEM FD/LS

11. FSN:	12. MFR PART NO.:
13. DRAWING NO.:	14. MFR:
15. QUANTITY:	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE: 5 MAR-31 MAR 1985

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
				a. INCIDENT		a. REPLACED
			X	b. INFORMATION		b. REPAIRED
X	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
	c. INSPECTION			b. MAJOR		e. REMOVED
	d. OTHER		X	c. MINOR		f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

A number of discrepancies have been noted in the operation of the anti-ice/de-ice fault detection and location system (FD/LS). The first problem area is the result of FD/LS software logic. When a fault is detected in a subsystem the FD/LS terminates its check of the remaining subsystems. For example, the FD/LS check of the de-ice controller includes a check of four subsystems: Tail rotor heater, main rotor heater, distributor and power controller (enclosure #1). If a fault in the tail rotor heater is detected, the FD/LS displays a tail rotor heater fail message and does not check the other subsystems. Therefore, the pilot can't determine the status of the main rotor heater, distributor, or the power controller using "on call" FD/LS.

The remaining subsystems are never checked and their status is therefore unknown. The test A/C presently displays a continuous "tail rotor heater no go" message because of an incorrect FD/LS software interface with the AEG-Telefunken De-ice Controller. Therefore, the status of the three other de-ice controller subsystems, (main rotor heater, distributor and power controller) is not available using FD/LS. Recommend the FD/LS be modified to allow a check of each subsystem irrespective of the status of the other

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
for JAMES M. ADKINS CW4, USA (805) 277-4786 Gary L Beeler	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 277-4643 SIGNED

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subsystems. Further recommend the FD/LS be modified to correctly interpret the tail rotor heater fail signal from the AEG-Telefunken De-ice Controller.

When A/C power is applied and the canopy heater switch is in the "off" position, the FD/LS will display a "canopy heater no-go, check heater switch" message. The message will appear under any outside temperature conditions. The keyboard switch must be rotated from "STBY" to FD/LS position and back to STBY to clear this message. The canopy heater position of the FD/LS is operating as designed. However, since modification of FD/LS software will be required to correct other discrepancies, recommend this subsystem will be altered to give a "cancpy heater no go" message only when the canopy heater switch is in the "on" position and the system has failed.

ENG. ANTI-ICE SYSTEM ENCLOSURE #2

When the ENG. anti-ice switch is placed in the "on" position, the following occurs:

a) Anti-ice bleed valve opens allowing bleed air to route to the forward ENG. inlet cowling.

b) The ENG. anti-ice thermal switch starts heating.

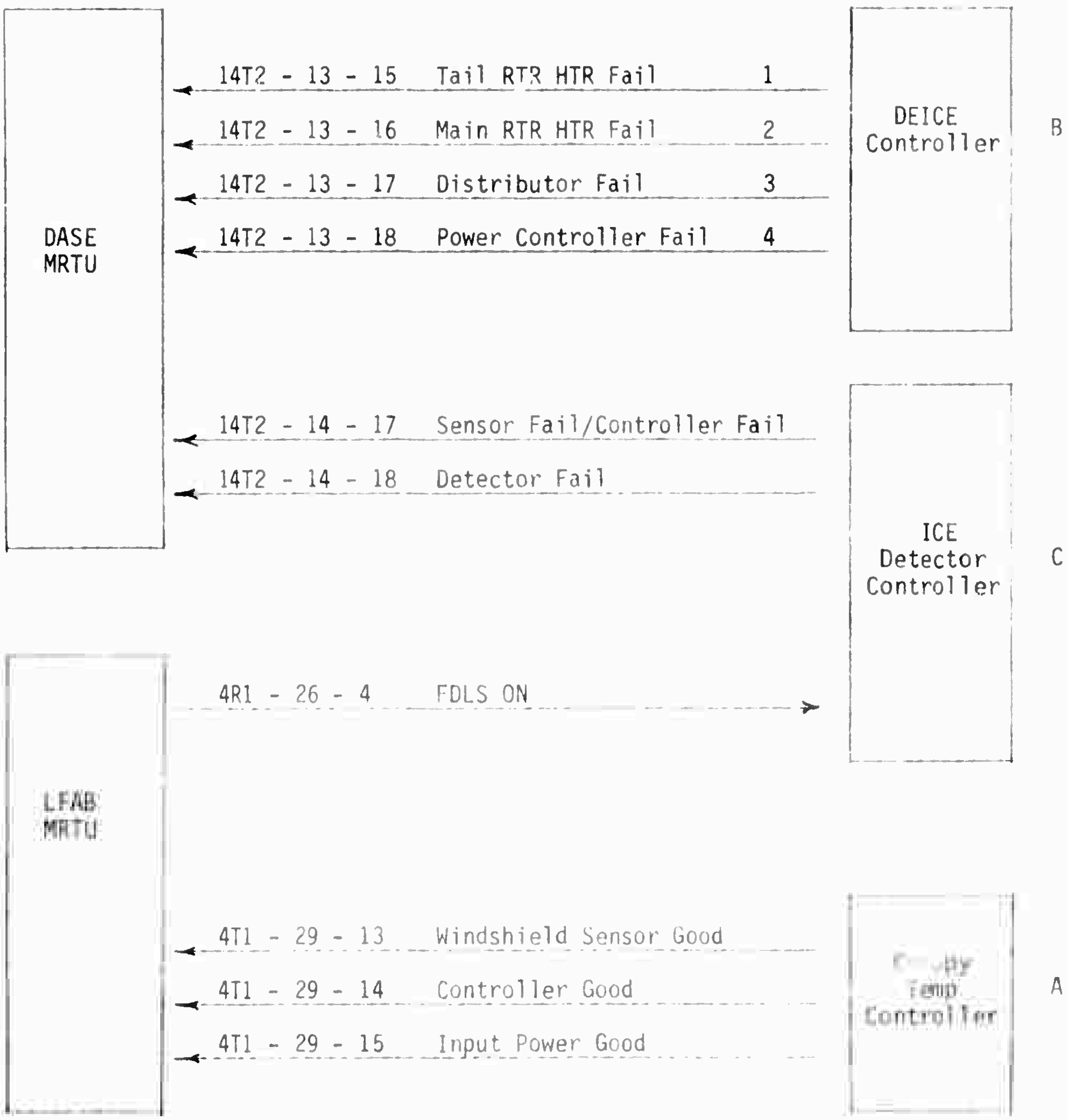
c) The nosegear box electro thermal heaters began heating.

The engine anti-ice caution lights (enclosure #3) and the master caution light illuminate until the nosegear box fairing temperature reaches 205° F to 235° F and the thermal switch reaches 155° C closing the switch contacts energizing the anti-ice fail relays. At this time the caution lights go out and the green engine inlet lights on the pilot's anti-ice panel (enclosure #2) illuminate. If for any reason the eng. anti-ice thermal switch did not close or the bleed valve did not open, the eng. anti-ice caution lights will remain illuminated. When the acft. enters icing conditions the engine anti-ice caution lights and the master caution lights cycle on and off continuously. It is believed that this condition is the result of the nosegear box fairing temperature dropping below 205° F. The caution light remains on for 3 to 5 minutes and then goes out when the nosegear box temperature again goes above 205° F. This intermittent illumination of the eng. anti-ice caution lights provides the pilot with a fail indication when in fact the system is functioning. Furthermore, the nosegear box fairing heaters are designed to operate at 30 AMPS. Circuit protection is provided through a 30 AMP circuit breaker which may result in the circuit breaker opening during normal system operation. This circuit breaker has not opened on any flights to date. However, suggest consideration be given to performing an electrical lcad survey of this system. It should also be noted that there is no provision to interrogate the engine anti-ice system through the on call or continuous FD/LS system. The addition of FD/LS software to check the engine anti-ice bleed valve, anti-ice thermal switch and nosegear box fairing temperature will aid in the troubleshooting of a system that has failed, because inflight conditions will be difficult to duplicate.

A copy of the proposed fix for the engine anti-ice caution lights cycling on and off is included on enclosure #4.

DEICE FDLS

A, B, C — Order in which major systems are checked by FDLS
1, 2, 3, 4 — Order in which subsystems are checked by FDLS



In Maintenance FDLS, if the Detector Fail Bit is high (14T2 - 14 - 18) the FCC commands FDLS ON (4R1 - 26 - 4). Then the FCC checks 14T2 - 14 - 17 to see if the sensor failed (Bit 17 = 1) or if the controller failed (Bit 17 = 0).

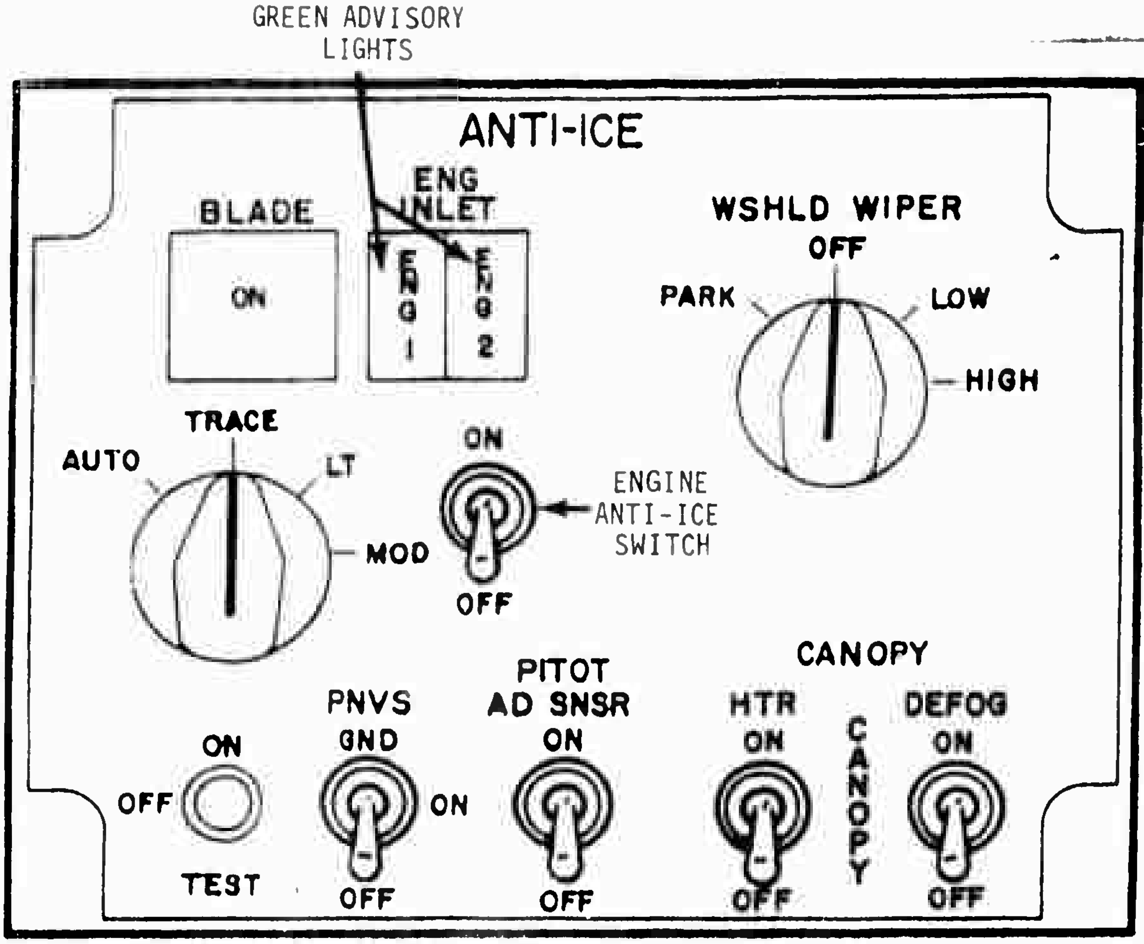
Enclosure I PM 4-9-84

DEICE FDLS

The FCC Software tests the status of the 14T2 - 13 message (Bits 15 - 18) and the 4T1 - 29 message (Bits 13 - 15) in both Continuous and Maintenance FDLS. Any error will result in a "NO-GO" for the individual failure.

Continuous FDLS tests the 14T2 14 - 18 Detector Fail message for a failure but does not check the 14T2 - 14 - 17 Sensor Fail/Controller Fail message.

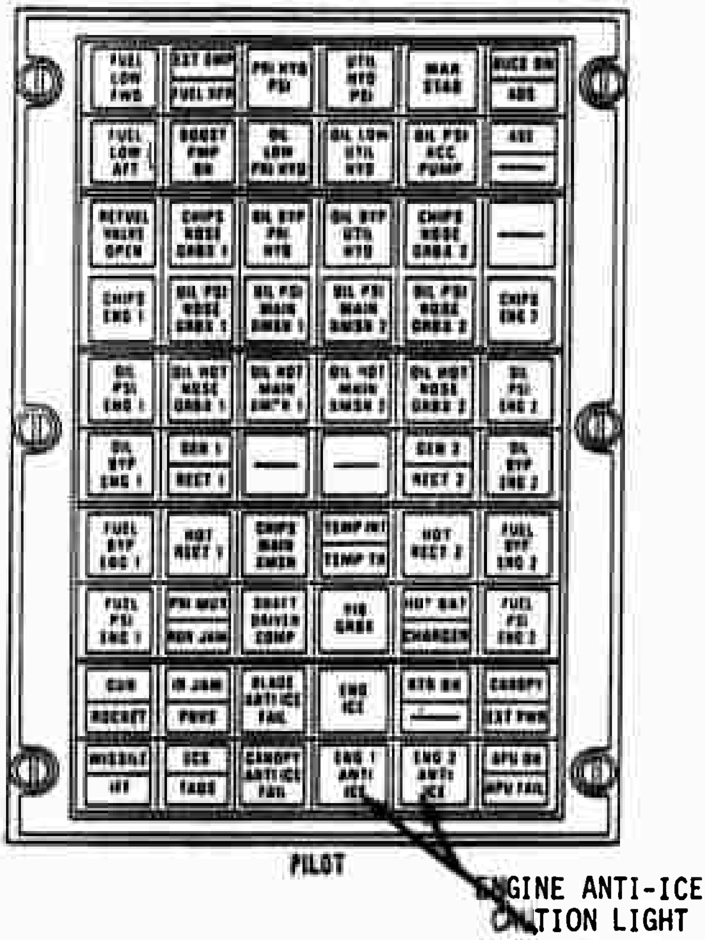
When the FCC detects an error it displays an error message and then exits the DEICE test. Any further DEICE test beyond the fail point will be bypassed.

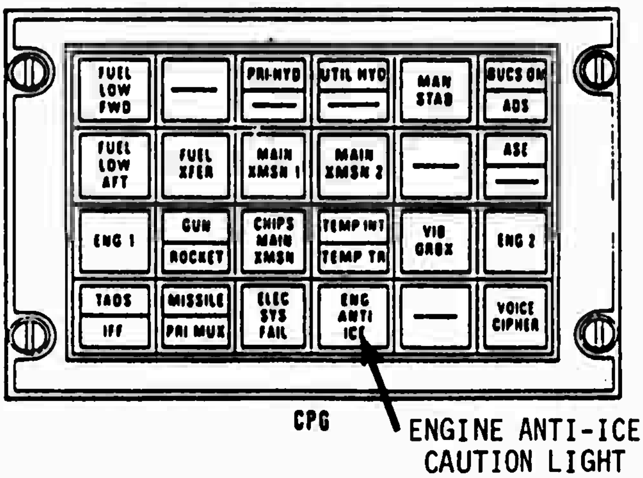


PILOT ANTI-ICE PANEL

Enclosure 2

Attachment 4 of EPR No. 84-23-16





Pilot and CPG Caution / Warning Panels

Enclosure 3

29 March 1985

When icing system is turned on, three things happen.

a. Anti-ice valve opens allowing engine hot air to be routed to nose gear box fairing.

b. Engine anti-ice thermal switch starts heating.

c. Nose gear box heaters are turned on (FULL). The ENG 1 & 2 Anti-ice Lite will come on until ENG Inlet Lite (Green) turns on.*

When Thermal switch reaches 155° F, switch contacts will close completing path to energize Anti-ice Fail Relays K 3-8 and K 1-8. If for any reason ENG Anti-ice valve does not open or Thermal switch did not close, the ENG Anti-ice Lite (CW) will come on.

When NGB Temperagure reaches 205° to 235°, the ENG Inlet Lite (Green) will come on. Power to heaters is reduced to approximately %. When aircraft goes into an icing condition, sensors of NGB Temperture may go below 205° F. IF this happens, the ENG Inlet Lite Green will go out and ENG Anti-ice Lite (CW) will come on.*

NGB Heater PWR will be increased and ENG Inlet Lite Green may or may not come on, depending on temperature.

*This is an undesirable condition and will soon be corrected by 7-332165033. The corrected method will be if the ENG Inlet Lite goes out because of Reduced temperature and not a failure. The (CW) ENG Anti-ice Fail Lite will not come on.

Enclosure 4

Attachment 6 of EPR No. 84-23-16

EQUIPMENT PERFORMANCE REPORT
(DARCOM/AMCR 700-38)

DATE: 10 April 1985

OFFICE SYMBOL: SAVTE-TA

TO:	FROM:
Commander US Army Aviation Systems Command ATTN: AMCPM-AAH-SE 4300 Goodfellow Blvd, St. Louis, MO 63120	Commander US Army Aviation Flight Activity ATTN: SAVTE-TA Edwards AFB, CA 93523-5000

1. EPR NO.:	2. TECOM/AVSCOM PROJ NO.:	3. TEST TITLE
84-23-17	USAAEFA Project #84-23	AH-64A Icing Testing

I MAJOR ITEM DATA

4. MODEL AH-64A	5. SERIAL NO.: 82-23356
6. QUANTITY: 1	7. LIFE PERIOD:
8. MFR: HUGHES	9. USA NO.:

II PART DATA

10. NOMENCLATURE/DESCRIPTION: AIRCRAFT ANTI-ICE DE-ICE SYSTEMS MAINTENANCE

11. FSN:	12. MFR PART NO.:
13. DRAWING NO.:	14. MFR: COX
15. QUANTITY:	16. NEXT ASSEMBLY:
17. MAC FUNCTIONAL GRP:	18. PART TEST LIFE:

III INCIDENT DATA

19. DATE OF OCCURRENCE:

22. MAINT SPT, ELM, CODE:

23. OBSERVED DURING:		24. TEST ENVIRONMENT:	20. TYPE OF REPORT:		21. ACTION TAKEN:	
				a. INCIDENT		a. REPLACED
			X	b. INFORMATION		b. REPAIRED
	a. OPERATION		25. INCIDENT CLASSIFICATION:			c. ADJUSTED
X	b. MAINTENANCE			a. CRITICAL		d. DISCONNECTED
	c. INSPECTION			b. MAJOR		e. REMOVED
	d. OTHER		X	c. MINOR	X	f. NONE

IV INCIDENT DESCRIPTION

26. DESCRIBE INCIDENT FULLY (INCLUDE IMPACT OF INCIDENT ON MAC CODE IDENTIFIED IN BLOCK 22):

When the Eng #1 or Eng #2 anti-ice fail lights (master caution panel) illuminate one of three conditions have occurred: (1) Eng. anti-ice thermal switch failed or is sensing less than 155° C. Three electrical signals which identify failed components in the cox eng. anti-ice system are available for maintenance troubleshooting but at present, are not connected to the FD/LS system (necessary wiring and software are not installed in the aircraft). These three electrical signals are (1) Nose gear box and cross shaft fairing heater blanket controller failure. (2) Nose gear box fairing heater failure (3) AC power failure. Failure to utilize these three electrical signals significantly increases time required to troubleshoot system components. In most cases, the only reliable method of repairing a failed engine anti-ice system is to systematically replace components until the system functions properly. TM 55-1520-238-T describes a number of troubleshooting procedures which are not consistent with system operation (Eng. anti-ice and blade de-ice systems). The number of errors or omissions are too numerous to list. Recommend a complete review of maintenance procedures for these systems.

INCIDENT CLASSIFICATION IS SUBJECT TO RECLASSIFICATION

27. DEFECTIVE MATERIAL SENT TO:

28. NAME, TITLE & TEL EXT OF PREPARER:	29. FOR THE COMMANDER:
JAMES M. ADKINS CW4, USA (805) 277-4786 SIGNED	MICHAEL A. GULICK CPT, AV, C, Plans & Programs (805) 277-4643 SIGNED

DARCOM FORM 1 SEP 76 2134 *Previous edition may be used until exhausted.*

Photo 1. Ice Detector - #1 Engine Inlet

Photo 2. Residual Ice Accumulation on Unprotected Main Rotor Head Components

Conditions: Environment - Artificial | Avg OAT - -9°C
Configuration - Hellfire and Rockets | Avg LWC - 1.19 gm/m^3
Flight - 2 | Time in Cloud - 30 minutes

