

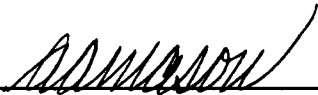
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FLYING QUALITIES AND PERFORMANCE
EVALUATION OF THE AH-64A
HELICOPTER

A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville

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ABSTRACT

The purpose of this evaluation was to assess the flying qualities and performance of the AH-64A helicopter for the USMC attack mission. The evaluation was completed during four flights totalling 6.0 flight hours and a 4.0 hour ground test. The Digital Automatic Stabilization Equipment (DASE) off flying qualities of the AH-64A will allow the pilot to continue the mission during a DASE failure with only a small increase in workload. In the event of a tail rotor failure, the engine chop collar will allow the pilot to reduce engine power to idle without removing his hand from the collective. The DASE off flying qualities and the engine chop collar are enhancing characteristics which should be incorporated in future designs. The excessive transient droop encountered during rapid collective application will result in settling of the aircraft into the ground or other obstacles to flight when conducting Nap of the Earth operations. The slow main rotor wind down following actuation of the rotor brake will result in excessive blade flapping and droop stop pounding during shipboard gusty wind conditions. The two previous deficiencies are Part I deficiencies which should be corrected as soon as possible. The extremely high lateral control sensitivity will result in an abrupt and jerky aircraft response to rapid lateral cyclic inputs and will result in pilot induced oscillations during

high gain tasks. The lateral control sensitivity is a Part II deficiency and nine additional Part II deficiencies were identified which should be corrected as soon as practicable. Eight Part III deficiencies were identified which should be avoided in future designs. Upon correction of the two Part I deficiencies identified during this evaluation, the AH-64A will have excellent potential for the USMC attack mission. A final determination of the suitability of the aircraft will be based on the results of a mission systems and a shipboard compatibility evaluation.

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LIST OF ABBREVIATIONS AND SYMBOLS

AC	Alternating current
A/C	Aircraft
ADSS	Air data sensor system
ADTA	Aviation Developmental Test Activity
AGL	Above ground level
AHRS	Attitude heading reference system
AOB	Angle of bank
APU	Auxiliary power unit
BO+F	Breakout plus friction
BUCS	Backup control system
CAS	Command augmentation system
CG	Center of gravity
Cp	Power coefficient
Ct	Thrust coefficient
DASE	Digital automatic stabilization equipment
DC	Direct current
DEP	Design eye position
ECU	Electrical control unit
EH	Electro-hydraulic
ESHP	Engine shaft horsepower
FLIR	Forward looking infrared system
FPM	Feet per minute
G	Normal acceleration due to gravity

GCA	Ground controlled approach
GSE	Ground support equipment
GW	Gross weight
HARS	Heading and attitude reference system
HAS	Hover augmentation system
HMU	Hydromechanical unit
Hp	Pressure altitude
HQR	Handling qualities rating
ICAO	International Civil Aviation Organization
IGE	In ground effect
IFR	Instrument flight rules
KIAS	Knots indicated airspeed
KTAS	Knots true airspeed
LDO	Lateral/directional oscillation
LDS	Load demand spindle
LVDT	Linear variable differential transducer
MIL PWR	Military power
MR	Main rotor
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
NATOPS	Naval air training and operating procedures standardization
NM	Nautical miles
NOE	Nap of the earth
Np	Free power turbine

NR	Main rotor speed in revolutions per minute
OAT	Outside air temperature
OGE	Out of ground effect
PA	Pressure altitude
PCL	Power control lever
PIO	Pilot induced oscillation
PWR	Power
RPM	Revolutions per minute
SAS	Stability augmentation system
SCU	Stabilator control unit
SFC	Specific fuel consumption
SN	Serial number
SR	Specific range
SHP	Shaft horsepower
SPADS	Shear pin actuated decoupling system
STD DAY	Standard day
TADS	Target acquisition and designation system
T/O	Takeoff
USMC	United States Marine Corps
USNTPS	United States Naval Test Pilot School
V_{cruise}	Cruise airspeed
V_h	Maximum level flight airspeed
V_{max}	Maximum airspeed
$V_{\text{max endur}}$	Maximum endurance airspeed
$V_{\text{max range}}$	Maximum range airspeed
V_{ne}	Never exceed airspeed

V_t	True airspeed
W	Aircraft weight
W_f	Fuel flow
XMSN	Transmission
σ	Density ratio
δ_{T_2}	Inlet pressure ratio
θ_{T_2}	Inlet temperature ratio
μ	Advance ratio
ρ	Density
ρ_a	Ambient air density
N_1	Engine RPM
T_5	Turbine inlet temperature
t	Time
Ω	Rotor speed in radians per second
R	Radius of the rotor disk
V_i hover	Induced velocity at hover
V_v	Vertical climb velocity (actual)
v'	Vertical climb velocity (calculated)
A_{disk}	Rotor disk area

I. INTRODUCTION

BACKGROUND

The U.S. Marine Corps is considering several aircraft to augment the current AH-1 attack helicopter fleet. The aircraft must demonstrate suitable hover, vertical climb and forward flight performance and handling qualities. The aircraft must also possess suitable hover and low speed agility, the ability to operate from amphibious ships, and the capability for night and IFR operations. One of the candidate aircraft is the AH-64A Apache helicopter, currently being fielded by the U.S. Army. The Marine Corps will evaluate each of the candidate aircraft in three primary areas: (a) flying qualities and performance, (b) mission systems, and (c) shipboard compatibility.

PURPOSE

The purpose of this evaluation was to assess the flying qualities and performance of the AH-64A helicopter for the USMC attack mission and to propose corrective action for any significant deficiencies identified.

DESCRIPTION OF TEST AIRCRAFT

The AH-64A helicopter, shown in Appendix C, Figure 1, is a dual piloted, night capable attack helicopter manufactured by McDonnell Douglas Helicopter Company. Two General Electric T700 engines, each rated at 1,694 SHP (at 100%

Torque) during sea level standard day conditions, provide power to the four bladed, fully articulated main rotor. The tail rotor is also four bladed and is of the teetering hub design. A rotor brake system is incorporated which is designed to hold the main rotor stationary during engine start and to stop the rotor during shutdown. The aircraft is equipped with a 30 mm gun, can carry 2.75 in. aerial rockets and Hellfire anti-tank missiles, and has a maximum gross weight of 17,650 lbs. Tandem seating is provided for the pilot and copilot/gunner and both cockpits incorporate conventional helicopter flight controls. The flight control system is boosted by two independent hydraulic systems. The aircraft incorporates digital automatic stabilization equipment including: a stability augmentation system (SAS), a command augmentation system (CAS), an attitude hold system, and a backup control system (BUCS) which provides an emergency fly by wire capability. The aircraft also incorporates a programmable horizontal stabilator. A detailed flight control description is presented in Appendix D. The primary weapons delivery system is the target acquisition and designation system (TADS) which is operated by the copilot/gunner. The pilot can also control weapons delivery utilizing his helmet mounted sight and display system. The helmet mounted display is also used in conjunction with the forward looking infrared (FLIR) system for night operations. An air data sensor system (ADSS) is

incorporated to provide inputs to the fire control computer. ADSS inputs are also utilized by the doppler navigation system and the SAS. A detailed description of the aircraft is contained in the AH-64A Operator's Manual, reference 1. The test aircraft, SN: 83-23788, was a representative production model AH-64A helicopter and was assigned to the Aviation Developmental Test Activity (ADTA) at Ft. Rucker, Alabama.

SCOPE OF TEST

The flying qualities and performance evaluation of the AH-64A was conducted during four hours of ground evaluation and four flights totalling six hours. The evaluation was conducted in the Ft. Rucker, Alabama local flying area under day visual meteorological conditions. A summary of tests and test conditions is presented in Appendix E, Table I. All operating procedures and limitations contained in the AH-64A Operator's Manual were observed. Additional test restrictions are presented in Table I.

Table I
SPECIAL TEST LIMITATIONS

Parameter	Limitations
Pitch Attitude	Maximum $\pm$ 45 deg.
Bank Angle	Maximum 90 deg.
Normal Acceleration	Min. 0.0 Max. +2.5 G's
Control Step Inputs	Maximum 2 in.
Sideslip	Refer to Appendix F, Figure 1

The test aircraft loading was a crew of two (pilot and copilot), full fuel, 500 rds. 30 mm ammo, 38 rockets and eight Hellfire missiles (inert ordnance). The aircraft takeoff gross weight was 16,344 lbs. with a CG of 205.1 in. (Aft). The aircraft configuration was engine bleed air systems off for all performance testing. The flying qualities and performance of the aircraft were evaluated against the requirements of MIL-D-23222A, reference 2, MIL-H-8501A, reference 3, MIL-C-5011A, reference 4, MIL-STD-250D, reference 5, MIL-C-18244A, reference 6, and the AH-64A detail specification, AMC-SS-AAH-H10000A, reference 7, hereinafter, collectively referred to as the Specification.

