FIRST ARTICLE PREPRODUCTION TESTS OF THE AH-64A HELICOPTER

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FINAL REPORT

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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
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First article preproduction flight testing was conducted in Mesa, Arizona (elevation 1387 feet) between 6 August and 29 August 1984. Twenty test flights were conducted for a total of 32.2 hours (22.1 hours productive). Level flight performance and handling qualities tests were conducted to verify that the production aircraft characteristics are the same as those of prototype YAH-64 as tested in Airworthiness and Flight Characteristics (Part 3) tests. The level flight performance and handling qualities of the production...
aircraft were not significantly different from those of the prototype. Additionally, the performance data presented in the operator's manual is a good representation of the production aircraft performance. One enhancing characteristic was found (the automatic contingency power feature of the T700-GE-701 engine). Two deficiencies were identified (the false indication of engine failures which was previously reported and the poor engine/airframe response characteristics which was previously reported as a shortcoming). Four shortcomings of the prototype aircraft have been corrected in the production aircraft while 13 remain. Additionally, 13 new shortcomings were found.
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SUBJECT: ERRATA Sheet, USAAEFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter.

1. Paragraph 2c of the AVSCOM Position Letter (located just after Table of Contents) was left out of the printed report. That paragraph should read as follows:
   c. Paragraph 60a - The 4/rev vertical vibration has been reported in several previous reports, and the Directorate maintains its concurrence in the shortcoming.

2. This ERRATA Sheet should be included in all copies of the subject report.
SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter

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1. The purpose of this letter is to establish the Directorate for Engineering position on the subject report. The objectives of this test were to obtain helicopter handling qualities and performance data on the first production AH-64A and compare these data to those obtained during previous tests of the prototype AH-64 helicopter.

2. This Directorate agrees with the report conclusions and recommendations, with the exceptions identified herein. Conclusions and recommendations are discussed by paragraph as indicated.

   a. Paragraph 59a - The contractor has developed a correction to the false indications of an engine failure. Engine torque is now an input signal to the engine out warning box along with engine rpm. This new software has been bench tested and flight tested satisfactorily. These actions correct this deficiency.

   b. Paragraph 59b - The undesirable engine/airframe response was previously reported under AEFA Project No. 80-17-3 as a shortcoming; however, at that time the full effects on NOE flight were not realized. This Directorate now agrees that the reported characteristic was a deficiency. In an effort to correct this deficiency, the engine and airframe contractors made modifications to the load demand spindle (LDS) bellcrank, engine electronic control unit (ECU), hydromecharlcal unit (HMU), and added a collective anticipator system. These modifications were flight tested by AEFA pilots subsequent to this report and have satisfactorily corrected this deficiency. A formal report of the correction and flight evaluation is forthcoming.

   d. Paragraph 60b - This Directorate agrees that the restricted pilot's field-of-view meets the definition of a shortcoming; however, this item is more truly the subject of the operational evaluation. It is important that
AMSAV-E
SUBJECT: Directorate for Engineering Position on the Final Report of USAAFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter

Qualitative assessments of the helicopter be made, but the degree of acceptability should remain with the operational evaluators.

e. Paragraph 60c - Access to the pilot's #1 fire handle will be improved by eliminating the offending floodlight with the incorporation of blue-green lighting.

f. Paragraph 60d - This directorate agrees that inconsistent HARS alignment accuracy is a shortcoming. HARS alignment accuracy has been demonstrated to be within specification requirements, however, the AAH mission requires dependable navigation, i.e. consistent alignment accuracy. The faults cited here could have been due to hardware or software errors. Improvements in these areas are currently being investigated by McDonnell Douglas Helicopters.

g. Paragraph 60e - The engine out warning (EOW) box circuitry used during these tests has been modified to prevent the high rotor speed activation. The new modified EOW box has been evaluated during subsequent flights and satisfactorily corrects this shortcoming.

h. Paragraph 60f - The Government has directed a split light on the fuel control panel to show both (1) a positive indication of fuel transfer and (2) a positive indication of crossfeed in progress. The contractor is preparing an ECP for incorporation of this fix. Upon incorporation, this ECP is expected to correct this shortcoming.

i. Paragraph 60g - AVSCOM agrees that the design of the pilot's cyclic grip is a shortcoming. The contractor is investigating relocating the HAS switch to the cyclic grip. The Government anticipates an ECP for relocation of the HAS switch upon successful completion of this investigation.

j. Paragraph 60h - The oil leakage that results in this accumulation is being remedied in production by changes to the sealant procedures and using a different "O" ring.

k. Paragraph 60i - The grease expulsion from the intermediate gearbox is being corrected by adding a hole in the baffle. An 80 hour test program has demonstrated the effectiveness of this fix. An ECP has been approved by the Government for incorporation in all aircraft, including retrofit. Additionally, a letter has been issued notifying users that these units must be shipped and handled in an upright position.

l. Paragraph 60j - This Directorate agrees that the inconsistent switch labeling between pilot and CPG fuel panels is a shortcoming and the Government has initiated action on this item. We also agree that the design of the pilot fuel control panel is a shortcoming. A redesign of the fuel control panel was agreed to by the Army and the contractor that called for the crossfeed and fuel ON/OFF switches to be of different design. While these new switches are different in appearance, they are indistinguishable by feel, especially while wearing gloves. The redesign efforts in this area are continuing.
m. Paragraph 60k - This Directorate does not agree that this is a shortcoming. When the longitudinal trim system is activated during commanded attitude changes with attitude hold ON the SAS system recenters the SAS longitudinal actuator. This is required for the attitude hold to operate and it operates in accordance with the system design. No action is planned to change the system operation.

n. Paragraph 60l - AVSCOM agrees with this finding and the main rotor system plunger assembly has been deleted from the production helicopter. A visual inspection procedure has been developed and is being used instead.

o. Paragraph 60m - Slack in the parking brake actuating cable allows the handle to be pushed in without releasing the parking brake. The Government is reviewing alternate designs and will request an ECP to correct this shortcoming.

p. Paragraph 60n - AVSCOM does not agree that this is a shortcoming. The crashworthy design features preclude additional longitudinal adjustment or seat padding of the pilot's seat. A lumbar support has been added to increase support but no additional action is planned.

q. Paragraph 60o - The total concept of the anti-ice and caution-warning panels is being reviewed by the Government and the contractor, with the intent of redesigning both. These efforts are intended to resolve this shortcoming.

r. Paragraph 60p - The location of the pilot engine control quadrant is not considered a shortcoming. The layout and location of this quadrant was carefully evaluated during design reviews and mock-up reviews. The quadrant is located as close as allowed with required collective control clearance. No action is planned.

s. Paragraph 60q - AVSCOM agrees that this is a shortcoming and the Program Management Office (PMO) has directed correction by the contractor. A redesign is currently being bench tested by the contractor. Army pilot evaluation will follow the contractor's tests.

t. Paragraph 60r - The location of the tailwheel lock/unlock light is not considered a shortcoming. The location of this light was evaluated during Government design and mock-up reviews and is acceptable in its present location.

u. Paragraph 60s - AVSCOM agrees that this is a shortcoming. The engine power lever assembly in the test helicopter had been incorrectly assembled and the cables were damaged during installation. The engine power lever control cable assembly has been redesigned to avoid this situation in the future. The redesign has been evaluated and approved by Army pilots and will be incorporated in the production helicopter (ECP 694).
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SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA Project No. 84-10, First Article Preproduction Tests of the AH-64A Helicopter

v. Paragraph 60t - The background noise has been eliminated through redesign and improvement of the grounding in the strobe power supply. The production test procedures have also been modified to insure optimum ADF alignment. No further action is planned.

w. Paragraph 60u - AVSCOM agrees that the lack of VDU roll symbology to show up/down direction is a shortcoming. The Government is initiating action to change the VDU display.

x. Paragraph 60v - An ECP (ECP 421R1) has been initiated to provide independent control of the CPG Marconi indicators.

v. Paragraph 60w - AVSCOM agrees that poor readability of the AN/APX-100(V) transponder code is a shortcoming; however, the design and location have been optimized for the crew station and no further action will be taken.

z. Paragraph 60x - The production helicopters number 52 and subsequent engine oil access doors will be fitted with a wing type camlock fastener to replace the slotted screw, thereby correcting this shortcoming. This change will not be retrofitted to other helicopters.

aa. Paragraph 60y - The caution-warning panel is in review by both AVSCOM and the contractor for redesign. This effort includes software changes to the rocket control panel. These efforts are expected to resolve this shortcoming.

bb. Paragraph 60z - The EMI in the caution-warning panel has been corrected through redesign of the Marconi test circuit. However, the indications that accompanied selection of number one and number two generator test switches were the result of incorrect procedure. The operator's manual (-10) is being corrected to avoid future occurrences. The pylon articulation has been corrected by ECP 707, which modified the Textron ESCS logic.

FOR THE COMMANDER:

[Signature]
Daniel M. McEwen
Director of Engineering
INTRODUCTION

BACKGROUND

1. The US Army contracted with Hughes Helicopters, Inc. (HHI) for production of the AH-64A Apache, Advanced Attack Helicopter. First article preproduction testing was required by the Production Airworthiness and Qualification Program contained in the system specification (ref 1, app A). The US Army Aviation Engineering Flight Activity (USAAEFA) has conducted several tests of the prototype YAH-64 (refs 2 through 9). In March 1984, the US Army Aviation Systems Command (AVSCOM) requested (ref 10) USAAEFA to conduct an evaluation of the first production aircraft to verify that the performance and handling qualities were not significantly different from those of the prototype.

TEST OBJECTIVES

2. The objective of this First Article Preproduction Test was to obtain quantitative flying qualities and performance data on the first article AH-64A. Additionally, a comparison was made of data obtained during this test to that gathered during the Airworthiness and Flight Characteristics Test, Part 3 (A&FC 3) (ref 9).

DESCRIPTION

3. The AH-64A (USA S/N 82-23355) is a two-place, tandem-seat, twin engine helicopter with four-bladed main and antitorque rotors and conventional wheel landing gear. The helicopter 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 (NOE)/Approach. A dummy 30mm gun, the Target Acquisition and Designation System (TADS), and the Pilot Night Vision System (PNVS) were installed on the aircraft. The helicopter has a wing with two store pylons (shorter than those on the YAH-64) on each side for carrying Hellfire missiles, 2.75-inch folding fin aerial rockets, or external fuel tanks. The test aircraft differed from a production aircraft in several respects. It was fully instrumented for handling qualities, performance, and structural tests. No fire control computer (FCC) was installed. The backup bus controller (BBC) provides some of the functions of the FCC when the FCC is not operational. However, to use the full capability of the Digital Automatic Stabilization Equipment (DASE) the FCC must be operational on a production aircraft. For this evaluation, the software in the BBC was modified to provide full DASE capability. A test airspeed
boom mounted on the nose of the aircraft and a round instrumenta-
tion canister mounted on the top of the main rotor mast were
items not found on production aircraft. Further description of
the helicopter may be found in appendix B of this report, in the
system specification (ref 1, app A), and in the operator's manual
(ref 11).

TEST SCOPE

4. Flight testing was conducted in Mesa, Arizona (elevation
1387 feet) between 6 August and 29 August 1984. Twenty test
flights were conducted for a total of 32.2 hours (22.1 hours
productive). HHI installed, calibrated, and maintained the test
instrumentation and performed all aircraft maintenance during the
test. Flight restrictions contained in the Airworthiness Release
issued by AVSCOM (ref 12) and the operator's manual were observed
during this evaluation. Test conditions are presented in table 1.
Where possible, flight test data were compared with the system
specification and A&FC 3. Performance data were also compared to
the data presented in the operator's manual.

TEST METHODOLOGY

5. Test methods are briefly discussed in the Results and Discus-
sion section of this report. Flight test data were obtained
from calibrated test instrumentation and were recorded on magnetic
tape. Real time telemetry was used to monitor selected parameters
throughout the flight test program. A detailed listing of the
test instrumentation is contained in appendix C. Test techniques
and data analysis methods are described in appendix D. The
Handling Qualities Rating Scale (HQRS) and the Vibration Rating
Scale (VRS), shown in appendix D, were used to quantify pilot
comments.
<table>
<thead>
<tr>
<th>Test</th>
<th>Calibrated Airspeed (knots)</th>
<th>Density Altitude (feet)</th>
<th>Gross Weight (lb)</th>
<th>Longitudinal Center of gravity (FS)</th>
<th>Remarks</th>
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<tbody>
<tr>
<td>Level flight Performance</td>
<td>41 to 139</td>
<td>7530</td>
<td>14,560</td>
<td>206.2</td>
<td>$C_T = 0.007991$</td>
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<td></td>
<td></td>
<td>9620</td>
<td>14,490</td>
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<td></td>
<td>10,800</td>
<td>14,770</td>
<td>206.7</td>
<td>$C_T = 0.008978$</td>
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<td></td>
<td>13,920</td>
<td>14,650</td>
<td>206.2</td>
<td>$C_T = 0.009525$</td>
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<td></td>
<td></td>
<td>13,660</td>
<td>13,920</td>
<td>206.1</td>
<td>$C_T = 0.010002$</td>
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<td>Control Positions in Trimmed Forward Flight</td>
<td>42 to 177</td>
<td>10,800</td>
<td></td>
<td></td>
<td>Level and Diving Flight</td>
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<td></td>
<td>59 to 133</td>
<td>7430</td>
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<td></td>
<td>Autorotational Descent</td>
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<tr>
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<td>0</td>
<td>70</td>
<td>7140</td>
<td></td>
<td>Level Flight</td>
</tr>
<tr>
<td></td>
<td>0</td>
<td>139</td>
<td>6880</td>
<td></td>
<td>Level Flight</td>
</tr>
<tr>
<td></td>
<td>139</td>
<td>7440</td>
<td></td>
<td></td>
<td>Autorotational Descent</td>
</tr>
<tr>
<td></td>
<td>139</td>
<td>7340</td>
<td></td>
<td></td>
<td>Level Flight</td>
</tr>
<tr>
<td>Static Lateral-Directional Stability</td>
<td>0</td>
<td>70</td>
<td>7250</td>
<td></td>
<td>Level Flight</td>
</tr>
<tr>
<td></td>
<td>0</td>
<td>139</td>
<td>7000</td>
<td></td>
<td>Level Flight</td>
</tr>
<tr>
<td></td>
<td>139</td>
<td>7250</td>
<td></td>
<td></td>
<td>Autorotational Descent</td>
</tr>
<tr>
<td></td>
<td>81</td>
<td>6630</td>
<td></td>
<td></td>
<td>Left and right turns pull ups and pushovers</td>
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<tr>
<td>Maneuvering Stability</td>
<td>135</td>
<td>7170</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Dynamic Stability</td>
<td>70 and 139</td>
<td>7100</td>
<td>14,700</td>
<td>206.4 (AFT)</td>
<td></td>
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<tr>
<td>Power Management</td>
<td>0 to 130</td>
<td>6150</td>
<td>to 7290</td>
<td></td>
<td>Quick Stops Pull ups Recoveries from Autorotation</td>
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<tr>
<td>Instrument Flight Capability</td>
<td>0 to 140</td>
<td>2500</td>
<td>to 8000</td>
<td></td>
<td>Enroute Navigation Holding Approaches Simulated Emergencies</td>
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<tr>
<td>DASE Evaluation</td>
<td>0</td>
<td>3700</td>
<td></td>
<td></td>
<td>HAS$^5$ ON and OFF</td>
</tr>
<tr>
<td>Simulated Single Engine Failures</td>
<td>70</td>
<td>2500</td>
<td></td>
<td></td>
<td>Attitude Hold ON and OFF Level and Turning Flight</td>
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<tr>
<td></td>
<td>100, 135</td>
<td>7450</td>
<td></td>
<td></td>
<td>Hover</td>
</tr>
<tr>
<td>Vibration Characteristics</td>
<td>42 to 177</td>
<td>10,800</td>
<td></td>
<td></td>
<td>Level Flight and Dives</td>
</tr>
</tbody>
</table>

**NOTES:**

1. Hellfire configuration, 100% rotor speed, DASE ON unless otherwise noted.
2. $C_T$ = Thrust coefficient.
3. IRP = Intermediate Rated Power.
4. DASE = Digital Automatic Stabilization Equipment.
5. HAS = Hover Augmentation System.
RESULTS AND DISCUSSION

GENERAL

6. Level flight performance and handling qualities tests were conducted to verify that the production aircraft characteristics are the same as those of the prototype YAH-64 as tested in the A&FC 3 tests. The level flight performance and handling qualities of the production aircraft were not significantly different from those of the prototype. Additionally, the performance data presented in the operator's manual is a good representation of the production aircraft performance. One enhancing characteristic was found (the automatic contingency power feature of the T700-GE-701 engine). Two deficiencies were also found (the false indication of engine failures which was previously reported and the poor engine/airframe response characteristic which was previously reported as a shortcoming). Four shortcomings of the prototype aircraft have been corrected in the production aircraft while 13 remain. Additionally, 13 new shortcomings were found.

LEVEL FLIGHT PERFORMANCE

7. Level flight performance tests were conducted at the conditions of table 1 to verify that the performance data in the AH-64A operator's manual (which was based on ref 9 testing) accurately represents the production aircraft performance. The aircraft was flown at zero sideslip and a main rotor speed of 290 RPM. All level flight performance tests were conducted with a dummy 30mm chain gun in the 11 degree elevation, and 0 degree azimuth position without the ammunition chute installed. TADS and PNVS sights were in the stowed position. The Hellfire missile launchers were set at the zero degree elevation position. The environmental control subsystem (ECS) was operated at a level for pilot and copilot comfort and the stabilator was in the automatic mode. The aircraft was flown in the 8-Hellfire configuration at an aft longitudinal center of gravity (cg). Previous data were gathered at a forward cg which is a more adverse condition for level flight performance. During this test, the aircraft could not be flown at a forward cg because of a fuel quantity restriction for the forward fuel cell. Test results are presented nondimensionally in figures 1 through 3, appendix E. Dimensional test results are presented in figures 4 through 9. A list of external items, either installed on the test aircraft and not included in the system specification (ref 1, app A), or defined in reference 1 and not on the test aircraft during this evaluation is presented in table 1, appendix B. No corrections for electrical load, variable power consumed by the ECS or external configuration differences between the test and production aircraft were applied to the data in this report.
8. The maximum cruise airspeed using maximum continuous power with the aircraft in the 8-Hellfire configuration at the primary mission gross weight of 14,694 pounds and at 4000 feet pressure altitude with an ambient temperature of 35°C and 289 rotor RPM (100 percent) was determined to be 144 knots true airspeed (KTAS). During A&FC 3, it was determined that a change in cg from fuselage station (FS) 202 to FS 206 caused a drag decrease of 3.1 ft², equivalent flat plate area (based on engine shaft horsepower change). Applying the correction to the A&FC 3 data, the maximum speed is 145 KTAS, which agrees with the current results (see fig. 9, app E). Comparisons between the current results and the operator's manual also indicated good agreement. The level flight cruise performance of the AH-64A is not significantly different from that of the YAH-64 and the operator's manual cruise data accurately represents AH-64A level flight performance.