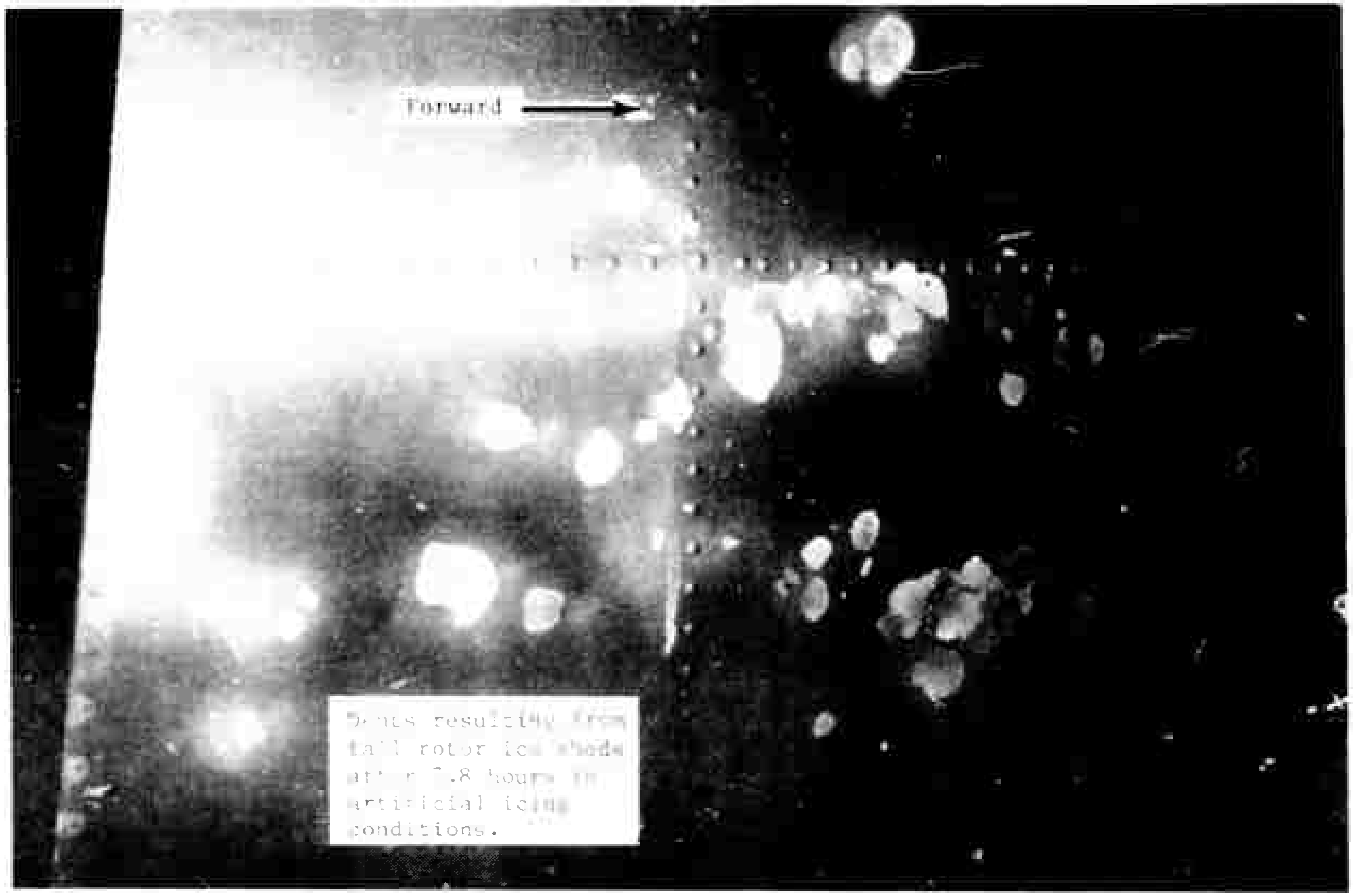


Photo 3. Left Stabilator - Top View

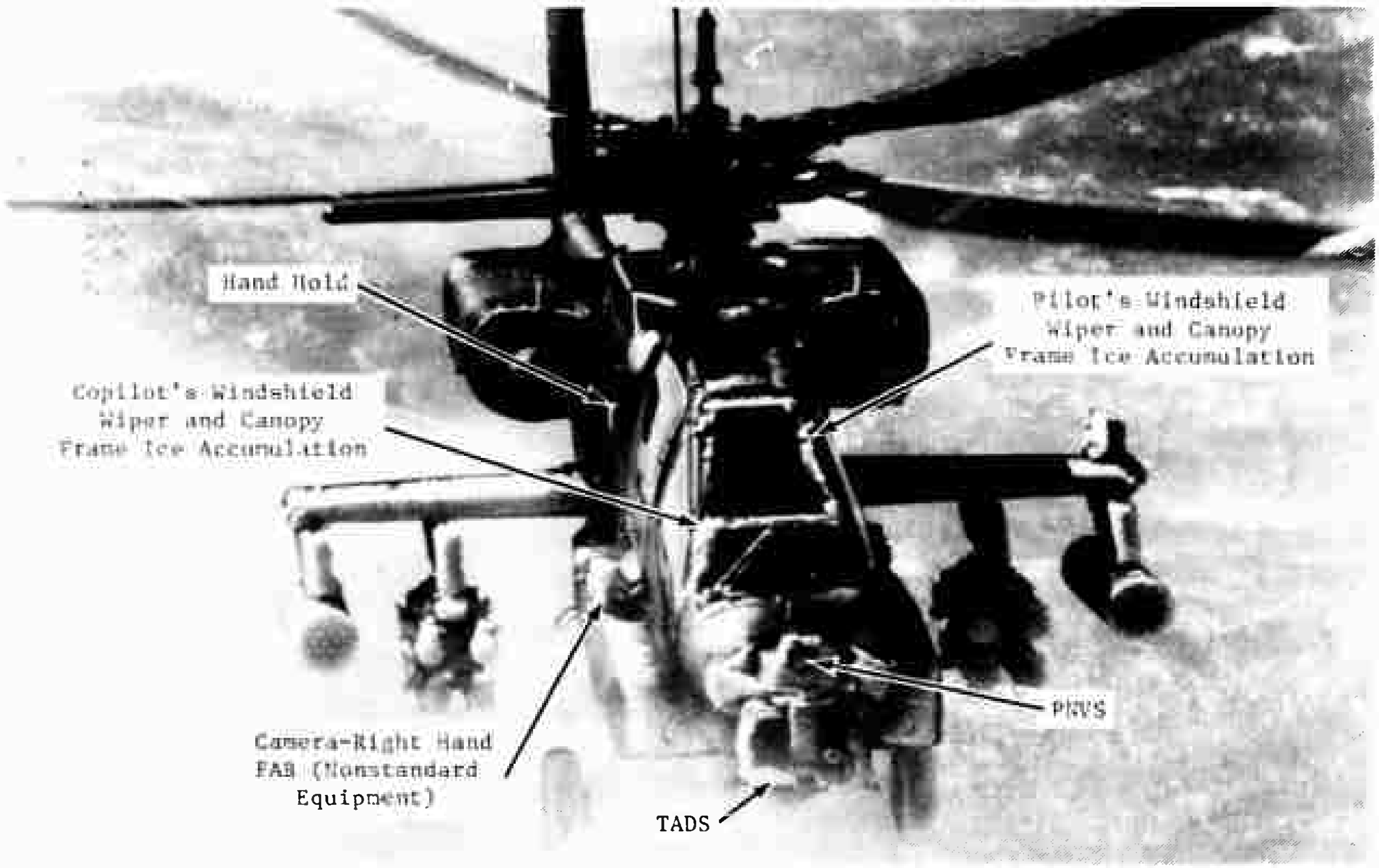


Photo 4. Typical Fuselage Ice Accretions

Conditions:	Environment - Artificial	Avg OAT - $-9.5°C$
	Configuration - Hellfire and Rockets	Avg LWC - $1.02\ gm/m^3$
	Flight - 4	Time in Cloud - 60 minutes

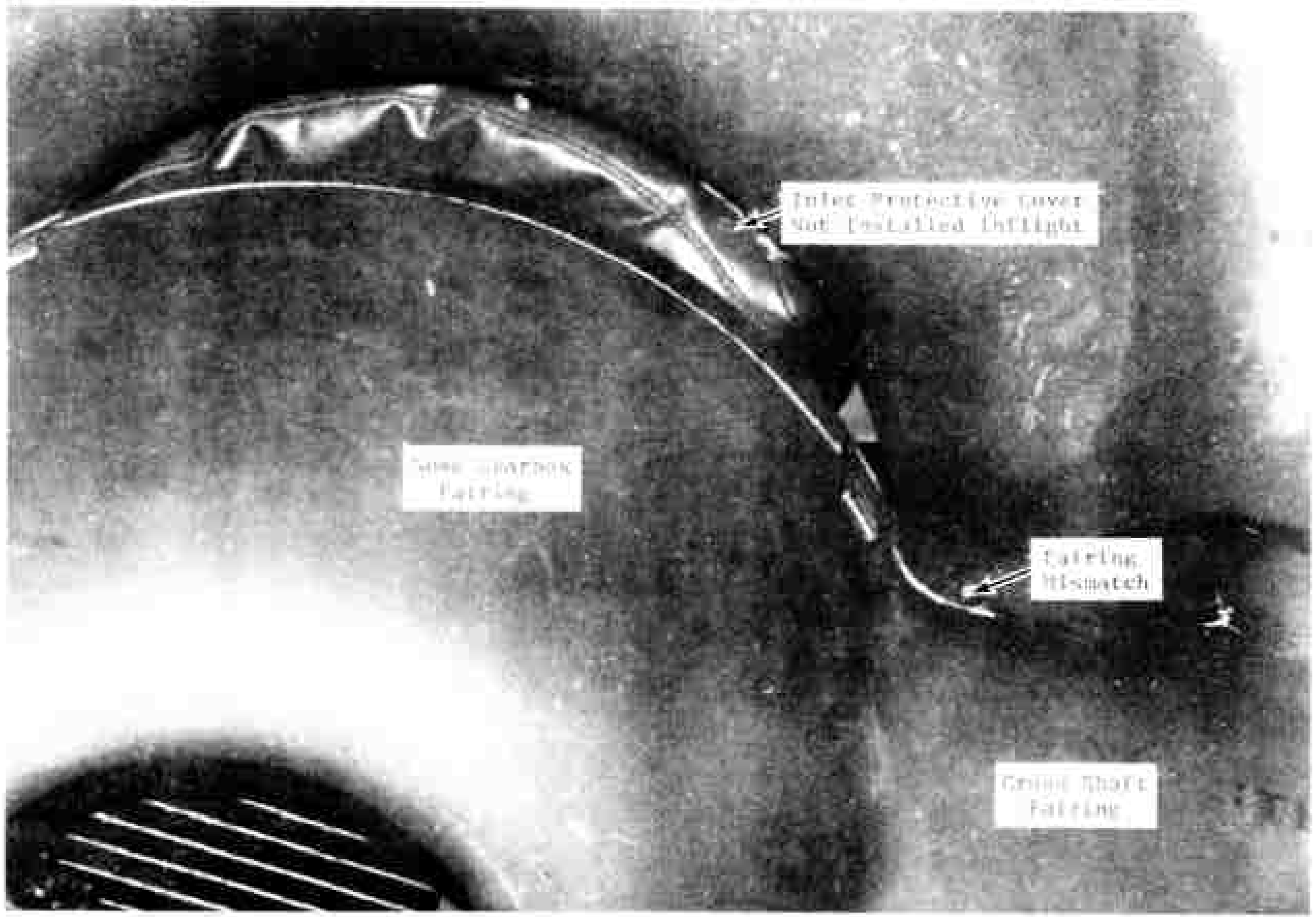


Photo 5. #2 Engine Inlet

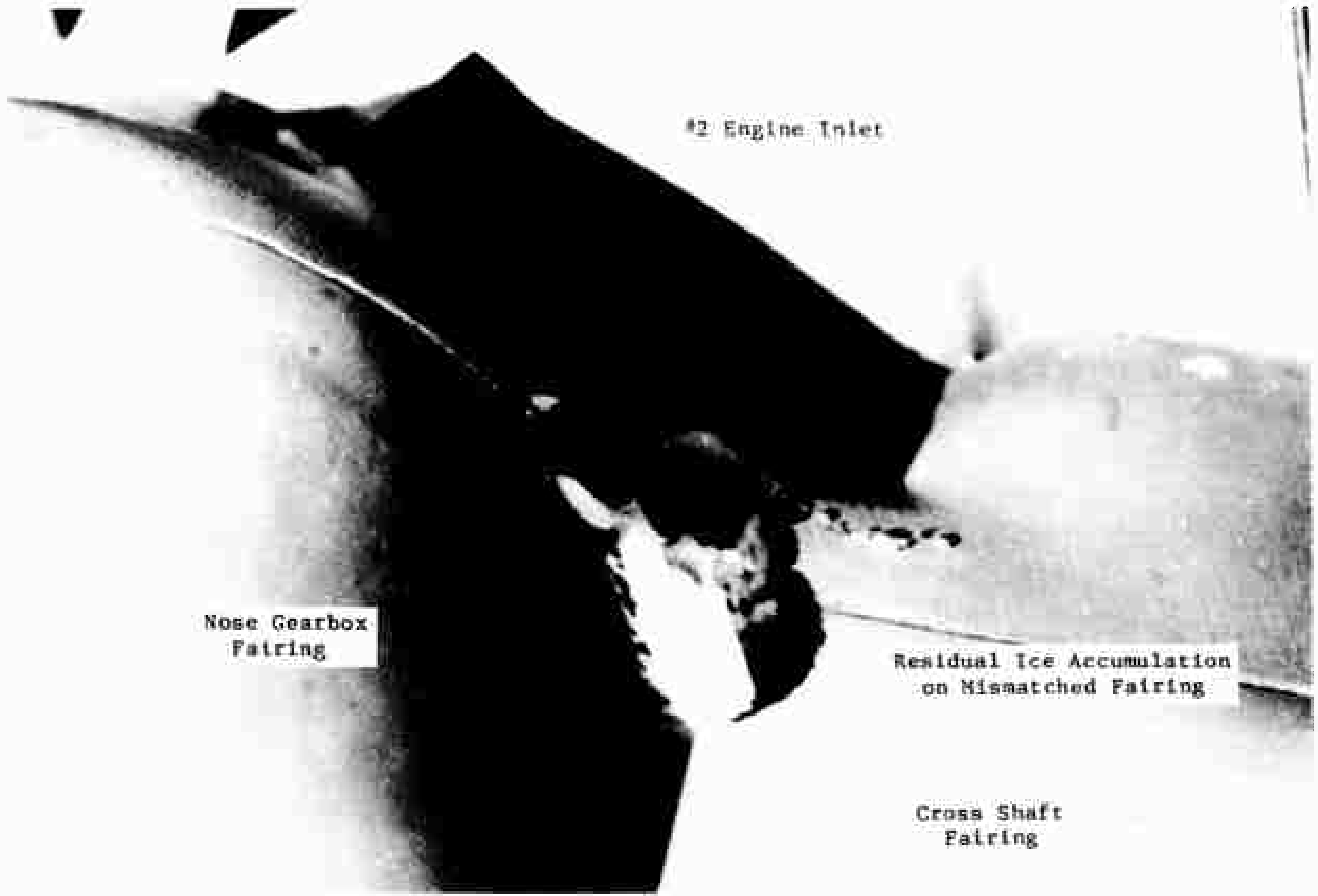


Photo 6. Residual Ice Accumulation on #2 Engine Nose Gearbox/Cross Shaft Fairing

Conditions: Environment - Artificial	Avg OAT - $-14.5°C$
Configuration - Hellfire and Rockets	Avg LWC - $1.02\ gm/m^3$
Flight - 13	Time in Cloud - 60 minutes

Photo 7. Ice Accumulation on Left Side Wing Stores - Pilot's Station Looking Aft

Conditions: Environment - Artificial
Configuration - Hellfire and Rockets
Flight - 4

Avg OAT - -9.5°C
Avg LWC - 1.21 gm/m^3
Time in Cloud - 60 minutes

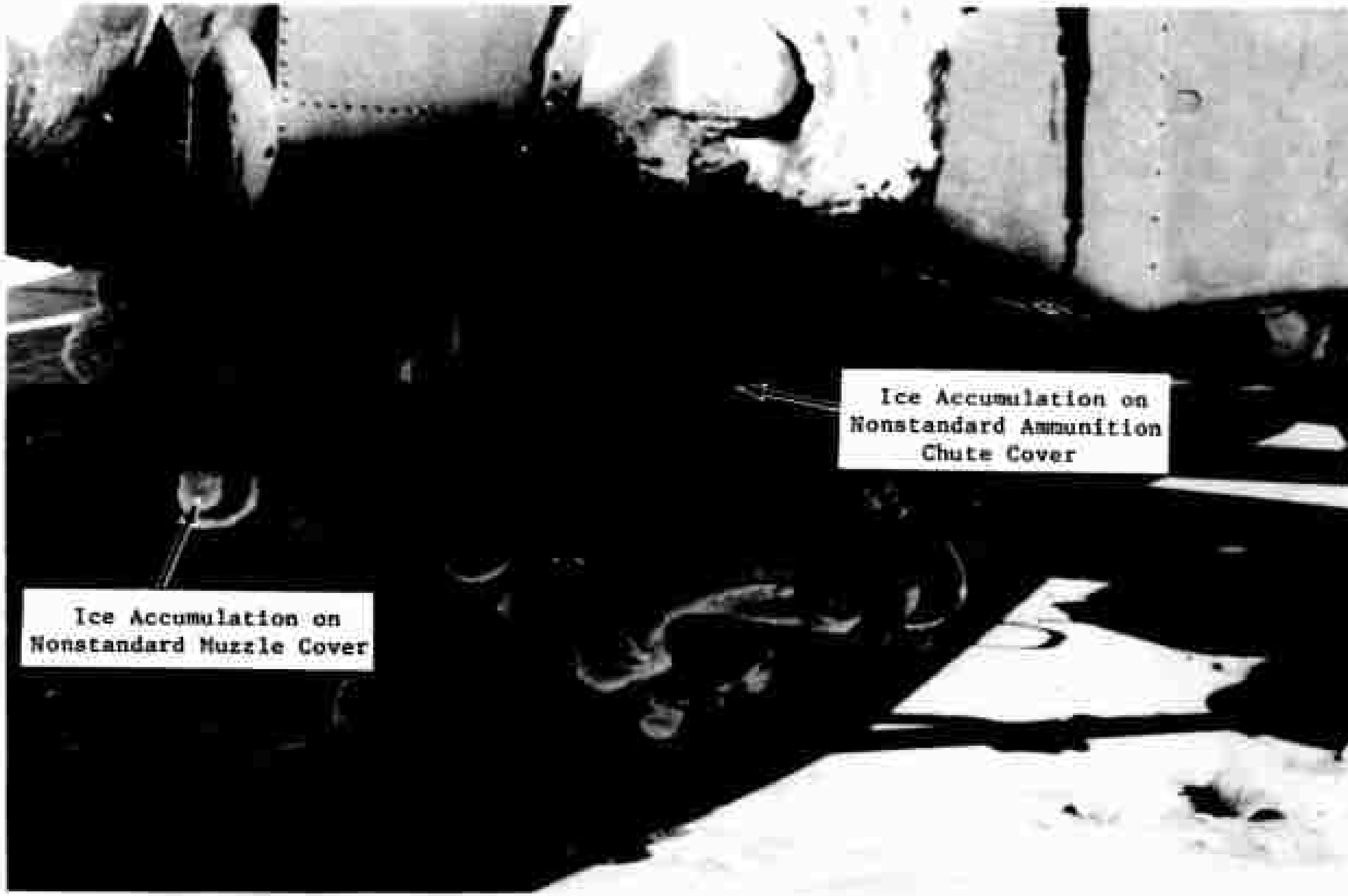


Photo 8. Residual Ice Accumulation on 30mm Area Weapon System

Condition: Environment - Artificial
Configuration - Hellfire and Rockets
Flight - 4

Avg OAT - -9.5°C
Avg LWC - 1.21 gm/m^3
Time in Cloud - 60 minutes

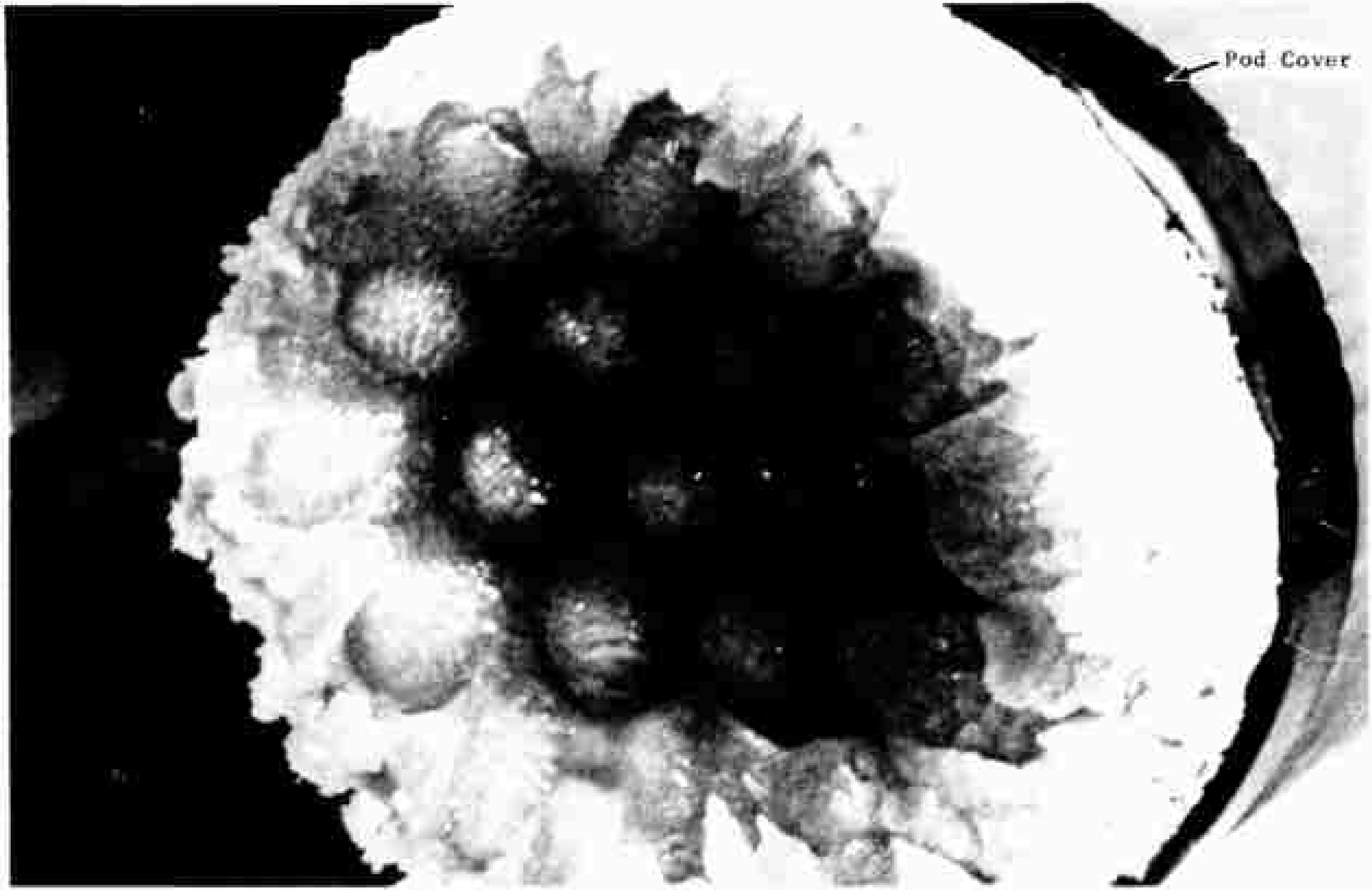


Photo 9. Residual Ice Accumulation on 2.75 inch Folding Fin Aerial Rocket Pod With Cover Installed

Conditions:	Environment - Artificial	Avg OAT - -9.5°C
	Configuration - Hellfire and Rockets	Avg LWC - 1.21 gm/m^3
	Flight - 4	Time in Cloud - 60 minutes

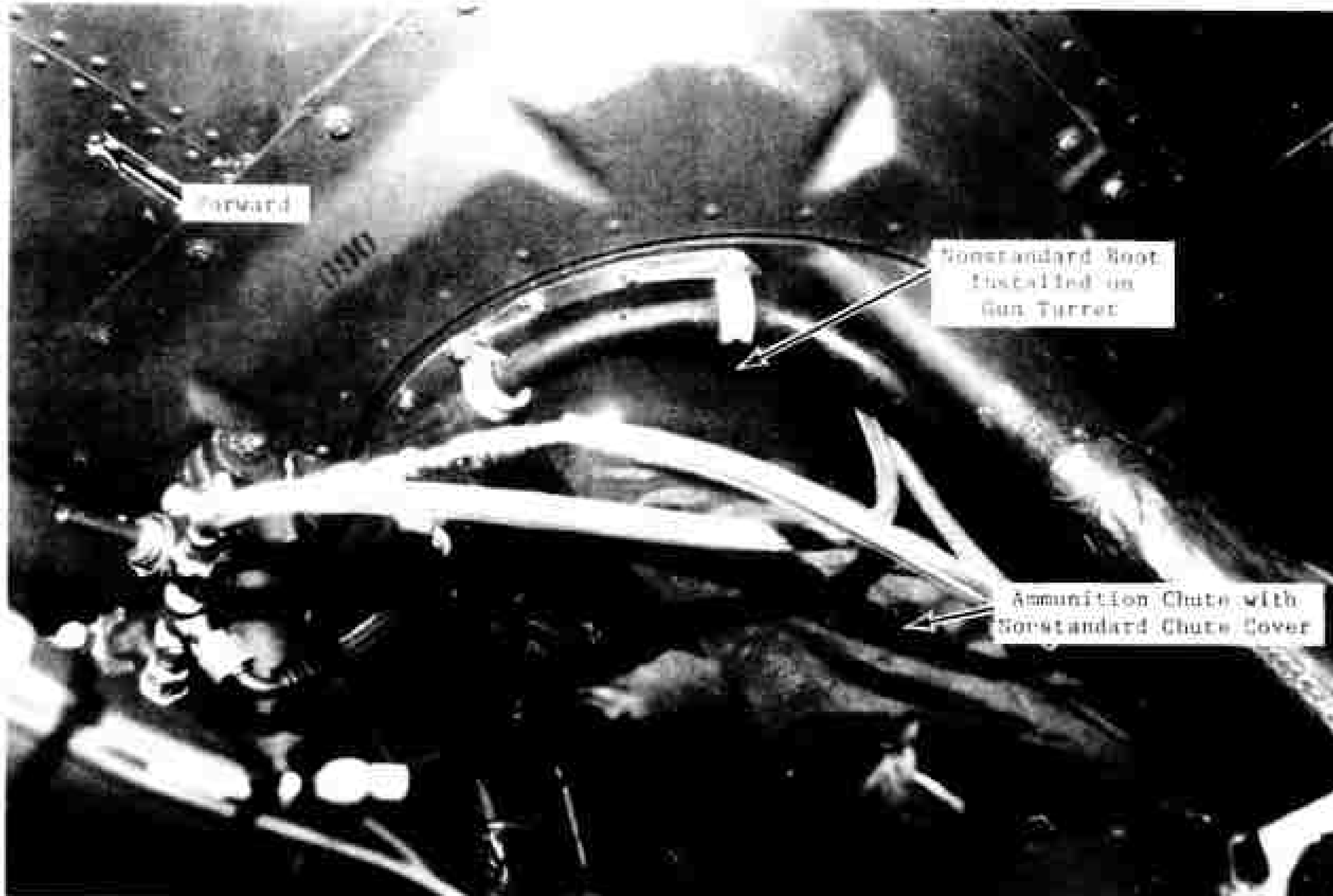


Photo 10. Aircraft Underside View - 30mm Chain Gun Turret

APPENDIX G. REPORT, *ARTIFICIAL AND NATURAL ICING TESTS OF THE AH-64A, USAAEFA PROJECT NO. 84-23 (PHASE II)*

DEPARTMENT OF THE ARMY
U.S. ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