METHOD OF TEST

Test methods and procedures were in accordance with United States Naval Test Pilot School Flight Test Manual, USNTPS-FTM-102, reference 8, USNTPS-FTM-105, reference 9, and TM-83-1-TP, reference 10. The data were recorded on hand-held data cards and a portable cockpit voice recorder. The data were reduced manually and with the Hewlett Packard 9845 computer and associated performance programs developed for USNTPS by Mr. J.J. McCue. Instrumentation used to supplement the normal aircraft instruments included: a hand-held force gauge to measure control forces, retractable tape measures to measure control positions and displacements, a hand-held control fixture to measure and restrict control inputs, a protractor and yaw string for sideslip angle measurement, a portable accelerometer to measure cockpit normal acceleration, and a hand-held electronic stopwatch for accurate time measurement. The onboard production air data sensor was utilized to measure true airspeed during the low airspeed tests. The onboard production radar altimeter was used to measure altitude AGL during the freeflight hover tests. Handling Qualities Ratings (HQR) were assigned in accordance with the Cooper-Harper handling qualities rating scale as published in reference 11, and presented in Appendix B, Figure 1. Vibration Assessment Ratings (VAR) were assigned in accordance with the Subjective Vibration Assessment Scale, reference 12, presented in Appendix B, Table I. Pilot

Induced Oscillations (PIO) were assessed in accordance with the Pilot Induced Oscillation Tendency Scale published in reference 9 and presented in Appendix B, Figure 2. The deficiencies identified were placed into one of three deficiency categories utilized by the Naval Air Test Center, Patuxent River, Maryland, which are defined in Appendix A.

II. RESULTS AND DISCUSSION

COCKPIT EVALUATION

GENERAL

An evaluation of the pilot's cockpit of the AH-64A helicopter was conducted during one static ground evaluation during daylight conditions for a total of 2.0 hours. The cockpit was also evaluated concurrently with the flight tests under the conditions presented in Appendix E, Table I. The evaluation pilot was attired in full summer flight clothing and equipment as listed in Table II. The evaluation pilot's specific anthropometric measurements are presented in Appendix E, Table II. The cockpit seat was positioned to place the pilot at design eye position (DEP) throughout the evaluation.

Table II
FLIGHT EQUIPMENT

Item		National Stock Number
Flight Suit	CWU-27P	9D 8415-01-043-9529
Gloves	GS/FRP-1	9D 8415-01-029-0112
Boots		9D 8430-00-624-3135
Survival Vest	SV-2B (with LPA)	9D 8415-00-139-6174
Helmet (AH-64 Integrated Display System)		N/A

EMERGENCY EGRESS

Emergency egress from the pilot's cockpit was evaluated during the static ground evaluation. The right door was placed in the up and locked position to simulate use of the canopy removal system. The canopy removal system is designed to jettison the canopy side panels through use of a detonation cord installation. During the emergency egress evaluation, canopy jettison procedures were simulated by grasping the canopy removal handle and rotating the handle clockwise 90 degrees. The push that is used to complete canopy jettison was simulated. The pilot's helmet hook ups and harness restraint were released, followed by egress out the right door. The egress procedure was completed within 20 seconds during each of the three evaluations. No snags on the pilot's clothing or equipment were encountered. The large right door provided an unobstructed easy egress route. The only obstruction to egress was the cyclic stick which required the pilot to slide his left leg behind the stick or step over it. The simple, relatively unobstructed emergency egress from the pilot's cockpit will allow the attack pilot to make easy, rapid emergency exits from the helicopter. Emergency egress from the pilot's cockpit of the AH-64A helicopter is satisfactory for the attack mission.

TAIL WHEEL LOCK SWITCH

The tail wheel lock switch was evaluated for proper location during actual ground operations prior to takeoff. The switch was located on the far left side of the pilot's instrument panel as depicted in Appendix C, Figure 2. The switch was located 1/2 inch outside the evaluation pilot's fully arm stretched functional reach with the shoulder harness locked. The pilot was required to unlock his shoulder harness to actuate the tail wheel lock switch and then relock his harness prior to takeoff. The attack pilot will experience delayed takeoffs while unlocking and relocking his shoulder harness to actuate the tail wheel lock switch. The location of the tail wheel lock switch outside the pilot's shoulder harness locked, fully arm stretched functional reach (zone 2) is a Part III deficiency which should be avoided in future designs. The location of the tail wheel lock switch met the requirements of the Specification.

ATTITUDE/HOVER HOLD SWITCH

The attitude/hover hold switch was evaluated for proper location during transitions from forward flight to hover operations under the conditions presented in Appendix E, Table I. The switch was located on the DASE panel as shown in Appendix C, Figure 3. It was not possible to keep the attitude/hover hold system engaged during the transition from forward flight to a hover, as the system is designed to

disengage when passing through 50 KTAS. In order to reengage the hover hold system, the pilot was required to release the collective and reach forward approximately 8 inches to actuate the ATTD/HOVER HOLD switch. When conducting IGE hover operations the attack pilot wishing to engage the hover hold system will be required to release the collective in close proximity to the ground which could result in impact with the ground or other obstacles. The requirement to release the collective lever to engage the hover hold system is a Part II deficiency which should be corrected as soon as practicable. The location of the attitude/hover hold switch met the requirements of the Specification. Recommendations for the correction of this deficiency and the other Part I and II deficiencies are presented in section III (Conclusions and Recommendations).

ENGINE FIRE PULL HANDLES

The engine fire pull handles were evaluated during simulated ground and inflight emergency procedures. The handles were located on the instrument panel as depicted in Appendix C, Figure 4. There was less than one inch of clearance between the tops of the handles and the instrument glare shield. During the simulated emergencies, actuating the pull handles was delayed by the pilot's flight gloves catching on the master caution panel and a grimes light installed on the underside of the glare shield. The flight gloves caught on the glare shield obstructions during four of five activation

attempts causing an approximate two second delay. When combating an actual engine fire, a delay in actuating the fire pull handle will result in spreading of the engine fire and damage to other aircraft components. The reduced accessibility of the engine fire pull handles, caused by glare shield obstructions, is a Part II deficiency which should be corrected as soon as practicable.

APU FIRE BOTTLE SELECTOR SWITCH

The APU fire bottle selector switch was evaluated while simulating APU fire emergency procedures. The switch was located on the APU control panel of the pilot's right hand console as depicted in Appendix C, Figure 4. The APU control panel was located on the aft portion of the console between the pilot seat armor and the canopy frame. The obstruction caused by the pilot's seat prevented reaching the switch from above and required the pilot to place his fingers around the APU fire pull handle to activate the fire bottle switch. While performing the simulated emergency procedures, reaching the switch was difficult and a delay of one to three seconds was experienced during five attempts. During an actual APU fire, a delay in actuating the fire bottle selector switch will delay discharging the extinguished agent which will allow the fire to spread and increase aircraft damage. The delay in activation of the APU fire bottle switch caused by obstruction of the fire pull handle is a Part II deficiency which should be corrected as soon as practicable.

FIELD OF VIEW

The external field of view from the pilot's cockpit of the AH-64A helicopter was evaluated quantitatively during the static ground evaluation and the results are presented in Appendix C, Figure 5. The primary obstructions to vision were the instrument panel and the canopy support frames. The pilot's canopy side panels were large and provided unobstructed view from 35 to 115 degrees right and left. Over the nose field of view was restricted by the instrument panel obstruction, but in the on deck level attitude was adequate. The field of view was also qualitatively evaluated during confined area approaches and Nap of the Earth (NOE) quickstops. The forward field of view was poor whenever the aircraft attitude was higher than five degrees nose up. The area of obstructed view varied from approximately 500 ft. ahead of the aircraft at five degrees nose up to 3000 ft. in the 15 degrees nose up attitude required for NOE quickstops. The copilot's forward field of view was also obstructed, therefore, the pilot could not rely on the copilot to watch for landing obstructions. During confined area approaches, the attack pilot will lose sight of the intended point of landing during the final portion of the approach often necessitating a wave off. When performing NOE quickstops, the attack pilot will use sideflare quickstops to improve his field of view, which will present the aircraft's broadside to threat weapons. The limited forward field of view of the

AH-64A helicopter during confined area approaches and NOE quickstops is a Part II deficiency which should be corrected as soon as practicable. The field of view of the AH-64A failed the requirements of MIL-STD-850B paragraph 8.2, in that unimpaired vision from the pilot's DEP of at least 25° down at 0° azimuth was not met by 16° (64%).

FLYING QUALITIES

FLIGHT CONTROL MECHANICAL CHARACTERISTICS

General

The mechanical characteristics of the AH-64 helicopter flight control system were evaluated during one ground session of 2.0 hours duration and inflight under the conditions presented in Appendix E, Table I. The ground evaluation was conducted with external electrical and hydraulic power to the aircraft. Additional test conditions were: rotors static, engines off, attitude hold off, SAS and trim on. Longitudinal, lateral, directional, and collective control positions and displacements were measured with a tape measure graduated to one sixteenth of an inch. All forces were measured with a hand-held calibrated force gauge. The data, recorded on hand-held data cards, is presented in accordance with Air Force Systems Command Technical Report, ASNF TN 68-3, "A Graphical Summary of Military Helicopter Flying and Ground Handling Qualities of MIL-H-8501A", dated 15 September 1968. A summary of the AH-64 flight control

mechanical characteristics is presented in Appendix E, Table III. Specific force measurement points for each control are depicted in Appendix C, Figure 6.