HANDLING QUALITIES

General

9. AH-64A handling qualities were evaluated with the production DASE software program No. 7-211D00005-11 (-11 version) except where noted. Additional testing included an evaluation of instrument flight capability in light turbulence and a night evaluation. System specification requirements were evaluated, where applicable. Both quantitative data and qualitative pilot comments were recorded during these tests. Tests were conducted in the 8-Hellfire configuration at the conditions specified in table 1.

Control System Characteristics

10. The control system mechanical characteristics were evaluated on the ground with external hydraulic and electric power applied to the aircraft and the rotors stopped. All measurements were taken at the pilot station with a digital force gauge and onboard control position indicators. Tests were performed with the trim feel system ON and OFF. The longitudinal control system was evaluated with the copilot/gunner (CPG) cyclic stick extended and retracted which affected the mass balance. Results were quantitatively verified with the auxiliary power unit ON (rotors static) and qualitatively verified in flight. Data are presented in figures 10 through 14, appendix E and a summary of control system mechanical characteristics is presented in table 2. The control system characteristics in the longitudinal and lateral axes appeared to be degraded compared to previous evaluations (refs 5 and 8, app A), due to the increased breakout plus friction forces.
Table 2. Control System Mechanical Characteristics

<table>
<thead>
<tr>
<th>Test Parameter</th>
<th>Longitudinal (CPG Stick Retracted)</th>
<th>Longitudinal (CPG Stick Extended)</th>
<th>Lateral</th>
<th>Directional</th>
</tr>
</thead>
<tbody>
<tr>
<td>Breakout Force (Plus Friction) (lb)</td>
<td>2.6 FWD, 3.4 AFT</td>
<td>3.0 FWD, 3.0 AFT</td>
<td>2.5 left, 2.0 right</td>
<td>9.0 left, 6.0 right</td>
</tr>
<tr>
<td>Full Control Travel (in.)</td>
<td>10.5</td>
<td>10.5</td>
<td>8.45</td>
<td>6.06</td>
</tr>
<tr>
<td>Control Oscillation</td>
<td>None</td>
<td>None</td>
<td>None</td>
<td>None</td>
</tr>
<tr>
<td>Freeplay</td>
<td>Negligible</td>
<td>Negligible</td>
<td>Negligible</td>
<td>Negligible</td>
</tr>
<tr>
<td>Force to Move Control 0.5 in. from Trim (lb)</td>
<td>3.3 FWD, 3.95 AFT</td>
<td>3.8 FWD, 3.4 AFT</td>
<td>2.9 left, 2.3 right</td>
<td>9.5 left, 7.5 right</td>
</tr>
<tr>
<td>Limit Control Force (lb)</td>
<td>17.0 FWD, 18.0 AFT</td>
<td>11.5 FWD, 17.6 AFT</td>
<td>9.1 left, 8.3 right</td>
<td>N/A</td>
</tr>
<tr>
<td>Control Centering</td>
<td>Positive</td>
<td>Positive</td>
<td>Positive</td>
<td>Positive</td>
</tr>
<tr>
<td>Trim Control Displacement Band (in.)</td>
<td>0.3</td>
<td>0.4</td>
<td>0.3</td>
<td>1.0</td>
</tr>
<tr>
<td>Control Jump</td>
<td>Negligible</td>
<td>Negligible</td>
<td>Negligible</td>
<td>Negligible</td>
</tr>
<tr>
<td>Control Forces Trimmable to Zero</td>
<td>No</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>Force Gradient (lb/in.)</td>
<td>1.45 FWD, 1.54 AFT</td>
<td>1.35 FWD, 1.45 AFT</td>
<td>0.79 left, 0.68 right</td>
<td>1.7 left, 2.6 right</td>
</tr>
<tr>
<td>Average Friction Band (lb)</td>
<td>1.5 FWD, 3.8 AFT</td>
<td>1.5 FWD, 0.5 AFT</td>
<td>2.6 left, 2.0 right</td>
<td>10.6 left, 10.2 right</td>
</tr>
</tbody>
</table>

NOTES:
1Rotoors static, primary hydraulic system and electrical power applied.
2Copilot gunner cyclic control stick.
3Force gradient increases sharply during last 3/8 of directional movement in both directions.
and the large mass imbalance in the longitudinal axis with the CPG cyclic stick retracted.

11. Longitudinal forces were measured at the lower third of the cyclic grip. With trim feel ON the control centering was positive with a small trim control displacement band (0.3 in.). In flight with the CPG cyclic stick retracted (normal mission configuration) the longitudinal control system showed stick mass imbalance in that slight aft pressure on the cyclic was required to maintain the desired trim attitude (para 28). On the ground with force trim OFF and the cyclic stick positioned 3.5 in. or less from the forward stop, the cyclic would fall forward. While airborne with force trim OFF the cyclic stick would rapidly fall toward the forward stop regardless of initial position as indicated in figure 15, appendix E. With the CPG cyclic stick extended mass imbalance was negligible. The longitudinal breakout force was high and demonstrated asymmetry (2.6 lb forward, 3.4 lb aft) with the CPG stick retracted. Although the asymmetry was not markedly objectionable to the pilot, the high cyclic control forces required constant retrimming during maneuvering flight. There was no mechanical coupling. The longitudinal control system characteristics failed to meet the following requirements of reference 1, appendix A.

a. 10.3.2.1.2 - the longitudinal breakout force (plus friction) in the fore and aft direction with the CPG cyclic stick retracted exceeded the maximum 10% asymmetry by 3%.

b. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the aft direction with the CPG cyclic stick retracted exceeded the 2.4 lb limit by 1.0 lb.

c. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the forward and aft directions with the CPG cyclic stick extended exceeded the 2.4 lb limit by 0.6 lb.

d. 10.3.2.5 - the longitudinal control force could not be trimmed to zero in flight with the CPG cyclic stick retracted.

12. The lateral control system forces were measured at the lower third of the cyclic grip. The lateral control system exhibited positive centering and a small (0.3 in.) trim control displacement band. The breakout force (plus friction) was higher left (2.5 lb) than right (2.0 lb) but the asymmetry was not noticeable to the pilot. However, during maneuvering flight and landing approaches, constant retrimming was required by the pilot to reduce control forces. Trim feel OFF operation was characterized by no force gradients or control centering. In the lateral
axis, the cyclic stick did not exhibit mass imbalance. The cyclic control system demonstrated force gradient harmony between the longitudinal and lateral axes. The lateral control breakout force (plus friction) failed the requirements of reference 1, appendix A, para 10.3.2.1.3 in that the 1.5 lb limit was exceeded by 1.0 lb (left) and 0.5 lb (right).

13. The directional control system characteristics were asymmetrical but not objectionable to the pilot. Control centering was positive but a large, noticeable trim displacement band existed (0.7 in.). The force gradient was approximately linear in both directions except at the last 3/4 in. where the gradient increased substantially due to the control tubes contacting guide sleeves in the tail boom area. No undesirable control oscillations or control jump was noted in this axis. The directional control mechanical characteristics failed the following requirements of reference 1, appendix A.

a. 10.3.2.1.3 - the left directional breakout force (plus friction) exceeded the 7.0 lb limit by 2.0 lb.

b. 10.3.2.2 - the left control force versus displacement gradient was less than the minimum value of 2.0 lb/in. by 0.3 lb/in.

c. 10.3.2.2.1 - the directional force gradient was not linear throughout the limit of control travel and was concave upward instead of downward during the last 3/4 in. of control travel in each direction.

14. Collective forces were measured at the forward edge of the grip, approximately one inch behind the throttle chop control. The average breakout (plus friction) was 3.3 lb down and 2.6 lb up at the mid range of collective movement. However, the friction was not constant throughout the range of collective control travel and an occasional ratcheting effect reduced force as much as one pound during movement. The poor design of the collective pitch control friction mechanism is a shortcoming previously reported. During maneuvering flight with positive g loading, the collective control consistently would drop up to one inch with friction set at mid level (5) or lower. The collective control system failed the requirements of reference 1, para 10.3.2.1.3 in that the collective breakout (plus friction) in the up direction was lower than the minimum value of 3.0 lb by 0.4 lb.

15. The control characteristics of the power control levers (PCL) were evaluated at the upper grip of each power lever. The breakout force (plus friction) of each engine control is summarized in table 3.
Table 3. Engine Control Breakout Forces

<table>
<thead>
<tr>
<th>Test Parameter</th>
<th>Breakout (Plus Friction) (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Forward</td>
</tr>
<tr>
<td>PCL/Position</td>
<td></td>
</tr>
<tr>
<td>#1 Idle</td>
<td>6.0</td>
</tr>
<tr>
<td>#1 Mid</td>
<td>6.0</td>
</tr>
<tr>
<td>#1 Fly</td>
<td>7.5</td>
</tr>
<tr>
<td>#2 Idle</td>
<td>6.0</td>
</tr>
<tr>
<td>#2 Mid</td>
<td>9.0</td>
</tr>
<tr>
<td>#2 Fly</td>
<td>8.0</td>
</tr>
</tbody>
</table>

NOTES:
1 Friction OFF
2 Power Control Lever (PCL)

With the engine control friction ON, the average breakout plus friction in either direction was approximately 17 lb for the #1 PCL and 18 lb for the #2 PCL. Precise engine power lever movement was difficult even with friction OFF and remains a shortcoming previously reported. The excessive control force required to operate the engine power control levers failed to meet the requirements of para 10.3.3.2.3 of reference 1 by up to 2.5 lb (friction OFF) and 10.5 lb (friction ON).

Control Positions in Trimmed Forward Flight

16. Control positions in forward flight were evaluated in level flight, intermediate rated power (IRP) climb, and autorotative descent at the conditions presented in table 1. Data are presented in figures 16 and 17, appendix E. No variation of longitudinal control position with calibrated airspeed was found between 40 and 60 knots calibrated airspeed (KCAS) in level flight. Above 80 KCAS the gradient increased to 0.03 in./knot up to maximum level flight airspeed at intermediate rated power (V\text{\textsubscript{H}}). The variation of lateral control position throughout the envelope was less than 1/4 in. Pitch attitude variation was essentially linear with increasing nose-down attitude with increased airspeed. In IRP climbs longitudinal control position was constant between 60 and 87 KCAS. Over 87 KCAS the control position moved forward with increased airspeed. In descent, longitudinal control position was constant between 60 and 75 KCAS, moved forward between 75 and 100 KCAS, and was approximately constant above 100 KCAS. The variation of lateral and directional control position was minimal throughout the airspeed range tested. Pitch attitude decreased slightly with airspeed in climbs and remained approximately constant in descents. Adequate control margins were available.
throughout the conditions tested. The control positions in
trimmed forward flight are satisfactory and met the requirements
of paragraphs 10.3.3.1.2, 10.3.4.1.1, and 10.3.5.2.4 of the
system specification (ref 1, app A).

**Static Longitudinal Stability**

17. The static longitudinal stability was evaluated in level
flight, IRP climbs, and autorotative descents at the conditions
presented in table 1. The collective was held fixed while airspeed
was varied 20 knots in 5-knot increments about trim with the
cyclic only. Data are presented in figures 18 through 21,
appendix E. For the low-speed trim point, approximately 70 KCAS,
the longitudinal control position gradient was essentially
neutral within 12 knots from trim varying less than 0.1 in.
At high speed (139 KCAS) the stick-fixed longitudinal stability
was weakly positive as indicated by the shallow control position
gradient. At both airspeeds, stick-free static longitudinal
stability was neutral within 10 knots of trim and slightly posi-
tive outside this range as indicated by the qualitative variation
of longitudinal control force with airspeed (increasing forward
force with increased airspeed). In IRP climbs at 81 KCAS, the
longitudinal control position gradient was essentially neutral
within 15 knots of trim airspeed. In descents at 81 KCAS the
gradient was positive within 10 knots of trim airspeed. Although
the neutral to weak longitudinal stability contributed to poor
trimmability and increased pilot workload in simulated instrument
meteorological conditions (IMC), as mentioned in paragraph 28 and
in previous evaluations, it is satisfactory for attack helicopters
operating in visual conditions. The static longitudinal stability
failed to meet the requirements of paragraph 10.3.4.1 of the
system specification (ref 1), in that the variation of longi-
tudinal control position with airspeed was neutral in level
flight at a trim airspeed of 70 KCAS and in IRP climb at 81 KCAS.

**Static Lateral-Directional Stability**

18. Static lateral-directional stability was evaluated in level
flight, IRP climbs, and descents at the conditions presented in
table 1. The collective control was held constant and sideslip
angle was varied in 5-degree increments (left and right) while
maintaining constant airspeed and heading. Data are presented in
figures 22 through 25, appendix E. At all conditions tested
the aircraft exhibited positive directional stability (as indicat-
ed by increased left directional control with increased right
sideslip). Positive dihedral effect (as indicated by increased
right lateral control with increased right sideslip) was also
exhibited at all conditions. Sideforce characteristics were weak
at 70 KCAS in level flight (approximately 0.14 deg roll attitude per deg of sideslip). At 139 KCAS in level flight sideforce characteristics were noticeably higher (approximately 1.0 deg roll attitude per deg of sideslip). The static lateral-directional stability is satisfactory and met the requirements of paragraphs 10.3.5.1.5, 10.3.5.1.6, and 10.3.5.1.7 of the system specification (ref 1, app A).

Maneuvering Stability

19. Maneuvering stability was evaluated in constant airspeed ball-centered turns and in pull ups and pushovers at the conditions specified in table 1. Data are presented in figure 26, appendix E. Maneuvering stability, as indicated by the variation of longitudinal control position and force with normal acceleration was positive (aft control movement and increasing aft force with increasing load factor) at 135 KCAS. The helicopter attitude was predictable and load factor could be quickly and easily commanded by the pilot. However, in constant airspeed steady turns of more than 60 deg of roll attitude, airspeed (+10 kt), bank angle (+5 deg), and pitch attitude (+10 deg) were difficult to maintain in that constant control inputs in all axes (+1/2 in) were required. During pullups at the never exceed airspeed with IRP the aircraft tended to "dig in" even with small longitudinal inputs. With a small power reduction this characteristic was eliminated. The maneuvering stability is satisfactory and met the requirements of paragraph 10.3.6 of the system specification (ref 1, app A) with the exception of 10.3.6.2 which was not evaluated.

Dynamic Stability

20. The longitudinal long-term dynamic stability was evaluated with the test DASE hardover computer installed (-11 software) at the conditions in table 1. Aircraft motion was induced by displacing longitudinal cyclic from trim and increasing or decreasing airspeed by 10 knots indicated airspeed (KIAS) then returning the cyclic slowly to trim. All controls were then held fixed until recovery was initiated. Tests were conducted with DASE ON (including attitude hold ON and OFF) and DASE OFF. Time history data are presented in figures 27 through 32, appendix E. With attitude hold ON, the response to an airspeed displacement was essentially deadbeat with the aircraft returning to the trim airspeed and attitude. With DASE ON and attitude hold OFF, the long-term response was dynamically unstable with a period of approximately 67 seconds at a trim airspeed of 60 KIAS (boom) and 65 seconds at 126 KIAS (boom) (VH). The amplitude increased moderately after the first cycle and approaching attitude/airspeed
limits required the pilot to initiate recovery after one cycle at 60 KTAS (boom) and two cycles at $V_H$. With attitude hold OFF and the CPG stick retracted, the long-term response was easily excited due to the combination of weak static stability, stick mass imbalance, and the lack of absolute positive centering (fig. 32). While this characteristic is undesirable and annoying to the pilot, it was not a factor during maneuvering flight and was only noticeable in cruise. The use of attitude hold prevented attitude/airspeed excursions during the enroute phase of flight. With the DASE OFF, the long-term oscillation amplitude increased rapidly at both airspeeds requiring recovery within one cycle due to rapid pitch rates (up to 20 deg/sec) and high g onset. The recovery to controlled flight was accomplished easily in any DASE configuration. However, with DASE OFF at $V_H$, load factors up to 2.5 g and high pitch rates were experienced, which required timely and positive control inputs by the pilot to prevent exceeding aircraft limits. The DASE ON long-term response is satisfactory. The DASE OFF long-term response is satisfactory for a degraded mode. The long-term response met the requirements of paragraph 10.3.4.2 of the system specification (ref 1, app A).

21. The short-term longitudinal and lateral-directional dynamic stability were evaluated at the conditions presented in table 1. Aircraft motion was induced by 1 inch 1/2 second pulses in all axes. The controls were then held fixed until the motion subsided or recovery was initiated. Time history data are presented in figures 33 through 40, appendix E. With attitude hold ON or OFF and DASE ON or OFF, the longitudinal short-term response was essentially deadbeat. The lateral-directional response was dynamically stable and damped out within one cycle with DASE ON (attitude hold ON or OFF). DASE OFF, the lateral-directional oscillations were neutrally damped but controllable. The short-term longitudinal and lateral-directional dynamic stability met the requirements of paragraphs 10.3.4.2 and 10.3.5.3 of the system specification (ref 1, app A).