REPLY TO
ATTENTION OF

JUL 2 1986

SAVTE-TA

SUBJECT: Report, Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23 (Phase II)

Commander
US Army Aviation Systems
Command
ATTN: AMSAV-ED
4300 Goodfellow Blvd.
St. Louis, MO 63120-1798

1. REFERENCES. a. Systems Specification, Hughes Helicopters, DRC-S-H10000B, 15 April 1982, with change No. 96, 4 June 1986.

b. Letter, USAAEFA, DAVTE-TA, 3 September 1982, subject: Letter of Effort, YAH-64 Icing Survey, USAAEFA Project No. 80-08.

c. Letter, AVSCOM, AMSAV-ED, 10 September 1984, subject: Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23. (Test Request)

d. Test Plan, USAAEFA Project No. 84-23, *Artificial and Natural Icing Tests of the AH-64A*, October 1984.

e. Test Plan, Hughes Helicopters, Report No. 77-FT-8005P-3, *Test Plan for Artificial and Natural Icing Tests of Production AH-64A Advanced Attack Helicopter*, January 1985.

f. Letter, USAAEFA, SAVTE-TA, 17 September 1985, subject: Report, Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23

g. Letter, AVSCOM, AMSAV-E, 16 January 1986, subject: Report, Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23.

h. Letter, AVSCOM, AMSAV-ED, 17 July 1985, subject: Artificial and Natural Icing Tests of the AH-64A, USAAEFA Project No. 84-23. (Phase II Test Request)

i. Technical Manual, TM55-1520-238-10, *Operator's Manual for Army AH-64A Helicopter*, 28 June 1984, with change 1, 15 October 1984.

j. Letter, AVSCOM, AMSAV-E, 25 February 1985, subject: Airworthiness Release for Artificial and Natural Icing Test of the AH-64A Helicopter, USA S/N 82-23356.

2. INTRODUCTION. a. Background. (1) The US Army requires the AH-64A helicopter to operate safely in an icing environment up to and including the moderate level of intensity (ref 1a). The AH-64A incorporates a rotor blade deicing system, anti-icing systems, and ice detection systems. Qualification tests are required to substantiate airworthiness for operation in a moderate icing environment.

(2) A preliminary investigation of the YAH-64 helicopter capability to operate in icing conditions was conducted during the 1981-82 icing season (ref 1b). Manual operation of the YAH-64 main and tail rotor deicing systems was satisfactory. The automatic function was inadequate. Since that evaluation, the AH-64A rotor deicing system has been redesigned. The US Army Aviation Systems Command (AVSCOM) tasked the US Army Aviation Engineering Flight Activity (USAAEFA) to conduct artificial and natural icing tests of the AH-64A during the winter of 1984-1985 (ref 1c), in accordance with the approved test plans (refs 1d and 1e). Results of this test (Phase I) are presented in reference 1f. The AVSCOM Directorate for Engineering position on these test results is presented in reference 1g. Failure of the ice detection/rate measuring system to accurately indicate the rate of ice accretion resulted in AVSCOM tasking USAAEFA to conduct further natural icing tests (Phase II) of the AH-64A during the winter of 1985-1986 (ref 1h).

b. Test Objectives. The objectives of this test were to conduct artificial and natural icing flight tests of the AH-64A helicopter to:

(1) Determine the effectiveness of the AH-64A ice protection and detection/rate measuring systems.

(2) Determine the impact of ice accumulation on performance and handling qualities.

(3) Determine the capability of the AH-64A to operate in moderate icing conditions.

c. Description. (1) The AH-64A is a two-place, tandem seat, twin engine helicopter with four-bladed main and anti-torque rotors and conventional wheel landing gear. The helicopter is manufactured by McDonnell Douglas Helicopter Company (MDHC) (previously called Hughes Helicopter Incorporated) and is powered by two General Electric T700-GE-701 turboshaft engines. The AH-64A has a movable horizontal stabilator. A 30mm gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has a wing with two pylons on each side for carrying Hellfire missiles, 2.75-inch folding fin aerial rockets and/or external fuel tanks. The test helicopter was USA serial number (S/N) 83-23811 (production vehicle 36). Further description of the helicopter may be found in the system specification (ref 1a), the operator's manual (ref 1i), and enclosure 1.

(2) The AH-64 anti-ice/deice systems are depicted in figure 2, enclosure 1. The deice system consisted of an outside air temperature (OAT) sensor, ice detector probe and signal processor unit, icing rate meter, blade deice control panel, slip rings for the main and tail rotor, and a deice controller. The main and tail rotor blades contained electrothermal resistive heating mats. Anti-ice systems were provided for sections of the windshield, pitot tubes, air data sensor, engines, engine inlets, nose gearbox/cross shaft fairings, and target acquisition designation system/pilot night vision system (TADS/PNVS). The frangible HELLFIRE deice system was not installed on the test aircraft since satisfactory system performance was previously demonstrated (ref 1b).

d. Test Scope. A previous phase of this program completed 8.3 hours of artificial and 6.5 hours of natural icing tests (ref 1f). The present evaluation included clear air and natural icing flight tests conducted in the vicinity of Duluth, Minnesota from 12 November through 20 December 1985 and 10 February through 11 April 1986. A joint contractor and USAAEFA test team was used for these tests. A total of 23 flights were conducted resulting in 17.4 productive hours. Of these flights, 3 were in clear air totaling 1.9 hours and 20 were in natural icing environment totaling 15.5 hours of cloud immersion. Specific icing conditions are presented in table 1, enclosure 4. The aircraft was flown in two different stores configurations. The first configuration consisted of 8 Hellfire (4 Hellfire dummy missiles on each inboard pylon) with 2.75-inch rocket pods outboard. The second configuration had external fuel tanks inboard with rocket pods outboard. Anti-ice and deice systems were operated continuously while in the icing environment. During 14 flights, the deice system operated in the automatic mode. During six flights totaling 3.6 hours in natural icing conditions, heater OFF times for the deice system were manually selected. Flight limitations contained in the operator's manual and the airworthiness release (ref 1j) were observed during testing.

e. Test Methodology. (1) Artificial icing using the Helicopter Icing Spray System was completed during Phase I testing in February and March 1985 (ref 1f). Results of the artificial icing tests provide an essential supplement to the natural icing data in determining the capability of the AH-64A to operate in moderate icing conditions. A detailed discussion of the test sequence, procedures, and results is contained in references 1d, 1e, and 1f.

(2) Natural icing tests were conducted during both phases of the evalution by flying in instrument meteorological conditions using instrument flight rules (IFR). The JU-21A chase aircraft configured with a cloud particle measuring system assisted in locating and documenting the icing conditions. Photos were taken from the JU-21A after the test aircraft exited the icing environment into visual meteorological conditions. Close coordination between air traffic control and the chase and test aircraft crews was required to find and stay in the icing environment and to implement in-flight aircraft rendezvous

for photographic documentation. Procedures included radar vectoring, navigational aid holding, and block airspace assignment. Time in the clouds was limited by the availability of the natural icing conditions and aircraft IFR fuel requirements. Video cameras were mounted on the test aircraft to document main rotor, tail rotor, engine inlet and nose gearbox/cross shaft fairing ice accretion and shedding characteristics both during and after cloud immersion.

(3) Test data were recorded on magnetic tape in frequency modulation format. A detailed description of special equipment and instrumentation is provided in enclosure 2.

(4) Test techniques and data analysis methods are presented in enclosure 3. These include methods used to determine cloud parameters and definitions of icing types and severities. Qualitative vibration assessment ratings were assigned in accordance with the Vibration Rating Scale (VRS).

3. RESULTS AND DISCUSSION. a. General. Natural icing tests were conducted during the winter of 1985-1986 as a continuation of the artificial and limited natural icing tests reported in reference 1f to establish a moderate icing envelope (through 1.0 gm/m^3 liquid water content (LWC)) for the AH-64A Apache helicopter. A joint contractor (MDHC) and USAAEFA test team performed the evaluation with MDHC having primary responsibility for the development phase of testing and the government having primary responsibility during the qualification effort. The government evaluation could not be completed due to the ice detection/rate measuring system (LWC probe and signal processor unit) development status, nonavailability of natural icing conditions, and grounding of the AH-64A fleet. A summary of the test conditions for each flight is presented in table 1, enclosure 4. The specific icing conditions in which the AH-64A was tested are presented in figure 1, enclosure 4. During these tests, the unaspirated LWC probe in the engine inlet was replaced with an aspirated LWC probe mounted on the upper fuselage. Ice accretions found in the engine inlet required the incorporation of heater blankets in the aft nose gearbox/cross shaft fairing. Since only 4 of the 26 natural icing flights encountered values of LWC greater than 0.3 gm/m^3, the effect of more severe conditions on various systems could not be thoroughly assessed. The AH-64A should be further evaluated in natural icing conditions at higher LWC (>0.3 gm/m^3) with outside air temperatures (OAT) between 0°C and -20°C to verify:

(1) Accuracy of the ice detection/rate measuring system.

(2) Main rotor ice shedding characteristics when the automatic rotor deice system is operating at the correct design blade heater on- and off-times.

(3) Satisfactory performance of the modified nose gearbox/cross shaft fairing heater blankets.

The cold aft cockpit environment during flight at approximately -20°C without the nonstandard boot installed on the 30mm turret (ref 1f) remains a deficiency. Additional deficiencies identified are:

(1) The high potential for engine foreign object damage (FOD) due to shed ice from the fore and aft nose gearbox/cross shaft fairing mismatch.

(2) Blade heater off-times that are inconsistent with actual icing conditions.

(3) Inaccurate indications on the pilot's icing severity meter.

(4) Rotor blade heater on-times which are inconsistent with the design schedule and actual OAT.

Three shortcomings were identified including two shortcomings previously identified in reference 1f.

b. Deice System Operation. (1) General. The AH-64A helicopter rotor deice system was previously evaluated in February through April 1985 (Phase I) for operational characteristics and effectiveness during 8.3 hours in the artificial icing environment and 6.5 hours in natural icing (ref 1f). Twenty flights totaling 15.5 hours in natural icing were conducted during this test. Specific test conditions for this test are shown in table 1, enclosure 4. Test conditions for the total 22.0 hours that the AH-64A has spent in natural icing (Phases I and II) are shown in figure 1, enclosure 4. Only four flights have been performed in stable LWC's above 0.3 gm/m^3. These four flights were conducted in ambient temperatures of -7°C to -9°C. Further natural icing tests should be conducted at higher LWC (>0.3 gm/m^3) and over a broader range of temperatures (0°C to -20°C).

(2) Rotor Blade Heater On-Time. The AEG-Telefunken deice controller determined blade heater element on-time as a function of OAT. Several modifications were required during previous icing tests (ref 1f) to adjust the on-time schedule and integrate the OAT signal for automatic system operation. The deice controller provided a blade heater on-time consistent with the OAT input signal and the design schedule during testing in November and December 1985. A modified deice controller which allowed the blade heaters to activate for a minimum of 2.25 seconds at temperatures colder than $+4.7 \pm 1$°C, was installed prior to testing in February 1986. Following this modification, on-times were as much as 2.0 seconds shorter than the design schedule for a given sensed OAT (fig. 2, encl 4). The main rotor blades did not appear to completely shed accumulated ice during the heater cycles as indicated by residual torque increases and vibration levels. During one flight in natural icing conditions at an OAT of -7.5°C, LWC of 0.15 gm/m^3 and with heater off-time manually selected, the short heater on-time resulted in indicated torque

required increases of up to 17% (collective fixed) and a noticeable increase in airframe vibrations (VRS 6). Maximum indicated torque increases of 8 to 10% were observed on previous flights under similar conditions when heater off-time was manually selected and the heater on-time was consistent with the design schedule and OAT. The cause of the short blade heater on-times following modification of the deice controller has not been determined. Rotor blade heater on-times which are inconsistent with the design schedule and actual OAT are a deficiency. Further natural icing tests should be conducted to verify correct blade heater on-time.

(3) Blade Heater Off-Time. The AEG-Telefunken deice controller determined rotor blade heater off-times as a function of LWC. Heater off-time controls the amount of ice accreted on the rotor blades between heating cycles. This in turn affects power required increases, vibration levels and potential aircraft damage. The controller provided a blade heater off-time consistent with the design schedule (fig. 3, encl 4) and input LWC signal. However, the ice detector/rate measuring system frequently provided an erroneous input signal (erratic and/or higher than actual LWC) resulting in short off-times (fig. 4, encl 4). Inconsistent performance of the ice detection/rate measuring system (para 3b(4)) resulted in the rotor deice system operating on the correct blade heater off-times for only 7.3 hours of the total 22.0 hours in natural icing conditions. Of these 7.3 hours, 3.6 hours were with manually selected off-times and only one flight (reported in ref 1f) was performed in LWC > 0.21 gm/m^3. Blade heater off-times affect the main rotor shedding characteristics described in paragraph 3b(5). Blade heater off-times that are inconsistent with actual icing conditions are a deficiency.

(4) Ice Detection/Rate Measuring and Indicating System. (a) The initial ice detection/rate measuring system consisted of an unaspirated Rosemount LWC probe (model 871FG1) in the left engine inlet, a signal processor unit, and an icing rate meter located on the pilot's instrument panel. To verify the ship system LWC readings, an aspirated Rosemount LWC probe (model 871FF2) was mounted on the upper fuselage of the test aircraft (photo 1, encl 5) with an ice rate meter/signal processor unit in the copilot/gunner (CPG) cockpit. Additionally, the JU-21A chase aircraft was used to document actual cloud conditions and further verify ship system LWC indications. Natural icing tests in stratiform clouds indicated that the ship system LWC probe and/or signal processor were providing an erroneous LWC signal (erratic and significantly higher than actual conditions) resulting in short blade heater off-times and high readings on the pilot's icing severity meter. To correct this problem the Rosemount signal processor unit was modified to incorporate the aspirated LWC probe located on the upper fuselage (photo 1, encl 5) in place of the engine inlet LWC probe. Correct system operation (LWC indications and heater off-time) was demonstrated during one natural icing flight (test 13). The aspirated probe was then moved to the proposed production location shown in photo 2, enclosure 5 and correct system operation was demonstrated during the next natural icing flight

(test 14). During the next series of natural icing flights (tests 16 through 29), the ice detection/rate measuring system provided erratic LWC readings believed to be the result of flow interference due to probe location. The probe was moved back to its original location on the upper fuselage and four test flights (tests 30 through 33) were conducted. The LWC readings were noticeably more consistent but indicated higher than the actual conditions. The aspirated LWC probe was moved to the far right side of the upper fuselage (photo 3, encl 5) to minimize possible flow interference. During the last test flight (test 34) the LWC readings appeared constant but higher than actual conditions. The grounding of the AH-64A fleet on 12 March 1986 prevented further testing. Extensive trouble-shooting and system analysis has failed to provide a satisfactory explanation for the ice detection/rate measuring system's inconsistent performance. The AH-64A should be further evaluated in more severe natural icing conditions to verify the accuracy of the ice detection/rate measuring system.

(b) An icing severity meter on the pilot's instrument panel is driven by the ice detection/rate measuring system signal processor unit. The erroneous LWC signal discussed in paragraph 3b(4)(a) results in high indications on the icing severity meter. Accurate icing severity indications are essential to allow the pilot to determine when icing conditions exceed the aircraft anti-ice/deice system limitations. High icing severity readings may create a hazardous situation when the pilot employs emergency procedures to depart what he believes to be severe icing conditions. Inaccurate indications on the pilot's icing severity meter are a deficiency.

(5) Main Rotor Ice Shedding Characteristics. Main rotor shedding characteristics were evaluated to determine potential aircraft damage due to shed ice, increases in power required between blade heater cycles, and changes in airframe vibration levels. Field repairable main rotor blade damage was discovered on post flight inspection following several flights in natural icing conditions. No other difficulties associated with main rotor ice shedding were observed during any test flight. At the lower LWCs and warmer ambient temperatures occasional, but not annoying vibration levels (VRS 5) were noted in the cockpit. The AH-64A main rotor ice shedding characteristics should be further evaluated in more severe natural icing conditions with the automatic rotor deice system operating at the correct blade heater on- and off-times.

(6) Tail Rotor. The tail rotor ice shedding characteristics and susceptibility to damage from shed main rotor ice were evaluated during these tests. Residual ice accumulations were noted on unheated portions of the tail rotor blades and hub assembly but no difficulties were associated with these accretions. Ice shed from the tail rotor repeatedly struck the stabilator causing minor dents (see para 3d(3)). Ice impact damage to the tail rotor was previously reported as a shortcoming in reference 1f. No tail rotor blade damage occurred during these tests but operation of the rotor deice system at the correct blade

heater off-times in higher LWC may alter the potential for tail rotor damage due to main rotor ice shedding. Susceptibility of the tail rotor to damage from shed main rotor ice should be further evaluated in more severe natural icing conditions when the automatic rotor deice system is operating at the correct blade heater on- and off-times.

c. Anti-Ice System Operation. (1) General. The AH-64A anti-ice systems were initially evaluated during February through April 1985 for operational characteristics and effectiveness during 8.3 hours of artificial icing and 6.5 hours of natural icing (ref 1f). An additional 15.5 hours of natural icing were performed during this test. Specific test conditions for this test are shown in table 1, enclosure 4. Anti-ice systems on the AH-64A helicopter protect the engines, engine inlet cowlings, nose gearbox and cross shaft fairings, windshields, pitot tubes, air data sensor (ADS), and TADS/PNVS. Ice accretions found in the engine inlet required the addition of heater blankets to the aft nose gearbox/cross shaft fairing. All anti-ice systems were activated prior to entering the icing environment and were operational for all icing flights. A detailed description of each system is presented in reference 1f and significant changes or system modifications are listed in enclosure 1.

(2) Engine. Engine anti-icing was accomplished by a combination of hot air from the axial compressor and heat transfer from the air/oil cooler in the engine frame. The system was controlled by one engine anti-ice switch located on the pilot's lower left subpanel (fig. 1). The engines were visually inspected (including borescope) daily and an engine health indication test was performed prior to each flight. No engine deterioration was noted during this program. There were no indications of ice accumulation in the engine. The engine anti-ice system demonstrated satisfactory operation in the icing environment for the conditions tested.

(3) Engine Inlet Cowling Ring. Engine bleed air heated the engine inlet cowling ring to prevent ice accumulation. The cowling ring anti-ice characteristics were evaluated throughout this test. Video documentation of the No. 2 engine inlet during three flights in natural icing revealed visible moisture running off the heated cowling ring and nose gearbox/cross shaft fairing into the engine inlet. This phenomenon may have contributed to the unacceptable performance of the engine inlet LWC probe (para 3b(4)(a)). There were no accumulations of ice noted on the cowling ring. The engine inlet cowling ring anti-ice system demonstrated satisfactory operation for the conditions tested.

(4) Nose Gearbox and Cross Shaft Fairings. The nose gearbox and cross shaft fairings were anti-iced electrically using heater elements inside the fairing. These electric heaters were activated by the same two-position switch on the pilot's anti-ice panel (fig. 1) that activates the engine air inlet bleed air system. Some ice accumulation was observed on the front of the nose

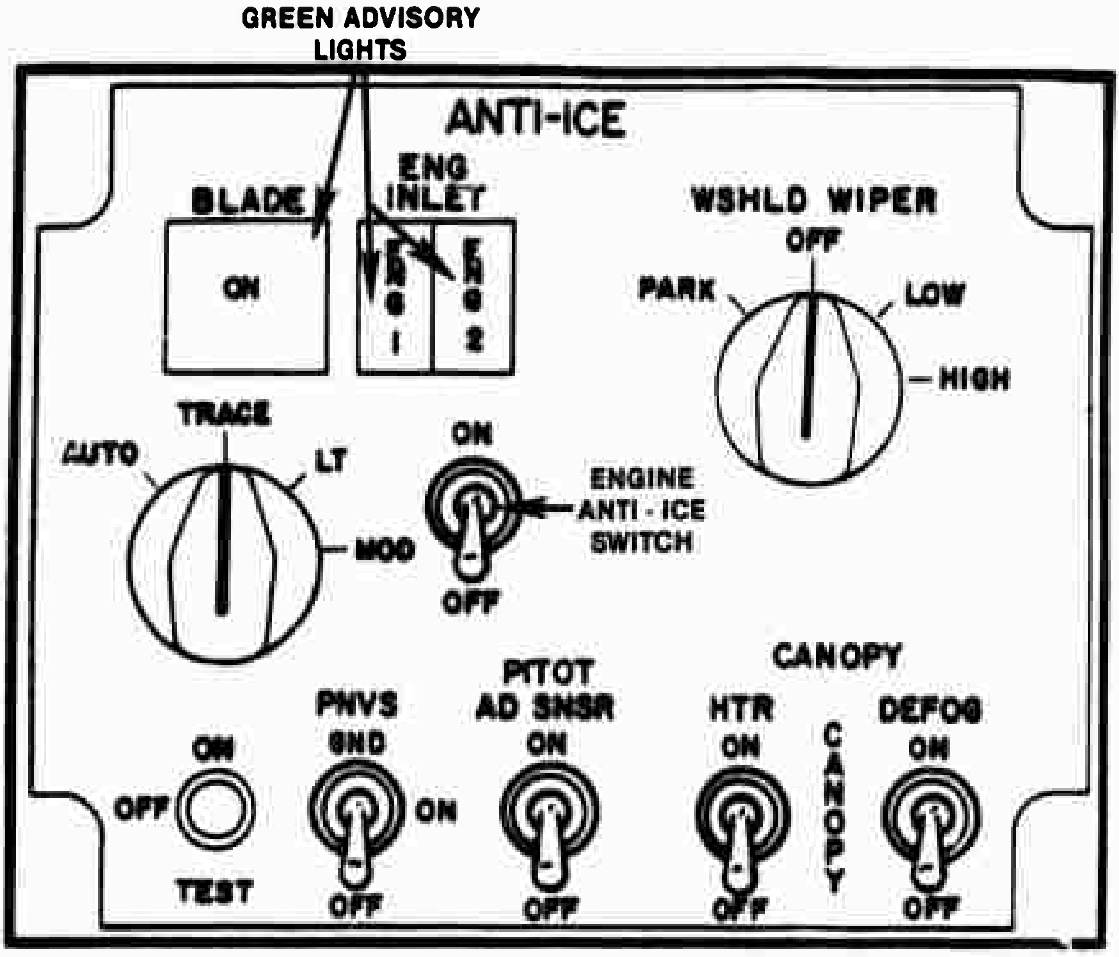


Figure 1. Pilot Anti-Ice Panel

gearbox oil cooler but no adverse effects were noted. No ice accretion was noted on the heated surfaces. Video documentation revealed visible moisture running off the heated fairing/cowling ring causing ice to accrete in the portion of the engine inlet formed by the unheated aft nose gearbox/cross shaft fairing (photo 4, encl 5). This ice accretion presented a substantial engine foreign object damage (FOD) hazard. The addition of heater blankets under a portion of the aft nose gearbox/cross shaft fairing appears to have corrected this problem at the conditions tested (LWC <0.3 gm/m^3). Additionally, the sharp edge created by poorly fitted fore and aft nose gearbox/cross shaft fairings (previously reported in ref 1f) accreted ice and may result in engine ice ingestion damage and is discussed in paragraph 3d(8). Operation of the modified nose gearbox/cross shaft fairing heaters should be further evaluated in more severe natural icing conditions.