Longitudinal Cyclic

The longitudinal cyclic mechanical characteristics were evaluated with the directional pedals in the neutral position, the collective full down, and the cyclic centered laterally. The data are presented in Appendix E, Table III and Appendix F, Figure 2. The average force required for the first one inch of longitudinal cyclic displacement was 5.0 lbs. per inch forward and 4.0 lbs. per inch aft, exceeding the breakout plus friction force of 2.8 lbs. forward and 2.2 lbs. aft. Beyond one inch the force gradient was essentially linear and averaged 2.4 lbs. per inch forward and 2.3 lbs. per inch aft. The gradients were positive throughout with no discontinuities. The limit control force reached 10.5 lbs. forward and 8.5 lbs. aft. The friction band was moderate measuring 2.4 lbs. forward and 1.4 lbs. aft. The longitudinal cyclic displayed a 1/8 to 1/4 inch stick jump following a one inch displacement and trim release. Moderately damped control dynamics were exhibited after a two inch control deflection and release as indicated by two small (1/16 - 1/8 in.) overshoots. The longitudinal cyclic control mechanical characteristics met the requirements of the Specification for which they were tested except:

a. Paragraph 10.3.2.1.1 in that the breakout force including friction for forward and aft cyclic exceeded 1.5 lbs. by 1.3 lbs. (87%) forward and 0.7 lbs. (47%) aft.

b. Paragraph 10.3.2.8 in that total system freeplay exceeded $\pm 1/16$ inch by $1/16$ inch (100%).

c. Paragraph 10.3.3.2.2 in that the requirement for no stick jump was exceeded by $1/4$ inch.

Lateral Cyclic

The lateral cyclic mechanical characteristics were evaluated with the directional pedals in the neutral position, the collective down, and the cyclic centered longitudinally. The data are presented in Appendix E, Table III, and Appendix F, Figure 3. Lateral cyclic total system freeplay was $\pm 1/16$ inch. The breakout force including friction was 1.5 lbs. to the right and 2.0 lbs. to the left. The average force gradients measured 1.1 lbs. per inch right and 0.8 lbs. per inch left and were essentially linear with no discontinuities. The limit control force reached 6.2 lbs. to the right and 5.5 lbs. to the left. Lateral cyclic centering was positive but not absolute as the right cyclic inputs stopped $1/4$ inch short of the trimmed position when slowly returned from a displacement. The non-absolute centering resulted in a $1/4$ inch trim control displacement band. When the cyclic was displaced laterally one inch and the trim release switch was actuated, a $1/8$ inch stick jump was

exhibited. When released from a two inch lateral displacement, the lateral control dynamics were heavily damped with no observed overshoots. The lateral cyclic control mechanical characteristics met the requirements of the Specification for which they were tested except:

a. Paragraph 10.3.2.1.1 in that the breakout force including friction for left cyclic exceeded 1.5 lbs. by 0.5 lbs. (33%).

b. Paragraph 10.3.3.2.2 in that the requirement for no stick jump was exceeded by 1/8 inch.

Cyclic Control Envelope

The cyclic control envelope was evaluated at six different collective and pedal positions with the collective ranging from full up to full down and full left pedal to full right pedal. The data are presented in Appendix E, Table III, and Appendix F, Figure 4. The maximum longitudinal travel was 5-3/4 inches and the maximum lateral travel was 7 inches. Control travel showed no effects of mechanical mixing. Collective and pedal position had no effect on the cyclic longitudinal or lateral control travel. The trim authority was 100% of the longitudinal and lateral control travel.

Directional Pedals

The directional control pedal mechanical characteristics were evaluated with the cyclic centered and the collective

full down. The data are presented in Appendix E, Table III, and Appendix F, Figure 5. Directional control total system freeplay was 1/16 inch. The breakout force including friction was 3.0 lbs. to the right and 4.0 lbs. to the left. The average force gradients measured 4.2 lbs. with right pedal and 4.3 lbs. with left pedal and remained positive with no discontinuities. The friction band both to the right and left was large with the average friction exceeding 4.0 lbs. Centering was positive but not absolute as the right pedal only returned to within 1/4 inch of the trimmed position and the left pedal to within 1/8 inch. A 3/8 inch trim control displacement band resulted. Control dynamics were heavily damped with no overshoots following release from a two inch displacement against trim. The directional control pedal mechanical characteristics met the requirements of the Specification.

Collective

The collective control mechanical characteristics were evaluated with the cyclic centered, pedals neutral, and the collective friction off. The data are presented in Appendix E, Table III, and Appendix F, Figure 6. The collective total system freeplay was 1/16 inch. The breakout force including friction was 5.8 lbs. up and 4.0 lbs. down. The force gradient for up collective averaged 0.6 lbs. per inch, however, the gradient was non-linear with discontinuities occurring between three to six inches up. The collective

control had neutral centering characteristics and a push force had to be applied to lower the collective. The average gradient for the down collective force was also 0.6 lbs. and discontinuities were present between 6 and 3-1/2 inches up collective. The limit control forces were 13.0 lbs. up and 9.0 lbs. down. The collective control mechanical characteristics met the requirements of the Specification for which they were tested except paragraph 10.3.2.2.1 in that undesirable discontinuities were present in the force gradients.

FORWARD FLIGHT

Trimmed Flight Control Positions

Level Flight

Trimmed flight control positions in level flight were evaluated from 41 to 188 KIAS under the conditions presented in Appendix E, Table I. The data are presented in Appendix F, Figure 7. Forward cyclic displacement was required with increasing airspeed until reaching the test day V_h (140 KIAS). During the dive to V_{ne} , longitudinal cyclic was positioned 1/4 in. aft of the V_h position. The total longitudinal cyclic displacement throughout the tested airspeed range was less than 3/4 in. Lateral cyclic and the directional control pedals remained essentially constant (less than 1/4 in. variation). Maintaining the desired trim airspeed (± 2 kts) between 41-120 KIAS was easy, requiring

only small (1/16 - 1/8 in.) longitudinal cyclic inputs at 3-5 second intervals (HQR-3). Cyclic control and directional control forces could be trimmed to zero using the cyclic trim release switch, however, the position of the switch on the cyclic stick grip required the pilot to relax his grip to reach and actuate the switch. This relaxed grip on the cyclic often resulted in a stick jump and an aircraft motion of 1-2 degrees of pitch or roll. Control margins were adequate with the minimum margin being 1.25 inches (26%) from full aft longitudinal cyclic at 41 KIAS. The field of view remained adequate throughout the test airspeed range. Low frequency vertical vibrations were present and increased from light (VAR 3) at 60-120 KIAS, to moderate (VAR 5) at 130-170 KIAS. While ingressing to the battle area in level forward flight the attack pilot will have adequate longitudinal, lateral, and directional control margins and will only be required to make small control inputs. The trimmed flight control positions of the AH-64A helicopter in level flight are satisfactory for the attack mission and met the requirements of the Specification.

Coordinated Level Turns

Trimmed control positions during coordinated level turns, up to 30 degrees angle of bank at 120 KIAS, were evaluated under the conditions presented in Appendix E, Table I. The data are presented in Appendix F, Figure 8. The lateral cyclic position for 15 degree angle of bank turns

was the same as the wings level position for both right and left turns. Increasing lateral cyclic into the turn was required for both right and left turns from 15 to 30 degrees angle of bank. A 1/8 in. pedal deflection into the turn was required until 15 degrees angle of bank and then the pedal position remained constant until 30 degrees of bank. Longitudinal control position remained essentially constant with the only change (1/8 in. aft) occurring at 30 degrees left angle of bank. Control margins were adequate, with the minimum margin being 1.5 in. (31%) from full aft longitudinal cyclic during the 30 degree angle of bank turn. The inherent sideslip angle remained constant at five degrees right. As bank angle increased, maintaining airspeed and bank angle became more difficult, however, at 30 degrees angle of bank, maintaining airspeed (± 2 KIAS) and bank angle (± 3 deg.) was still relatively easy requiring small (1/8 in.) longitudinal and lateral cyclic inputs every 2-3 seconds (HQR-3). Vibrations were light, low frequency, vertical vibrations that remained constant (VAR 3). The attack pilot will easily perform coordinated level turns while orbiting a rendezvous point and will have adequate control margins. Within the scope of this test, the trimmed flight control positions of the AH-64A helicopter in coordinated level turns are satisfactory for the attack mission and met the requirements of the Specification.

Climbs and Descents

Trimmed control positions in forward climbing and descending flight were evaluated at 90 KIAS under the conditions presented in Appendix E, Table I. The data are presented in Appendix F, Figure 9. The lateral control positions varied less than 1/2 inch throughout the test torque range (2-98%). Directional control pedal movements were even smaller varying only 3/8 inch. The longitudinal control position varied less than 1/4 in. from 2-80% torque, however, once torque was increased above 92-95% a 1/2 inch aft cyclic input was required to maintain airspeed. This aft cyclic input was apparently required due to pitot-static errors. As power increased to 90% torque, the airspeed indication fluctuated ± 10 kts. and then as power was further increased above 95% torque, the airspeed indication stabilized at a value 10 knots faster than trim. Maintaining airspeed (± 2 kts) during the maximum power climb was moderately difficult requiring frequent (2 sec.) small (1/8 in.) longitudinal inputs (HQR-4). The minimum control margin was 1 inch (21%) aft longitudinal cyclic remaining and occurred during the maximum power climb. Field of view forward was adequate except in the maximum power climb where the pitch attitude reached seven degrees nose up. When performing maximum power climbs, the attack pilot will be annoyed by airspeed fluctuations and will be required to make frequent longitudinal inputs to maintain airspeed. The

fluctuation in airspeed encountered above 92% torque is a Part III deficiency which should be avoided in future designs. The trimmed control positions of the AH-64A helicopter during climbing and descending flight met the requirements of the Specification. Further testing is recommended to determine the source and magnitude of the aircraft's pitot static system errors.