Power Management

22. The power management features were evaluated at the conditions shown in table 1. Tests included engine/airframe response characteristics, functional checks, and the contingency power capability. The functional checks and contingency power capability remained unchanged from A&FC 3 and are satisfactory and enhancing, respectively. The undesirable engine/airframe response is essentially unchanged from A&FC 3, however, due to the additional poor engine/rotor matching characteristics noted during pull up maneuvers the previously noted shortcoming has been downgraded to a deficiency.
23. The engine/airframe response tests included power recoveries from autorotation, collective control reversals, and various mission representative maneuvers (quick stops, rapid lateral control inputs, and pull ups). Time history data are presented in figures 41 through 47, appendix E. Following collective applications from below 6% torque during recoveries from autorotations and quickstops, the most common response noted was a main rotor speed droop accompanied by activation of the low main rotor speed warning and subsequent engine/airframe oscillations (figs. 41 and 42). Main rotor speed varied from 94 to 104 percent and oscillations damped in approximately 12 seconds over four cycles. Significant engine/airframe response was also noted during the recovery from pull ups from below one g (symmetrical pull ups). When initiating this maneuver with the collective fixed at above 90% gas producer speed \( (\text{Ng}) \) (approximately 40% torque), no objectionable adverse engine/airframe response was noted (figs. 43 and 44). When initiating the same maneuver from below this initial power condition (40% torque), main rotor speed droop and activation of the low main rotor warning was observed accompanied by engine/airframe oscillations (figs. 45 and 46). The oscillations damped in approximately 15 seconds and 3 cycles. Main rotor speed varied from 90 to 105 percent (maximum transient limit 104 percent) and yaw rates up to 8 deg/sec in each direction were noted. Initial trim airspeed prior to the pull up maneuver did not appear to be a major factor in the subsequent response. Any down collective movement by the pilot during the pull up greatly increased main rotor overspeed and during the recovery phase significant droop generally occurred. Figure 47 is a representative time history of a 2.2 g pull up initiated at 120 KIAS (boom) and approximately 35% torque. During the pull up, the collective was unintentionally lowered 1.8 in. from the initial trim position of 5.1 in. The subsequent main rotor speed reached 114% during g onset and then drooped to 88% accompanied by the low main rotor speed warning and both ENG OUT warning lights (para 36). Maintaining simultaneous heading (+5 deg), roll attitude (+5 deg), and rotor speed (94 to 104%) tolerances required in the NOE environment were not possible even with large control inputs in all axes (HQRS 7). The engine/airframe responses observed during this test will be most distracting during NOE or contour flight while performing recoveries from low power descents, quickstops, and pull up type evasive maneuvers where ground clearance and power required are critical. The pilot will be required to redirect his attention inside the cockpit to compensate for the main rotor speed fluctuations and airframe oscillations, resulting in the degradation of the masking and maneuvering capability of the aircraft. This problem will severely reduce mission effectiveness and may result in the helicopter unintentionally striking the ground (with loss of...
aircraft and crew) during rapid descent recoveries when threat avoidance or combat maneuvering tactics are employed. The undesirable engine/airframe response during power applications from a zero torque condition and during pull ups from below 40% torque at less than one g is a deficiency which should be corrected prior to operational use of the helicopter.

24. During the evaluation of mission type maneuvers, it was noted that rapid left lateral inputs larger than 2 in. produced large torque spikes (more than 30% above trim power) (fig. 48, app E). This torque spike was more pronounced at high forward airspeeds (near $V_H$) at high power settings. The dual engine torque transient limit is 115% and could easily be exceeded during evasive left rolling maneuvers while operating above 85% torque. The following CAUTION should be placed in the operator's manual.

**CAUTION**

Rapid left roll inputs greater than 2 inches could exceed dual engine transient torque limits when operating above a trimmed torque setting of more than 85%.

**Instrument Flight Capability**

25. A limited IMC evaluation was performed in daytime with light turbulence and at night with calm air conditions. Turbulence reporting criteria are defined in table 1, appendix D. Maneuvers performed included climbs, descents, climbing and descending turns, changing airspeed, retrimming in level flight and turns, simulated instrument approaches and instrument takeoffs. IMC conditions were simulated using a helmet mounted hood. The pilot's Visual Display Unit (VDU) (fig. A) was used as the primary flight reference to evaluate the symbology available to the pilot through the Integrated Helmet and Display Sighting System. The VDU provided pitch and roll attitude, vertical speed indicator (VSI) with digital altitude readout below 1500 ft above ground level, aircraft heading, torque, and true airspeed information. Aircraft flight and navigation indicators were used to provide other information as required and to confirm VDU information. The skid/slip ball on the VDU display was inoperative. The pitch attitude reference was not incremented and provided only trend information. Additionally, the VSI and VDU display showed rate of climb and descent approximately 50% greater than the pilot's VSI. The automatic direction finding (ADF) navigation system failed to give reliable indications when the aircraft was more than 3 miles from the station. Furthermore, the aural
Figure A. Visual Display Unit (VDU)
Identification on the ADF receiver was unreadable due to loud static in the background. The lack of reliable audio signal and azimuth information from the AN/ARN 89B ADF set is a shortcoming.

26. Instrument approaches were executed using the Precision Approach Radar at Luke Air Force Base, Arizona, Auxiliary Field Number One and using the Doppler to simulate automatic direction finding stations. The PNVS system was not operational in the test aircraft. Maneuvers were flown DASE ON with attitude hold ON and OFF and with DASE OFF.

27. Flight with attitude hold ON, in light turbulence required minimal pilot inputs to maintain aircraft pitch attitude within ±3 degrees of trim in light turbulence (HQRS 3). Retrimming in level flight and turns (attitude hold ON) caused the stability augmentation system (SAS) actuators to recenter resulting in uncommanded pitch and roll inputs which required the pilot to apply corrective inputs in all three axes. Several large longitudinal inputs of 1 to 1-1/2 inch were required to obtain the desired pitch attitude and airspeed (HQRS 4) due to the recentering of the SAS actuator. The pilot tended to fly against the force gradients rather than rettrim when small airspeed changes were made. Pilot workload increased significantly from calm air to light turbulence due to the increased requirement to rettrim the aircraft (particularly in pitch). The excessively large longitudinal cyclic inputs required when rettrimming, during IMC flight with attitude hold ON remain a shortcoming (previously reported in A&FC 3).

28. Flight with DASE ON and attitude hold OFF in light turbulence increased the pilot workload significantly requiring frequent rettrimming to maintain ±5 KIAS, ±10 degrees heading, and ±100 ft altitude (HQRS 5). Weak longitudinal static stability (para 17), trim control displacement band (para 11), and longitudinal stick mass imbalance (CG cyclic retracted) (para 11) resulted in poor trimmability. The cyclic tended to creep forward requiring the pilot to position the cyclic aft of the desired position to compensate. Extending the CPG cyclic to the full up position reduced this problem. The IMC flight characteristics with DASE ON and attitude hold OFF are satisfactory since this is a degraded mode of operation. The trim system failed to meet the requirements of paragraph 10.3.2.5 of the specification (ref 1, app A) in that cyclic control would not maintain the zero force position selected by the pilot. A comment should be added to the operator's manual stating "Intentional flight in IMC with attitude hold OFF is not recommended".
29. DASE OFF flight in light turbulence resulted in an extremely high pilot workload to maintain +10 KIAS due to the divergent long-term characteristics (para 20). Continuous cyclic, collective, and pedal inputs were required to execute instrument approaches within safe limits. The IMC flight characteristics with DASE OFF are satisfactory for a degraded mode of operation. DASE OFF flight met the requirements of paragraph 10.3.2.7.3 of the specification (ref 1). A change should be made to the operator's manual to prohibit intentional flight in IMC with DASE OFF.

**DASE Evaluation**

30. Tests were conducted to evaluate the production DASE software, (-11 version). The production DASE configuration is described in appendix B. Gust response was evaluated in forward flight with attitude hold ON and OFF and at a hover with the Hover Augmentation System (HAS) ON and OFF. The aircraft gust response in forward flight and hover is essentially unchanged from that reported in A&FC 3 and remains satisfactory.

31. Tests were performed to investigate aircraft response to SAS actuator recentering in three axes (simulated DASE failure). Additionally, single axis SAS actuator hardovers were evaluated using a production DASE modified to include a hardover box. Test conditions are shown in table 1.

32. Simulated three axis DASE failures were performed during forward flight (attitude hold ON and OFF) and hover (HAS ON and OFF) by depressing the DASE release switch on the pilot cyclic grip. The aircraft was stabilized in trim prior to DASE disengagement (SAS actuators recentering). All controls were held fixed for a minimum of 3 seconds following disengagement. The most abrupt response in forward flight occurred during disengagement with attitude hold ON. The aircraft was easily returned to stable level flight following a 3 second delay. The most abrupt response during disengagement at a hover was with HAS ON. The aircraft was easily returned to a stable hover following a 3 second delay. The aircraft response to simulated DASE failures in forward flight and hover is satisfactory. The aircraft response to abrupt disengagement of the automatic stabilization equipment in forward flight and hover met the requirements of paragraph 10.3.2.7.1 of the specification (ref 1, app A).

33. Single axis DASE hardover conditions were evaluated during forward flight and hover using a production DASE modified to drive the SAS actuators in the pitch, roll, and yaw axes to full travel on command. Following the hardover condition, the DASE
disengaged only in the affected axis and the respective SAS actuator recentered. All controls were held fixed for a minimum of 3 seconds following the hardover condition. During forward flight and hover, the aircraft responded more abruptly to SAS actuator recentering than the initial hardover condition. The aircraft was easily returned to stable forward flight or hover following a 3 second delay. Aircraft response to single axis hardovers in the pitch, roll, and yaw axes is satisfactory.

Simulated Single Engine Failures

34. Simulated single engine failures were evaluated by retarding one power lever to the idle position. Test conditions are shown in table 1. Collective control reduction was initiated after 2.0 seconds delay time or 1.0 second after activation of the low rotor speed warning light and audio tone for specification compliance. The low rotor speed warning system was designed to activate at 94%. During the test, the low rotor speed system activated at 96 to 97% rotor speed (para 37). A representative time history is presented in figure 49, appendix E.

35. Initial aircraft response to a simulated single engine failure was a slight left yaw. Due to this mild aircraft response, the first indication of engine failure at high power settings may often be the activation of the engine out/low rotor speed warning audio tone. A collective control delay of 1.0 second after activation of the low main rotor speed warning resulted in a droop of the operating engine power turbine speed (Np) to 89% during a simulated single engine failure at V_H. This was 5.0% below the minimum power off transient Np limit of 94%. No adverse handling qualities were observed at main rotor speeds down to 85%. Main rotor speed increased from 85% to normal operating range (98 to 100%) within 4 seconds after initial collective reduction. Aircraft response to simulated single engine failure is satisfactory. The aircraft response to simulated engine failure failed to meet the requirements of paragraph 10.3.8.1.1 of the System Specification (ref 1, app A) in that the available collective control delay time at V_H was less than 2.0 seconds.

36. The engine out warning system was evaluated at high power settings by retarding one power lever to the idle position to determine if a false indication of a dual engine failure would occur. This was a previously reported deficiency during A&FC, part 3 (ref 9). The engine out warning system was designed to activate at 89% Np speed with the power levers at the fly position. The engine out warning system was modified for this test so that the system was not deactivated when the power levers
were out of fly. At high power settings ($N_C$ speeds above 95%), indications of a single engine failure were a low main rotor speed warning (horn and light) followed shortly by illumination of both engine out warning lights as main rotor speed and $N_p$ of the operating engine drooped below 89%. This condition could result in the pilot executing the dual engine out emergency procedure during terrain flight operations and committing the aircraft to an autorotative landing when it may have been possible to establish single engine flight with the contingency power feature of the YT700-GE-701 engine. The possibility of a false indication of engine failure remains a deficiency. The engine out warning system should be modified to insure a timely warning of engine failure under all flight conditions and should not activate unless an actual engine failure has occurred.

37. The low main rotor speed warning system was evaluated to determine if the system activated at the design threshold of 94%. During this test, the low main rotor speed warning system consistently activated at 96 to 97% during simulated single engine failures. On several occasions the warning system activated at 99 to 100% rotor speed during normal flight maneuvers. Replacing the engine out warning box failed to correct activation of the warning system at excessively high rotor speed. Activation of the low main rotor speed warning system at excessively high rotor speed is a shortcoming.

VIBRATION

38. The forward flight vibration characteristics of the AH-64A were evaluated in the 8-Hellfire configuration. Quantitative data were recorded during level flight tests at the conditions shown in table 1. Quantitative vibration data are presented in figures 50 through 52, appendix E. Qualitative tests were conducted during climbs, descents, and low speed flight. Vibration assessment ratings were assigned in accordance with the VRS shown in figure 4, appendix D. Significant and objectionable 4 per revolution (4/rev) vibration accelerations (19.3 Hz) became noticeable to both crewmembers below 60 KCAS and above 110 KCAS (VRS 4). The 4/rev vibrations increased from VRS 4 to VRS 5 at 120 KCAS and VRS 6 above 140 KCAS. 4/rev vibrations were also noticeable during IRP climbs at 80 KCAS (VRS 5) and during normal approaches to a hover (VRS 5 below 35 KCAS). Vibrations observed during this evaluation were most noticeable at the pilots station as they have been during previous tests. The 4/rev vibration characteristics are essentially unchanged from those previously noted in part 3 of the A&FC report (ref 9, app A). The objectionable 4/rev vibration remains a shortcoming. Vertical vibration
characteristics failed to meet requirements of paragraph 3.2.11.5.1 of the System Specification (ref 1) in that 4/rev vibration intensities exceeded the specification limit below 80 KCAS and at 126 KCAS; the 8/rev vibration intensities exceeded 0.2g at 83 KCAS.

HUMAN FACTORS

Cockpit Evaluation

39. The pilot cockpit of the AH-64A was evaluated throughout the test program including an evaluation of cockpit design with primary emphasis on instrument displays, switch functions, control designs, and ease of operation of system controls. Pilot comfort was also considered. Uncorrected shortcomings previously reported are shown in paragraph 55.

40. Ease of system controls operation was evaluated during this test. Access to the pilot's #1 fire handle is partially obstructed by an instrument panel flood light mounted on the underside of the glare shield at the upper left side of the instrument panel. This flood light was removed from the test aircraft but is installed in current production aircraft. The position of the flood light will cause the pilot difficulty in reaching and pulling the #1 fire handle. Obstructed access to the pilot's #1 fire handle is a shortcoming.

41. When attempting fuel transfer or crossfeed operation (use fuel from one tank only) no immediate indication was available to show direction of transfer or that crossfeed operation was in progress. The primary means of monitoring fuel transfer or crossfeed operations is fuel quantity indications on the CPG digital indicator. The operator's manual (ref 11) indicates the fuel transfer light will be on during fuel transfer operations. Crew attention will easily be diverted from the fuel quantity indicators by normal crew duties allowing more than the desired amount of fuel to be transferred/crossfed and could easily result in exceeding aircraft cg limits or engine flame out (fuel starvation) when operating in a low fuel condition. The lack of a fuel transfer/crossfeed light is a shortcoming. The fuel transfer light fails to meet the requirements of paragraph 3.7.7.1.3.10(C) (ref 1) in that the light does not illuminate when fuel transfer is in progress.

42. The pilot's cyclic grip (photo 1) incorporated eight different functions activated by thumb switches. In addition, some additional functions are required to be activated simultaneously such as the weapons action switch and the trigger switch. An
Extreme reach was required for operation of the force trim release switch. A pilot with an average sized hand (glove size 9) will have difficulty in operating the switches on the cyclic and will frequently be required to reposition his hand on the cyclic grip in order to perform the necessary functions. Frequent use of the force trim release switch was required during all flight and hovering maneuvers. Furthermore, the pilot was unable to utilize the force trim release switch and radio/intercom switch simultaneously. This severely limited internal and external communications during high gain tasks requiring control retrimming. The mission pilot will prefer to fly with force trim OFF during maneuvering flight and hover. Flight with force trim OFF is less than desirable due to the stick mass imbalance and loss of the Command Augmentation System (CAS). The poor anthropometric design of the pilot's cyclic grip remains a shortcoming.

43. The pilot fuel control panel (redesigned since A&FC, Part 3) shown in photo 2, was evaluated throughout the test. Switch labeling was not consistent between front and aft cockpit in that the "CROSSFEED" switch on the pilot's panel is labeled "TK SEL" on the CPG's panel (fig. 30, app B). Engine fuel switches were difficult to see with engine power control levers in the full aft position. Normal operation of the fuel system required frequent pilot attention to the fuel control panel in order to maintain comparable fuel levels in each tank. No immediate indications were available to show direction of transfer or that crossfeed operation was in progress (see para 41). Switch identification and determination of switch position were extremely difficult during night operations. The close proximity of the fuel transfer and crossfeed switches to the engine fuel switches and similar manner of switch operation will result in a high probability of inadvertent shutdown of an engine during attempted fuel crossfeed or transfer operation. The poor design of the pilot's fuel control panel remains a shortcoming.

44. The engine anti-ice system was qualitatively evaluated during this test. On two occasions an uncommanded open anti-ice/start bleed valve, with no positive cockpit indications, resulted in reduced power available and higher operating engine turbine gas temperature (TGT) at a given power setting. The anti-ice/start bleed valve and the engine inlet anti-ice valve are spring loaded open and electrically closed by the ENGINE INLET SWITCH on the pilot's anti-ice panel. ENG 1 and ENG 2 advisory lights on the anti-ice panel are designed to illuminate when the ENG INLET switch is placed on and the electrical circuit is complete. An anti-ice caution light for each engine is designed to illuminate when an engine nose gearbox fairing overheats (120 to 180°F). The lack of a positive indication of anti-ice/start bleed valve
position is a shortcoming. The engine anti-ice system fails to meet the requirements of paragraph 3.7.5.4.1 of the System Specification (ref 1, app A) in that failure of the anti-ice/start bleed valve to close does not illuminate a caution light.

45. The VDU roll attitude reference was evaluated during night and daytime simulated instrument conditions. VDU symbology for roll attitude reference is a symmetrical dashed line (fig. A) which fails to identify upward from downward direction. During unusual attitude recovery from bank angles approaching 90 degrees, the pilot was unable to determine the appropriate direction of lateral control input required to recover the aircraft to wings level, upright flight using the VDU roll attitude reference. A standby attitude indicator is available on the instrument panel. However, the pilot will be using the integrated helmet and display sight system (IHADSS) display as the primary reference for aircraft attitude when inadvertent IMC conditions are encountered. There is a high probability of unusual aircraft attitudes during the initial entry into inadvertent IMC conditions. The lack of VDU roll attitude symbology to distinguish upward from downward direction is a shortcoming.

46. Failure of the ECS to provide adequate crew station cooling was a shortcoming previously reported in A&FC, Part 3 and resulted in a system redesign. The ECS also provides cooling air for the forward avionics bay (FAB). Mission equipment normally installed in the FAB was not operational on the test aircraft. The ECS light illuminated (indicating temperatures greater than 27°C in the FAB) during ground operations with outside ambient temperatures of 40°C and above. A deviation/change to normal operating procedures for production aircraft described in reference 13, appendix A, requires the crew to close the crew station outlets for 5 minutes or until the ECS light goes out to prevent damage to mission equipment in the FAB. This procedure was not implemented during the test due to the absence of mission equipment in the FAB. Crew station cooling air was adequate for the test aircraft. Further testing of a fully equipped mission aircraft should be accomplished to insure adequate cooling air is available for the crew station and FAB simultaneously.