(5) Heated Windshields. The two panels forming the pilot and copilot/gunner's (CPG) windshield were anti-iced electrothermally. The windshield temperature was regulated between 65° and 85°F (18° to 29°C) which kept the heated areas ice free throughout these tests. Ice accretion characteristics of the AH-64A heated windshield are satisfactory for the conditions tested.

(6) Pitot Tubes. The dual, wing mounted pitot tubes were anti-iced electrically and controlled by a single switch on the pilot's anti-ice control panel (fig. 1). The pitot heat was activated for all flights in icing conditions. No ice accretion was observed and the system operated without failure. The ice accretion characteristics of the pitot tubes are satisfactory for the conditions tested.

(7) Air Data Sensor (ADS). The ADS was located above the main rotor and attached to a standpipe through the main rotor mast. The rotating hub and arms of the ADS were heated. Control of the anti-ice function of the system was accomplished by the same switch as the pitot heat. Ice did not accrete on the heated portions of the ADS. The ADS anti-ice system is satisfactory for the conditions tested.

(8) Target Acquisition and Designation/Pilot Night Vision System (TADS/PNVS). a. The TADS/PNVS anti-ice provisions included window, window frame, and selected turret surface panel heating. During natural icing tests the PNVS was coupled to the pilot's helmet while the TADS turret was fixed forward and only slewed periodically to determine the effects of ice accretions on range of motion. During one flight, the PNVS anti-ice system failed approximately 30 minutes after entering natural icing conditions. The failure was believed to coincide with movement of the pilot's function switch from the NVS position to STBY and back to NVS. Post flight inspection revealed no apparent reason for this failure. Several changes were made to the TADS/PNVS anti-ice system during the evaluation in an attempt to improve performance. The final configuration tested provided adequate anti-icing under the natural icing

conditions tested (LWC < 0.3 gm/m^3). The TADS/PNVS anti-ice system should be further evaluated in more severe natural icing conditions.

(9) A proposed production TADS laser safety shield (Part No. 79923857-1/2/3) was installed on the test aircraft (photo 5, encl 5) on 3 March 1986. No natural icing flights were performed after installation of this shield. Ice accretion and shedding characteristics of the new TADS laser safety shield should be evaluated in natural icing conditions.

d. Unprotected Aircraft Components. (1) General. Ice accretion and shedding characteristics of the unprotected aircraft components were evaluated at the test conditions shown in table 1 and figure 1, enclosure 4. Ice formed on all unprotected stagnation areas and sharp protrusions from the airframe and main rotor. Large ice formations accreted on the canopy frames, windshield wipers, handholds/steps, and nose gearbox/cross shaft fairing mismatches. The accretions create the potential for engine FOD and/or aircraft damage due to shed ice.

(2) Main Rotor. Minimal amounts of ice accumulated on unprotected blade surfaces inboard of the heater mats and on the blade retention mechanisms. No restriction of any movable component on the main rotor head was noted. No aircraft damage resulted from ice shedding characteristics of the unprotected portion of the main rotor and rotating components.

(3) Stabilator. The ice accretion and shedding characteristics of the stabilator were evaluated throughout the program. Minor ice buildups were observed on the stabilator leading edge. No problems were identified due to these accretions. A crack in the stabilator skin (photo 6, encl 5) was discovered after the test aircraft returned to the MDHC facility at Mesa, Arizona. The cause of the crack has not been determined and is under investigation. Further evaluation of the potential for stabilator damage due to shed tail rotor ice in natural icing should be accomplished.

(4) Fuselage. The fuselage ice accretion and shedding characteristics were evaluated during these tests. Typical fuselage ice accretions are shown in photos 7 and 8, enclosure 5. Large accumulations of ice were noted on the wing leading edges, nose, handholds/steps, landing gear, landing gear struts, tail wheel, vertical stabilizer/tail rotor pylon leading edge and many other surface irregularities on the fuselage. No operating difficulties were identified due to these ice accumulations. All windows, doors and access panels remained functional after these icing encounters. Ice accretion and shedding from the fuselage handholds/steps create the potential for engine FOD and aircraft damage and should be further evaluated in natural icing conditions with LWC > 0.3 gm/m^3.

(5) Unprotected Windows. No ice protection was provided for any window panels except the two electrically heated forward facing windows discussed in

paragraph 3c(5). During flight in the icing environment, ice accretion on these unprotected window areas was minimal. No obstruction to the field of view resulted from these minor accretions.

(6) Canopy Frames. The framework supporting the essentially flat plate canopy was not anti-iced. The forward canopy frames accreted large quantities of ice (photo 9, encl 5). The CPG windshield wiper was used for removal of water which pooled on the heated glass window. This water and the accreted ice (particularly around the CPG window) combined to form significant ice accumulations on the canopy frames and windshield wiper arms. These ice formations became large enough to break away during flight and could impinge on the nose gearbox or be ingested by an engine. Ice accretion and shedding from the canopy frames should be further evaluated in a more severe natural icing environment.

(7) Windshield Wipers. The CPG windshield wiper was occasionally used at warmer temperatures (-5°C) during natural icing conditions and demonstrated satisfactory performance. Use of the pilot's windshield wiper was never required in natural icing conditions. Ice formations on both the pilot and CPG windshield wipers (photos 9 and 10, encl 5) became large enough to break away and could impinge on the nose gearbox or be ingested by an engine. Ice accretion and shedding from the windshield wipers should be further evaluated in natural icing conditions with LWC > 0.3 gm/m^3.

(8) Engine Ice Ingestion. The potential for engine ice ingestion was evaluated during this test. A video camera mounted to observe No. 2 engine inlet recorded several instances of the engine ingesting small pieces of ice. No unusual cockpit indications were observed and borescope engine inspections failed to reveal any compressor damage. The most probable areas from which this ice shed are the handhold on the right side of the aircraft (photo 8, encl 5), the canopy frames, windshield wipers, or the camera mount (a non-standard configuration) on the right forward avionics bay (FAB). Ice accretion on the mismatch between the fore and aft nose gearbox/cross shaft fairings was previously documented during artificial icing tests (ref 1f). No permanent modification to correct this problem has yet been implemented. The fairing edges were smoothed during this test (photos 11 and 12, encl 5) to eliminate the mismatch and prevent engine FOD. The high potential for engine FOD due to shed ice from the mismatch between the fore and aft nose gearbox/cross shaft fairings is a deficiency. The susceptibility of the engines to damage from shed ice should continue to be investigated in natural icing conditions with LWC > 0.3 gm/m^3 and OAT between 0°C and -20°C.

(9) Antennas. The ice accretion and shedding characteristics of the aircraft antennas were evaluated throughout these tests. Many AH-64A antennas are flush mounted and thus accrete little if any ice. Exceptions to this are the transponder antenna located on the cabin overhead and the two forward

facing radar warning antennas located on the front of each FAB. Ice shed from these antennas presented no operational difficulties. No degradation in aircraft radio transmission or reception was noted on any communications or navigation radios, although no specific tests were conducted to evaluate these characteristics. A study should be conducted to determine the possible degradation of reception with ice accumulation on the forward facing radar warning antennas.

(10) M-130 Chaff Dispenser. The ice accretion and shedding characteristics of the M-130 chaff dispenser were evaluated throughout these tests. Two dispensers were installed on the test aircraft, one on each side of the tailboom. The M-130 system was not operated during this evaluation. No ice accretions or subsequent sheds were observed which would interfere with the operation of the system during or after an icing encounter.

(11) ALQ-144 Infrared (IR) Countermeasures Device. The ice accretion and shedding characteristics of a simulated ALQ-144 IR Countermeasures device were evaluated during these tests. The device was approximately the same shape as an ALQ-144 and was located just aft of the main rotor mast on the top of the fuselage. This location offered the advantage of masking some of the water drops which would normally impinge and accumulate on the forward surfaces of the device. The resulting accumulations did not adversely effect the operation of the aircraft. Future natural icing tests should use an operational ALQ-144 to verify test results.

(12) Hellfire Missile System. Ice was accreted on the dummy Hellfire missiles and launcher racks at the specific flight conditions shown in table 1, enclosure 4. All sharp edges and stagnation points on the missiles and launcher racks accreted ice. The frangible Hellfire deice system was not installed on the test aircraft since satisfactory system performance was previously demonstrated (ref 1b).

(13) Wing Pylon Articulation. Satisfactory performance of the wing pylon articulation system was previously demonstrated (ref 1f). No restriction of pylon travel was noted during these tests.

(14) Area Weapon System. The ice accretion and shedding characteristics of the 30mm chain gun area weapon system were evaluated in natural icing conditions. Ice accreted on all stagnation points and sharp edges (photo 13, encl 5). Weapons firing was not attempted following flight in icing conditions. Although live fire operations were not performed, the accreted ice did not hamper the traversing operation of the weapon. The gun was traversed inflight through its full range in azimuth and elevation to verify unrestricted motion. For these flights, a temporary muzzle protector made of cloth prevented ice buildups inside the gun barrel. A nonproduction cover was also fitted over the ammunition chute. This cover was modified from previous tests (ref 1f) and incorporated

a 30 mm turret boot (see para 3i(2)). Without this cover ice would have accumulated inside the chute preventing proper ammunition feed. An ammunition chute cover should be incorporated as a permanent installation and testing should be conducted to verify proper system operation during live fire exercises with the cover installed.

(15) Rocket Launcher, M200. Rocket launchers for the 2.75 inch folding fin aerial rockets were installed for 20 flights in natural icing conditions. Forward facing protective covers were installed on the rocket pods. These covers were made of a thick, black plastic material and appeared to be very durable. Ice accreted on those protective covers. Tests of these covers and their separation characteristics have not been accomplished. Further testing should be conducted on the 2.75 inch aerial rocket launcher system with the pod covers installed to determine the consequences of the covers departing the launcher, with and without accumulated ice.

(16) Radar Jammer Antennas. A radar jammer transmitter antenna was mounted on the TADS/PNVS capsule (photo 14, encl 5) and a radar jammer receiver antenna was mounted on the upper fuselage (photo 1, encl 5). Operation of these antennas were not evaluated during the artificial icing tests (ref 1f). The 20 natural icing flights with these antennas installed included only two flights with LWC > 0.3 gm/m^3. Large ice accretions on the unprotected radar jammer transmitter antenna could possibly restrict PNVS capsule motion. The potential for restriction of PNVS capsule range of movement by large ice accretions on the unprotected radar jammer transmitter antenna should be further evaluated in more severe natural icing conditons.

(17) Proposed Wire Strike Protection System. Installation of proposed Wire Strike Protection System (WSPS) would mount wire cutters on the front of the aircraft on some unspecified upper and lower fuselage sections. The proposed WSPS should be evaluated in natural icing conditions to determine possible impact on existing anti-ice/deice systems and the potential for aircraft damage (i.e., engine FOD) due to shed ice.

e. Performance. (1) Level Flight Performance. Level flight performance characteristics of the AH-64A helicopter were evaluated at the specific natural icing test conditions listed in table 1, enclosure 4. Collective control position was fixed at pre-immersion trim position, altitude was maintained and airspeed was allowed to vary as necessary during the encounter. Indicated power required increases between deice cycles ranged from 4 to 10 percent for the majority of the natural icing conditions. During one flight at 0.15 gm/m^3 and -7.5°C, indicated power required increases of 17% were observed between blade heater cycles (para 3b(2)). This increase in power required was believed to be the result of short blade heater on-times. However, only one natural icing flight was performed with LWC > 0.3 gm/m^3 while the blade deice system was operating on the correct on- and off-time schedules. Further tests should be conducted in more severe natural icing conditions to document level flight performance degradation due to ice accretion when the automatic deice system blade heaters are operating at the correct on- and off-times.

(2) Power Loss with Operation of Anti-Ice Systems. The engine performance characteristics were recorded with the bleed air anti-ice systems OFF and ON for comparison (ref 1f). An approximate 50°C increase in measured gas temperature was observed with anti-ice system use. The power available losses with activation of the anti-ice systems were significant. Performance data incorporating power losses due to operation of the anti-ice systems should be published in the operator's manual for preflight planning information.

f. Handling Qualities. A qualitative handling qualities evaluation was performed during natural icing flights at the conditions listed in table 1, enclosure 4. The evaluation was accomplished by performing typical instrument flight maneuvers with and without ice on the aircraft. No degradation of aircraft handling qualities were noted as a result of aircraft ice accretion.

g. Vibration. The aircraft vibration characteristics were monitored throughout these evaluations. Qualitative crew comments were compiled during the icing immersions and flights to home base with residual ice. No significant increase in vibration levels other than those previously discussed was noted during flight in natural icing conditions. Further evaluation of vibration characteristics should be conducted under more severe natural icing conditions with the automatic deice system blade heaters operating at the correct on- and off-times.

h. Reliability and Maintainability. (1) Tail rotor elastomeric bearings. The four bladed semi-rigid teetering tail rotor has four elastomeric bearings which connect each delta hinged hub assembly to a titanium fork and provide the teetering axis for each pair of blades. The elastomeric bearing is bonded to the fork assembly. Debonding of the tail rotor elastomeric bearings was previously reported as a shortcoming (ref 1f). As a result of a new bonding procedure no evidence of tail rotor elastomeric bearing debonding was observed during this evaluation.

(2) Fault Detection/Location System (FD/LS). The FD/LS is a function of the fire control computer (FCC) program. The FD/LS incorporates continuous monitoring of selected subsystems to advise the pilot of system status during aircraft operation. Additionally, the "on command" FD/LS program allows maintenance personnel and/or aircrew to perform system checks and troubleshoot failed or partially failed systems by entering a two digit code for that system on the data entry keyboard. The latest version of the FCC software (part no. 7-319200001-23) was installed for two flights (tests 33 and 34) to evaluate the FD/LS operation. Four FD/LS shortcomings previously reported in reference 1f had been corrected and one additional shortcoming was identified.

(a) FD/LS Software Logic. The failure of the FD/LS software logic to allow interrogation of all subsystems in a faulty system has been corrected. An additional blade deice system FD/LS software logic shortcoming was identified (see para 3h(2)(e).

(b) Tail Rotor Heater. The erroneous tail rotor heater no-go status message has been corrected.

(c) Canopy Heater. Previous FCC software allowed the FD/LS to display a canopy heater no-go message when the heater switch was placed in the OFF position, regardless of system status. The FD/LS has been corrected to display a no-go status only when the system is activated and has failed.

(d) Engine Anti-Ice System. The periodic illumination of the engine anti-ice fail/master caution lights during normal operations has been corrected. Additionally, failure of the FD/LS to provide the capability to interrogate the engine anti-ice subsystems has been corrected by addition of bleed valve advisory lights on the anti-ice control panel.

(e) Blade Deice System. The FD/LS monitors four subsystems through the blade deice controller. These four subsystems are the tail rotor heaters, main rotor heaters, distributor, and power controller. When AC electrical power is applied and the blade deice switch located on the pilot's anti-ice control panel is in the OFF position, the FD/LS displays no-go (fail) messages for each subsystem, regardless of system status. These fail messages are continuously displayed as long as the blade deice control switch remains in the OFF position and requires the crew to scroll through several pages of displayed no-go messages to identify the status of other systems. Additionally, display of no-go messages with the system turned off is contrary to the design philosophy of other anti-ice/deice systems (see para 3h(2)(c). No-go status fail messages for the blade deice subsystems resulting from the blade deice switch being placed in the OFF position are a shortcoming. The continuous FD/LS function should be modified to display blade deice system no-go messages only when the blade deice switch is in the ON position and a subsystem has failed.

(3) Main and Tail Rotor Deice Controller. The AEG-Telefunken controller has the inherent capability of preventing a blade or blade element from overheating above a temperature limit of 55° C. The requirement to trim the deice controller to the electrical resistance characteristics of each main and tail rotor blade was documented during previous tests (ref 1f). The controller measures blade element temperature by the change in electrical resistance as the element is heated requiring the controller to be "trimmed" for the variation in resistance in each blade/blade element. This procedure is required each time a main or tail rotor blade is changed. The specific procedure and equipment required was not contained in the -23 series technical manuals. The

procedure and the peculiar ground support equipment required was documented only in the MDHC rotor blade deice production test procedure (PTP), PTP 77-TP-7541-3, Rev A, Main and Tail Rotor Blade Deice system. Furthermore, charts are required which are contained only in the AEG-Telefunken manual. The required equipment and charts are only available at MDHC in Mesa, Arizona. Since main and tail rotor blade changes are a unit level maintenance function, this will cause a system maintainability problem. A modified deice controller is proposed to correct this problem but has not been incorporated in production aircraft. The requirement to manually adjust the deice controller to the rotor system electrical resistance characteristics following a main or tail rotor blade change remains a shortcoming.

i. Human Factors. (1) Ice Protection Systems Advisory Light Location. The pilot's anti-ice control panel (fig. 1) was located at the aft end of the pilot's lower left console. Two advisory lights (deice system ON and inlet anti-ice system ON) were located on this panel and are not in the pilot's normal instrument scan pattern. The ice protection system control panel and/or the advisory lights should be located in the pilot's normal instrument scan pattern. The poor location of the ice protection systems advisory lights and anti-ice control panel as previously reported in reference 1f remains a shortcoming. Consideration should be given to moving the advisory lights and/or the pilot's anti-ice control panel to a more suitable location.

(2) Cabin Environment. The cabin environment at both crew stations was qualitatively evaluated. To resolve a previously reported cabin environment problem (ref 1b), the emergency air door in the pilot's compartment was modified for these tests and a nonstandard boot was installed around the ammunition chute on the 30mm gun turret (photo 13, encl 5). The modified pilot's emergency air door was satisfactory. The 30mm turret boot and 30mm ammunition chute cover (para 3d(14)) previously evaluated in reference 1f were combined into a single unit for this test. MDHC reported that this configuration failed the live fire tests and a proposed production configuration is undergoing further modification/evaluation. With the modified emergency air door and nonstandard turret boot, the cabin environment was satisfactory at all ambient temperature conditions. The cold aft cockpit environment during flight at approximately -20°C (ref 1f) without the nonstandard boot modification on the 30mm gun turret constitutes a safety of flight hazard and remains a deficiency. A permanent boot should be installed on the 30mm gun turret to eliminate the undesirable aft cockpit environment in cold weather flight.

4. CONCLUSIONS. a. General. The following general conclusions were reached as a result of the partially completed artificial and natural icing tests of the AH-64A helicopter. Potential problem areas requiring further evaluation are described under recommendations (para 5).

(1) Five deficiencies were found.

(2) Three shortcomings were found.

(3) Sixteen Test Incident Reports were submitted.

b. Deficiencies. The following deficiencies have been identified and are listed in relative order of importance.

(1) The high potential for engine FOD due to shed ice from the mismatch between the fore and aft nose gearbox/cross shaft fairings (para 3d(8)).

(2) Blade heater off-times that are inconsistent with actual icing conditions (para 3b(3)).

(3) Inaccurate indications on the pilot's icing severity meter (para 3b(4(b)).

(4) Rotor blade heater on-times which are inconsistent with the design schedule and actual OAT (para 3b(2)).

(5) The cold aft cockpit environment during flight at approximately -20°C without the nonstandard boot installed on the 30mm gun turret constitutes a safety of flight hazard (para 3i(2)).

c. Shortcomings. The following shortcomings have been identified and are listed in relative order of importance.

(1) The requirement to manually adjust the deice controller to the rotor system electrical characteristics following a main or tail rotor blade change (para 3h(3)).

(2) No-go (fail) status messages for the blade deice subsystems resulting from the blade deice switch being in the OFF position (para 3h(2)(e).

(3) The poor location of the ice protection system advisory lights/anti-ice control panel (para 3i(1)).

5. RECOMMENDATIONS. The following recommendations are made:

a. Correct the deficiencies and verify proper system operation prior to clearing the AH-64 for operational use in a moderate icing environment.

b. Correct the shortcomings as soon as practical.

c. The AH-64 should be further evaluated in natural icing conditions at higher LWC (>0.3 gm/m^3) and over a broader range of temperatures (0°C and -20°C) to evaluate and verify:

(1) Correct blade heater on-time (para 3b(2)).

(2) Accuracy of the ice detection/rate measuring system (para 3b(4)(a)).

(3) Main rotor ice shedding characteristics when the automatic rotor deice system is operating at the correct blade heater on- and off-time (para 3b(5)).

(4) Susceptibility of the tail rotor to damage from main rotor shed ice (para 3b(6)).

(5) Operation of the modified nose gearbox/cross shaft fairing heaters (para 3c(4)).

(6) TADS/PNVS anti-ice system operation (para 3c(8)).

(7) Ice accretion and shedding characteristics of the new TADS laser shield (para 3c(9)).

(8) The potential for stabilator damage due to shed tail rotor ice (para 3d(3)).

(9) The potential for engine FOD and aircraft damage due to ice accretion and shedding from:

(a) The fuselage handholds/steps (para 3d(4)).

(b) The canopy frames (para 3d(6)).

(c) The windshield wipers (para 3d(7)).

(10) Susceptibility of the engines to damage from shed ice (para 3d(8)).

b. A study should be conducted to determine possible degradation of reception with ice accumulation on the forward facing radar warning antennas (para 3d(9)).

c. Future natural icing tests should use an operational ALQ-144 (para 3d(11)).

d. Incorporate a 30mm ammunition chute cover as a permanent installation and conduct testing to verify proper system operation during live fire exercises with the cover installed (para 3d(14)).

e. Further testing should be conducted on the 2.75 inch aerial rocket launcher system with the pod covers installed to determine the consequences of the covers departing the launcher, with and without accumulated ice (para 3d(15)).

f. The potential for large ice accretions on the unprotected radar jammer transmitter antenna to restrict PNVS capsule range of motion should be further evaluated in natural icing conditions (para 3d(16)).

g. The proposed WSPS impact on existing anti-ice/deice systems and potential to cause engine FOD and aircraft damage due to ice accretion/shedding characteristics should be evaluated in natural icing conditions (para 3d(17)).

h. Level flight performance degradation (power required increases) due to ice accretion when the automatic deice system blade heaters are operating at the correct on- and off-times should be further evaluated in more severe natural icing conditions (para 3e(1)).

i. Performance data incorporating power losses due to operation of the anti-ice systems should be published in the operator's manual for preflight planning information (para 3e(2)).

j. Further evaluation of vibration characteristics should be conducted in more severe natural icing conditions with the blade heaters operating at the correct on- and off-times (para 3g).

k. The FD/LS should be modified to display blade deice system no-go messages only when the blade deice switch is in the ON position and a subsystem has failed (para 3h(2)(e).

l. Consideration should be given to moving the advisory and/or the pilot's anti-ice control panel to a more suitable location (para 3i(1)).

m. Incorporate a permanent boot installation on the 30mm gun turret to eliminate the undesirable aft cockpit environment in cold weather (para 3i(2)).