Static Longitudinal Stability

The static longitudinal stability characteristics were evaluated about a trim airspeed of 120 KIAS under the conditions presented in Appendix E, Table I. The data are presented in Appendix F, Figure 10. The aircraft exhibited neutral to slightly positive longitudinal stability. No forward cyclic was required with increasing airspeed, however, aft cyclic was required to decrease airspeed from the trimmed value. The gradient of the cyclic position with airspeed was extremely shallow and varied only 1/4 in. (5%) from 110-130 KIAS. Force and displacement cues to an off-trim condition were non-existent with increasing airspeed and were slight with decreasing airspeed. Adequate force and displacement cues occurred at 10 knots below the trim airspeed. When relying on time distance checks during low level missions, the attack pilot will have to closely monitor the airspeed indicator or the airspeed readout on his helmet mounted display to remain on airspeed which will divert his attention from other mission essential tasks. The small

cyclic force and displacement cues to off-trim conditions about a trim speed of 120 KIAS is a Part III deficiency which should be avoided in future designs. The static longitudinal stability characteristics of the AH-64A failed the requirements of the Specification paragraph 10.3.4.1 in that no forward displacement and push force were required for a 10 knot increase in airspeed from trim.

Static Lateral Directional Stability

General

The static lateral directional stability characteristics were evaluated under the conditions presented in Appendix E, Table I. The evaluation was conducted utilizing steady heading sideslips to a maximum of 17 degrees right and left. Lateral cyclic only and pedal only turns were conducted to establish the relative strengths of effective dihedral and static directional stability.

Effective Dihedral

The data are presented in Appendix F, Figure 11. The aircraft exhibited slightly positive effective dihedral at 120 KIAS as indicated by the requirement for increasing lateral cyclic displacement in the direction of the sideslip. During directional pedal only turns the aircraft was qualitatively evaluated to have weak to moderate effective dihedral. The roll response to left pedal inputs was sluggish and required one full ball width of sideslip (approximately 15°) to

develop a 2 degree/second roll rate. Effective dihedral was stronger in turns to the right with equal pedal deflections resulting in double the roll rate. During a GCA, the attack pilot will be able to make small directional pedal only heading corrections without developing large bank angles. Within the scope of this test, the effective dihedral characteristics of the AH-64A are satisfactory for the attack mission and met the requirements of the Specification.

Directional Static Stability

The data are presented in Appendix F, Figure 11. The aircraft exhibited positive directional static stability at 120 KIAS, in both directions, as indicated by the requirement for increasing left pedal displacement with increasing right sideslip and vice versa. Moderate directional stability was exhibited during cyclic only turns to the right and left, in that sideslip angles returned to within three to five degrees of the wings level inherent sideslip. No pitot static errors were detected during the steady heading sideslips or the cyclic only turns. While performing descending 30 degree angle of bank turns during normal approaches, maintaining airspeed (± 5 kts.), angle of bank (± 5 deg.), and coordinated flight ($\pm 1/4$ ballwidth) was relatively easy requiring only small (less than $1/8$ in.) cyclic inputs every 3-5 seconds laterally and at only 15 to 20 seconds longitudinally (HQR-3). After an initial $1/4$ inch pedal input into the turn, no further pedal inputs were required to maintain

coordinated flight. The directional stability characteristics of the AH-64A will allow the attack pilot to easily make coordinated turns without the requirement for large directional pedal inputs. Within the scope of this test, the directional stability characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Sideforce Characteristics

The data are presented in Appendix F, Figure 11. Positive sideforce characteristics were indicated by the requirement for angle of bank into the sideslip during steady heading sideslips. At 120 KIAS the bank angles required to balance sideforce at sideslips other than inherent were small (approximately 5 deg. AOB/10 deg. sideslip) and provided the pilot with limited proprioceptive cues to an out-of-balanced flight condition. The pilot could not readily detect an out-of-balanced flight condition until approximately 10 degrees of sideslip without reference to ball position. The trim ball provided accurate and timely indications of coordinated flight. In addition to the mechanical trim ball underneath the pilot's video display, an electronic trim ball was provided with the helmet mounted display. While straight and level, coordinated flight was easily maintained, however, the inherent sideslip angle was large (5 deg. right) and noticeable. While maintaining an external scan, the attack pilot will not be provided adequate sideforce cues to maintain

balanced flight without reference to the trim ball. The limited sideforce cues to an out-of-balanced flight condition is a Part III deficiency which should be avoided in future designs.

Maneuvering Stability

The maneuvering stability characteristics were evaluated in steady turns and symmetrical pullups and pushovers at a trim airspeed of 120 KIAS under the conditions presented in Appendix E, Table I. The data are presented in Appendix F, Figures 12 and 13. The helicopter exhibited positive maneuvering stability as indicated by the requirement for aft longitudinal control displacements with increasing normal acceleration. The gradients of the longitudinal control position versus normal acceleration curves were shallow and measured 0.7 in/g for the steady turns and 1.0 in/g for the pullups and pushovers. The increasing force requirements (2.5 lbs. per g) provided the pilot with sufficient force cues to maintain normal acceleration within $\pm .2$ g's. The longitudinal cyclic force and displacement requirements with increasing normal acceleration provided the pilot with the tactile cues necessary to maneuver the helicopter predictably, with a high degree of precision, without the tendency to exceed "g" limitations. When performing evasive maneuvers the attack pilot will receive adequate cyclic force and displacement cues to avoid exceeding the normal acceleration envelope of the aircraft. Within the scope of this test,

the maneuvering stability characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Dynamic Stability

Long Term Response

The long term response characteristics were evaluated at a trim airspeed of 120 KIAS under the conditions presented in Appendix E, Table I and the data are presented in Appendix F, Figure 14. With attitude hold on, the long term response was not excited under the test day calm air conditions either naturally or with 10 knots from trim artificial excitation. With the attitude hold system off, the long term response was easily excited naturally with the flight controls fixed. The aircraft response was oscillatory, and divergent with a period of 15 seconds. The time to double amplitude was 33 seconds. The pilot was able to avoid the long term response with minimal compensation. Maintaining 120 KIAS (± 2 kts.) required small ($\pm 1/16$ - $1/8$ in.) longitudinal cyclic inputs at 4-5 second intervals (HQR-3). While performing level flight the attack pilot will easily avoid the long term response of the aircraft with small longitudinal cyclic inputs or through use of the attitude hold system. Within the scope of this test, the long term response characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Short Term Response

The short term response characteristics were evaluated at a trim airspeed of 120 KIAS under the conditions presented in Appendix E, Table I. The short term response was excited by a one inch longitudinal cyclic doublet at the approximate natural frequency of the aircraft (1.0 cycles/sec). The short term response was heavily damped with only one detectable overshoot. Changing pitch attitude (5 degrees $\pm$ 1) quickly (less than 1 second) was easy with 1/4 inch longitudinal cyclic inputs (HQR-2). The aircraft response was precise and predictable. During formation flights the attack pilot will make quick precise pitch attitude adjustments to maintain position. Within the scope of this test, the short term response of the AH-64A helicopter is satisfactory for the attack mission and met the requirements of the Specification.

Lateral Directional Oscillation

The lateral directional oscillation characteristics were evaluated in level forward flight at 120 KIAS under the conditions presented in Appendix E, Table I. The aircraft was flown in calm, smooth air conditions which did not naturally excite a lateral directional oscillation. A one inch directional pedal doublet, a one inch lateral cyclic doublet, and a release from a steady heading sideslip were used to artificially excite a lateral directional

oscillation. The aircraft exhibited a heavily damped lateral directional response with all three methods of excitation as indicated by a maximum of one visually detectable overshoot in roll or yaw. The attack pilot will be able to perform mission maneuvers without the irritation of, or workload required, to counteract lateral directional oscillations. Within the scope of this test, the lateral directional oscillation characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Spiral Stability

The spiral stability characteristics were evaluated at 120 KIAS under the conditions presented in Appendix E, Table I. The data are presented in Table III.

Table III
SPIRAL STABILITY

Initial AOB (Deg.)	Time to Half Amplitude (Sec.)
Right 15	6
Right 30	12
Left 15	5
Left 30	9

When the cyclic was returned to trim from lateral cyclic only turns, the aircraft exhibited convergent spiral stability as indicated by the return towards wings level flight. The spiral stability to the right was slightly weaker than to the left requiring 1-3 seconds longer to reach half amplitude. In controls fixed level forward flight, the roll attitude remained nearly constant, oscillating less than one degree right or left. The effect of the attitude hold system on the aircraft's spiral stability was also evaluated qualitatively. The positive spiral stability was increased as indicated by a more rapid roll rate during the return to wings level. The convergent spiral stability characteristics will allow the attack pilot to maintain straight and level flight without having to constantly monitor the aircraft roll attitude. Within the scope of this test, the convergent spiral stability characteristics are satisfactory for the attack mission and met the requirements of the Specification.