Night Evaluation

47. The AH-64A was equipped with an An/APX-100(V) transponder and RT-1296 transponder control panel. The control head is mounted on the right hand console (photo 3) and requires the pilot to look down and to the right rear when setting or reading the transponder code numbers. The code numbers are displayed in small windows recessed below a cover plate which partially obscures
Photo 3. Pilots Right Console
the numbers being displayed. Additionally, a small amount of
dust collected on the recessed windows which made the control
numbers unreadable at night with available cockpit lighting. To
adequately clean the code windows, the cover plate must be re-
moved. In dusty conditions, the code windows must be cleaned
daily. The partially obscured code numbers and dust collection
on the code windows also made it difficult to read the numbers
under daylight conditions. Poor readability of the transponder
code numbers is a shortcoming.

48. Instrument panel and indicator lighting at both crew stations
was evaluated during this test. Light intensity of both the
pilot and CPG Marconi indicators is set with a single knob at the
pilot station since the CPG does not have a separate light adjust-
ment in the front cockpit. The lack of an individual light inten-
sity adjustment for the CPG Marconi indicators is a shortcoming.

RELIABILITY AND MAINTAINABILITY

General

49. The reliability and maintainability features of the AH-64A
aircraft were evaluated throughout the test. Seventeen Equipment
Performance Reports (EPR's) (appendix F) were prepared and
submitted during this test. This section is intended only to
document undesirable reliability and maintainability features
encountered during the test and not previously reported. Previous-
ly reported shortcomings are discussed in paragraph 35.

50. The Heading and Attitude Reference System (HARS) failed to
align accurately during this test. Alignments, both the normal
and fast mode, were inconsistent and indicated headings varied
from 0 to 5 degrees from actual runway heading. Additionally,
indicated heading errors varied from 0 to 9 degrees following
flights of 1.0 to 1.5 hours. Attempts to realign the system (on
the ground with rotors turning) using the 8 minute normal align
feature were unsuccessful. The inconsistent and inaccurate
alignment of HARS is a shortcoming.

Main Rotor Plunger

51. The main rotor system incorporated a plunger assembly
(photo 4) designed to determine that the main rotor shaft nut
and retention ring are installed. Presently, a daily special
inspection procedure requires depression of the main rotor
plunger. The reason for this inspection is not clear since the
plunger does not indicate that there is torque on the retaining
screws, nor does it check the position of the nut or retaining ring. During pilot training in preparation for this test, a portion of the main rotor plunger on the test aircraft broke off inside the main rotor assembly. Extensive maintenance was required to remove the broken piece and insure no internal damage had occurred to the main rotor assembly. Depression of the main rotor plunger on other production aircraft showed varying degrees of movement and failed to provide any conclusive information as to the position of the main rotor shaft nut and retention ring. The poor design of the main rotor bearing plunger assembly is a shortcoming. Consideration should be given to removing the main rotor bearing plunger assembly from production aircraft. The position of the main rotor shaft nut and retaining ring as well as torque on retaining screws should be determined by maintenance inspection procedures at the time of main rotor assembly and installation.

**Engine Oil Level Access Doors**

52. To open the engine oil level access doors required a common screwdriver. Without a tool to open the doors (photos 5 and 6), the oil level cannot be checked during the preflight inspection. Additionally, the doors are hinged so they open into the airflow, with the potential of being torn from the aircraft in flight if left unsecured. The poor design of the engine oil level access doors is a shortcoming.

**Gearbox Grease Expulsion**

53. During postflight inspections, large amounts of grease (approximately 6 ounces) were found on the tail rotor intermediate gearbox. The grease was apparently being forced out of the top breather port of the gearbox (EPR 84-10-10, app G). Since there are no provisions for servicing the intermediate gearbox (ref TM 55-1520-238-23), loss of grease will require replacement of the gearbox. Because of the repeated expulsion of grease, the gearbox was replaced 125.5 hours after its installation in the test aircraft. The replacement gearbox was modified with a baffle to prevent further loss of grease through the breather port. By the conclusion of this evaluation the new gearbox had accumulated 9.8 hours with no loss of grease. The excessive expulsion of grease from the top breather port of the intermediate gearbox is a shortcoming. The modified gearbox should be observed during further testing to evaluate the long-term effect of the modification.
Photo 6. #1 Engine Oil Level Access Door
Electromagnetic Interference (EMI)

54. Throughout the evaluation, Electromagnetic Interference (EMI) in the caution panel was observed during ground and flight operations. During ground runup with the number one generator on line, moving the number two generator to the test position illuminated the "RECT" and "HOT BAT" lights and articulated the wing pylons. Moving the number one generator to test position resulted in the number one "RECT" and "MAN STAB" lights illuminating. Additionally, moving the engine instrument test switch to the test position would result in illumination of the number one "ENG OIL PSI" light. In flight, the master caution light would occasionally illuminate for one to two seconds. Due to the short illumination time, it was not possible to identify the associated caution panel light. The momentary master caution and caution panel light illumination was not repeatable or predictable. EMI in the caution panel is a shortcoming.

MISCELLANEOUS

55. Testing was accomplished to determine the status of shortcomings previously reported during A&FC 3. As a result of this evaluation, four were found to have been corrected. The following shortcomings were found to still exist:


   b. The restriction to the pilot's field-of-view caused by window edge distortion, the overhead circuit breaker panel, canopy reflection, CPG helmet, and the PNVS turret.

   c. The poor anthropometric design of the pilot cyclic grip.

   d. The excessive accumulation of oil on the main transmission deck and in the upper fairing access area.

   e. The poor design of the pilot fuel control panel.

   f. The excessively large longitudinal cyclic inputs required when rettrimming during IMC flight with attitude hold ON.

   g. The lack of reliable indication of parking brake status.

   h. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls.

   i. The poor location of the pilot engine control quadrant.
j. The poor design of collective pitch control friction mechanism.

k. The poor location of the tail wheel unlock light.

l. The high inherent friction of the engine power control levers.

m. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight.

AIRSPEED CALIBRATION

56. The pilot and copilot pitot-static systems were calibrated in level flight and dives using a T-28 pace aircraft as a calibrated airspeed reference. The data are presented in figures 53 and 54, appendix E.
CONCLUSIONS

GENERAL

57. The following conclusions were reached upon completion of The First Article Preproduction Test of the AH-64A helicopter:

   a. The level flight performance and handling qualities were not significantly different from the prototype.

   b. The level flight performance presented in the operator's manual satisfactorily represents the performance of the production AH-64A (para 8).

   c. One enhancing characteristic was found.

   d. Two deficiencies were found, one of which was previously reported on the prototype YAH-64.

   e. Four previously reported shortcomings have been corrected.

   f. Twenty-six shortcomings were found, thirteen of which were previously reported on the prototype YAH-64.

   g. Fifteen items of specification noncompliance were found.

   h. Seventeen equipment performance reports were submitted.

ENHANCING CHARACTERISTIC

58. The automatic contingency power feature of the T700-GE-701 engine is an enhancing characteristic (para 22).

DEFICIENCIES

59. The following are deficiencies of the AH-64A (see app D for definition of deficiency used in this report):


   b. The undesirable engine/airframe response during power applications from zero torque condition and during pull ups from less than 40% torque at less than one g (para 23).
SHORTCOMINGS

60. The following shortcomings (listed in order of relative importance) have been identified. Those shown with asterisks were reported during previous USAAEFA tests of the prototype YAH-64 and still exist (see app D for definition of shortcoming used in this report).

a. The objectionable 4/rev (19.3 Hz) vertical vibration characteristics (para 38).*

b. The restriction to the pilot's field-of-view caused by window edge distortion, the overhead circuit breaker panel, canopy reflection, CPG helmet, and the PNVS turret (para 55).*

c. The obstructed access to the pilot's #1 fire handle (para 40).

d. The failure of the HARS to consistently align with the correct magnetic heading (para 50).

e. The activation of the low rotor speed warning system at an excessively high rotor speed (96 to 97%) (para 37).

f. The lack of fuel transfer/crossfeed indications (para 41).

g. The poor anthropometric design of the pilot cyclic grip (para 42).*

h. The excessive accumulation of oil on the main transmission deck and in the upper fairing access area (para 55).*

i. The excessive expulsion of grease from the top breather port of the intermediate gearbox (para 53).

j. The poor design of the pilot fuel control panel (para 43).*

k. The excessively large longitudinal cyclic inputs required when retrimming during IMC flight with attitude hold ON (para 27).*

l. The poor design of the main rotor bearing plunger assembly (para 51).

m. The lack of reliable indication of parking brake status (para 55).*
n. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls (para 55).*

o. The lack of engine anti-ice/start bleed valve position indications (para 44).

p. The poor location of the pilot engine control quadrant (para 55).*

q. The poor design of the collective pitch control friction mechanism (para 14).*

r. The poor location of the tail wheel unlock light (para 55).*

s. The high inherent friction of the engine power control levers (para 15).*

t. The lack of reliable audio signal and azimuth information from the AN/ARN 89B ADF set (para 25).*

u. The lack of VDU roll attitude symbology to distinguish upward from downward direction (para 45).

v. The lack of individual light intensity adjustment of the CPG Marconi indicators (para 48).

w. The poor readability of the transponder code numbers (para 47).

x. The poor design of the engine oil level access doors (para 52).

y. The washout of the rocket panel display, Marconi instrument indications and caution, warning, and advisory panel segment lights in direct sunlight (para 55).*

z. The EMI in the caution panel (para 54).

SPECIFICATION NONCOMPLIANCE

61. The AH-64A was found to be not in compliance with the following paragraphs of the system specification (ref 1, app A). Additional specification noncompliance, beyond the scope of this evaluation may exist.
a. 10.3.2.1.2 - the longitudinal breakout force (plus friction) in the fore and aft direction with the CPG cyclic stick retracted, exceeded the maximum 10% asymmetry by 3.0% (para 11).

b. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the aft direction with the CPG cyclic stick retracted exceeded the 2.4 lb limit by 1.0 lb (para 11).

c. 10.3.2.1.3 - the longitudinal breakout force (plus friction) in the forward and aft directions with the CPG cyclic stick extended exceeded the 2.4 lb limit by 0.6 lb (para 11).

d. 10.3.2.5 - the longitudinal control force could not be trimmed to zero in flight with the CPG cyclic stick retracted (paras 11 and 28).

e. 10.3.2.1.3 - the lateral control breakout force (plus friction) exceeded the 1.5 lb limit by 1.0 lb left and 0.5 lb right (para 12).

f. 10.3.2.1.3 - the left directional breakout force (plus friction) exceeded the 7.0 lb limit by 2.0 lb (para 13).

g. 10.3.2.2 - the left control force versus displacement gradient was less than the minimum value of 2.0 lb/in. by 0.3 lb/in. (para 13).

h. 10.3.2.1 - the directional force gradient was not linear throughout the range of control travel and was concave upward instead of downward during the last 3/4 inch of control travel in each direction (para 13).

i. 10.3.2.1.3 - the collective breakout force (plus friction) in the up direction was lower than the minimum value of 3.0 lb by 0.4 lb (para 14).

j. 10.3.3.2.3 - the engine power control lever breakout force (plus friction) exceeded the maximum value by 2.5 lb (friction OFF) and 10.5 lb (friction ON) (para 15).

k. 10.3.4.1 - the static longitudinal stability was neutral about trim airspeeds of 70 KCAS in level flight and 81 KCAS in IRP climbs (para 17).

l. 10.3.8.1.1 - the collective control delay time following a single engine failure at maximum level flight airspeed was less than the minimum of 2.0 seconds by 1.0 second (para 35).
m. 3.2.11.5.1 - the 4/rev and 8/rev vertical vibration at the pilot station exceeded the specification maximum in certain airspeed ranges (para 38).

n. 3.7.7.1.3.10(c) - there is no advisory light to indicate when fuel transfer is in progress (para 41).

o. 3.7.5.4.1 - the failure of anti-ice/start bleed valve to close does not illuminate a caution light (para 44).
RECOMMENDATIONS

61. The following recommendations are made:

a. Correct the deficiencies prior to operational use (para 58).

b. Correct the shortcomings as soon as practical (para 59).

c. The following CAUTION should be placed in the operator’s manual:

CAUTION

Rapid left roll inputs greater than 2 inches could exceed dual engine transient torque limits when operating above a trimmed torque setting of 85%.

d. The following statement should be added to the operator’s manual: Intentional flight in IMC with attitude hold OFF is not recommended (para 28).

e. The operator’s manual should be changed to prohibit intentional flight in IMC with DASE OFF (para 29).

f. Further testing of a fully equipped mission aircraft should be accomplished to insure adequate cooling air is available for the crew stations and the forward avionics bays simultaneously (para 46).

g. Further observation of the intermediate gearbox for grease expulsion should be accomplished (para 53).
APPENDIX A. REFERENCES


# APPENDIX B. DESCRIPTION

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<tr>
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GENERAL

1. The AH-64 Advanced Attack Helicopter (fig. 1) is a tandem, two-place, twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopter Incorporated (HHI). The main rotor is a four-bladed fully articulated system supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a dual-teetering, semi-rigid, delta-hinged system incorporating elastomeric flapping bearings. The rotors are driven by two General Electric T700-GE-701 engines through the power train shown in figure 2. An AiResearch GTCP 35-55 (H) auxiliary power unit (APU), is installed to drive the accessory section of the main transmission when the rotors are not turning. This provides pneumatic power for engine starting as well as electrical and hydraulic power for aircraft systems. The aircraft is designed to carry ordnance stores internally in the ammunition bay and externally on four wing store positions. The AH-64 is designed to operate during day, night, and marginal weather combat conditions using the Martin-Marietta Target Acquisition Designation System (TADS)/Pilot Night Vision Sensor (PNVS). The test aircraft, SN 82-23355, was configured with an aero-dynamic mockup of the TADS/PNVS, 30mm chain gun, and eight Hellfire missiles to represent the primary mission configuration (photos 1 through 8). The major modifications and external configuration differences between the YAH-64 (AV05) flown during the airworthiness and flight characteristics (A&FC) Part 3 tests and the aircraft configuration are presented in table 1. Drag estimates provided by HHI are included in this table for the items not on the test aircraft but described in the system specification for the program and for any external test instrumentation equipment. The backup control system (BUCS) was not operational for this test.

Table 1. Configuration of Test Aircraft (USA S/N 82-23355)

Configuration differences from prototype aircraft (AV05) flown during A&FC Part 3:

a. Installed digital automatic stabilization equipment (DASE) production software (program no. 7-211 D00005-11)

b. Wing pylons shortened 6 inches

c. Maintenance step installed on left side of aircraft

Items not on the test aircraft but defined in the system specification for the production program:
Figure 1. Aircraft Dimensions
Figure 2. Powertrain
Photo 1. Front View, AH-64A (Apache) Helicopter
Photo 4. Right Three-Quarter View, AH-64A (Apache) Helicopter
Photo 5. Rear View, AH-64A (Apache) Helicopter
Photo 7. Left View, AH-64A (Apache) Helicopter
a. 30mm ammunition chute

b. Fire control computer

Items not in the system specification but installed on the test aircraft:

<table>
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<tr>
<th>Estimated Drag</th>
</tr>
</thead>
<tbody>
<tr>
<td>AFe (SQ FT)</td>
</tr>
<tr>
<td>----------------</td>
</tr>
<tr>
<td>a. Airspeed boom with angle of attack and sideslip vanes</td>
</tr>
<tr>
<td>b. Instrumentation canister located on main rotor hub</td>
</tr>
<tr>
<td>c. Telemetry antenna brackets located on scissors assembly and telemetry antenna and bracket located on underside of tail cone.</td>
</tr>
<tr>
<td>d. Instrumented main and tail rotor pitch change links</td>
</tr>
<tr>
<td>e. Instrumented main rotor blade</td>
</tr>
<tr>
<td>f. Instrumented tail rotor blade</td>
</tr>
</tbody>
</table>

**DIMENSIONS AND GENERAL DATA**

**Main Rotor**

<table>
<thead>
<tr>
<th>Dimension</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter (ft)</td>
<td>48.0</td>
</tr>
<tr>
<td>Blade chord (in.)</td>
<td>21.0*</td>
</tr>
<tr>
<td>Main rotor total blade area (sq ft)</td>
<td>166.5</td>
</tr>
<tr>
<td>Main rotor disc area (sq ft)</td>
<td>1809.56</td>
</tr>
<tr>
<td>Main rotor solidity (thrust weighted no tip loss)</td>
<td>0.092</td>
</tr>
<tr>
<td>Airfoil</td>
<td>HH-02**</td>
</tr>
<tr>
<td>Twist (deg)</td>
<td>-9</td>
</tr>
<tr>
<td>Number of blades</td>
<td>4</td>
</tr>
<tr>
<td>Rot. r speed at 100 percent NR (rpm)</td>
<td>289.3</td>
</tr>
<tr>
<td>Tip speed at 100 percent NR (ft/sec)</td>
<td>727.09</td>
</tr>
<tr>
<td>Gear ratio (engine to main rotor)</td>
<td>72.424322</td>
</tr>
</tbody>
</table>

*Includes tips

**Outer 20 inches swept 20 degrees and transitioned to a NACA 64A006 airfoil**

53
### Tail Rotor

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter (ft)</td>
<td>9.17</td>
</tr>
<tr>
<td>Chord; constant (in.)</td>
<td>10</td>
</tr>
<tr>
<td>Tail rotor total blade area (sq ft)</td>
<td>14.89</td>
</tr>
<tr>
<td>Tail rotor disc area (sq ft)</td>
<td>66.0</td>
</tr>
<tr>
<td>Tail rotor solidity</td>
<td>0.2256</td>
</tr>
<tr>
<td>Airfoil</td>
<td>NACA 63-414 (modified)</td>
</tr>
<tr>
<td>Twist</td>
<td>-8.8</td>
</tr>
<tr>
<td>Number of blades</td>
<td>4</td>
</tr>
<tr>
<td>Rotor speed at 100 percent $N_R$ (rpm)</td>
<td>1403.4</td>
</tr>
<tr>
<td>Distance from main rotor mast centerline (CL) (ft)</td>
<td>29.67</td>
</tr>
<tr>
<td>Tip speed at 100 percent $N_R$ (ft/sec)</td>
<td>673</td>
</tr>
<tr>
<td>Teetering angle (deg)</td>
<td>10 soft/15 Hard</td>
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<tr>
<td>Maximum blade angle (deg)</td>
<td>27</td>
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### Horizontal Stabilator