6. AUTHORS. This report was prepared by CW4 James M. Adkins, Project Officer, Autovon 527-4986, Commercial (805) 277-4986, and Mr. Ralph Woratschek, Project Engineer, Autovon 527-4786, Commercial (805) 277-4786.

6 Encl

Al R. Todd

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SYSTEM DESCRIPTION

GENERAL

1. The deice system consisted of an outside air temperature (OAT) sensor, ice detector/rate measuring sensor and signal processor unit, icing rate meter, slip rings for the main and tail rotor, and deice controller. The main and tail rotor blades contained electrothermal resistive heating mats. Anti-ice systems were provided for sections of the windscreen, pitot-static tubes, air data sensor, engines, engine inlets, and nose gearbox/cross shaft fairings, target acquisition designation system, and pilot night vision system. The deice/anti-ice systems were activated by the pilot and copilot/gunner anti-ice control panels (fig. 1). The anti-ice/deice systems on the test aircraft S/N 83-23811 shown in figure 2 were identical to those described in reference 1f except as noted below.

2. During this test, several major modifications were made to the deice/anti-ice systems. The unaspirated ice detector/rate measuring probe mounted in the No. 1 engine inlet and previously tested in February-April 1985 (ref 1f) was replaced with an aspirated Rosemount probe mounted on the upper fuselage. Several different probe locations were evaluated (see para 3b(4)). Additionally, the blade heater on-time schedule was modified and additional heater blankets were added to the aft nose gearbox/cross shaft fairing at the engine inlet.

ROTOR BLADE DEICE SYSTEM

3. The rotor blade deice system is described in reference 1f. The modified blade heater on-time schedule is shown in figure 2, enclosure 4 and the off-time schedule is shown in figure 3, enclosure 4.

ICE DETECTION/RATE MEASURING SYSTEM

4. The ice detection/rate measuring system consisted of an aspirated Rosemount liquid water content (LWC) probe, Model 871FF2 and a Model 524Y4 signal processor unit. The LWC probe is of the ice accretion type employing a vibrating sensing probe which changes frequency as ice is accreted. The probe is aspirated by heated compressor air and was mounted on the right side of the upper fuselage for the most recent configuration as shown in figure 2. The system includes a signal processor to output an analog voltage signal proportional to LWC.

5. The ice detector rate probe is powered by dc current and is on whenever ship's power is on. The icing rate signal is generated whenever ice buildup on the probe is between 0.015 inch and 0.060 inch thick. At the 0.060 inch thickness point, the probe is electrothermally deiced and the cycle repeated. The ENG ICE caution light illuminates at the 0.015 inch point and remains on for a minimum of 90 $\pm$10 seconds. If no further icing were encountered, the light would go out at the end of that time. If icing is continually encountered, the light will remain on.

Encl 1

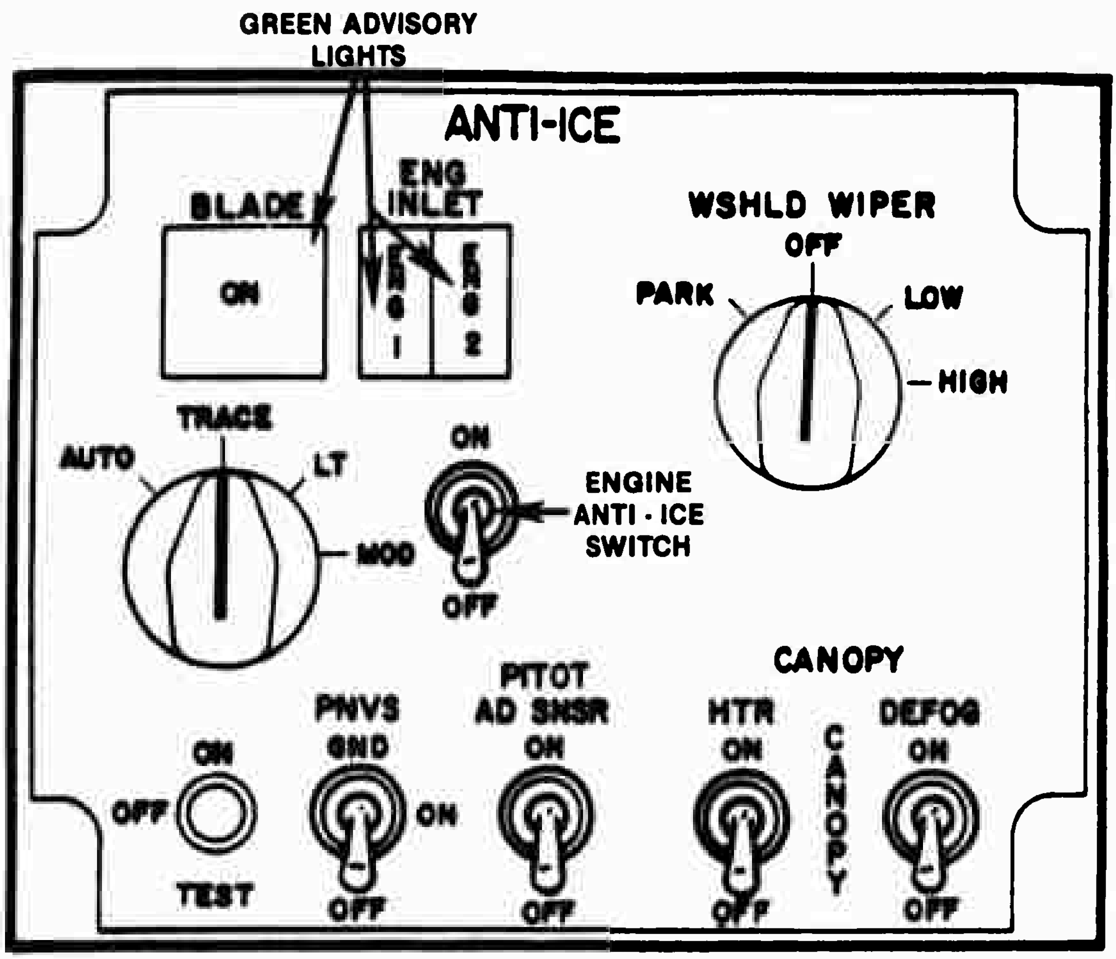


Figure 1. Pilot Anti-Ice Panel

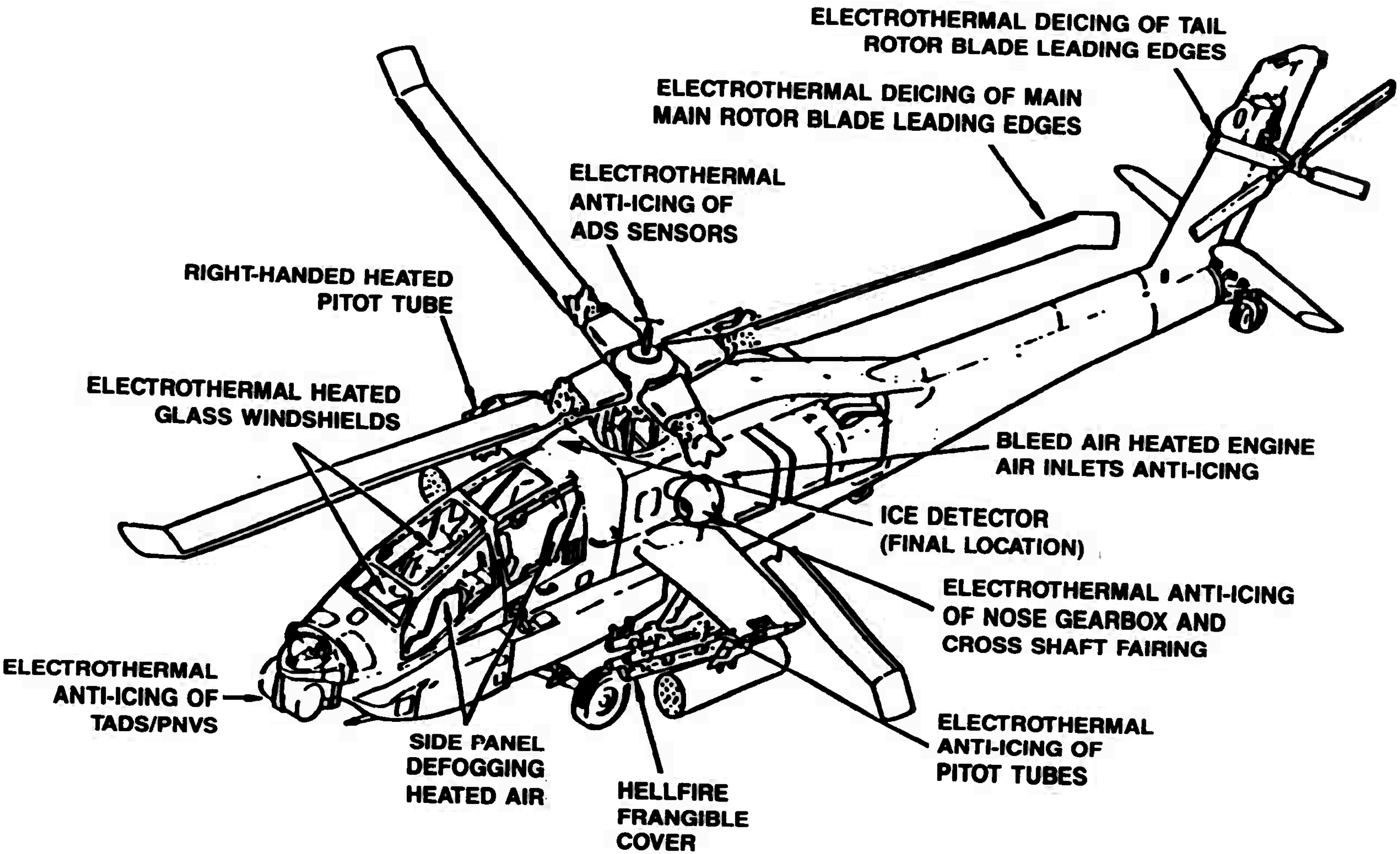


Figure 2. AH-64 Anti-Ice/Deice Systems

6. During the time the sensing probe is deiced and recovering from its 6 second heater application, a hold circuit is employed that maintains the LWC signal at its last value. The hold circuit operates until the probe has thermally recovered and releases when accreted ice thickness again reaches 0.015 inch or 60 $\pm$10 seconds of time has elapsed. Thus, if the aircraft emerges from an icing environment prior to reaching the 0.015 inch thickness point, the LWC signal would go to zero after this time interval and indicate a zero LWC condition. The probe recovery time (time the detector is off line) varies with LWC and OAT. Typical maximum recovery times for a probe velocity of 112 knots are 75 seconds at LWC = 0.12 gm/m^3, 15 seconds at 0.5 gm/m^3, and 10 seconds at 1.0 gm/m^3.

7. The output of the ice detection/rate measuring system will vary with air temperature and local flow velocity over the probe at a fixed LWC. In addition, the accuracy will vary with LWC at a fixed temperature and flow velocity. The ice detection/rate measuring sytem is calibrated in an icing tunnel to read LWC $\pm$10 percent for a local flow velocity of 112 knots. At higher and lower local flow velocities, the system will indicate slightly higher or lower than actual LWC values.

8. The deice system included an icing rate meter which displayed LWC as measured by the ice detector probe and a signal processor. The LWC meter display is mounted on the pilot's instrument panel.

NOSE GEARBOX/CROSS SHAFT FAIRING

9. The forward nose gearboxes and cross shafts from the engine to the transmission are covered with a fairing that incorporates electro-thermal heating blankets for anti-icing. The heater blankets are controlled automatically by a temperature controller which maintains the surface at a temperature sufficient to prevent the moisture that impinges on the fairing from freezing. Additional heater blankets were added to the aft nose gearbox/cross shaft fairing at the engine inlet to prevent runback and refreezing.

10. The electric heaters are turned on by the same two-position switch on the pilot's anti-ice panel that activates the engine air inlet bleed air system. Overheat sensors are provided that activate the ENG ANTI-ICE segment on the pilot's and the CPG's caution and warning panels and shut the electric heaters off automatically. The overheat sensor is a controlling-type sensor which allows the system to continue operation at the higher temperature without further overheat protection.

INSTRUMENTATION AND SPECIAL EQUIPMENT

GENERAL

1. In addition to standard aircraft instruments, McDonnell Douglas Helicopter Company installed and maintained calibrated instrumentation. Data from the cockpit instrumentation was hand recorded on flight cards. The instrumentation system recorded frequency modulation (FM) data on magnetic tape. Additional equipment installed at the pilot's station included full instrument landing system instrumentation (localizer, glide slope, and marker beacon). Special equipment installed at the copilot/gunner station included an icing rate indicator for the aspirated system and a control panel for operation of the video cameras and data system.

FM DATA PARAMETERS

2. The following data were recorded on magnetic tape in FM format.

Time code
Total air temperature
Observed air temperature (rotor deice system)
Blade deice current phase A
Icing severity (aspirated system)
Icing severity (engine inlet probe)
Target Acquisition Designation System Temperatures
- Top of dayside window
- Bottom of dayside window
- Dayside window frame
- Nightside window
- Nightside window frame
- Nightside shroud main heater
- Nightside shroud heater element

Pilot Night Vision System Temperatures
- Window
- Fairing

CAMERA SYSTEMS

3. Four video cameras were located on the test aircraft to document ice accretion characteristics. One camera was mounted on the right forward avionics bay to photograph the main rotor advancing blade leading edge and lower surface (photo 5, encl 5). Another camera located on the top of the left wing tip photographed the tail rotor and horizontal stabilator (photo 7, encl 5). The third camera photographed the left engine inlet (photo 7, encl 5). A fourth camera was mounted on the transmission access panel for three flights to document ice accretion on the right engine aft nose gearbox cross shaft fairing (photo 15, encl 5).

Encl 2

4. An additional video camera was located on board the chase aircraft to document ice accretion on the test aircraft after exit from icing encounters. Single lens reflex 35mm cameras were used for still photo (color prints and slides) documentation both in the air and on the ground following icing flight.

CLOUD SAMPLING EQUIPMENT

5. For cloud measurements in the natural environment, the US Army Aviation Engineering Flight Activity employs a JU-21A fixed-wing aircraft, US Army S/N 66-18008, equipped with a cloud measurement package. This package consists of the following equipment: a Particle Measuring System (PMS) forward scattering spectrometer probe (model FSSP-100), a PMS optical array cloud droplet spectrometer probe (model OAP-200X), Rosemount outside air temperature sensor and display, Cambridge model 137 chilled mirror dew point hygrometer and display, Cloud Technology ice detector unit, Leigh MK 12 ice detector unit, and a Small Intelligent Icing Data System.

TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. The deice system on the AH-64A helicopter was functionally tested prior to each icing flight. All anti-ice systems (i.e., pitot heat, windshield anti-ice, and engine air induction system anti-ice) were activated prior to cloud immersion. The JU-21A icing scout aircraft located and documented the icing condition before the test aircraft entered the icing environment. The JU-21A then loitered in the area to facilitate a post-immersion rendezvous with the test aircraft for photographic documentation. The liquid water content (LWC), median volumetric diameter, and ambient temperature were documented by the JU-21A aircraft. The Rosemount icing rate meter in the test aircraft was also used to monitor LWC in natural clouds.

ICE ACCRETION AND SHEDDING

2. Ice accretions were documented using hand-held video from the chase aircraft and four video cameras mounted on the test aircraft. Post-flight photographs were made to document the ice remaining on the individual components of the airframe and rotors.

3. The vibration characteristics of the aircraft before and after ice shedding were evaluated using the Vibration Rating Scale shown in figure 1.

WEIGHT AND BALANCE

4. Prior to testing, the aircraft gross weight and longitudinal and lateral centers of gravity (cg) were determined by using calibrated scales. The aircraft was weighed with full instrumentation onboard, without fuel, and with auxiliary fuel tanks installed inboard and rocket pods outboard. The aircraft basic weight was 12,101 pounds with a longitudinal cg location at fuselage station 208.4 and a lateral cg location at buttline -0.3 left.

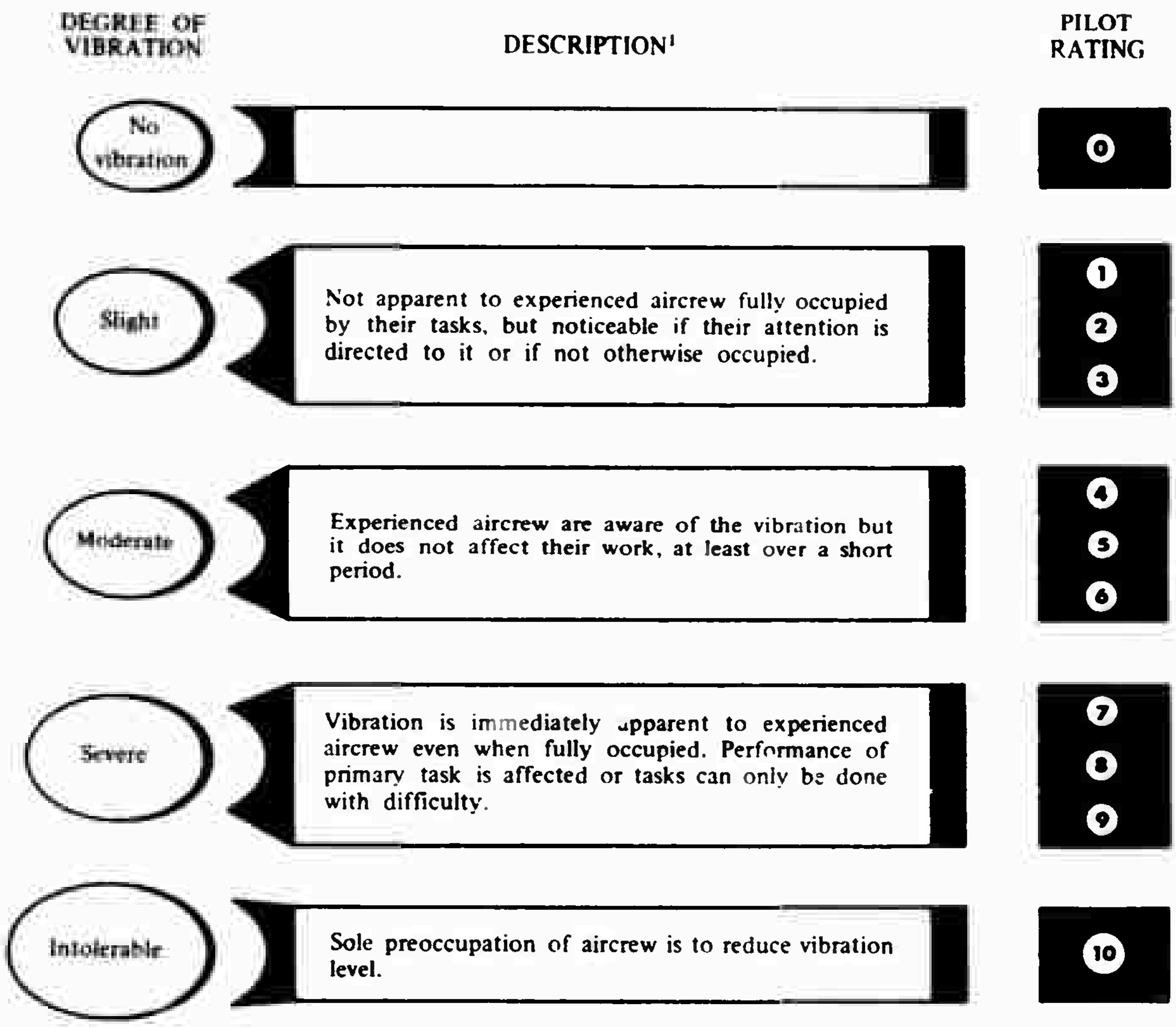
DEFINITIONS

5. Icing characteristics were described using the following definitions of icing severity. These definitions may be found in FM 1-230.

a. Trace Icing: Ice becomes perceptible. Rate of accumulation slightly greater than rate of sublimation. It is not hazardous even though deicing equipment is not used, unless encountered for an extended period of time (over 1 hour).

b. Light Icing: The rate of accumulation may create a problem if flight is prolonged in this environment (over 1 hour). Occasional use of deicing/anti-icing equipment removes/prevents accumulation. It does not present a problem if the deicing/anti-icing equipment is used.

Encl 3



[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 1. Vibration Rating Scale

c. Moderate Icing: The rate of accumulation is such that even short encounters become potentially hazardous and use of deicing/anti-icing equipment or diversion is necessary.

6. Results were categorized as deficiencies or shortcomings in accordance with the following definitions.

Deficiency: A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipments operational capability.

Shortcoming: An imperfection or malfunction occurring during the life cycle of equipment, which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the usability of the material or end product.