Adverse/Proverse Yaw

The adverse/proverse yaw characteristics were evaluated at 120 KIAS in conjunction with lateral cyclic only turns under the conditions presented in Appendix E, Table I. The aircraft response to slow (1 in./sec.) lateral cyclic inputs yielded no perceivable adverse or proverse yaw. Rapid lateral cyclic inputs (3 in./sec.) also yielded no perceivable adverse or proverse yaw. During the rapid inputs

a slight delay (1 sec.) was noted before the aircraft yawed in the direction of the turn. The neutral adverse/proverse yaw characteristics will allow the attack pilot to rapidly enter evasive maneuver turns without having a counter uncommanded yaw responses. The neutral adverse/proverse yaw characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Gust Response

The gust response characteristics were evaluated under the conditions presented in Appendix E, Table I. No natural wind gusts or thermal action were encountered during the calm air test day conditions; therefore, a one inch collective pulse was utilized to simulate turbulence and excite the aircraft's gust response. The collective pulse caused vertical accelerations variations, a small yaw oscillation, and a slight roll excursion. No perceptible pitch excursions or oscillations resulted. Normal acceleration variations were small (between 0.8 and 1.2 G) and the attitude excursions were quickly damped with no pilot effort. The aircraft's ride quality was evaluated to be smooth and comfortable. The attack pilot will be able to fly in light turbulence without having to make large attitude and heading corrections which will reduce pilot workload. The gust response characteristics of the AH-64A helicopter

are satisfactory for the attack mission and met the requirements of the Specification.

Control Response

Longitudinal

The longitudinal control response characteristics were evaluated at 120 KIAS under the conditions presented in Appendix E, Table I, and the data are presented in Appendix F, Figure 15. No perceptible time delay was encountered during any of the control step inputs. The control sensitivity was qualitatively evaluated to be high as the initial pitch accelerations were rapid, however, no objectionable roughness or "g" spikes resulted. The damping was also qualitatively evaluated to be high as the time to steady state pitch rate was short (less than 1 sec.). The steady state pitch rate increased with the size of the input from 5 deg./sec. for a 1/2 inch aft input to 19 deg./sec. for a 1-1/2 inch input. The rates developed were adequate for rapid pitch attitude changes. The aircraft response to rapid (1 sec.) pitch attitude changes (10°) was consistent, predictable and there was no tendency for pilot induced oscillations (PIO 1). During NOE flight the attack pilot will be able to precisely control pitch attitudes and rates while maneuvering to avoid obstacles. The longitudinal control response characteristics of the AH-64A helicopter are satisfactory for the attack

mission and met the requirements of the Specification.

Lateral

The lateral control response characteristics were evaluated at 120 KIAS under the conditions presented in Appendix E, Table I, and the data are presented in Appendix F, Figure 16. There was no perceptible time delay in the response to any of the step inputs. The lateral sensitivity was evaluated to be very high. The initial acceleration from the inputs was extremely quick which resulted in an abrupt, jerky response to rapid lateral control inputs. There was also a tendency for pilot-induced oscillations during high gain tasks such as slope landings (refer to the Slope Landings paragraph). Damping in the lateral axis was moderate to high as the roll rates reached steady state values in approximately one second. The steady state roll rates increased proportionally with the size of the input varying from 6 deg./sec. for a 1/2 inch right cyclic input to 27 deg./sec. for a 1-1/2 inch left cyclic input. The roll rates generated were more than adequate for the mission and predictable. The attack pilot will command an abrupt and jerky aircraft response to rapid lateral cyclic inputs and will be prone to pilot-induced oscillations during shipboard landings. The extremely high lateral control sensitivity of the AH-64A helicopter is a Part II deficiency which should be corrected as soon as practicable. The lateral control

response characteristics met the requirements of the Specification.

LOW AIRSPEED

Trimmed Flight Control Positions

Forward and Rearward Flight

The forward and rearward trimmed flight control positions were evaluated in 10 knot increments from 0-30 KTAS. The data are presented in Appendix F, Figure 17. Longitudinal cyclic variation was small (7/8 in.) from 30 KTAS forward to 30 KTAS rearward. The control position changes were essentially linear with no discontinuities. Control margins were adequate throughout the test airspeed range with the minimum margin being 1.38 in. (28%) of aft longitudinal cyclic remaining at 30 KTAS rearward. No coupling was observed throughout the test. Control forces could be trimmed to zero with the trim release switch on the cyclic control grip. Pitch attitude changes were small (less than 5 deg. nose up or down) which provided adequate field of view. Vibrations were generally light low frequency vertical vibrations (VAR 3); however, the vibrations became moderate at 20 KTAS forward and aft in the area of translational lift (VAR 4). The attack pilot will have adequate control margins to hover the aircraft in headwinds or tailwinds up to 30 knots. Within the scope of this test, the trimmed flight control positions of the AH-64A helicopter during forward or rearward flight up to 30 KTAS are satisfactory for the attack mission and met the requirements of the Specification.

Sideward Flight

The sideward flight trimmed flight control positions were evaluated in 10 knot increments from 0-30 KTAS. The data are presented in Appendix F, Figure 18. The lateral cyclic requirements were increasing right lateral cyclic with increases in right sideward airspeed and vice versa to 20 KTAS and then remained constant to 30 KTAS. Aerodynamic control coupling was evident, however, the longitudinal cyclic and pedal changes were not objectionable. The minimum control margin was one inch (27%) right pedal remaining and occurred at 30 KTAS left. Control forces were trimmed to zero with the cyclic trim release switch. The roll attitude changes were small (max. 6 deg. left) and did not affect field of view which remained adequate. Vibrations were light low frequency vertical vibrations (VAR 3) except at 20 KTAS right and left where the vibrations increased to moderate (VAR 4). The attack pilot will have adequate control margins to hover the aircraft in crosswinds (90° and 270° relative) up to 30 KTAS. The trimmed flight control positions of the AH-64A helicopter during sideward flight up to 30 KTAS are satisfactory for the attack mission and met the requirements of the Specification.

Critical Azimuth

A critical azimuth determination was conducted at 20 KTAS in 45 degree relative azimuth increments. The data are

presented in Appendix F, Figures 19 and 20. Control position changes with changes in relative azimuth were not excessive. The minimum control margin was one inch (27%) right pedal remaining and occurred at both the 270 degree and 315 degree relative azimuths. Control forces could be trimmed to zero and the field of view was adequate for all tested azimuths. Vibrations were moderate low frequency vertical vibrations (VAR 4) that reached a maximum intensity (VAR 5) at the 225 degree relative azimuth. While flying at 20 KTAS on the 225 degree and 270 degree relative azimuths, the aircraft developed a heading oscillation. The frequency of the oscillation was approximately one cycle/second and the amplitude was ± 3 degrees. The frequency of the oscillation was too high to be effectively countered with pedal inputs so the pedals were held in a fixed position. The critical azimuths were determined to be 225 to 270 degrees due to the heading oscillation encountered. However, further testing is required prior to assigning a deficiency classification to this problem. Testing will have to be conducted during actual 20 kt wind conditions to determine the ability of the hover augmentation system to counter the heading oscillation. The hover augmentation system disengages at 15 kt ground speed and therefore, could not be evaluated with the normal low airspeed test technique.

Static Stability

Static Longitudinal Stability

The low airspeed static longitudinal stability characteristics were evaluated about a 0 KTAS trim point through a range of ± 10 knots. The data are presented in Appendix F, Figure 21. The aircraft exhibited essentially neutral static longitudinal stability as evidenced by the longitudinal cyclic change of only 1/4 inch over the tested airspeed range. Longitudinal cyclic force and displacement cues about trim were very small providing poor indications to an off-trim condition; however, maintaining a hover position (± 1 ft.) with the hover augmentation system engaged was very easy requiring only occasional (8-10 sec.) small (1/8 in.) longitudinal and lateral cyclic inputs (HQR-2). The neutral longitudinal stability characteristics will allow the attack pilot to easily maintain a stable hover position while preparing for a Hellfire missile launch during gusty wind conditions. The neutral low airspeed static longitudinal stability characteristics of the AH-64A are satisfactory for the attack mission. The neutral static longitudinal stability characteristics failed the requirements of the Specification paragraph 10.3.3.1.2 in that positive static stability was not exhibited.

Static Lateral Directional Stability

The low airspeed static lateral directional stability characteristics were evaluated about a 0 KTAS trim point

through a range of ± 10 knots. The data are presented in Appendix F, Figure 22. The aircraft exhibited slightly positive to neutral static stability as indicated by the small lateral cyclic displacement in the direction of flight and small pedal displacement opposite to the direction of travel. The average control position gradients were shallow; lateral (0.2 in./10 kts.), directional (0.2 in./10 kts.). Force and displacement cues were very slight. Bank angle cues were also slight as the maximum roll attitude was 3 degrees. The static lateral directional stability characteristics of the aircraft will allow the attack pilot to easily maintain position over a deck spot while preparing for a shipboard landing. The static lateral directional stability characteristics of the AH-64A helicopter are satisfactory for the attack mission and met the requirements of the Specification.