<table>
<thead>
<tr>
<th>Parameter</th>
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</tr>
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<tbody>
<tr>
<td>Weight (lb)</td>
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</tr>
<tr>
<td>Area (ft)</td>
<td>33.36</td>
</tr>
<tr>
<td>Span (ft)</td>
<td>11.17</td>
</tr>
<tr>
<td>Tip chord (ft)</td>
<td>2.65</td>
</tr>
<tr>
<td>Root chord (ft)</td>
<td>3.67</td>
</tr>
<tr>
<td>Geometric aspect ratio</td>
<td>3.41</td>
</tr>
<tr>
<td>Sweep of leading edge (deg)</td>
<td>2.89</td>
</tr>
<tr>
<td>Sweep of trailing edge (deg)</td>
<td>-7.23</td>
</tr>
<tr>
<td>Dihedral (deg)</td>
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### Vertical Stabilizer

<table>
<thead>
<tr>
<th>Parameter</th>
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<tbody>
<tr>
<td>Area (from boom CL) (sq ft)</td>
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</tr>
<tr>
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</tr>
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<td>Root chord (at boom CL) (in.)</td>
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</tr>
<tr>
<td>Geometric aspect ratio</td>
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<tr>
<td>Airfoil</td>
<td>NACA 4415 (modified)</td>
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<tr>
<td>Sweep of leading edge (deg)</td>
<td>28.1</td>
</tr>
<tr>
<td>Vertical stabilizer trailing edge deflection</td>
<td>16 deg left above</td>
</tr>
<tr>
<td></td>
<td>W.L. 196.0</td>
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</table>

### Wing

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Span (ft)</td>
<td>16.33</td>
</tr>
<tr>
<td>Mean aerodynamic chord (in.)</td>
<td>45.9</td>
</tr>
</tbody>
</table>
Total area (ft²) 61.59
Airfoil at root NACA 4423

Aircraft
Fuel quantity (gal) 375
Basic design gross weight (lb) 14,660
Maximum gross weight (lb) 17,650
Primary mission gross weight (lb) 14,694
Maximum alternate gross weight (lb) 17,650
Extended range gross weight 20,100

FLIGHT CONTROL DESCRIPTION

General

2. The AH-64A helicopter employs a hydromechanical irreversible flight control system which is mechanically activated with conventional cyclic, collective and directional pedal controls, through a series of push-pull tubes attached to four airframe mounted hydraulic servoactuators. The four hydraulic servoactuators control longitudinal cyclic, lateral cyclic, main rotor collective, and tail rotor collective pitch. Cyclic and directional servoactuators incorporate integral stability augmentation system (SAS) actuators. Hydraulic power is supplied by two independent 3000 psi hydraulic systems which are powered by hydraulic pumps mounted on and driven by the accessory gearbox of the main transmission to allow full operation under a dual-engine failure condition. A DASE system is installed to provide rate damping and command augmentation. The DASE is limited to ±10 percent of control authority in roll and yaw. The longitudinal cyclic hydraulic servoactuator allows 20 percent forward and 10 percent aft control authority in the pitch axis. The DASE also provides attitude hold and a Hover Augmentation System (HAS). An electrically actuated horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. Movement of the stabilator can be controlled either manually or automatically. A trim feel system is incorporated in the cyclic and directional controls to provide a control force gradient with control displacement from a selected trim position. A 3 position trim release switch located on the pilot's cyclic grip, provides discrete off, or momentary interruption of the force gradient in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new reference trim position. Full control travel is 10.5 inches in the longitudinal control, 8.45 inches in the lateral control, 11.73 inches in the collective control and 6.06 inches in the directional pedals.
Cyclic Control System

3. The cyclic control system (fig. 3) consists of dual-tandem cyclic sticks attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable differential transducers (LVDTs) designed to measure electrically the longitudinal and lateral cyclic position for the DASE computer inputs. A series of push-pull tubes and bellcranks transmit the motion of the cyclic control to the servoactuators and the mixer assembly. Motion of the mixer assembly positions the nonrotating swashplate, which transmits the control inputs to the main rotor pitch change links through the rotating swashplate (fig. 4). The cyclic stick grips are shown in figure 5. A stick fold linkage is provided to allow the copilot/gunner (CPG) to lower the cyclic stick to prevent interference when operating the weapons systems.

Collective Control System

4. The collective pitch control system (fig. 6) consists of dual-tandem collective sticks which transmit collective inputs to the main rotor through a series of push-pull tubes and bellcranks attached to the collective servoactuator. Motion of the servoactuator is transmitted through the mixer assembly to the swashplate to control the main rotor blades in collective pitch. Collective inputs are also transmitted from the collective servoactuator to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide compensation for collective pitch. Located at each collective control base assembly are the primary control stop, an LVDT, and a one g balance spring. Redundant LVDTs on the collective servoactuator supply electrical inputs to the stabilator control units.

5. Each collective stick (fig. 7) incorporates a switch box assembly, an engine chop collar, a stabilator control panel and an adjustable friction control. The engine chop collar allows rapid deceleration of both engines to flight idle, primarily to allow immediate action in the event of a tail rotor failure.

Directional Control System

6. The directional control system (fig. 8) consists of a series of push-pull tubes and bellcranks which transmit directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly is a primary tail rotor control stop and one LVDT. Two sets of wheel brake cylinders are attached to the directional pedals and a 360 degree swiveling tail wheel is incorporated.
Figure 4. Main Rotor Swashplate Assembly
1 TRIM RELEASE SWITCH  
   (MOMENTARY ONLY CPG)
2 WEAPONS ACTION SWITCH
3 FLIGHT MODE SYMBOLOGY SWITCH
4 TO BE DETERMINED
5 BASE RELEASE SWITCH
6 GUARDED TRIGGER SWITCH
7 REMOTE TRANSMITTER SELECTOR SWITCH  
   (PILOT GRIP ONLY)
8 RADIO, ICS ROCKER SWITCH

Figure 5. Cyclic Stick Grips
1 NIGHT VISION SWITCH
2 BORESIGHT HMD SWITCH
3 ENGINE CHOP COLLAR
4 COLLECTIVE GRIP
5 STABILATOR MANUAL CONTROL SWITCH
6 AUTOMATIC OPERATION/AUDIO WARNING
   RESET BUTTON
7 BUOS TRIGGER (CPG ONLY)

Figure 7. Collective Stick Control
Figure 8. Directional Controls
The tail wheel may be locked in the trailing position by means of a switch located on the pilot instrument panel.

Trim Feel System

7. A trim feel system is incorporated in the longitudinal, lateral, and directional control systems. The system uses individual magnetic brake clutch assemblies in each of the control linkages. Trim feel springs are incorporated to provide a control force gradient and positive control centering. The electromagnetic brake clutch is powered by 28 VDC and is protected by the TRIM circuit breaker. A complete DC electrical failure will disable the trim feel system and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the CPG grip allows momentary release of the trim feel system. A three position trim release switch on the pilot's grip allows the system to be ON, discrete OFF, or momentarily released.

Horizontal Stabilator

8. A horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. A block diagram of the stabilator control system is shown in figure 9. A dual series, 28 VDC electromechanical actuator allows incidence changes of +35 trailing edge down (TED) to -10 degrees TED of travel. Safety features include an automatic shutdown capability which allows operation in the manual mode by means of a stabilator control panel located on each collective stick. An audio tone is associated with the failure of the automatic mode of operation. There are three modes of stabilator operation: the automatic mode, the Nap-of-the-earth (NOE)/Approach mode and the manual mode. The stabilator is controlled in the automatic mode by two stabilator control units (SCUs). Each SCU controls one side of the dual actuator. Both SCUs receive collective control position information from redundant LVDTs located on the collective servoactuator. Two independent pitch rate gyro's provide pitch rate information to the SCUs (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCUs. Additionally, the left-hand pitot-static system supplies airspeed to one SCU and the right-hand system supplies airspeed to the other SCU. Both SCUs receive position information from both sides of the dual actuator.

9. The automatic mode is operational when the aircraft has normal AC and DC electrical power applied. Automatic positioning of the stabilator during flight is primarily a function of airspeed and collective position. Software in the SCU limits the incidence change in the automatic mode from +25 deg to -5 deg TED.
Figure 9. Stabilator Control Block Diagram
10. The NOE/Approach Mode is selected through the NOE/APPR mode switch on the pilot DASE panel. This mode is selectable at any airspeed when the automatic mode is engaged. The mode becomes operational below 80 knots indicated airspeed (KIAS) and will move the stabilator to 25 degrees TED. The NOE/approach mode can be disengaged by manual mode selection below 80 knots, activation of the DASE release, by the AUTO STAB reset switch or by turning the NOE approach mode switch off. Acceleration through 80 KIAS will engage the automatic schedule and the stabilator will move to the automatic mode schedule position. Failure to revert to automatic schedule will result in system disengagement with both visual and aural indications.

11. The manual mode can be selected below 80 KIAS through the pilot and CPG manual control switch on either collective stick. Manual control selection will result in MAIN STAB caution light illumination. Selection of automatic mode can be accomplished by pressing the AUTO STAB reset switch on the pilot or CPG collective stick. Acceleration through 80 KIAS in manual mode will engage the automatic mode and the stabilator to the automatic mode schedule position.

Flight Control Rigging

12. A flight control rigging end to end check was performed in accordance with procedures outlined in HHI Production Acceptance Test Procedure PATP 00007. The horizontal stabilator schedule is shown in figure 10. Tables 2 and 3 present the collective and cyclic rigging. Tail rotor rigging is shown below:

- Full right pedal: 14.5 degrees thrust to left
- Full left pedal: 27.9 degrees thrust to right

Digital Automatic Stabilization Equipment

13. The DASE provides rate damping (SAS), command augmentation (CAS), HAS, attitude hold, and turn coordination. A block diagram of the DASE is provided in figure 11. The DASE is controlled by the digital automatic stabilization equipment computer (DASEC). The DASEC receives information from several sources. The heading and attitude reference system (HARS) provides the DASEC with aircraft angular velocities (3 axes), aircraft attitudes (pitch, roll, and heading), and inertial horizontal and lateral velocities (measured by the Doppler radar). The ADS provides longitudinal airspeed and sideslip angle. The LVDTs provide longitudinal, lateral, collective and directional control position information. The DASEC processes this information and commands control inputs.
Figure 10
Level Flight Performance
AH-64A USA S/N 82-23355

□ 8x Collective
△ 50x Collective
+ 100x Collective

Notes:
1. Voltages input to stabilator control unit to simulate inputs from air data system for airspeeds to 50 knots.
2. Left and right pitot systems pressurized to obtain airspeeds above 50 knots.
3. Lines represent design schedule.

Horizontal Stabilator Angle (Degrees)

0 20 40 60 80 100 120 140 160 180 200

Indicated Airspeed (Knots)
<table>
<thead>
<tr>
<th>Blade Azimuth Position (deg)</th>
<th>Item</th>
<th>Rig Pins</th>
<th>Stick position</th>
<th>Pitch Housing Angle</th>
<th>Leading Edge Up or Down</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Collective</td>
<td>Longitudinal Cyclic</td>
<td>Lateral Cyclic</td>
<td>Collective</td>
</tr>
<tr>
<td>$\psi = 90$</td>
<td>1</td>
<td>In</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>In</td>
<td>Out</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>In</td>
<td>Out</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>In</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>Out</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>Out</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>7</td>
<td>In</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td>$\psi = 180$</td>
<td>8</td>
<td>In</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>9</td>
<td>In</td>
<td>In</td>
<td>Out</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>10</td>
<td>In</td>
<td>In</td>
<td>Out</td>
<td>Rig</td>
</tr>
<tr>
<td></td>
<td>11</td>
<td>In</td>
<td>In</td>
<td>In</td>
<td>Rig</td>
</tr>
</tbody>
</table>
Figure 11. Digital Automatic Stabilization Equipment Block Diagram
Table 3. Computation of Blade Angle Travel
Pilots Collective and Cyclic Controls

<table>
<thead>
<tr>
<th>Computation</th>
<th>Travel (deg)</th>
<th>Requirements (deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>LONITUDINAL CYCLIC</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1. Forward (Item 2 - Item 1) =</td>
<td>20.2</td>
<td>20 (minimum)</td>
</tr>
<tr>
<td>(If Item 1 is leading edge up, add item 2 to Item 1)</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2. Aft (Item 3 - Item 1) =</td>
<td>10.8</td>
<td>10 (minimum)</td>
</tr>
<tr>
<td>(If Item 1 is leading edge down, add Item 3 to Item 1)</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>LATERAL CYCLIC</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3. Left (Item 9 - Item 8) =</td>
<td>11.1</td>
<td>10.3 (minimum)</td>
</tr>
<tr>
<td>(If Item 8 is leading edge up, add Item 9 to Item 8)</td>
<td></td>
<td></td>
</tr>
<tr>
<td>4. Right (Item 10 - Item 8) =</td>
<td>7.7</td>
<td>7 (minimum)</td>
</tr>
<tr>
<td>(If Item 8 is leading edge down, add Item 10 to Item 8)</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>COLLECTIVE</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>5. Full pitch travel (Item 5 - Item 6) =</td>
<td>18.7</td>
<td>18 (minimum)</td>
</tr>
<tr>
<td>(If Item 6 is leading edge down, add Item 5 to Item 6)</td>
<td></td>
<td></td>
</tr>
<tr>
<td>6. Collective pitch full down</td>
<td>-8</td>
<td>-10 to -7</td>
</tr>
<tr>
<td>Measured at pitch housing</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
through the electrohydraulic servo valves on the longitudinal, lateral, and directional servoactuators. The DASE authority is limited in the lateral and directional axes to ±10 percent of full control authority while the longitudinal axis is limited to 20 percent forward and 10 percent aft.

14. The SAS function of the DASE system provides rate damping in pitch, roll, and yaw axes. Each axis is separately engageable through a magnetically held toggle switch on the DASE control panel shown in figure 12. The CAS is used to augment the pilot control inputs and is an automatic function of the DASE whenever pitch and roll SAS are selected and yaw SAS is selected below 60 knots true airspeed (KTAS). The yaw CAS function is automatically disengaged during ground operations. Schematic diagrams showing gains and transfer functions of SAS/CAS are provided as figures 13 through 17.

15. A limited authority HAS mode is provided through pitch and roll DASE channels using rates, attitudes and doppler corrected inertial velocities from the HARS. HAS is used to reduce pilot workload by assisting the pilot in maintaining a desired hover position. HAS is engaged below 15 knots ground speed and 50 KTAS with trim feel system ON, ATTD/Hover Hold switch ON, and the pitch and roll DASE channels are engaged. Additionally, a heading hold mode is provided through the yaw DASE channel using aircraft heading information from the HARS. This function is engaged whenever the yaw DASE channel and HAS is engaged. Schematic diagrams are shown in figures 18 through 20.

16. A limited authority attitude hold mode is provided through pitch and roll SAS. Attitude Hold is engageable above 60 KTAS when the ATTD/Hover Hold switch is on, trim feel system ON, and pitch and roll SAS are engaged. Attitude Hold will automatically disengage whenever the airspeed is decreased to 50 KTAS. Schematic diagrams are shown in figures 20 through 22.

17. A limited authority turn coordination function is provided through yaw SAS using sideslip information from the ADS. This function is automatically provided above 60 KTAS whenever yaw SAS is engaged. A schematic diagram is shown in figure 23.

**HYDRAULIC SYSTEM**

**General**

18. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic
Figure 12. Pilots DASE Control Panel
Figure 13. Pitch SCAS Block Diagram
Figure 14. Roll SCAS Block Diagram
Figure 15. Mechanical and SAS Servo Loop
Figure 16. Yaw SCAS Low Speed Mode Block Diagram
Figure 17. Yaw SCAS Ground Mode Block Diagram
Figure 18. Pitch Hover Augmentation Mode Block Diagram
Figure 19. Roll Hover Augmentation Mode Block Diagram
Figure 20. Heading Hold Mode Block Diagram
Figure 21. Pitch Attitude Hold Mode Block Diagram
Figure 22. Roll Attitude Hold Mode Block Diagram
Figure 23. Yaw SCAS Turn Coordination Block Diagram
systems. Each servoactuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of a single hydraulic system failure. The two systems (primary and utility) are driven by the accessory gearbox utilizing variable displacement pumps, independent reservoirs and accumulators. The APU drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotors are not turning. The accessory gearbox is driven by the main transmission during flight and provides for normal operation of both hydraulic systems during autorotation. An emergency hydraulic system is provided to allow emergency operation of the flight controls in the event of a dual system failure.

**Primary Hydraulic System**

20. The primary hydraulic system (fig. 24) consists of a one-pint capacity reservoir, which is pressurized to 30 psi using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that directs the fluid to the lower side of the four servoactuators. The primary system also provides the hydraulic pressure for operation of the DASE functions.

**Utility Hydraulic System**

21. The utility hydraulic system (fig. 25) consists of an air pressurized 1.3 gallon reservoir and a 3000-psi accumulator which drives the APU starting motor. The utility manifold directs fluid to the upper side of the servoactuators, the stores pylon system, tail wheel lock mechanism, area weapon turret drive, and rotor brake. Other manifold functions include an auxiliary isolation check valve which isolates the area weapon turret drive and external stores actuators when either a low pressure or low fluid condition exists; a low pressure sensor isolates the accumulator as an emergency hydraulic source for the servoactuators in the event of a dual hydraulic system failure. The accumulator assembly stores enough fluid for emergency operation of the flight controls through four full strokes of the collective stick and one 180 degrees heading change. The emergency system may be activated by either the pilot or CPG emergency switch. An electrically activated emergency shutoff valve is designed to isolate the utility side of the directional servoactuator and the tail wheel lock mechanism when a low fluid condition exists.
Figure 24. Primary Hydraulic System
Figure 25. Utility Hydraulic System
Servoactuators

22. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig. 26) consists of a ballistically tolerant housing, a single actuator rod and dual frangible pistons, a BUCS plunger, and various parts for routing of both primary and utility hydraulic fluid. The system is designed to accommodate all flight loads with a failure of either system, however, some control authority will be lost in the directional servoactuator system. DASE and BUCS functions would be lost with failure of the primary system.