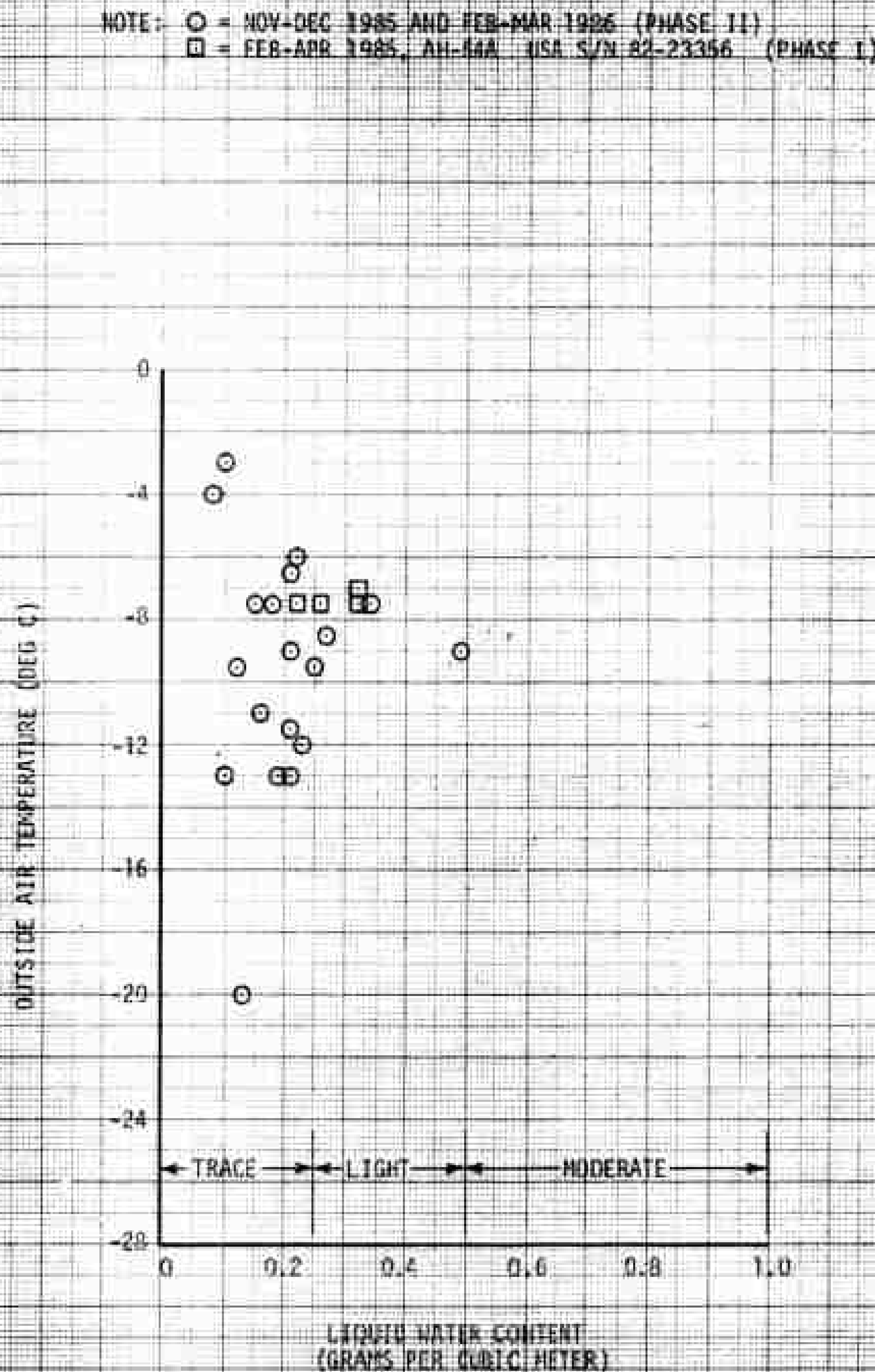
Table 1. Specific Natural Icing Test Conditions[1]

MDHC[2] Test Number	Date	Average Gross Weight (lb)	Average Longitudinal Center of Gravity Location (FS)	Average True Airspeed (knots)	Average Density Altitude (feet)	Average OAT (°C)	Time in Cloud (minutes)	Median Volumetric Diameter (microns)	Average Aspirated Rosemount LWC[3] (gm/m^3)	U-21 Instrumentation: Average FSSP[4] LWC (gm/m^3)	U-21 Instrumentation: Average CT[5] LWC (gm/m^3)	U-21 Instrumentation: Average Leigh Mk 12 LWC (gm/m^3)
8	16 Nov 85	14,950	204.0	118	3630	-6.0	60	20	0.27	0.26	0.14	--
9	19 Nov 85	15,590	204.0	116	2430	-9.0	30	16	0.50	0.48	0.48	--
10	19 Nov 85	15,400	204.0	118	2420	-11.5	11	12	0.20	0.19	0.23	--
11	20 Nov 85	15,260	204.0	126	6740	-20.0	29	15	0.17	0.12	0.09	--
12	26 Nov 85	14,620	204.5	61-137	1150	-12.0	60	15	0.23	0.21	0.25	--
13	26 Nov 85	14,590	204.6	116	910	-13.0	60	--	0.08	0.08	0.14	--
14	4 Dec 85	14,100	204.5	118	2650	-13.0	85	13	0.22	0.18	0.18	--
16	8 Dec 85	14,570	204.6	119	2920	-9.5	45	19-300[6]	0.55	0.25	0.15	--
23	17 Feb 86	14,560	204.9	114	2690	-7.5	44	17	0.52	0.23	0.20	0.40
24	17 Feb 86	14,870	205.0	112	2040	-6.5	30	--	0 - 1.30[7]	0.19	0.15	0.30
25	17 Feb 86	14,180	204.5	117	1800	-8.5	75	16	0.50 - 2.00[7]	0.23	0.17	0.40
26	18 Feb 86	14,280	204.5	118	2170	-9.0	44	--	0.60 - 1.45[7]	--	0.12	0.30
27	19 Feb 86	14,600	204.6	115	4140	-7.5	45	--	0.20 - 1.30[7]	0.09	0.06	0.40
28	19 Feb 86	14,770	205.1	116	4380	-6.0	50	--	0.30 - 1.00[7]	0.16	0.11	0.40
29	19 Feb 86	14,800	205.1	117	5420	-7.5	60	15-20	0 - 0.25[7]	0.13	0.09	0.24
30	25 Feb 86	14,690	204.7	119	2670	-3.0	13	14	0.28	0.12	0.08	--
31	26 Feb 86	14,700	204.7	122	4710	-4.0	20	17	0 - 1.0[8]	0.11	0.05	--
32	26 Feb 86	14,470	204.6	116	1050	-9.5	30	--	0.15	--	0.09	--
33	28 Feb 86	14,200	204.5	125	6260	-13.0	70	16	0.30	0.15	0.11	0.38
34	3 Mar 86	14,570	204.4	116	4230	-11.0	50	14	0.27	0.15	0.13	0.20

NOTES:

[1]Rotor speed = 100%, Configuration: Auxiliary fuel tanks inboard, rocket pods outboard.
[2]McDonnell Douglas Helicopter Co.
[3]Liquid water content.
[4]Forward scattering spectrometer probe.
[5]Cloud Technology, Inc. (formerly Johnson-Williams).
[6]Mixed icing and freezing rain condition.
[7]Erratic readings possibly due to probe location.
[8]Erratic readings resulting from an open electrical circuit.

FIGURE 1
AH-64 NATURAL ICING TEST CONDITIONS
AH-64A USA S/N 83-23801



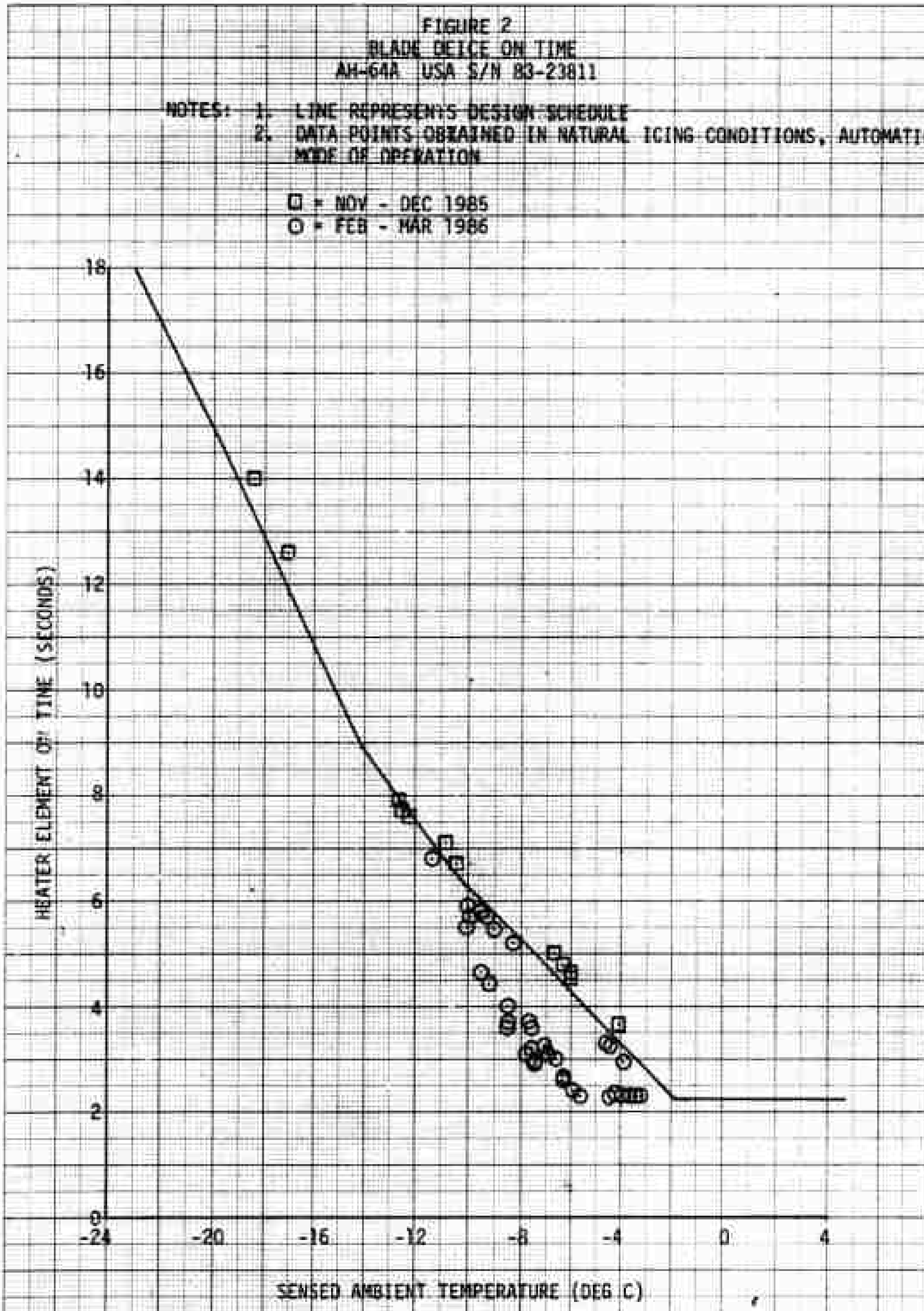


FIGURE 2
BLADE DEICE ON TIME
AH-64A USA S/N 83-23811
NOTES: 1. LINE REPRESENTS DESIGN SCHEDULE
2. DATA POINTS OBTAINED IN NATURAL ICING CONDITIONS, AUTOMATIC MODE OF OPERATION
□ = NOV - DEC 1985
○ = FEB - MAR 1986
HEATER ELEMENT ON TIME (SECONDS)
18
16
14
12
10
8
6
4
2
0
-24
-20
-16
-12
-8
-4
0
4
SENSED AMBIENT TEMPERATURE (DEG C)

FIGURE 3
BLADE DEICE OFF TIME
AH-64A USA S/N 83-23811

NOTES: 1. LINE REPRESENTS DESIGN SCHEDULE
2. DATA POINTS OBTAINED IN NATURAL ICING CONDITIONS, AUTOMATIC MODE OF OPERATION
3. SHADED SYMBOLS DENOTE MANUAL MODE OF OPERATION

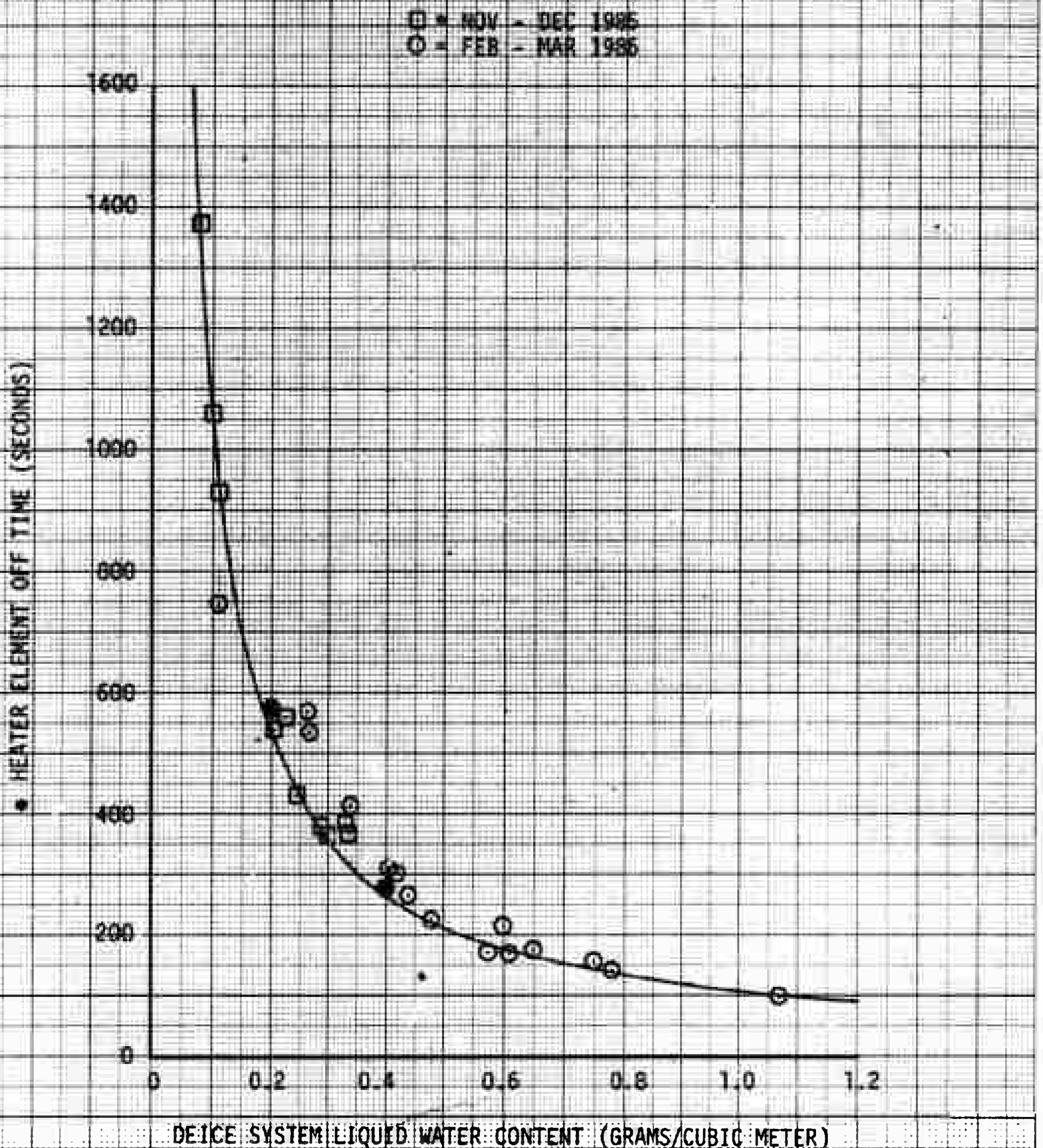
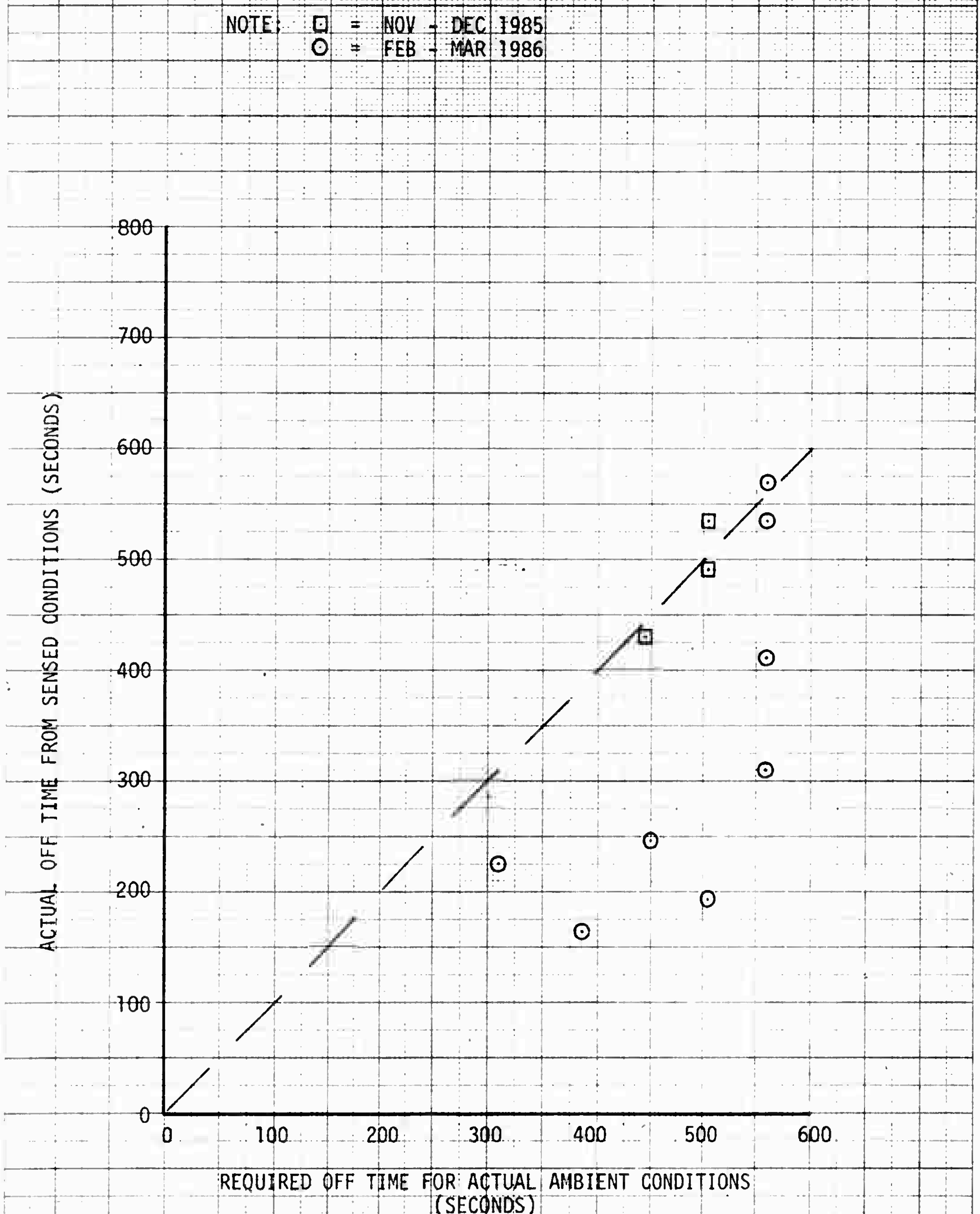