Coupled Dynamic Response

The coupled dynamic response characteristics were evaluated under the conditions presented in Appendix E, Table I. With the hover augmentation system off, the aircraft response was easily excited (within 10 sec.) in calm air conditions. The aircraft response was oscillatory, divergent in both the pitch and roll axes. The oscillations did not exhibit a consistent period. The response was easily suppressed as maintaining an OGE hover position was easy requiring only small (1/8 in.) lateral and longitudinal

cyclic inputs at 3-5 second intervals (HQR-3). With the hover augmentation system engaged, the coupled dynamic response of the aircraft was not excited. When performing OGE hovering operations, the attack pilot will avoid the oscillatory, divergent coupled dynamic response with small cyclic inputs or through use of the hover augmentation system. The coupled dynamic response characteristics of the AH-64A are satisfactory for the attack mission and met the requirements of the Specification.

MISSION TASKS

Slope Landings

The handling qualities of the AH-64A helicopter during slope landings were evaluated under the conditions presented in Appendix E, Table I. The evaluation was conducted to an eight degree, left sloping, grass surface. The maneuver was initiated by a 1/8 inch collective reduction to start a slow controlled rate of descent. While attempting to bring the right wheel in contact with the ground, a lateral pilot-induced oscillation was produced (PIO 4). The cyclic was then held in a fixed position laterally to allow the oscillations to dampen out. The main landing gear struts were soft which made judging when the right main landing gear touchdown occurred difficult. As the collective reduction was continued, a left roll of the aircraft confirmed that the right gear was on deck and tail wheel contact was felt. The

cyclic was positioned 1/4 in. right (into the slope) and the collective was lowered to bring the left main gear to touch-down. The tendency to overcontrol laterally once again surfaced and the pilot was required to stop all lateral inputs until the left main gear was on the ground. While performing the landing, maintaining position (drift $\pm$ 1 ft.) was moderately difficult requiring frequent (2 sec.) small ($\pm$ 1/16 - 1/8 in.) longitudinal cyclic inputs and small ($\pm$ 1/8 in.) pedal inputs at 5-7 sec. intervals (HQR-4). When conducting slope landings, high lateral control sensitivity will cause the attack pilot to overcontrol and result in pilot-induced oscillations. The pilot-induced oscillations incurred during slope landings is a Part II deficiency which should be corrected as soon as practicable. Further testing is recommended to evaluate the effects of the AH-64A's lateral control sensitivity and main landing gear on the aircraft's shipboard landing capability.

Hovering

The hovering flight characteristics were evaluated under the conditions presented in Appendix E, Table I. The maneuvers were flown with the hover augmentation system engaged which provided pitch, roll, and heading hold. Maintaining heading ($\pm$ 1°) and hover position over the desired spot ($\pm$ 1 ft.) was easy requiring only occasional (8 sec.) very small ($\pm$ 1/8 in.) longitudinal and lateral cyclic inputs (HQR-2). Directional pedal inputs were negligible. However,

maintaining the selected hover altitude (± 1 ft. IGE, ± 5 ft. OGE) was moderately difficult requiring frequent (3-5 sec.) small ($1/8 - 1/4$ in.) collective inputs (HQR-4). Because of the frequent collective inputs, hovering was a moderate workload maneuver and was assigned HQR-4. During prolonged hovering operations, the attack pilot will become fatigued and will be distracted from other mission essential tasks while concentrating on holding altitude. The frequent collective inputs required during hover operations is a Part II deficiency which should be corrected as soon as practicable.

Level Accelerations

The flying qualities of the AH-64A helicopter were evaluated during level accelerations under the conditions presented in Appendix E, Table I. The accelerations were initiated at 40 KIAS with a moderate (5-7 sec.) collective increase to 100% torque. A 10 degree nose down attitude was set with a $3/4$ inch forward cyclic input and $1/4 - 1/2$ inch of left pedal was applied. The initial acceleration to 80 KIAS was rapid, however, the acceleration slowed considerably passing 100 KIAS prompting the pilot to increase the nose down attitude (15-20 deg.). The increased nose down attitude made holding altitude difficult requiring frequent (2 sec.) small ($1/4$ in.) longitudinal cyclic inputs (HQR-5). To assist in maintaining altitude, the aircraft attitude was reduced to 13-14 degrees nose down which resulted in a

sluggish acceleration from 120 to 140 KIAS. When conducting level accelerations, the attack pilot will be required to make frequent longitudinal cyclic inputs to maintain altitude and often lose altitude in an attempt to increase the acceleration past 100 KIAS. The frequent longitudinal cyclic inputs required to maintain altitude during level accelerations is a Part III deficiency which should be avoided in future designs.

Ground Taxi

The handling qualities of the AH-64A helicopter during ground taxi were evaluated under the conditions presented in Appendix E, Table I. The ability to maintain precise ground tracks along taxiway lines and the ability to stop at desired positions were qualitatively evaluated. Ground taxi was initiated by increasing the collective three inches and moving the cyclic forward one inch. Once aircraft movement began, maintaining the desired ground track was easy requiring only occasional (5-7 sec.) small ($1/8 - 1/4$ in.) pedal inputs (HQR-3). No additional collective or cyclic inputs were required. Response to pedal inputs was rapid and predictable which resulted in precise directional control. To stop at a desired location; one inch of aft cyclic was applied, the collective was positioned full down, and the brakes were applied. The pilot was required to lift his feet off the deck so that the balls of his feet were placed at the top of the directional pedals to actuate the brakes. The

response to brake pedal inputs was sluggish and required approximately four inches of pedal displacement to achieve adequate braking action. The large brake inputs coupled with the pilot's feet being off the deck resulted in inadvertent directional pedal movement and subsequent aircraft heading changes ($\pm 3^\circ$). When applying the main landing gear brakes during ground taxi operations, the attack pilot will be required to make large brake pedal inputs which will result in inadvertent directional pedal movement. The large brake pedal inputs required for adequate braking action is a Part III deficiency which should be avoided in future designs.

DEGRADED MODES

DASE Failure

The flying qualities of the AH-64A helicopter during a complete DASE failure were evaluated at 100 KIAS under the conditions presented in Appendix E, Table I. While established in straight and level flight with the attitude hold system off, the DASE release button on the cyclic stick grip was depressed. An immediate release of the pitch, roll, and yaw SAS channels occurred. No detectable aircraft response resulted until approximately five seconds later when a small nose down pitch rate began. During level flight with the DASE disengaged maintaining airspeed (± 2 kts.) altitude (± 25 ft.) and heading ($\pm 2^\circ$) was easy requiring only small ($1/8$ in.) longitudinal cyclic inputs at 3-5 second intervals (HQR-3). With the exception of the small ($\pm 2^\circ$) pitch

oscillation, the aircraft remained a smooth, stable platform. Roll attitude and heading stayed essentially constant, oscillating less than one degree right and left. When experiencing a complete DASE failure, the attack pilot will not be confronted with a significant degradation in the flying qualities of the aircraft and will continue the mission with only a small increase in workload. Within the scope of this test, the DASE off flying qualities of the AH-64A helicopter are an enhancing characteristic which should be incorporated in future designs.

TRIMMABILITY

The trimmability characteristics were evaluated in conjunction with the tests and test conditions presented in Appendix E, Table I. Cyclic control and directional control forces could be trimmed to zero utilizing the TRIM REL switch on the cyclic stick grip. The position of the TRIM REL switch is depicted in Appendix D, Figure 8. The switch was in a position that required the pilot to release his grip on the cyclic and rotate his hand 1/2 inch in order for his thumb to reach the switch. Whenever the cyclic had been displaced against the trim force prior to the TRIM REL switch activation, stick jump occurred (refer to the Longitudinal and Lateral Cyclic paragraphs). Due to the pilot's relaxed grip on the cyclic, the jump was at times 1/4 inch in magnitude which resulted in an appropriate aircraft response to a 1/4 inch cyclic input. When attempting to trim control

forces to zero, the attack pilot will be annoyed by uncommanded cyclic inputs caused by stick jump. The poor location of the TRIM REL switch on the pilot's cyclic grip resulting in stick jump is a Part II deficiency which should be corrected as soon as practicable.

VIBRATIONS

The vibration characteristics of the AH-64A helicopter were evaluated concurrently with the other tests under the conditions presented in Appendix E, Table I. The vibrations varied under the different test conditions. Generally, vibrations increased with increasing airspeed, increasing angle of bank, and during translational lift. From 60 to 120 KIAS in level forward flight the vibrations were light, low frequency, vertical vibrations (VAR 3). These vibrations increased to a moderate level above 120 KIAS (VAR 4) and reached the maximum intensity between 130-170 KIAS (VAR 5). Whenever the aircraft was established in a turn greater than 45 degrees angle of bank and during translational lift, the vibration level was also moderate. The attack pilot will be fatigued by the moderate level vibrations which will reduce the number of missions that can be safely flown on a single day. The moderate low frequency, vertical vibration of the AH-64A helicopter, during flight above 120 KIAS, at bank angles greater than 45 degrees, and during translational lift, is a Part III deficiency which should be avoided in future designs. The aircraft failed to meet the

requirements of the Specification (paragraph 3.7.1) in that objectionable vibrations were exhibited.