Engines

23. The AH-64A helicopter, for this test, was powered by two General Electric T700-GE-701 front drive turboshaft engines, rated at 1690 shaft horsepower (shp) at sea level, standard day, unassembled. The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: a cold section, a hot section, a power turbine, and an accessory section. Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high-pressure gas generator turbine, a two-stage uncooled power turbine and self-contained lubrication and electrical systems. In order to reduce sand and dust erosion, and foreign object damage, an integral particle separator operates when the engine is running. The T700-GE-701 engine also incorporates a history recorder which records total engine events. Engines S/N 374115 and 374114 were installed in the left and right positions, respectively. Both engines were equipped with GO2J Electrical Control Units (ECUs). ECUs S/N CAOE2019 and CAOE2066 were installed on the left and right engine, respectively. The following engine data are provided:

<table>
<thead>
<tr>
<th>Model</th>
<th>T700-GE-701</th>
</tr>
</thead>
<tbody>
<tr>
<td>Type</td>
<td>Turboshaft</td>
</tr>
<tr>
<td>Rated power (intermediate)</td>
<td>1690 shp sea level, standard day, unassembled</td>
</tr>
<tr>
<td>Output speed (at 100 percent N_r)</td>
<td>20,952 RPM</td>
</tr>
<tr>
<td>Compressor</td>
<td>5 axial stages, 1 centrifugal stage</td>
</tr>
<tr>
<td>Variable geometry</td>
<td>Inlet guide vanes, stages 1 and 2 stator vanes</td>
</tr>
<tr>
<td>-------------------</td>
<td>-----------------------------------------------</td>
</tr>
<tr>
<td>Combustion chamber</td>
<td>Single annular chamber with axial flow</td>
</tr>
<tr>
<td>Gas generator turbine stages</td>
<td>2</td>
</tr>
<tr>
<td>Power turbine stages</td>
<td>2</td>
</tr>
<tr>
<td>Direction of rotation (aft looking forward)</td>
<td>Clockwise</td>
</tr>
<tr>
<td>Weight (dry)</td>
<td>423 lb</td>
</tr>
<tr>
<td>Length</td>
<td>47 in.</td>
</tr>
<tr>
<td>Maximum diameter</td>
<td>25 in.</td>
</tr>
<tr>
<td>Fuel</td>
<td>MIL-T-5624 (JP-4 or JP-5)</td>
</tr>
<tr>
<td>Lubricating oil</td>
<td>MIL-L-7808 or MIL-L-23699</td>
</tr>
<tr>
<td>Electrical power requirements for history recorder and Np overspeed protection</td>
<td>40W, 115 VAC, 400 Hz</td>
</tr>
<tr>
<td>Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector</td>
<td>1 amp, 28 VDC</td>
</tr>
</tbody>
</table>

**INFRARED (IR) SUPPRESSION SYSTEM**

24. The IR suppression system consists of finned exhaust pipes attached to the engine outlet and bent outboard to mask hot engine parts. The finned pipes radiate heat which is cooled by rotor downwash in hover and turbulent air flow in forward flight. The engine exhaust plume is cooled by mixing it with engine cooling air and bay cooling air (fig. 27). The exhaust acts as an eductor, creating air flow over the combustion section of the engine providing engine cooling. Fixed louvers on the top and bottom of the aft cowl and a door on the bottom forward cowling
provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

**FUEL SYSTEM**

25. The AH-64A fuel system has two fuel cells located fore and aft of the ammunition bay. The system includes a fuel boost pump in the aft cell for starting and for high-altitude operation, a fuel transfer pump for transferring fuel between cells, a fuel crossfeed/shutoff valve, and provisions for pressure and gravity fueling and defueling. Additionally, provisions exist for external, wing-mounted fuel tanks. Figure 28 is a schematic of the fuel system. Figure 29 shows the locations and capacities of the two internal fuel cells.

26. By using the crossfeed (pilot) or tank select (CPG) switch on the fuel control panel (fig. 30), the pilot or CPG can select either or both tanks from which the engines will draw fuel. With the crossfeed or tank select switch in the NORM position, the left (No. 1) engine will draw fuel from the forward fuel cell and the right (No. 2) engine will draw from the aft cell. When FWD tank (pilot) or FROM FWD (CPG) is selected the two fuel crossfeed/shutoff valves are positioned so that both engines draw fuel from the forward tank. The AFT tank (pilot) or FROM AFT (CPG) position allows the engines to draw fuel from the aft tank only. The air-driven boost pump operates automatically during engine start and may be activated by the switch on the pilot or CPG fuel control panel only when the tank select switch (CPG) or crossfeed switch (pilot) is in the aft position.

27. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels. Moving the fuel transfer switch out of the OFF position closes the refuel valve and activates the air-driven pump which transfers fuel in the selected direction.

28. Fuel is supplied to the APU from the AFT fuel cell through the APU shutoff valve and APU boost pump. The APU shutoff valve is controlled by the APU off-run-start switch located on the APU control panel (fig. 31). When this switch is placed to the run position the shutoff valve is electrically opened and fuel is supplied to the APU boost pump.

29. External refueling provisions are provided on the right side of the aircraft. The forward and aft cells may be gravity filled separately. The single point adapter or closed circuit adapter allow the fore and aft cells to be pressure refueled separately or
Figure 28. Fuel System Major Components
Figure 29. Fuel Cell Locations

FORWARD FUEL CELL
155 GAL

AFT FUEL CELL
220 GAL
Figure 30. Pilot and Copilot/Gunner Fuel Control Panels
Figure 31. APU Control Panel
simultaneously using the external refueling panel shown in figure 32.
Figure 32. External Refueling Panel
APPENDIX C. INSTRUMENTATION

1. An airborne data acquisition system was installed and maintained by Hughes. The system utilized pulse code modulation (PCM) encoding. Magnetic tape was used to record parameters aboard the aircraft. The various external instrumentation drag items are shown in photos 1 through 7.

2. A boom was mounted on the aircraft extending 52 inches forward of the nose. A pitot-static tube, an angle-of-attack sensor, and an angle-of-sideslip sensor were mounted on the boom (photo 1).

3. Instrumentation and related special equipment installed are presented in the following lists. The pilot and copilot-gunner stations are shown in photos 8 and 9.

**Pilot Station (aft cockpit displays)**

- Pressure altitude (boom)
- Airspeed (boom)
- Vertical rate of climb
- Main rotor speed
- Engine torque (both engines)
- Engine measured gas temperature (both engines)
- Engine power turbine speed (both engines)
- Engine gas producer speed (both engines)
- Angle of sideslip
- Event switch
- Radar altitude

**Control Positions**

- Longitudinal
- Lateral
- Directional
- Collective

- Stabilator incidence angle
- Normal acceleration (cg)
- Horizontal situation indicator with Doppler interface
- Primary attitude reference
- Turn needle and ball
- Sensitive cg lateral acceleration indicator

**Copilot Station Displays**

- Airspeed (ship)
- Altitude (ship)
- Main rotor speed
- Engine torque (both engines)
- Engine measured gas temperature (both engines)
- Engine gas producer speed (both engines)
Photo 1. Instrumentation Boom
Photo 5. Accelerometer Located on Right Engine Infrared Suppressor
Photo 6. PCM Antenna (Located at Approximately Station 480)
Photo 9. Copilot-Gunner Station
Fuel used (both engines)
Fuel used (APU)
Total air temperature
Time code display
Event switch
Data system controls
Doppler

PCM Parameters (Magnetic Tape)

Time code
Event
Main rotor speed
Fuel temperature (both engines)
Fuel used (both engines)
Engine fuel flow rate (both engines)
Fuel used (APU)
Engine torque (both engines)
Engine measured gas temperature (both engines)
Engine gas producer speed (both engines)
Engine power turbine speed (both engines)
Airspeed (boom)
Airspeed (ship, left and right)
Altitude (boom)
Altitude (ship, left and right)
Total air temperature (boom)
Angle of attack (boom)
Angle of sideslip (boom)
Control positions
  Longitudinal cyclic
  Lateral cyclic
  Pedal
  Collective
Stabilator incidence angle
Aircraft attitudes
  Pitch
  Roll
  Yaw
Aircraft angular velocities
  Pitch
  Roll
  Yaw
Vibration Accelerometers
  Pilot seat (3 axes)
  Copilot seat (3 axes)
  Aircraft cg (3 axes)
Stability augmentation system actuator positions
  Longitudinal
Lateral
Directional
Control actuator positions
Tail rotor
Collective pitch
Cyclic pitch
Cyclic roll
Air data system
Longitudinal velocity
Lateral velocity
Pressure altitude
Outside air temperature
Angle of sideslip
Resultant airspeed
Radar altitude
CG normal acceleration
CG lateral acceleration
Rotor azimuth

4. Position error of the boom airspeed system was determined and is presented in figure 1.
FIGURE 1
AIRSPEED CALIBRATION
AH-64A USA S/N 82-23355
BOOM PITOT-STATIC SYSTEM

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUINAL CG LOCATION (AFT)</th>
<th>AVG DENSITY (FEET)</th>
<th>AVG ALTITUDE (FEET)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG ROTOR SPEED (RPM)</th>
<th>AVG FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14600</td>
<td>206.4</td>
<td>7250</td>
<td>20.5</td>
<td>200</td>
<td>200</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

CORRECTION TO BE ADDED (KNOTS)

NOTE: SHADED SYMBOLS DENOTE DIVES

CALIBRATED AIRSPEED (KNOTS)

INSTRUMENT CORRECTED AIRSPEED (KNOTS)

NOT FOR HANDBOOK USE

LINE OF ZERO ERROR
APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Performance data were obtained using the methods described in Army Materiel Command Pamphlet AMCP 706-204 (ref 14, app A) as a guide. Handling qualities data were evaluated using test methods described in Naval Air Test Center Flight Test Manual FTM No. 101 (ref 15) as a guide.

AIRCRAFT WEIGHT AND BALANCE

2. The aircraft was weighed in the 8-Hellfire configuration, as instrumented for test, with full oil, residual fuel after draining, and ballast required for the test. In this configuration, the aircraft weighed 13,078 pounds with a longitudinal center of gravity (cg) at fuselage station (FS) 209.2. The lateral cg was at butt line (BL) -0.9. The forward fuel tank was limited to a maximum of 900 pounds of fuel by the Airworthiness Release. This restriction caused all tests to be done at an aft cg.

LEVEL FLIGHT PERFORMANCE

General

3. Level flight performance was generalized by the use of non-dimensional coefficients as follows:

a. Coefficient of power (Cp):

\[
C_p = \frac{SHP (550)}{\rho A (\text{AR})^3}
\]

(1)

b. Coefficient of thrust (CT):

\[
C_T = \frac{GW}{\rho A (\text{AR})^2}
\]

(2)

c. Advance Ratio (\(\alpha\))

\[
\alpha = \frac{V (1.6878)}{T}
\]

(3)
Where:

- **SHP** = Engine output shaft horsepower (both engines)
- **ρ** = Ambient air density (lb·sec²/ft⁴)
- **A** = Main rotor disc area = 1809.56 ft²
- **Ω** = Main rotor angular velocity = 30.26 radians/sec (at 289 RPM)
- **R** = Main rotor radius = 24.0 ft
- **GW** = Gross weight (lb)

\[
V = \text{True airspeed (kt)} = \frac{V}{\sqrt{\frac{\rho}{\rho_0}}}
\]

- 1.6878 = Conversion factor (ft/sec)/kt
- \(\rho_0 = 0.0023769\) (lb·sec²/ft⁴)

For a rotor speed of 289 rpm the following constants were used:

- **fK** = 726.34
- \(A(ΩR)^2 = 9.54657879 \times 10^8\) ft⁴/sec²
- \(A(ΩR)^3 = 6.934025959 \times 10^{11}\) ft⁵/sec³

4. Specific range (NAMPP) data were calculated from test data using the following equation:

\[
\text{NAMPP} = \frac{V}{W_f} \tag{4}
\]

where:

- **NAMPP** = Nautical airmiles per pound of fuel
- **V** = True airspeed (kt)
- **W_f** = Fuel flow (lb/hr)
Corrections to Test Data

5. No corrections were made to the test data presented in this report. Possible sources of error were: a. the drag of the test airspeed boom and the rotating PCM canister; b. the power consumed by the environmental control unit which was used during the test, and; c. the difference in electrical power consumption between the test instrumentation and the mission equipment which would be used during an operational flight.

Shaft Horsepower Required

6. The engine output shaft torque was determined by use of the engine torque sensor. The power turbine shaft twists an amount proportional to the torque in the shaft. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine shaft at the rear end. The relative rotation at the rear end of the two shafts is a measure of the torque being transmitted by the power turbine shaft. The two shafts have reference teeth at the rear end which pass by a magnetic pickup. This pickup sends an electrical pulse to the engine electronic control unit (ECU) each time a reference tooth passes by. From these pulses, the ECU calculates the twist in the shaft and converts that twist to torque. A calibration of the torque measuring system was performed in an engine test cell during the engines' acceptance runs. These data are presented in figures 1 and 2 and were used to correct the indicated torque readings during the performance testing. The output SHP was determined from the engine output shaft torque and rotational speed by the following equation:

$$\text{SHP} = \frac{2\pi N_p Q}{33,000}$$

(5)

Where:

- \(Q\) = Engine output shaft torque (ft-lb)
- \(N_p\) = Engine output shaft rotational speed (RPM)
- 33,000 = Conversion factor (ft-lb/min)/SHP

Test Method

7. Level flight performance was determined by using equations 1, 2, and 3. Each speed-power was flown at a constant thrust...
FIGURE 1
ENGINE TORQUEMETER CALIBRATION
T700-GE-701  S/N 374-115

NOTES: 1. POWER TURBINE SPEED = 20452 RPM
        2. ENGINE INSTALLED ON LEFT SIDE OF AIRCRAFT

CORRECTION TO BE ADDED
(FT-LB)

DYNAMOMETER TORQUE
(FT-LB)

LINE OF ZERO ERROR

ENGINE INDICATED TORQUE
(FT-LB)
FIGURE 2
ENGINE TORQUEMETER CALIBRATION
1700-GE-791 S/N 374-114

NOTES: 1. POWER TURBINE SPEED = 20952 RPM
2. ENGINE INSTALLED ON RIGHT SIDE OF AIRCRAFT

CORRECTION TO BE ADDED (FT-LB)

DYNAMOMETER TORQUE (FT-LB)

LINE OF ZERO ERROR

ENGINE INDICATED TORQUE (FT-LB)
coefficient and rotor speed. To maintain a constant ratio of gross weight to air density ratio (i.e., constant thrust coefficient) altitude was increased as fuel was consumed. Test day level flight power was corrected to standard day conditions by assuming that the test day dimensionless parameters, $C_{p_t}$, $C_{t_t}$, and $\mu_t$ were independent of atmospheric conditions. Consequently, the standard day dimensionless parameters $C_{p_s}$, $C_{t_s}$, and $\mu_s$ were identical to $C_{p_t}$, $C_{t_t}$, and $\mu_t$, respectively. From equation 1, the following relationship can be derived.

$$\text{SHP}_s = \frac{\rho_s}{\rho_t} \times \text{SHP}_t$$

where:

- $t$ = Test day
- $s$ = Standard day

**HANDLING QUALITIES**

8. Stability and control data were collected and evaluated using standard test methods as described in reference 15, appendix A. The Handling Qualities Rating Scale presented in figure 3 was used to augment pilot comments relative to handling qualities. Turbulence reporting criteria used during this evaluation are presented in table 1.

**VIBRATION**

9. The vibration data were reduced by means of a fast Fourier transform from the analog flight tape. Peak vibration amplitudes were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale, presented in figure 4, was used to augment crew comments on aircraft vibration levels.

**AIRSPEED SYSTEM CALIBRATION**

10. The boom airspeed system was calibrated by using a pace aircraft to determine the airspeed and altimeter position errors.
Figure 3. Handling Qualities Rating Scale
<table>
<thead>
<tr>
<th>DEGREE OF VIBRATION</th>
<th>DESCRIPTION</th>
<th>PILOT RATING</th>
</tr>
</thead>
<tbody>
<tr>
<td>No vibration</td>
<td>Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.</td>
<td>0</td>
</tr>
<tr>
<td>Slight</td>
<td>Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.</td>
<td>1 2 3</td>
</tr>
<tr>
<td>Moderate</td>
<td>Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.</td>
<td>4 5 6</td>
</tr>
<tr>
<td>Severe</td>
<td>Sole preoccupation of aircrew is to reduce vibration level.</td>
<td>7 8 9</td>
</tr>
<tr>
<td>Intolerable</td>
<td></td>
<td>10</td>
</tr>
</tbody>
</table>

1 Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 4. Vibration Rating Scale
Calibrated airspeed was obtained by correcting indicated values for instrument error and position error (fig. 1, app C). Altitude position error was calculated from the airspeed position error by assuming that all errors in the pitot-static system were in the static side. The following equation was used:

\[
\Delta p_a = 1.4 \, P_o \left( \frac{V_{ic}}{a_o} \right) \left[ 1 + 0.2 \left( \frac{V_{ic}}{a_o} \right)^2 \right]^{2.5} \left( \frac{\Delta V_{pc}}{a_o} \right)
\]

+ 0.7 \, P_o \left[ 1 + 0.2 \left( \frac{V_{ic}}{a_o} \right)^2 \right]^{1.5} \left[ 1 + 1.2 \left( \frac{V_{ic}}{a_o} \right)^2 \right] \left( \frac{\Delta V_{pc}}{a_o} \right)^2
\]

Where:

- \( \Delta p_a \) = Static position error
- \( P_o \) = Atmospheric pressure at standard-day sea level
- \( V_{ic} \) = Instrument corrected indicated airspeed
- \( a_o \) = Speed of sound at standard-day sea level
- \( \Delta V_{pc} \) = Measured airspeed position error.