FIGURE 4
HEATER ELEMENT OFF TIME COMPARISON
AH-64A USA S/N 83-23811



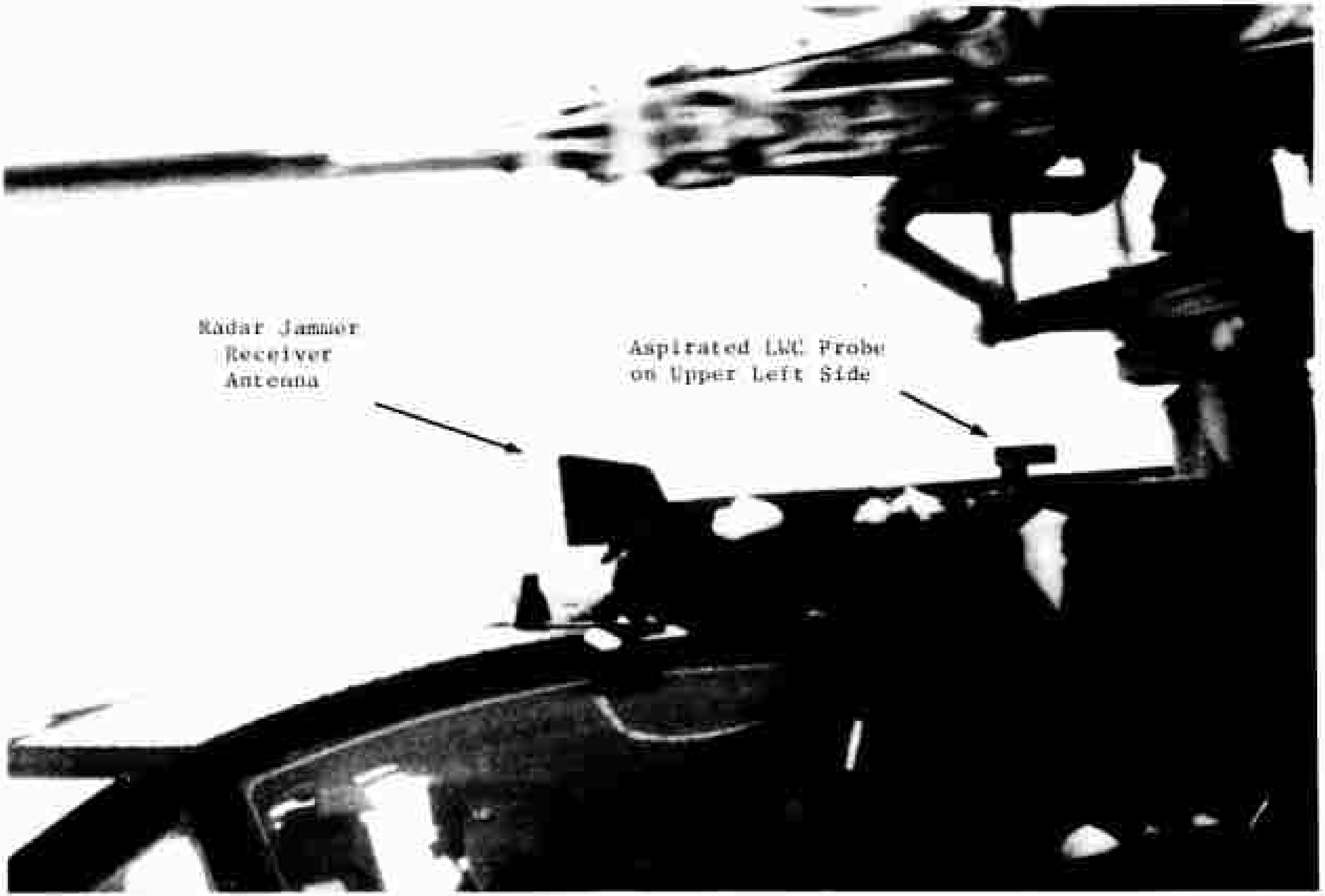


Photo 1. AH-64 Left Side - Upper Fuselage

Encl 5

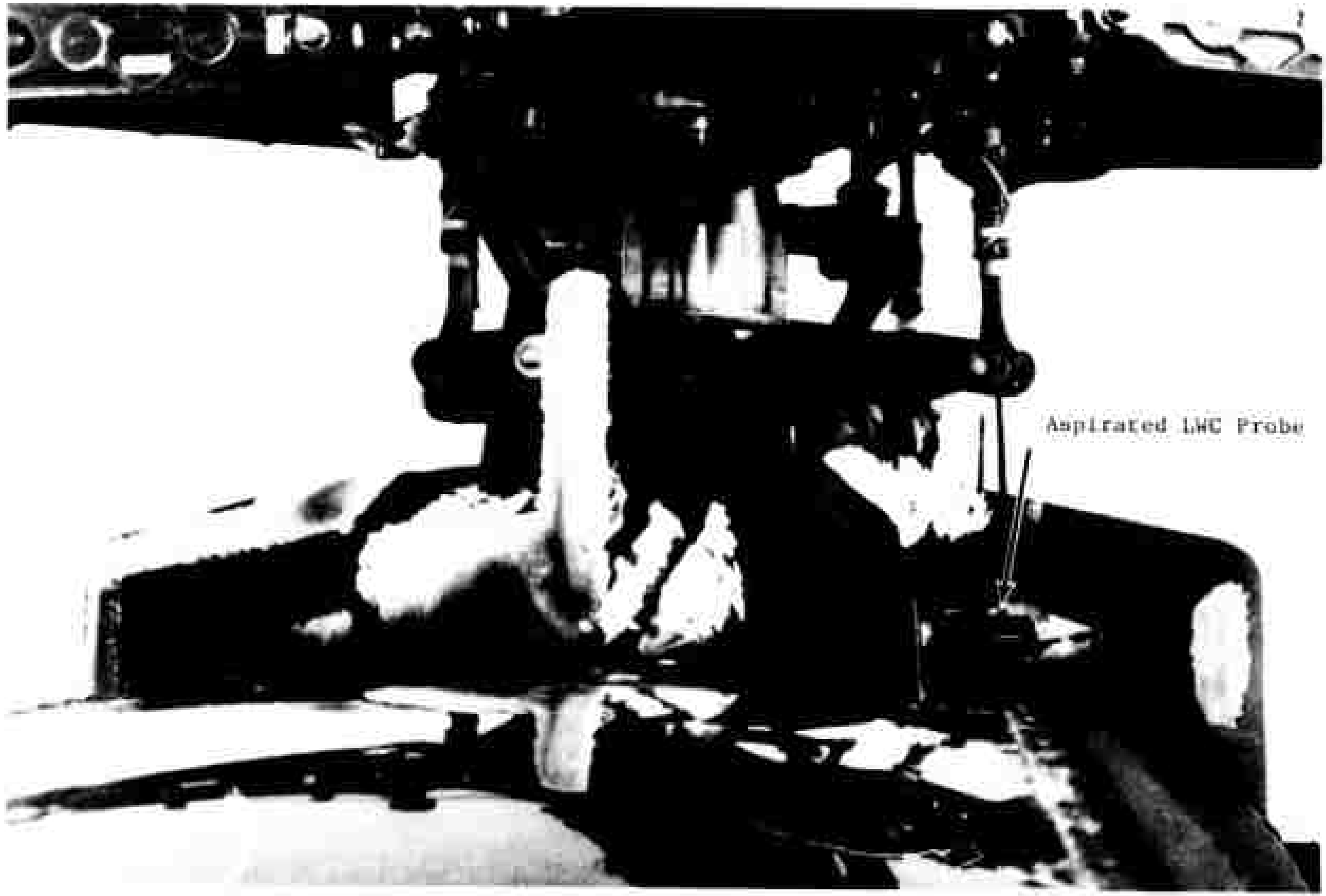


Photo 2. AH-64 Front View - Upper Fuselage

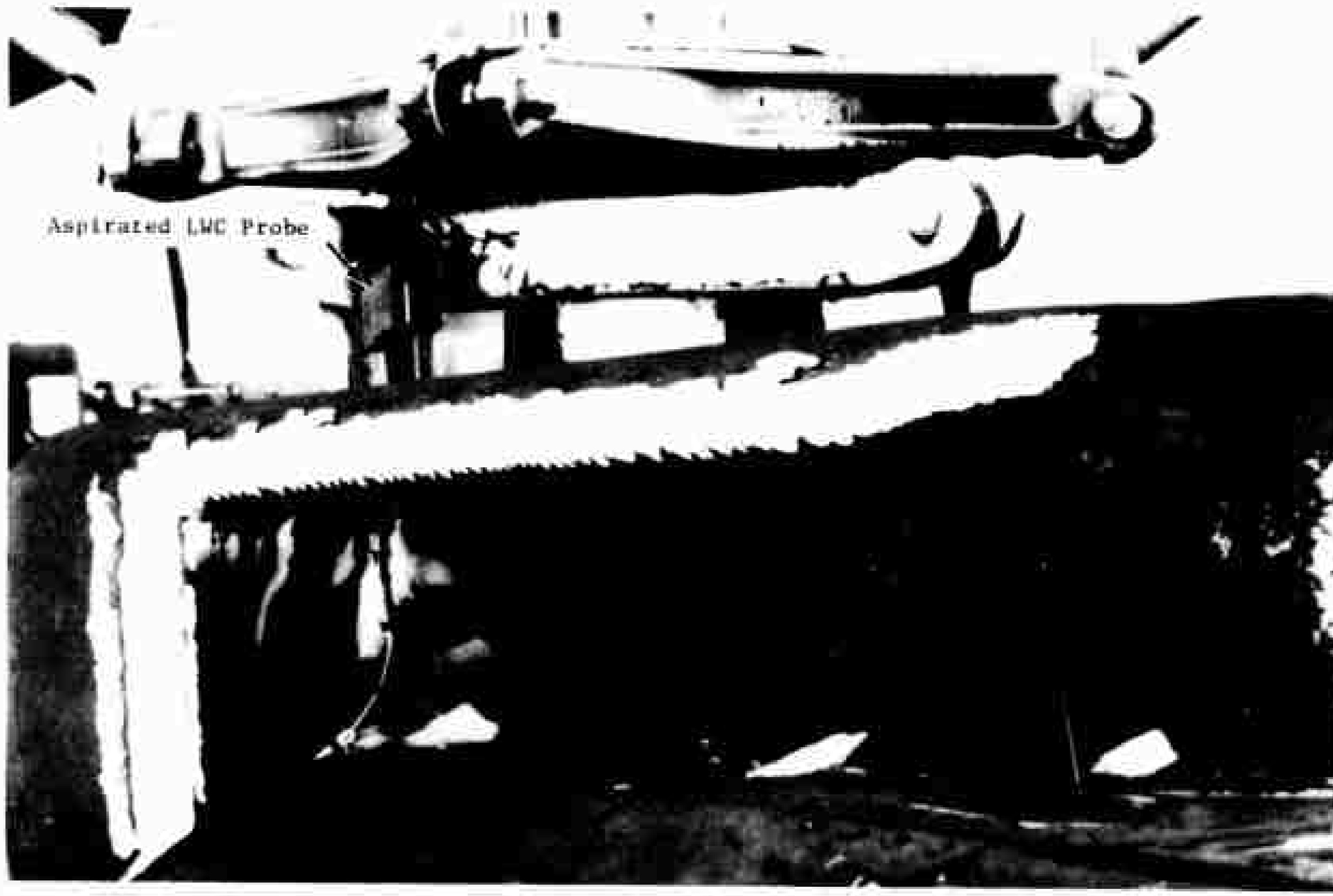


Photo 3. AH-64 Upper Fuselage - Quartering Right Front View

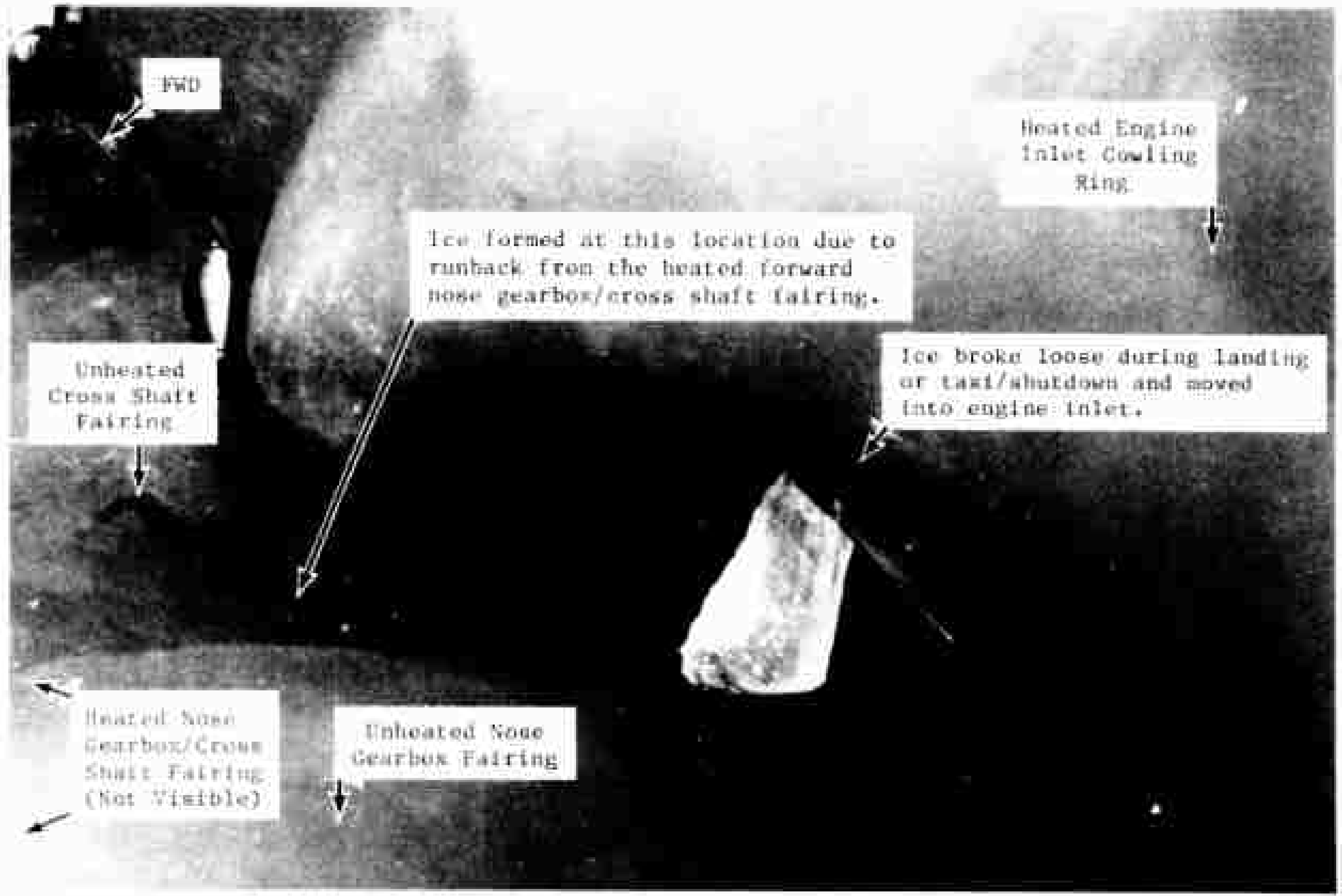


Photo 4. AH-64 No. 1 Engine Inlet

Photo 5. AH-64 - Right Front View

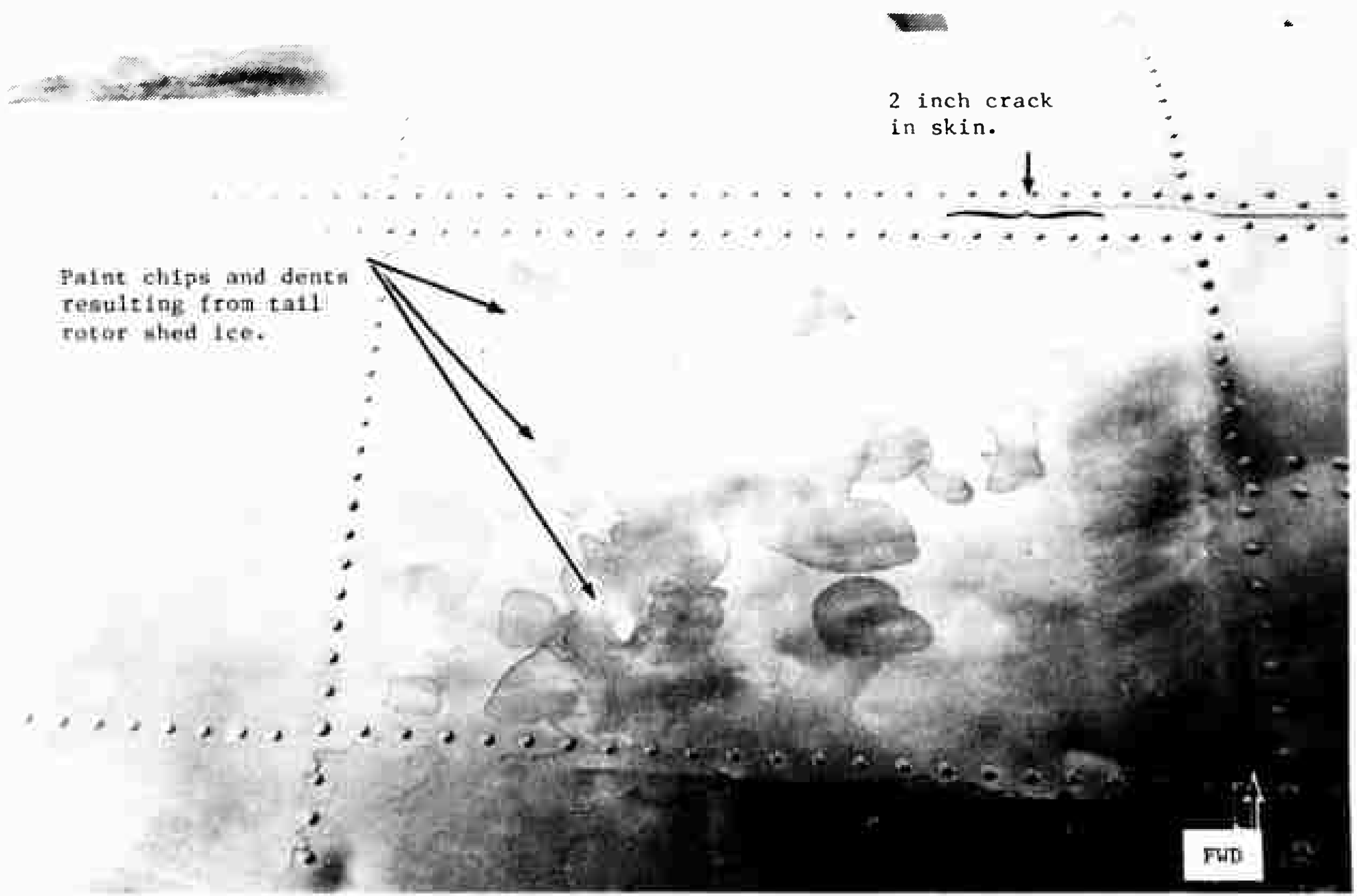


Photo 6. AH-64 Left Stabilator

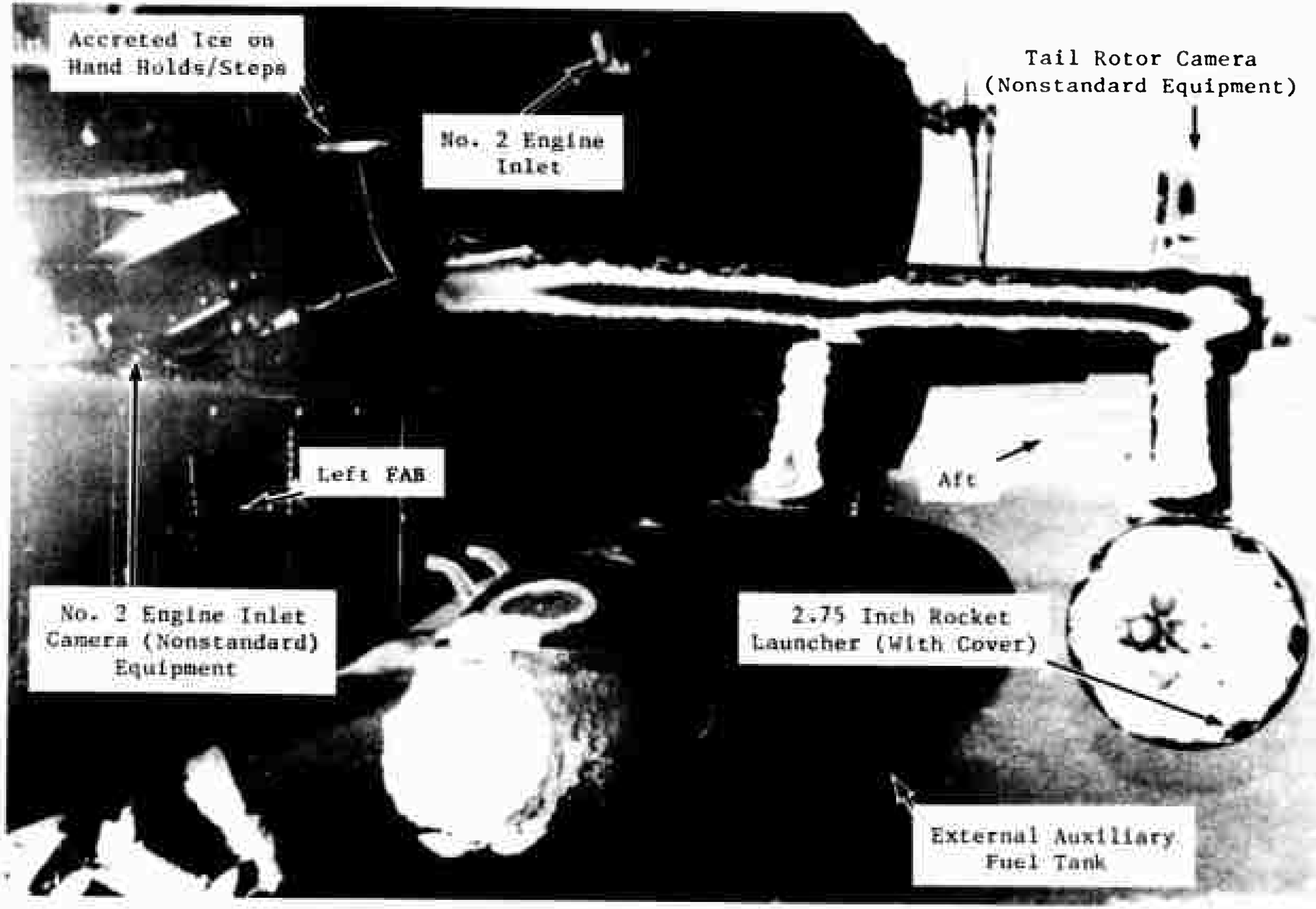


Photo 7. AH-64 Fuselage Ice - Left Side

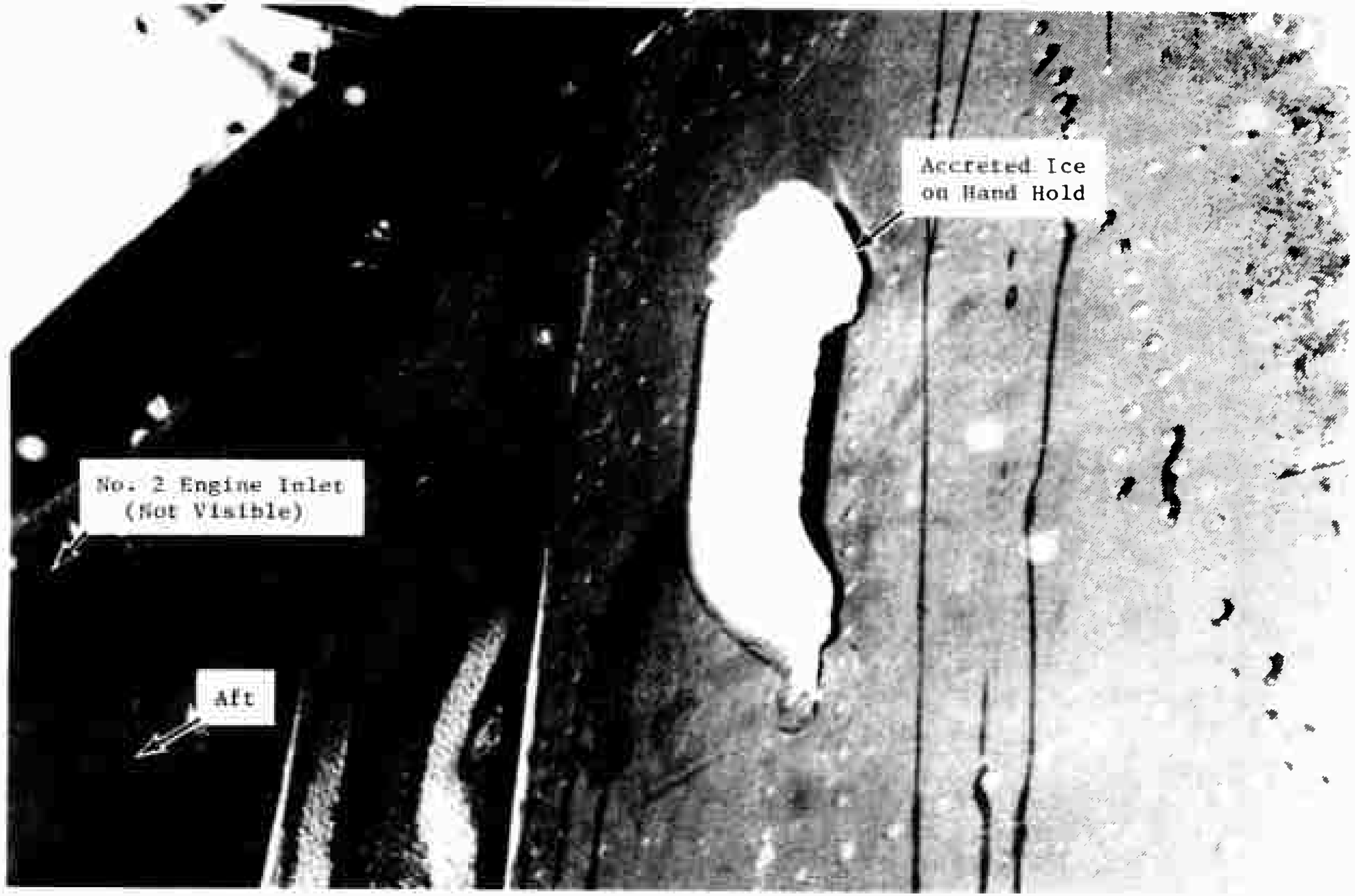


Photo 8. AH-64 Fuselage - Right Side