PERFORMANCE

GENERAL

Within the performance section of this report, frequent reference will be made to three different aircraft weights. The first of these is the U.S. Army Specification primary mission weight of 14,694 lbs. which includes an ordnance load of eight Hellfire missiles and 320 rds. of 30 mm ammunition. The second weight is the alternate mission weight of 16,344 lbs. which includes eight Hellfire missiles, 38 2.75 in. rocket, and 500 rds. of 30 mm ammunition. Based on the ordnance loads, the alternate mission weight has more application for Marine Corps missions and therefore, was the aircraft loading flown during the evaluation. The third weight referred to is the maximum mission gross weight of 17,650 lbs. In this report, standard day refers to ICAO Standard Atmosphere and hot day refers to MIL-STD-210A Hot Day. Actual fuel flow data was not available during the evaluation; therefore, fuel flow for the performance calculations was obtained from the AH-64A Operator's Manual performance section and was based on engine torque values.

ENGINE ASSESSMENT

General

The engine assessment evaluation was conducted concurrently with other tests under the conditions presented in Appendix E, Table I. Sixteen data points were collected ranging from a minimum power autorotation to a military power climb. Corrected engine variables were computed utilizing the HP 9845B computer. The variables were plotted on the generalized engine performance graphs which are corrected to sea level, standard day conditions. The generalized engine performance graphs are presented in Appendix F, Figures 23 through 27 and are included for future engineering analysis.

Power Available

Engine power available was calculated from the generalized engine performance graphs and the results are presented in Appendix F, Figures 28 and 29. The standard day power available was calculated to be 1694 ESHP at military power (30 min.) and 1723 ESHP at contingency power (2-1/2 min.). At military power the engine shaft horsepower remained constant until reaching an altitude of 10,600 ft. pressure altitude where a linear decrease with increasing pressure altitude began. The contingency power curve displayed a similar characteristic; however, the horsepower decrease did not begin until 14,200 ft. pressure altitude. Under hot day conditions at sea level, contingency power equalled the standard day power of 1723 ESHP. Under the same

conditions, military power only reached 1659 ESHP which is 35 ESHP (2%) below the standard day power. During attack missions conducted under standard or hot conditions, the AH-64A will have adequate power available. The power available of the AH-64A is satisfactory for the attack mission and met the requirements of the Specification.

Engine Chop Collar

The engine chop collar was evaluated for functional utility during the post-start engine checks. The collar was located on the pilot and the copilot/gunner's collective stick grip as depicted in Appendix C, Figure 7. The collar is designed to provide the pilot with a means to rapidly bring both engines to the idle position without removing his hand from the collective lever. The chop collar was actuated by grasping the collar, moving it forward 1/2 inch to release the safety lock mechanism, and rotating the collar 90° clockwise. The engine response to use of the chop collar was a smooth and rapid deceleration to idle. A CHOP warning light on the pilot's master caution panel illuminated immediately upon chop collar actuation. During a tail rotor failure or other instance requiring immediate engine power reduction, the attack pilot will quickly and easily accomplish the power reduction to idle without removing his hand from the collective. The engine chop collar, which allows the pilot to reduce engine power to idle without removing his hand from

the collective, is an enhancing characteristic which should be incorporated in future designs.

Engine Power Levers

The engine power levers were evaluated for functional utility during the Engine Control Unit ECU lockout check of the post-start checklist. The levers were located on the pilot's power level quadrant as depicted in Appendix C, Figure 8. The friction on the engine condition levers was very high even with the friction adjustment completely off. The force required to actuate the levers was estimated to be 15 lbs. which made small precise engine adjustments difficult. Torque overshoots (3-5%) of the desired value often resulted. When operating an engine in ECU lockout, the attack pilot will be required to use a high force to actuate the engine condition lever which will often result in overshooting the desired torque setting. The high force required to adjust the engine condition levers during ECU lockout operation is a Part III deficiency which should be avoided in future designs.

Transient Droop

The transient droop characteristics were evaluated from an initial condition of 100% NR and stabilized torque of 10%. Four collective increases to 80% torque were accomplished. The time increment for collective application to 80% torque was decreased on each run starting at 18 seconds for run one.

The evaluation was completed with a five-second application on run four. During the collective pulls, the engine governors initially displayed overcancel as NR increased to 104%. As the collective pull continued, NR decreased to approximately 98% followed by one to three RPM oscillations of 2-4%. During the final portion of the collective increase, NR drooped to the minimum level which normally reached a minimum of 96%. The maximum droop occurred during the five-second collective pull with NR falling to 93%. Activation of the low rotor RPM audio warning horn occurred at 94%. When performing NOE quickstops and high speed approaches into attack positions, the attack pilot will make rapid collective applications which will result in excessive main rotor RPM droop and settling of the aircraft into the ground or other obstacles to flight. The excessive transient droop encountered during rapid collective application is a Part I deficiency which should be corrected as soon as possible.

Engine Torque Transients

Engine torque transients of the AH-64A helicopter were evaluated during evasive maneuver roll reversals. Moderate rate (25-30 deg./sec.) rolls were accomplished from 60 degrees angle of bank right to left and vice versa. The level flight transmission torque setting was approximately 80% (40% per engine). During the left rolls transmission, torque increases as high as 100% occurred. During actual

evasive maneuvering, high transmission torque settings (above 80%) and rapid roll rates will be required. The attack pilot entering a rapid left roll with torque set at 95-100% could easily exceed the 115% transient transmission torque maximum resulting in transmission or drive train component damage. The excessive engine torque transient during left rolling maneuvers is a Part II deficiency which should be corrected as soon as possible.

Rotor Brake

The main rotor brake was evaluated for functional utility during normal engine shutdown procedures. The data are presented in Table IV.

Table IV
ROTOR BRAKE SHUTDOWN

Rotor RPM at Brake Actuation (%)	Main Rotor Wind Down Time (sec.)
40	57
38	53
40	59

The response to rotor brake application was sluggish and the main rotor wind down times were approximately one minute during all three shutdowns. During the one minute main rotor wind down period after rotor brake engagement, gusting wind conditions across the deck will result in extreme rotor blade

flapping and droop stop pounding which will cause main rotor head damage and present a hazard to flight deck personnel. The slow main rotor wind down following rotor brake application is a Part I deficiency which should be corrected as soon as possible.

HOVER

General

The hover performance of the AH-64A helicopter was evaluated utilizing the free hover method under the conditions presented in Appendix E, Table I. The generalized hover performance data are presented in Appendix F, Figures 30 and 31. The hover performance data presented in Appendix F, Figure 35 through 40 were unrefereed from the generalized data and are presented for standard day and hot day conditions. The non-dimensional hover performance graphs, presented in Appendix F, Figures 32 and 33, are submitted for future engineering analysis.

IGE/OGE Determination

The transition height between In Ground Effect (IGE) and Out of Ground Effect (OGE) was determined by recording the engine power required to hover at 10 ft. incremental hover heights from 10 to 70 ft. AGL. The data are presented in Appendix F, Figure 34. The IGE/OGE transition occurs where the power required to hover remains essentially constant with increasing hover height. From the graph in Appendix F,

Figure 34, it can be seen that the transition occurs at 50 ft.

IGE Hover Performance

The results of the 10 ft. IGE hover performance evaluation are presented in Appendix F, Figures 35, 36, and 37. At the primary mission weight, the standard day 10 ft. IGE hover ceiling was calculated to be in excess of 20,000 ft. pressure altitude. Even with a five percent power margin, to account for engine deterioration, the hover ceiling was 19,000 ft. pressure altitude. At the alternate mission weight, the standard day 10 ft. IGE hover ceiling was calculated to be 18,000 ft. pressure altitude and 15,700 ft. with a five percent power margin. The hot day, five percent power margin, hover ceiling at the primary and alternate mission weight was calculated to be 13,500 ft. and 9,100 ft. pressure altitude, respectively. At the alternate mission weight of 16,344 lbs., the AH-64A will have the capability to conduct IGE hovering operations under hot day conditions and pressure altitudes up to 9,000 ft., which will allow summertime mountainous operations with full fuel and a 1,950 lb. ordnance load. Within the scope of this test, the 10 ft. IGE hover performance of the AH-64A helicopter is satisfactory for the attack mission. The 10 ft. IGE standard day hover ceiling of 21,500 ft. PA at the primary mission weight

exceeded the requirement of the Specification by 6,900 ft. (47%).

OGE Hover Performance

The results of the OGE hover performance evaluation are presented in Appendix F, Figures 38, 39, and 40. At the primary mission weight, the standard day OGE hover ceiling was calculated to be 18,200 ft. pressure altitude and 15,500 ft. PA with a five percent power margin. At the alternate mission weight, the standard day OGE hover ceiling was calculated to be 15,400 ft. pressure altitude. Under hot day conditions, the hover ceilings were reduced which lowered the alternate mission weight hover ceiling to 2,400 ft. pressure altitude. Under standard day conditions, the AH-64A will be capable of performing OGE hovers to 18,200 ft. PA at the primary mission weight which will allow employment of NOE tactics during mountainous area operations. Within the scope of this test, the OGE hover performance of the AH-64A helicopter is satisfactory for the attack mission. The OGE standard day hover ceiling of 18,200 ft. PA at the primary mission weight exceeded the requirement of the Specification by 7,000 ft. (62%).