### DEFINITIONS

20. The following definitions of deficiencies and shortcomings were used during this evaluation.

a. 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 equipment's operational capability.

b. 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 useability of the material or end product.
<table>
<thead>
<tr>
<th>Intensity</th>
<th>Aircraft Reaction</th>
<th>Reaction Inside Aircraft</th>
<th>Reporting Term-Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>Light</td>
<td>Turbulence that momentarily causes slight erratic changes in altitude and/or attitude (pitch, roll, yaw). Report as Light Turbulence.</td>
<td>Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly.</td>
<td></td>
</tr>
<tr>
<td></td>
<td>or Turbulence that causes slight, rapid and somewhat rhythmic humpiness without appreciable changes in altitude or attitude. Report as Light Chop</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Moderate</td>
<td>Turbulence that is similar to Light Turbulence but of greater intensity. Changes in altitude and/or attitude occur but the aircraft remains in positive control at all times. It usually causes variations in indicated airspeed. Report as Moderate Turbulence.</td>
<td>Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged.</td>
<td>Occasional - Less than 1/3 of the time</td>
</tr>
<tr>
<td></td>
<td>or Turbulence that is similar to Light Chop but of greater intensity. It causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude. Report as Moderate Chop.</td>
<td></td>
<td>Intermittent - 1/3 to 2/3</td>
</tr>
<tr>
<td>Severe</td>
<td>Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control. Report as Severe Turbulence.</td>
<td>Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about.</td>
<td>Continuous - More than 2/3</td>
</tr>
<tr>
<td>Extreme</td>
<td>Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage. Report as Extreme Turbulence.</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
APPENDIX E. TEST DATA

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<td>10 through 15</td>
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<td>30 and 31</td>
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<td>50 through 52</td>
</tr>
<tr>
<td>Airspeed Calibration</td>
<td>53 and 54</td>
</tr>
</tbody>
</table>
FIGURE I
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 62-23355

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 290 RPM
3. AVG LONGITUDINAL CB LOCATION FS 206.4 (AFT)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 4 THROUGH 8

THrust COEFFICIENT, CT X 10^4 = \frac{6W}{\rho A(OR)^2} X 10^4
NOTE: 1. H-HELFIRE CONFIGURATION  
2. MAIN ROTOR SPEED = 200 RPM  
3. AVG. LONGITUDINAL CG LOCATION FS 206.4 (AFT)  
4. ZERO SIDESLIP  
5. CURVES DERIVED FROM FIGURES 4 THROUGH 8

POWER COEFFICIENT, 
\( CP \times 10^5 = \frac{SW \times 550}{\rho (A DR)^3} \)

THRUST COEFFICIENT, 
\( CT \times 10^4 = \frac{GW}{\rho A (DR)^2} \times 10^4 \)
FIGURE 8
Nondimensional Level Flight Performance
AH-64A USA S/N 82-23355

Notes:
1. B-HELLFIRE CONFIGURATION
2. MAIN ROTOR SPEED = 298 RPM
3. AVG LONGITUDINAL CG LOCATION FS 206.4 (AFT)
4. ZERO SIDESLIP
5. CURVES DERIVED FROM FIGURES 4 THROUGH 8

Power Coefficient, $C_p = \frac{550}{\rho AC_0 R^2} \times 10^5$

Thrust Coefficient, $C_T \times 10^4 = \frac{6W}{\rho AC_0 R^2} \times 10^4$
FIGURE 4

LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-29355

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>GROSS</td>
<td>LONGITUDINAL</td>
<td>DENSITY</td>
<td>ALTITUDE</td>
<td>OAT</td>
<td>SPEED</td>
</tr>
<tr>
<td>WEIGHT</td>
<td>DG LOCATION</td>
<td>(FEET)</td>
<td>(DEG:C)</td>
<td>(RPM)</td>
<td>CONFIGURATION</td>
</tr>
<tr>
<td>(LB)</td>
<td>(FS)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>14500</td>
<td>208.2 (CAFT)</td>
<td>7630</td>
<td>18.6</td>
<td>290</td>
<td>0.087991</td>
</tr>
</tbody>
</table>

NOTE: ZERO SIDESLIP

SHAFT HORSEPOWER REQUIRED

TRUE AIRSPEED (KNOTS)

CURVE DERIVED FROM FIGURES 1 THROUGH 3
**FIGURE 5**
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL LOCATION (FT)</th>
<th>AVG DENSITY (PSI)</th>
<th>AVG ALTITUDE (FEET)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG ROTOR SPEED (RPM)</th>
<th>AVG THRUST THRU COEFF.</th>
<th>CONFIGURATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14400</td>
<td>206.6 (AFT)</td>
<td>9820</td>
<td>16.8</td>
<td>290</td>
<td>0.006408</td>
<td>8-HELLFIRE</td>
<td></td>
</tr>
</tbody>
</table>

**NOTE:** ZERO SIDESLIP

The diagram shows a curve derived from figures 1 through 3.

**TRUE AIRSPEED (KNOTS)**
FIGURE 8
LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>GROSS WEIGHT (LB)</td>
<td>LONGITUDINAL CG LOCATION (PS)</td>
<td>ALTITUDE (FEET)</td>
<td>DENSITY (DEG C)</td>
<td>OAT (RPH)</td>
</tr>
<tr>
<td>14/70</td>
<td>206.7 (AFT)</td>
<td>10000</td>
<td>13.5</td>
<td>200</td>
</tr>
<tr>
<td>THRUST COEFFICIENT</td>
<td>CONFIGURATION</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>0.000973</td>
<td>0-HELFIRE</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

NOTE: ZERO SIDESLIP

![Graph of specific range vs. true airspeed](true airspeed (knots))

Curve derived from figures 1 through 3
FIGURE 7
LEVEL FLIGHT PERFORMANCE
AH-64A USA 57N 82-23355

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL CG LOCATION (FT)</th>
<th>AVG DENSITY (PSI)</th>
<th>AVG ALTIMETER (FT)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG Rotor Speed (RPM)</th>
<th>AVG THRUST COEFF.</th>
<th>CONFIGURATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14650</td>
<td>200.2 (AFT)</td>
<td>12420</td>
<td>0.5</td>
<td>290</td>
<td>0.080525</td>
<td>8-HELLFIRE</td>
<td></td>
</tr>
</tbody>
</table>

NOTE: ZERO SIDESLIP

CURVE DERIVED FROM FIGURES 1 THROUGH 3
**FIGURE 8**

LEVEL FLIGHT PERFORMANCE
AH-64A USA S/N 02-2355

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL MOMENT (PS)</th>
<th>AVG DENSITY (FT3/SL)</th>
<th>AVG ALTITUDE (FT)</th>
<th>AVG OAT DEG C</th>
<th>AVG SPEED (KNOTS)</th>
<th>AVG COEFF. THRUST</th>
<th>AVG CONFIGURATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>15900</td>
<td>200.1 (AFT)</td>
<td>13600</td>
<td>8.0</td>
<td>200</td>
<td>0.018082</td>
<td>B-HELLFIRE</td>
<td></td>
</tr>
</tbody>
</table>

**NOTE:** ZERO SIDESLIP

SHAFT HORSEPOWER REQUIRED

CURVE DERIVED FROM FIGURES 1 THROUGH 3

TRUE AIRSPEED (KNOTS)

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### Figure 9

**Level Flight Performance**

AH-64A USA S/N 82-23356

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL CO LOCATION (Ft)</th>
<th>AVG DENSITY (FEET)</th>
<th>AVG GAT (DEG C)</th>
<th>AVG SPEED (KPH)</th>
<th>AVG THRUST COEFF.</th>
<th>CONFIG</th>
</tr>
</thead>
<tbody>
<tr>
<td>14684</td>
<td>206.4 (AFT)</td>
<td>7128</td>
<td>35.0</td>
<td>260</td>
<td>0.000618</td>
<td>0-HELLFIRE</td>
</tr>
</tbody>
</table>

**Note:** Dashed line denotes ASFC's data corrected for a difference in longitudinal CG (≈0.1 square feet, equivalent flat plate area)

---

**Graph:**

- **Maximum Continuous Power**
- **True Airspeed (Knots)**
- **Shaft Horsepower Required**

---

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FIGURE 10
LIMITS OF CYCLIC CONTROL TRAVEL
AH-64A USA S/N 82-23355

NOTES:
1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED
   BY GROUND POWER UNITS
3. CONTROL POSITIONS MEASURED AT CENTER OF GRIP.
FIGURE 11
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N B2-23355

NOTES: 1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED
   BY GROUND POWER UNITS
3. FORCES AND POSITIONS MEASURED AT LOWER THIRD OF GRIP
4. AVERAGE FORCE GRADIENT NEAR TRIM: 1.45 LB/IN.
   FWD, 1.54 LB/IN. AFT
5. BREAKOUT INCLUDING FRICTION: 2.6 LB FWD, 3.4 LB AFT
6. AVERAGE FRICTION BAND: 1.5 LB FWD, 0.8 LB AFT
7. COPILOT STICK RETRACTED

LONGITUDINAL CONTROL POSITION
(INCHES FROM FULL FORWARD)
NOTES:
1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED
   BY GROUND POWER UNITS
3. FORCES AND POSITIONS MEASURED AT LOWER THIRD OF GRIP
4. AVERAGE FORCE GRADIENT NEAR TRIM: 1.35 LB/IN.
   FWD, 1.45 LB/IN. AFT
5. BREAKOUT INCLUDING FRICTION: 3.0 LB FWD AND AFT
6. AVERAGE FRICTION BAND: 1.5 LB FWD, 0.5 LB AFT
7. COPILOT STICK EXTENDED

LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355
FIGURE 13
LATERAL CONTROL SYSTEM CHARACTERISTICS
AH-64A USA S/N 82-23355

NOTES:
1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED
   BY GROUND POWER UNITS
3. FORCES AND POSITIONS MEASURED AT LOWER THIRD OF GRIP
4. AVERAGE FORCE GRADIENT NEAR TRIM: 0.78 LB/IN.
   LEFT, 0.66 LB/IN. RIGHT
5. BREAKOUT INCLUDING FRICTION: 2.6 LB LT, 2.0 LB RT
6. AVERAGE FRICTION BAND: 2.6 LB LEFT, 2.0 LB RIGHT

LATERAL CONTROL FORCE
(GROUNDS)

LATERAL CONTROL POSITION
(INCHES FROM FULL LEFT)
FIGURE 14
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
AI-04A USA S/N 62-22355

NOTES: 1. ROTORS STATIC
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED
   BY GROUND POWER UNITS
3. FORCES AND POSITIONS MEASURED AT CENTER OF PEDAL
4. AVERAGE FORCE GRADIENT NEAR TRIM: 1.7 LB/IN.
   LEFT, 2.0 LB/IN. RIGHT
5. BREAKOUT INCLUDING FRICTION: 9.8 LB LT, 8.8 LB RT
6. AVERAGE FRICTION BAND: 10.6 LB LEFT, 10.2 LB RIGHT
FIGURE 16
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
AH-64A USA S/N 88-0356

| AVG GROSS | AVG LON. | AVG ALTITUDE | AVG AT. | AVG ROTOR |
| WEIGHT | LOCATION | (Ft) | SPEED | (FOOT) |
| 1477 | 200.7 (AFT) | 18000 | 18.6 | 200 |

NOTES:
1. SHADeD SYMBOLS DENOTE DIVING FLIGHT
2. Open symbols denote level flight
3. Dash on, attitude hold off
4. S-HELLFIRE CONFIGURATION.
NOTES: 1. DAE ON, ATTITUDE HOLD OFF
2. 8-HELLFIRE CONFIGURATION

TOTAL COLLECTIVE CONTROL TRAVEL = 11.75 INCHES

TOTAL DIRECTIONAL CONTROL TRAVEL = 9.30 INCHES

TOTAL LATERAL CONTROL TRAVEL = 0.45 INCHES

TOTAL LONIGITUDINAL CONTROL TRAVEL = 10.5 INCHES
FIGURE 10
STATIC LATERAL STABILITY
AH-64A USA 3/4 82-2355

<table>
<thead>
<tr>
<th>AVG LONG CG</th>
<th>AVG DENSITY</th>
<th>AVG ROTOR</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>(FPS)</td>
<td>(FT)</td>
<td>(DEG C)</td>
<td>(KNOTS)</td>
</tr>
<tr>
<td>206.5 (AFT)</td>
<td>7140</td>
<td>24.0</td>
<td>290</td>
</tr>
<tr>
<td>LEVEL</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

NOTES:
1. SHAPED SYMBOLS DENOTE TRIM
2. BASE ON ATTITUDE HOLD OFF
9. S-HELLFIRE CONFIGURATION

TOTAL DIRECTIONAL CONTROL TRAVEL = 8.06 INCHES

TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES

TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES

CALIBRATED AIRSPEED (KNOTS)

60 70 80 90 100
FIGURE 10
STATIC LATERAL STABILITY
AH-64D USA S/N 82-23355

<table>
<thead>
<tr>
<th>AVG WEIGHT (LB)</th>
<th>AVG LONG C G (FT)</th>
<th>AVG DENSITY (PSI)</th>
<th>AVG Rotor Speed (RPM)</th>
<th>AVG FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14578</td>
<td>208.7 (AFT)</td>
<td>6888</td>
<td>25.8</td>
<td>267</td>
</tr>
</tbody>
</table>

NOTES: 1. SHADEd SYMBOLS DENOTE TRIM
2. BASE ON, ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

TOTAL DIRECTIONAL CONTROL TRAVEL = 6.96 INCHES

TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES

TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES

CALIBRATED AIRSPEED (KNOTS)
<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LOAD CE LOCATION (PS)</th>
<th>AVG DENSITY (FEET)</th>
<th>AVG ALTITUDE (DEG C)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG RPM (RPM)</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14736</td>
<td>266.5 AFT</td>
<td>7449</td>
<td>24.0</td>
<td>289</td>
<td>CLIMB</td>
<td></td>
</tr>
</tbody>
</table>

**NOTES:**
1. SHADED SYMBOLS DENOTE TRIM
2. BASE ON, ATTITUDE HOLD OFF
3. B-HELLFIRE CONFIGURATION

**TOTAL DIRECTIONAL CONTROL TRAVEL = 6.06 INCHES**

**TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES**

**TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES**
Figure 21
STATIC LATERAL STABILITY
AH-64A USA S/N 92-23965

Table:

<table>
<thead>
<tr>
<th>AVG WEIGHT</th>
<th>AVG LOCATION</th>
<th>AVG DENSITY</th>
<th>AVG ALTITUDE</th>
<th>AVG DAY</th>
<th>AVG SPEEED</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>23124 LBS</td>
<td>200.8 FT AGL</td>
<td>7.52</td>
<td>7340 FT</td>
<td>23-5</td>
<td>294</td>
<td>DESCENT</td>
</tr>
</tbody>
</table>

Notes:
1. Shaded symbols denote trim.
2. Base on attitude hold off.
3. 8-Hellfire configuration.

Graphs:
- Total directional control travel = 6.96 inches
- Total lateral control travel = 0.45 inches
- Total longitudinal control travel = 10.5 inches

Calibrated airspeed (knots)
## Figure 22
### Static Lateral-Directional Stability
AH-64A USA S/N 82-2355

<table>
<thead>
<tr>
<th>Average Longitudinal CG Location (ft)</th>
<th>Average Rotor Speed (RPM)</th>
<th>Trim</th>
<th>Flight Condition</th>
</tr>
</thead>
<tbody>
<tr>
<td>395.5 (AST)</td>
<td>176</td>
<td>108</td>
<td>70</td>
</tr>
<tr>
<td>266.6 (AST)</td>
<td>226</td>
<td>208</td>
<td>70 (level)</td>
</tr>
</tbody>
</table>

**Notes:**
1. Shaded symbols denote trim
2. Case ON, attitude hold off
3. 8-HELLFIRE CONFIGURATION

**Total Longitudinal Control Travel:** 19.5 inches

**Total Lateral Control Travel:** 3.45 inches

**Total Directional Control Travel:** 6.08 inches
FIGURE 23
STATIC LATERAL-DIRECTIONAL STABILITY
AH-64A USA S/N 82-23335

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT</th>
<th>AVG L/G LOCATION</th>
<th>AVG ALTITUDE</th>
<th>AVG OAT</th>
<th>AVG ROTOR SPEED</th>
<th>AVG AIRSPEED</th>
<th>TRIM CALIBRATION</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14578</td>
<td>706.7 LFTS</td>
<td>7050</td>
<td>20.6</td>
<td>207</td>
<td>130</td>
<td>LEVEL</td>
<td></td>
</tr>
</tbody>
</table>

NOTES:
1. SHADED SYMBOLS DENOTE TREM
2. BASE ON, ATTITUDE HOLD OFF
3. B-HELLFIRE CONFIGURATION

**ROLL ANGLE**
TOTAL LATERAL CONTROL TRAVEL = 18.5 INCHES

**LONGITUDINAL ATTITUDE**
TOTAL LONGITUDINAL CONTROL TRAVEL = 8.45 INCHES

**LATERAL CONTROL POSITION**
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.06 INCHES

**ANGLE OF SIDESLIP**
(Degrees)
### FIGURE 24

**STATIC LATERAL-DIRECTIONAL STABILITY**

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LBS)</th>
<th>AVG LONG CO LOCATION (FTS)</th>
<th>AVG DENSITY (FTS)</th>
<th>AVG GAT ATTITUDE (DEG)</th>
<th>AVG ROTOR SPEED (RPM)</th>
<th>AVG AIRSPEED (KTS)</th>
<th>TRIM CALIBRATION</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14670</td>
<td>260.3 (AFD)</td>
<td>7250</td>
<td>21.5</td>
<td>280</td>
<td>81</td>
<td>CLIMB</td>
<td></td>
</tr>
</tbody>
</table>

**NOTES:**
1. SHAPED SYMBOLS DENOTE TRIM
2. BASE ON ATTITUDE HOLD OFF
3. 8-HELLFIRE CONFIGURATION

**TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES**

**TOTAL DIRECTIONAL CONTROL TRAVEL = 6.66 INCHES**

**TOTAL LONGITUDINAL CONTROL TRAVEL = 13.5 INCHES**
FIGURE 25
STATIC LATERAL-DIRECTIONAL STABILITY
AH-64A USA 571 92-23955

AVG  AVG  AVG  AVG  TRIM
GROSS  LONG. WGT  DENSITY  AVG  Rotor  C.G.  FLIGHT
WEIGHT  LOCATION  ALTITUDE  OAT  SPEED  AIRSPEED  CONDITION
(LB)  (FT)  (PSI)  (F)  (RPM)  (KT)  DESCENT
14576  286.3 (left)  0800  21.5  280  86  DESCENT

NOTES:
1. SHADED SYMBOLS DENOTE TRIM
2. BASE ON ALTITUDE HOLD OFF
3. G-HELLFIRE CONFIGURATION

TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES

TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES

TOTAL DIRECTIONAL CONTROL TRAVEL = 6.06 INCHES

ANGLE OF SIDESLIP
(DEGREES)
Figure 26
MANEUVERING STABILITY
AH-64A USA 3/N 82-2335

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>CG LOCATION (Ft)</th>
<th>DENSITY (PSI)</th>
<th>OAT (°F)</th>
<th>SPEED (KNOTS)</th>
<th>AVGS</th>
<th>Rotor Calib</th>
</tr>
</thead>
<tbody>
<tr>
<td>14900</td>
<td>208.1</td>
<td>71.6</td>
<td>-23.6</td>
<td>290</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

- RIGHT TURN
- LEFT TURN
- PULL UP
- PUSH OVER

TOTAL LATERAL CONTROL TRAVEL = 8.45 INCHES

TOTAL LONGITUDINAL CONTROL TRAVEL = 10.5 INCHES
FIGURE 27
LONG TERM DYNAMIC STABILITY
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14690</td>
<td>206.4 (AFT)</td>
<td>7370</td>
<td>29.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

NOTE: DASE ON ATTITUDE HOLD ON

PITCH ATTITUDE

INDICATED AIRSPEED

PITCH RATE

ANGLE OF ATTACK

PITCH SAS POSITION

HORIZONTAL STABILIZER INCIDENCE

LONGITUDINAL CONTROL POSITION
FIGURE 28
LONG TERM DYNAMIC STABILITY
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY (LT)</th>
<th>ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14690</td>
<td>206.4 (AFT)</td>
<td>7030</td>
<td>29.0</td>
<td>290</td>
<td>LEVEL</td>
<td></td>
</tr>
</tbody>
</table>

NOTE: DASE ON, ATTITUDE HOLD OFF

- PITCH ATTITUDE
- INDICATED AIRSPEED
- PITCH RATE
- PITCH SAS POSITION
- ANGLE OF ATTACK
- HORIZONTAL STABILIZER INCIDENCE
- LONGITUDINAL CONTROL POSITION
FIGURE 29
DYNAMIC STABILITY
AH-64A (USA) S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14690</td>
<td>206.4 (AFT)</td>
<td>6820</td>
<td>29.5</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

NOTE: BASE OFF.