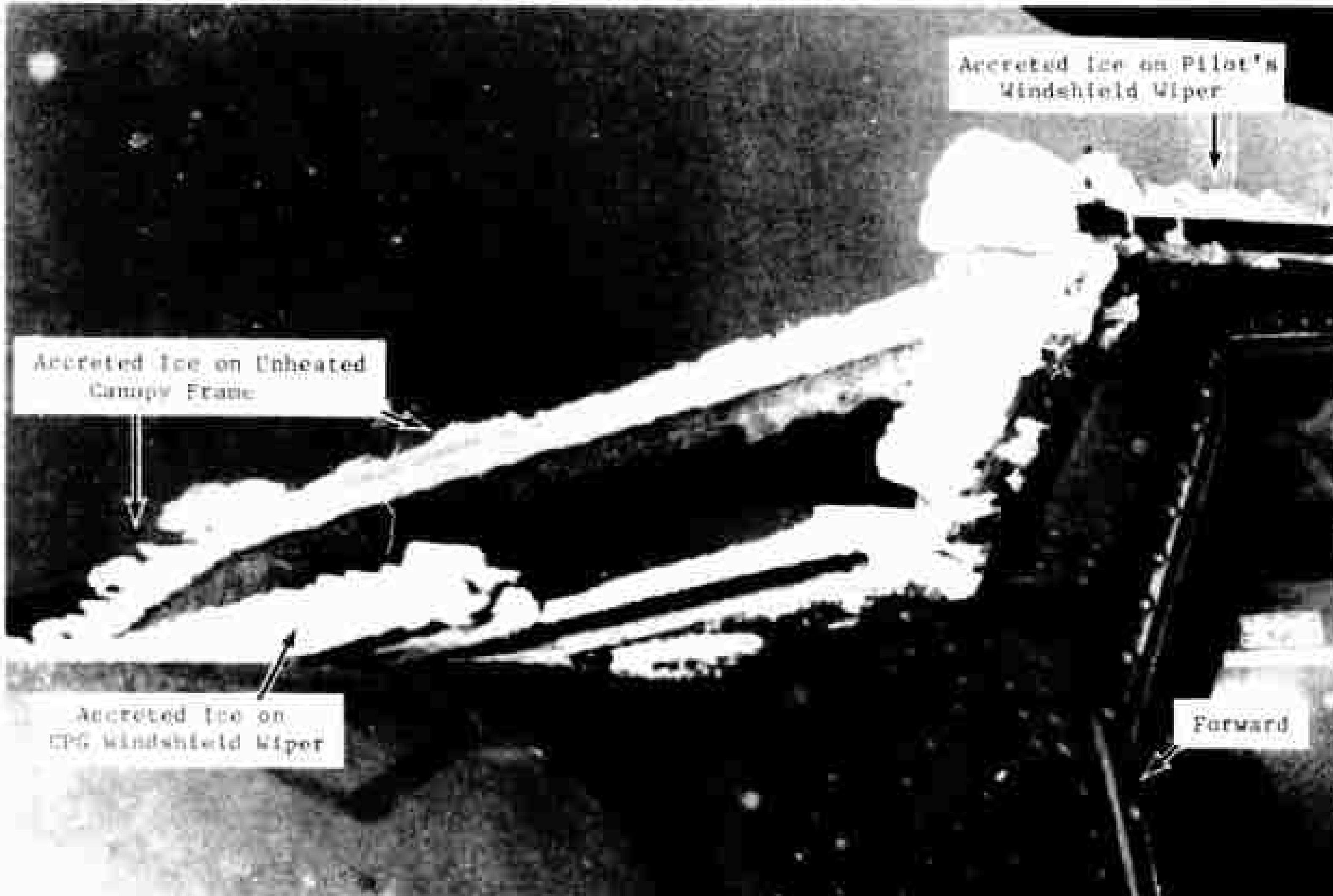


Photo 9. Ah-64 Forward Canopy

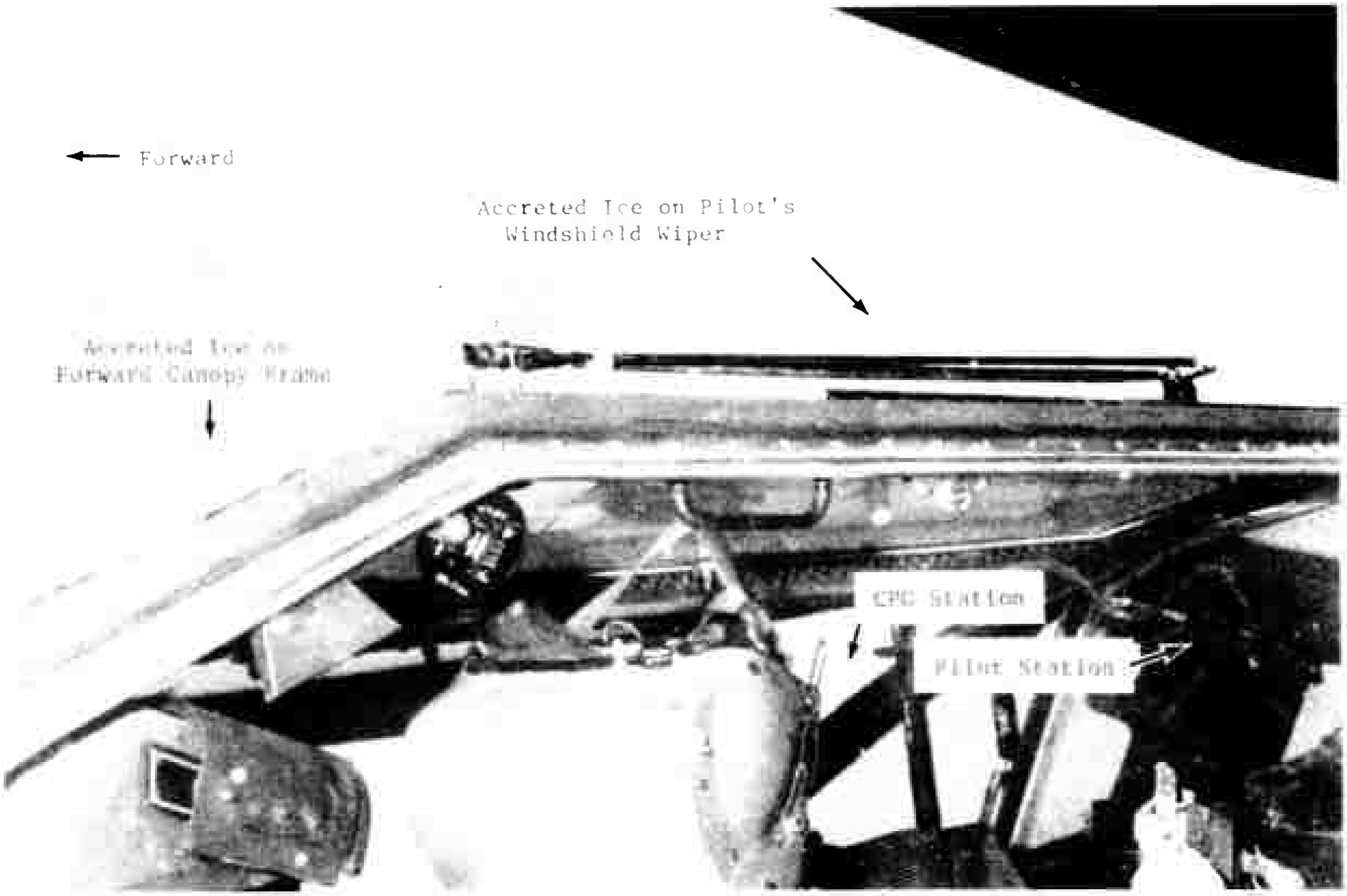


Photo 10. AH-64 Canopy - Left Side View

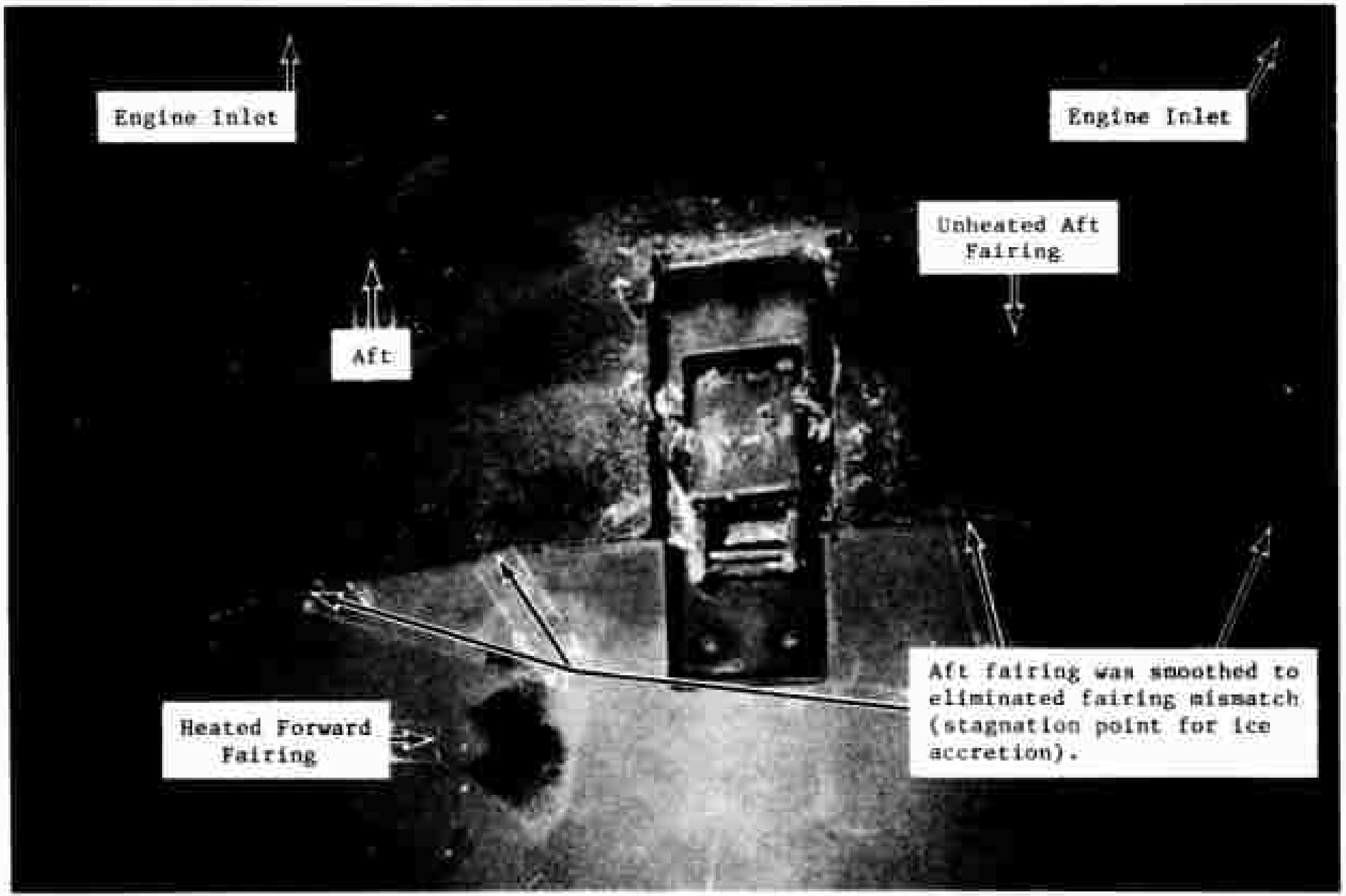


Photo 11. AH-64 No. 2 Engine Nose Gearbox - Top View

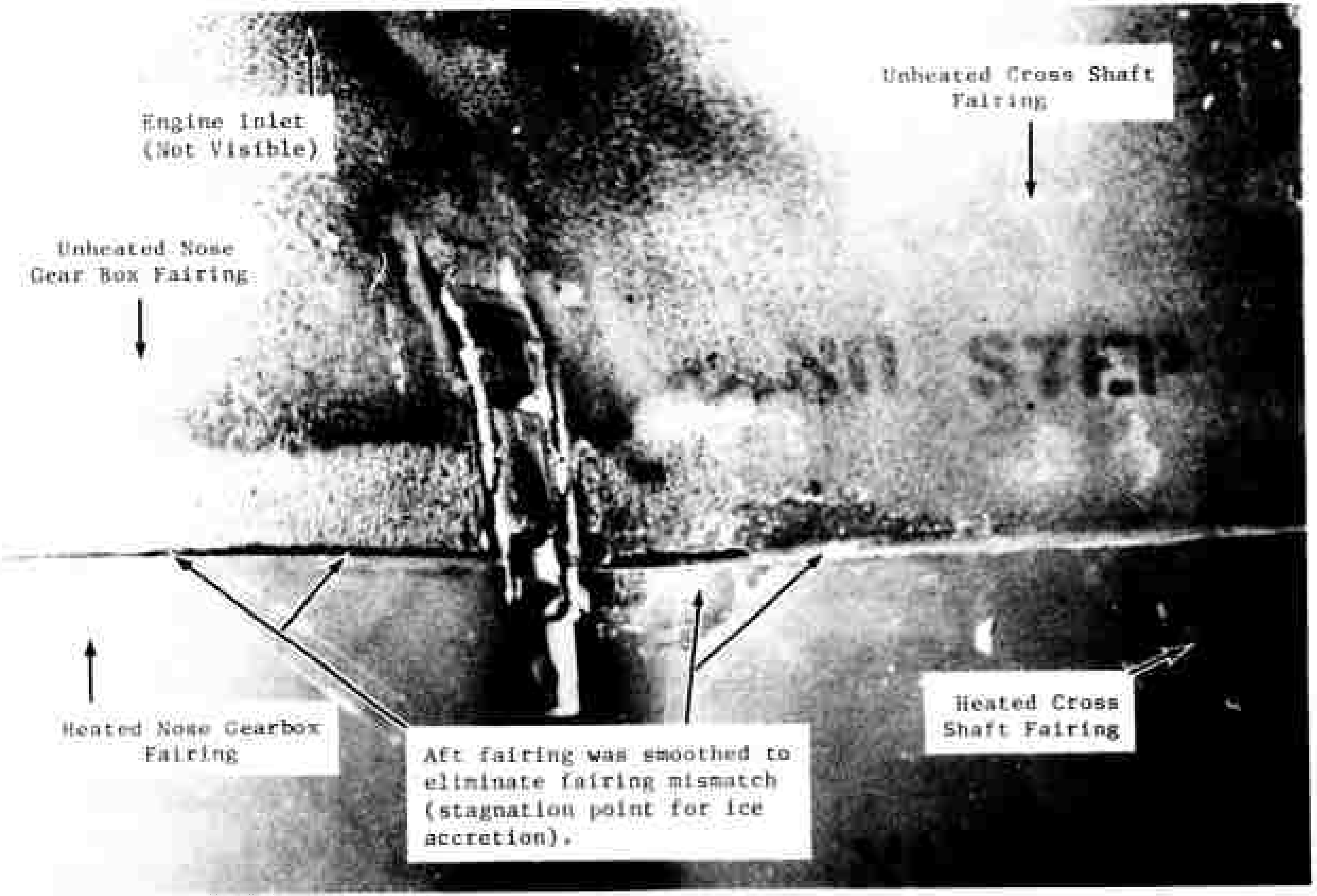


Photo 12. AH-64 No. 2 Engine Nose Gearbox/Cross Shaft Fairing - Top View

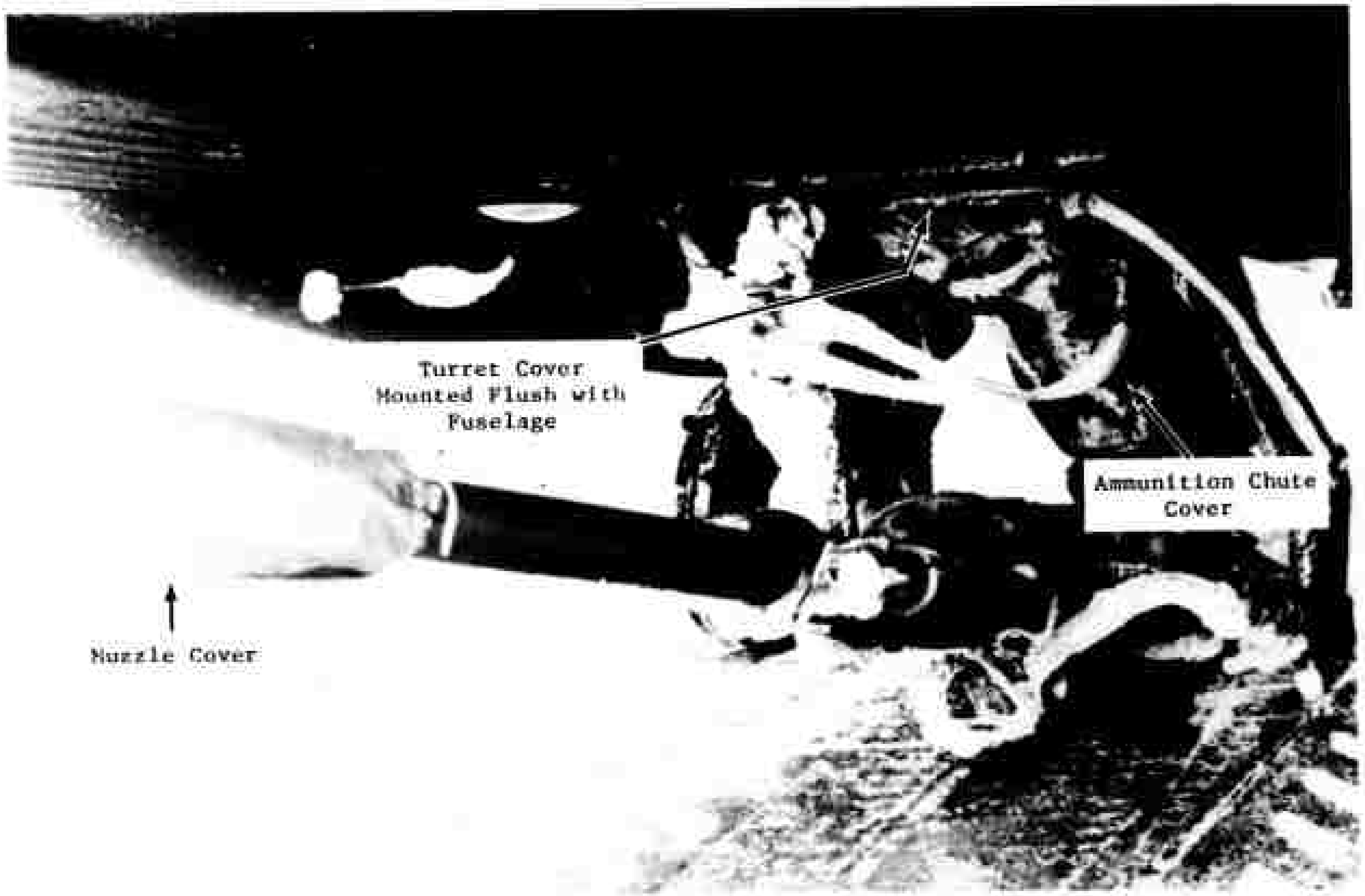


Photo 13. AH-64 30mm Area Weapon on Fuselage Underside

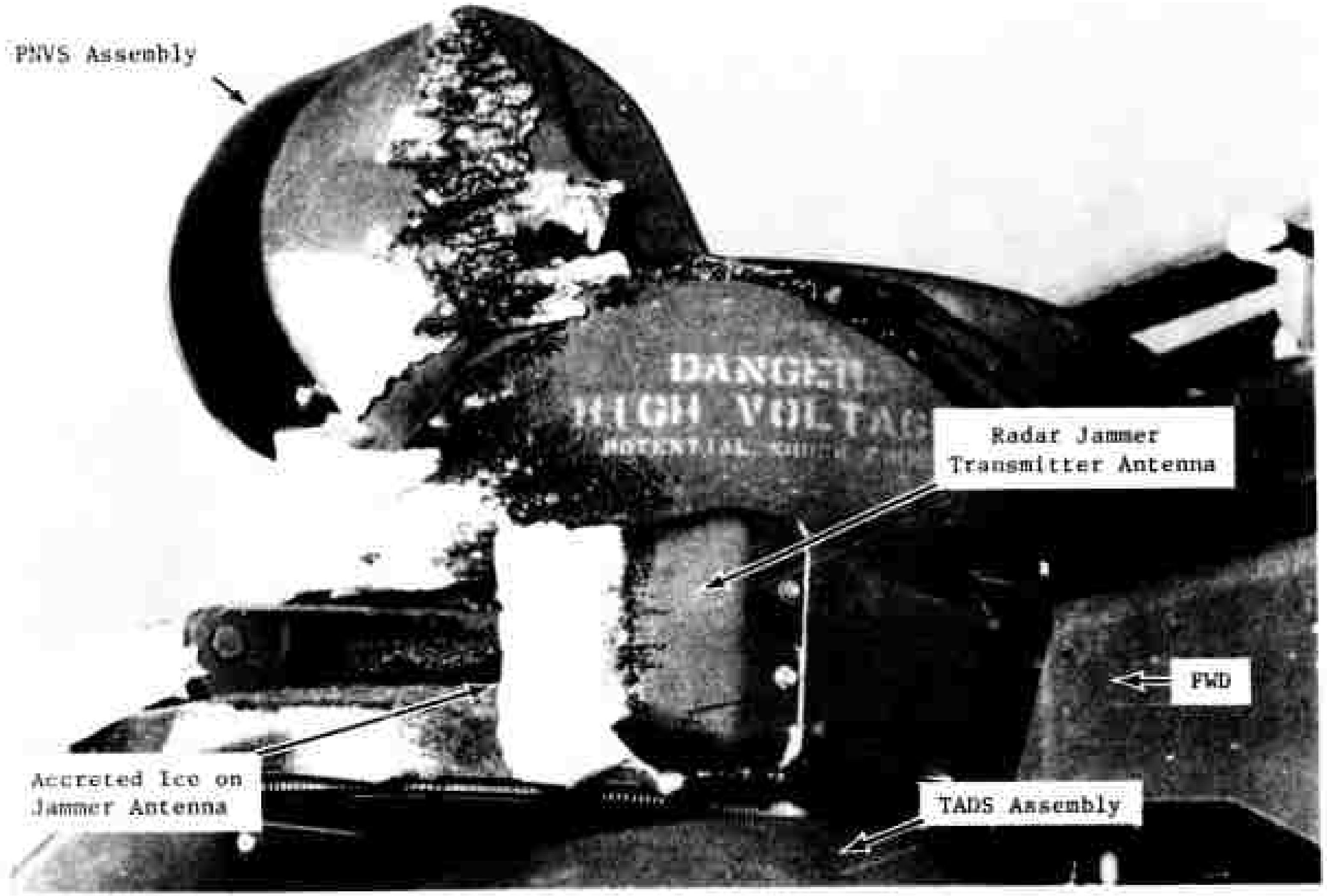


Photo 14. AH-64 TADS/PNVS - Left Front View

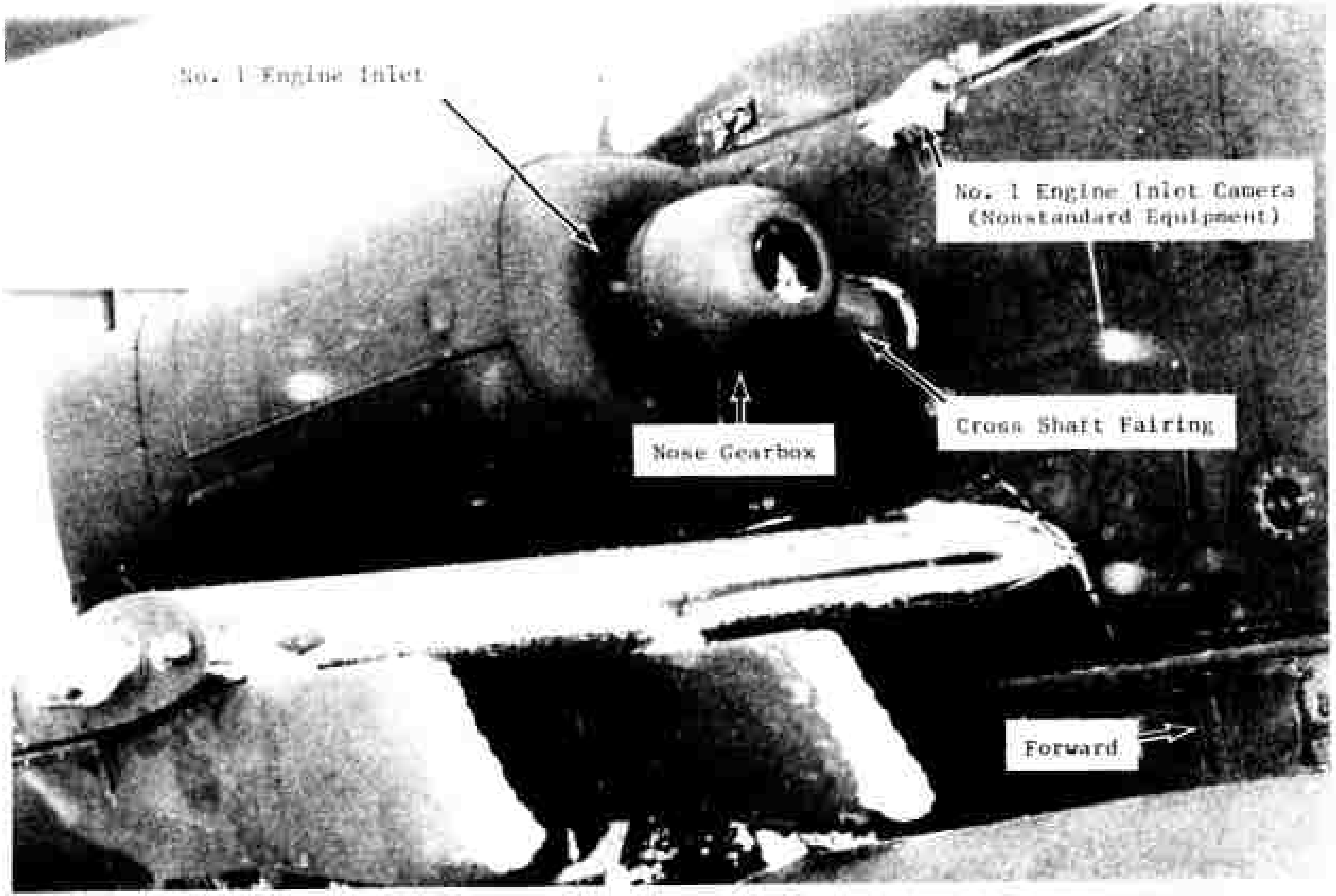


Photo 15. AH-64 Fuselage - Right Side View

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