PITCH ATTITUDE
INDICATED AIRSPEED
PITCH RATE
ANGLE OF ATTACK
PITCH SAS POSITION
HORIZONTAL STABILIZER INCIDENCE
LONGITUDINAL CONTROL POSITION

RECOVERY.
### Table: Dynamic Stability

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>ALTITUDE (FT)</th>
<th>DENSITY</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14690</td>
<td>206.4 (AFT)</td>
<td>6990</td>
<td></td>
<td>28.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

**Figure 30**

**Dynamic Stability**

AH-64A USA S/N 82-23355

- **Gross Weight (LB):** 14690
- **Long. CG Location (FS):** 206.4 (AFT)
- **Altitude (FT):** 6990
- **Density:**
- **OAT (DEG C):** 28.0
- **Rotor Speed (RPM):** 290
- **Trim Flight Condition:** LEVEL

**Note:** Base on, attitude hold on.
FIGURE 31
DYNAMIC STABILITY
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14690</td>
<td>206.4 (AFT)</td>
<td>6820</td>
<td>29.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

- PITCH ATTITUDE
- INDICATED AIRSPEED
- PITCH RATE
- ANGEL OF ATTACK
- PITCH SAS POSITION
- HORIZONTAL STABILIZER INCIDENCE
- LONGITUDINAL CONTROL POSITION
**Figure 32**

**Control System Imbalance**

AH-64A USA S/N 82-2335

<table>
<thead>
<tr>
<th>Gross Weight (LB)</th>
<th>Long. CG Location (FS)</th>
<th>Density Altitude (FT)</th>
<th>OAT (Deg C)</th>
<th>Rotor Speed (RPM)</th>
<th>Trim Flight Condition</th>
</tr>
</thead>
<tbody>
<tr>
<td>14770</td>
<td>206.4 (AFT)</td>
<td>7620</td>
<td>24.0</td>
<td>290</td>
<td>Level</td>
</tr>
</tbody>
</table>

*Graph showing各项参数随时间的变化图：*

- Pitch Attitude
- Indicated Airspeed
- Pitch Rate
- Angle of Attack
- Pitch SAS Position
- Horizontal Stabilizer Incidence
- Longitudinal Control Position

*Graph axes:*
- Time (Seconds)
- Pitch Attitude (Degree)
- Indicated Airspeed (Knots)
- Pitch Rate (Degree/Second)
- Angle of Attack (Degree)
- Pitch SAS Position (Percent)
- Horizontal Stabilizer Incidence (Degree)
- Longitudinal Control Position (Percent)
**FIGURE 33**
AFT LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FTS)</th>
<th>DENSITY AT ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14650</td>
<td>206.7 (AFT)</td>
<td>6960</td>
<td>-27.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

**NOTE:** DASE ON, ATTITUDE HOLD ON

- **PITCH ATTITUDE**
- **INDICATED AIRSPEED**
- **PITCH RATE**
- **PITCH SAS POSITION**
- **ANGLE OF ATTACK**
- **HORIZONTAL STABILIZER INCIDENCE**
- **LONGITUDINAL CONTROL POSITION**

**ELAPSED TIME SECONDS**
FIGURE 35
FORWARD LONGITUDINAL PULSE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14650</td>
<td>206.4 (AFT)</td>
<td>7110</td>
<td>21.5</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

NOTE: DASE ON, ATTITUDE HOLD OFF
FIGURE 38B
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB) | LONG. CG LOCATION (FS) | DENSITY ALTITUDE (FT) | OAT (DEG C) | ROTOR SPEED (RPM) | TRIM FLIGHT CONDITION
---|---|---|---|---|---
14760 | 206.6 (AFT) | 7390 | 20.5 | 290 | LEVEL

NOTE: DASE ON, ATTITUDE HOLD ON

ROLL ATTITUDE

ROLL RATE

ROLL SAS POSITION

YAW RATE

LATERAL CONTROL POSITION

ELAPSED TIME SECONDS
FIGURE 39B
RIGHT PEDAL PULSE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FT)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14650</td>
<td>206.4 (AFT)</td>
<td>6950</td>
<td>22.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

NOTE: DASE ON, ATTITUDE HOLD OFF

ROLL ATTITUDE

ROLL RATE

ROLL SAS POSITION

YAW RATE

LATERAL CONTROL POSITION (HORSES FROM ROLL LEFT)
Figure 39c
Right Pedal Pulse
AH-64A USA S/N 82-23355

Gross Weight (LB) 14650
Long. CG Location (FS) 206.4 (AFT)
Density Altitude (FT) 6950
OAT (DEG C) 22.0
Rotor Speed (RPM) 290
Trim Flight Condition LEVEL

Note: Base ON, Attitude Hold Off

Yaw Attitude

Yaw Rate

Yaw SAS Position

Roll Rate

Directional Control Position
NOTE: DASE OFF
FIGURE 40C
LEFT PEDAL PULSE
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
<th>ROTOR SPEED (RPM)</th>
<th>TRIM FLIGHT CONDITION LEVEL</th>
</tr>
</thead>
<tbody>
<tr>
<td>14760</td>
<td>206.6 (AFT)</td>
<td>7270</td>
<td>21.0</td>
<td>290</td>
<td>LEVEL</td>
</tr>
</tbody>
</table>

**NOTE:** BASE OFF

**YAW ATTITUDE**

**YAW RATE**

**YAW SAS POSITION**

**ROLL RATE**

**DIRECTIONAL CONTROL POSITION**
FIGURE 41A
RECOVERY FROM AUTOROTATION
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>15090</td>
<td>206.5 (AFT)</td>
<td>7180</td>
<td>22.0</td>
</tr>
</tbody>
</table>

LEFT ENGINE T4.5

SIDE SLIP ANGLE

RIGHT ENGINE T4.5

PITCH ATTITUDE

RIGHT ENGINE Ng

COLLECTIVE CONTROL POSITION

LEFT ENGINE Ng
FIGURE 41B
RECOVERY FROM AUTOROTATION
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>15090</td>
<td>206.5 (AFT)</td>
<td>7180</td>
<td>22.0</td>
</tr>
</tbody>
</table>

- **RIGHT ENGINE FUEL FLOW**
- **MAIN ROTOR SPEED**
- **LEFT ENGINE FUEL FLOW**
- **RIGHT ENGINE Np**
- **RIGHT ENGINE TORQUE**
- **LEFT ENGINE Np**
- **LEFT ENGINE TORQUE**
FIGURE 42
QUICK STOP
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY (FT)</th>
<th>OAT (DEG C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>14800</td>
<td>206.1 (AFT)</td>
<td>7150</td>
<td>22.5</td>
</tr>
</tbody>
</table>
FIGURE 43A
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)  14690
LONG. CG LOCATION (FS)  206.4 (AFT)
DENSITY ALTITUDE (FT)  7290
OAT (DEG C)  29.0

LEFT ENGINE FUEL FLOW

CG NORMAL ACCELERATION

MAIN ROTOR SPEED

LEFT ENGINE Ng

LEFT ENGINE Np

PILOT EVENT MARKER

COLLECTIVE CONTROL POSITION
Figure 45C
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT
(LB) 14690

LONG. CG LOCATION (FS) 206.4 (AFT)

DENSITY ALTITUDE (FT) 6500

OAT (DEG C) 30.0

ROLL ATTITUDE

ROLL RATE

INDICATED AIRSPEED

YAW RATE

LATERAL CONTROL POSITION

Time (Sec)
FIGURE 45D
PULL UP
AH-64A USA S/N 82-23355

GROSS
WEIGHT (LB) 14690
LONG. CG LOCATION (FS) 206.4 (AFT)
DENSITY ALTITUDE (FT) 6500
OAT (DEG C) 30.0
FIGURE 46B
PULL UP
AH-64A USA S/N 82-23355

GROSS WEIGHT (LB)  15090
LONG. CG LOCATION (FS)  206.5 (AFT)
DENSITY ALTITUDE (FT)  7030
OAT (DEG C)  22.0

RIGHT ENGINE FUEL FLOW

MAIN ROTOR SPEED

LEFT ENGINE FUEL FLOW

RIGHT ENGINE Np

RIGHT ENGINE TORQUE

LEFT ENGINE Np

LEFT ENGINE TORQUE

TIME (SEC)

FUEL FLOW (GAL/HR)
<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>14770</td>
<td>206.4 (AFT)</td>
<td>6460</td>
<td>25.0</td>
</tr>
</tbody>
</table>

**Figure 47A**
PULL UP
AH-64A USA S/N 82-23355

- **LEFT ENGINE Np**
- **CG NORMAL ACCELERATION**
- **RIGHT ENGINE TORQUE**
- **COLLECTIVE LOWERED 1.8 IN.**
- **COLLECTIVE CONTROL POSITION**
- **LEFT ENGINE TORQUE**
- **RIGHT ENGINE Np**
- **MAIN ROTOR SPEED**
FIGURE 47B
PULL UP
AH-64A USA S/N 82-23355

<table>
<thead>
<tr>
<th>GROSS WEIGHT (LB)</th>
<th>LONG. CG LOCATION (FS)</th>
<th>DENSITY ALTITUDE (FT)</th>
<th>OAT (DEG C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>14770</td>
<td>206.4 (AFT)</td>
<td>6460</td>
<td>25.0</td>
</tr>
</tbody>
</table>

- PITCH ATTITUDE
- INDICATED AIRSPEED
- PITCH RATE
- ANGLE OF ATTACK
- PITCH SAS POSITION
- HORIZONTAL STABILIZER INCIDENCE
- LONGITUDINAL CONTROL POSITION
Figure 50
VIBRATION CHARACTERISTICS
AH-64A USA S/N 82-23355
PILOT SEAT

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG CG LOCATION (FS)</th>
<th>AVG DENSITY (FEET)</th>
<th>AVG ALTITUDE (DEG C)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG SPEED (RPM)</th>
<th>FLIGHT LEVEL</th>
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<tbody>
<tr>
<td>14778</td>
<td>206.7 (AFT)</td>
<td>18800</td>
<td>13.5</td>
<td>200</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

- □ VERTICAL
- △ LATERAL
- ◆ LONGITUDINAL

NOTES:
1. SHADED SYMBOLS DENOTE DIVES
2. LINES DENOTE SYSTEM SPECIFICATION LIMITS

- 8/REV = 38.5 Hertz
- 4/REV = 19.3 Hertz
- 2/REV = 9.6 Hertz
- 1/REV = 4.8 Hertz

CALIBRATED AIRSPEED (KNOTS)
FIGURE 51
VIBRATION CHARACTERISTICS
AH-64A USA S/N 82-29355
COPILOT SEAT

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL CG LOCATION (IN)</th>
<th>AVG DENSITY (PSI)</th>
<th>AVG ALTITUDE (FT)</th>
<th>AVG OAT (°F)</th>
<th>AVG R.P.M.</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14770</td>
<td>200.7 (AFT)</td>
<td>10600</td>
<td>19.5</td>
<td>290</td>
<td>LEVEL</td>
<td></td>
</tr>
</tbody>
</table>

NOTES: 1. SHAPED SYMBOLS DENOTE DIVES
2. LINES DENOTE SYSTEM SPECIFICATION LIMITS

- □ VERTICAL
- ▲ LATERAL
- ◇ LONGITUDINAL

- 8/REV = 38.5 Hertz
- 4/REV = 19.9 Hertz
- 2/REV = 9.6 Hertz
- 1/REV = 4.8 Hertz

CALIBRATED AIRSPEED (KNOTS)

187
**Figure 52**

**Vibration Characteristics**

**AH-64A USA S/N 82-23955**

**Aircraft Center of Gravity**

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL CG LOCATION (FT)</th>
<th>AVG DENSITY (FT/SEC²)</th>
<th>AVG ALTITUDE (FT)</th>
<th>AVG OAT (DEG C)</th>
<th>AVG ROTOR SPEED (RPM)</th>
<th>FLIGHT LEVEL</th>
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</thead>
<tbody>
<tr>
<td>14770</td>
<td>200.7 (AFT)</td>
<td>18800</td>
<td>13.5</td>
<td>200</td>
<td>200</td>
<td>VERTICAL</td>
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</table>

Note: Shaded symbols denote dives.

- **8/REV = 38.5 Hertz**
- **4/REV = 19.3 Hertz**
- **2/REV = 9.6 Hertz**
- **1/REV = 4.8 Hertz**

**Calibrated Airspeed (Knots)**

188
FIGURE 53
Airspeed Calibration
AH-64A USA S/N 92-23355
Pilot Pitot-Static System

<table>
<thead>
<tr>
<th>AVG GROSS WEIGHT (LB)</th>
<th>AVG LONGITUDINAL CG LOCATION (Ft)</th>
<th>AVG DENSITY (FEET)</th>
<th>AVG DAT (DEG C)</th>
<th>AVG ROTOR SPEED (RPM)</th>
<th>FLIGHT CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>14800</td>
<td>206.4 (AFT)</td>
<td>7250</td>
<td>20.5</td>
<td>290</td>
<td>LEVEL</td>
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</table>

Diagram showing correction to be added (knots) versus instrument corrected airspeed (knots).

Note: Shaded symbols denote dives.

Line of zero error.
FIGURE 54
Airspeed Calibration
AH-64A USA S/N 82-23355
Copilot Pitot-Static System

<table>
<thead>
<tr>
<th>Avg Gross Weight (LB)</th>
<th>Avg Longitudinal CG Location (PS)</th>
<th>Avg Density (FEET)</th>
<th>Avg OAT (DEG C)</th>
<th>Avg Rotor Speed (RPM)</th>
<th>Flight Condition</th>
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</thead>
<tbody>
<tr>
<td>14860</td>
<td>286.4 (AFT)</td>
<td>7250</td>
<td>28.5</td>
<td>290</td>
<td>Level</td>
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</table>

Correction to be added (knots)

Note: shaded symbols denote dives

Calibrated Airspeed (knots)

Instrument Corrected Airspeed (knots)

Line of Zero Error
# APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

<table>
<thead>
<tr>
<th>NUMBER</th>
<th>SUBJECT</th>
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<tbody>
<tr>
<td>84-10-1</td>
<td>SDC Failure</td>
</tr>
<tr>
<td>84-10-2</td>
<td>Primary hydraulic pump failure</td>
</tr>
<tr>
<td>84-10-3</td>
<td>Engine anti-ice valve failure</td>
</tr>
<tr>
<td>84-10-4</td>
<td>SDC failure</td>
</tr>
<tr>
<td>84-10-5</td>
<td>Cracks in vertical stabilizer support mounts</td>
</tr>
<tr>
<td>84-10-6</td>
<td>RPM warning box</td>
</tr>
<tr>
<td>84-10-7</td>
<td>Aft fuel tank contamination</td>
</tr>
<tr>
<td>84-10-8</td>
<td>Heading attitude reference system</td>
</tr>
<tr>
<td>84-10-9</td>
<td>DASH computer FD/LS check failures</td>
</tr>
<tr>
<td>84-10-10</td>
<td>IGB grease expulsion</td>
</tr>
<tr>
<td>84-10-11</td>
<td>Cracks in rubber of #2 blade damper</td>
</tr>
<tr>
<td>84-10-12</td>
<td>Lack of pitch reference in VDU</td>
</tr>
<tr>
<td>84-10-13</td>
<td>EMI in caution panel</td>
</tr>
<tr>
<td>84-10-14</td>
<td>Intermittant failure of the tail wheel to unlock</td>
</tr>
<tr>
<td>84-10-15</td>
<td>Failure of the HARS to consistently align with correct magnetic heading</td>
</tr>
<tr>
<td>84-10-16</td>
<td>The failure of the ADF set to provide aural identification and accurate bearing information</td>
</tr>
<tr>
<td>84-10-17</td>
<td>Failure of the #1 nose gear box</td>
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US Army Aviation Research and Technology Activity (AVSCOM) 2

Aviation Applied Technology Directorate (SAVRT-TY-DRD SAVRT-TY-TSC (Tech Library))
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<tr>
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<tr>
<td>Assistant Technical Director for Projects, Code: CT-24</td>
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<td>(Mr. Joseph Dunn)</td>
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<td>4520 Test Group (ENML)</td>
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<tr>
<td>Commander, Naval Air Systems Command (AIR 5115B, AIR 5301)</td>
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<td>Defense Intelligence Agency (DIA-DT-2D)</td>
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