VERTICAL CLIMB

The vertical climb performance of the AH-64A helicopter was evaluated under the conditions presented in Appendix E, Table I. The generalized vertical climb data and climb

correction factor graph are presented in Appendix F, Figures 41 and 42. The vertical climb performance graphs presented in Appendix F, Figures 43 and 44 were unreferred from the generalized data and are presented for both standard and hot day conditions. The vertical climb performance was calculated to be 2020 FPM at the primary mission weight under standard day sea level conditions. At the alternate mission weight, the rate of climb decreased to zero by 18,200 ft. pressure altitude at the primary mission weight and by 14,200 ft. pressure altitude at the alternate mission weight. The vertical rate of climb to zero point decreased by 7,000 ft. under hot day conditions and was calculated to be 11,200 ft. pressure altitude at the primary mission weight and 7,200 ft. at the alternate mission weight. During hot day sea level conditions, the AH-64A will be capable of operating with a full load of fuel, almost 2,000 lbs. of ordnance and perform a vertical climb at 1050 FPM. Within the scope of this test, the vertical climb performance of the AH-64A helicopter is satisfactory for the attack mission. The vertical climb rate of 2020 FPM at the primary mission weight under standard day sea level conditions failed the requirements of the Specification by 340 FPM (14%).

LEVEL FLIGHT

General

The level flight performance of the AH-64A helicopter was evaluated under the conditions presented in Appendix E,

Table I. The w/σ test technique was utilized with constant w/σ values determined inflight using an HP-41CX hand-held calculator. The results of the evaluation are summarized in Table V.

Table V
LEVEL FLIGHT PERFORMANCE

Parameter	Result
V_H	128 KTAS
$V_{\max}$ range	115 KTAS
V_{cruise}	123 KTAS
$V_{\max}$ endurance	65 KTAS
Combat Radius	103 NM
Maximum Endurance	2.67 HR

The generalized level flight performance data is presented in Appendix F, Figure 45, and the generalized level flight engine performance (W_f corr vs. ESHP corr) is presented in Appendix F, Figure 46. The nondimensional level flight performance graph presented in Appendix F, Figure 47 is submitted for future engineering analysis.

Maximum Level Flight Airspeed

The data are presented in Appendix F, Figure 48. The maximum level flight airspeed (V_H) under sea level standard day conditions at maximum gross weight (17,650 lbs.) was

calculated to be 128 KTAS. At the alternate mission gross weight (16,344 lbs.) 128 KTAS would be the maximum level flight airspeed at 2,500 ft. pressure altitude and standard day. Current Marine Corps medium and heavy lift helicopters are capable of speeds greater than 140 KTAS. When performing helicopter escort missions, the attack aircraft will have a limited dash capability which will restrict maneuvering in and around the assault helicopter formation. The aircraft will not have a sufficient airspeed margin to suppress enemy ground fire received enroute and catch up to the formation. The low maximum level flight airspeed of the AH-64A helicopter is a Part II deficiency which should be corrected as soon as practicable. The V_H of 128 KTAS met the requirements of the Specification.

Combat Radius

The combat radius of the AH-64A helicopter was calculated based on the fuel requirements listed in Table VI.

Table VI

COMBAT RADIUS DETERMINATION

Flight Phase	Fuel Required (lbs.)
10 min. Warmup and T/O	150
Cruise Flight at V_{cruise} (each way)	1025
10% Fuel Reserve	<u>244</u>
Total	2444

Based on the 123 KTAS (V_{cruise}) specific range of .1005, the combat radius at 17,650 lbs. is 103 NM under standard day sea level conditions. The calculations from Table VI represent a single target attack and return to base or in effect no time on station. The graph presented in Appendix F, Figure 50 shows the effect of time on station on the combat radius of the aircraft. The time on station was calculated based on V_{max} endurance fuel flow. With one hour on station, the combat radius will be reduced to 65 NM. The combat radius of the aircraft will allow operations from secure areas and ships well behind the forward edge of the battle area and still provide adequate time on station. Within the scope of this test, the combat radius of the AH-64A helicopter is satisfactory for the attack mission and met the requirements of the Specification.

Range

The range of the AH-64A helicopter was also calculated based on the requirements of Table VI; however, flown at V_{max} range (115 KTAS). The maximum range at 17,650 lbs. under sea level standard day conditions, was calculated to be 208 NM. The range at the alternate mission weight, 2,500 ft. pressure altitude, and standard day temperature, was also 208 NM. When deploying from a ship for operations ashore, the attack aircraft can launch well out to sea without bringing the ship within the range of shore defenses. Within the scope of this test, the range of the AH-64A helicopter is

satisfactory for the attack mission and met the requirements of the Specification.

Endurance

The data are presented in Appendix F, Figure 49. Endurance airspeed ($V_{\text{max endurance}}$) at the alternate mission weight, 2,500 ft. pressure altitude, and standard day temperature was calculated to be 65 KTAS. Based on a fuel flow rate of 912 lbs./hr. the maximum endurance was calculated to be 2.67 hrs. with zero fuel reserve. The fuel flow curve is steep which only provides a 10 knot airspeed band (60-70 KTAS) for maximum endurance. Maintaining 60-70 KTAS (± 2 kts.) was easy, requiring only small (1/16 - 1/8 in.) longitudinal cyclic inputs at 3-5 second intervals (HQR-3). When conducting on-call close-in fire support missions, the AH-64A will be able to hold at maximum endurance for 1.5 hours and still maintain a 45 NM combat radius. Within the scope of this test, the endurance of the AH-64A helicopter is satisfactory for the attack mission and met the requirements of the Specification.

FORWARD FLIGHT CLIMB AND DESCENT

Climb Performance

The forward flight climb performance was evaluated under the conditions presented in Appendix E, Table I, and the data are presented in Appendix F, Figure 51. The maximum rate of climb under the test day conditions was 2350 FPM and occurred

at 80 KIAS. Flying at an airspeed 10 knots faster than the maximum rate of climb airspeed reduced the rate of climb by 100 FPM. Flying at an airspeed 10 knots slower than the maximum rate of climb airspeed had an even greater effect, reducing rate of climb by 150 FPM. When conducting terrain flight operations at the alternate mission weight, between 2,000 to 4,000 ft. pressure altitude, and 24°C, the aircraft will have adequate climb performance to avoid terrain and obstacles, and the pilot will be able to maximize rate of climb by flying at 80 KIAS. Within the scope of this test, the forward flight climb performance of the AH-64A helicopter is satisfactory for the attack mission.

Descent Performance

The forward flight descent performance was evaluated under the conditions presented in Appendix E, Table I, and the data are presented in Appendix F, Figure 51. The minimum rate of descent under the test day conditions was 2,250 FPM and occurred at 60 KIAS. During the descent, maintaining 60 KIAS (± 2 kts.) and balanced flight ($\pm 1/4$ ball width) was easy requiring only small ($1/8$ in.) longitudinal cyclic inputs every 3 to 5 seconds. No collective or pedal inputs were required after the initial collective reduction to full down and the $1/2$ in. right pedal input. The rate of descent was qualitatively evaluated to be high. During a low altitude dual engine failure, the attack pilot will easily maintain the minimum rate of descent airspeed but due to the

rapid descent will have very little time to select a landing spot. The high minimum rate of descent of the AH-64A helicopter is a Part III deficiency which should be avoided in future designs.

III. CONCLUSIONS AND RECOMMENDATIONS

GENERAL

Upon correction of the two Part I deficiencies identified during this evaluation, the AH-64A will have excellent potential for the USMC attack mission. A final determination of the suitability of the aircraft will be based on the results of a mission systems and a shipboard compatibility evaluation.

SPECIFIC

ENHANCING CHARACTERISTICS

The following features of the AH-64A helicopter were determined to be enhancing characteristics which should be incorporated in future designs:

1. the DASE off flying qualities;
2. the engine chop collar which allows the pilot to reduce engine power to idle without removing his hand from the collective.

PART I DEFICIENCIES

General

As presented in Appendix A, Part I deficiencies are those problems which adversely affect airworthiness of the aircraft, the ability of the aircraft to accomplish its mission, or the effectiveness of the aircrew. These Part I deficiencies should be corrected as soon as possible and are listed below:

1. the excessive transient droop encountered during rapid collective application;
2. the slow main rotor wind down following rotor brake actuation.

Transient Droop

The transient droop experienced by the AH-64A helicopter is apparently the result of the T700 engines not being able to accelerate rapidly enough to avoid a reduction in rotor speed when a large and/or rapid collective increase is made. The T700 engine is designed to respond to power demands automatically through interaction of the hydromechanical unit (HMU) and the electrical control unit (ECU). The HMU provides fuel to the engine combustion system through the use of a high pressure fuel pump and a metering valve. The HMU provides the fuel based on inputs received from the load demand spindle (LDS). The LDS is connected to the collective pitch linkage and mechanically provides collective position information to the HMU. Additionally, the HMU responds to signals from the ECU to interpret the proper fuel flow for all operating conditions. This is accomplished by the ECU engine governing channel which monitors an engine speed sensor and generates an output signal to the HMU. The primary recommended corrective action to this deficiency would be installation of a system to provide collective rate information to the engine ECU/HMU. Currently, the engine only receives collective position information which is

inadequate to avoid rotor droop during rapid collective increases. The CH-46 and the AH-1W aircraft incorporate collective anticipators to reduce transient droop. The collective anticipators provide an electrical signal to the ECU, or power management system for the CH-46, based on collective rate of change. On the AH-1W, which also incorporates T700 engines, the collective anticipator is a linear voltage differential transformer connected to the LDS linkage which provides the collective rate of change signal to the ECU. If the addition of a collective anticipator does not sufficiently correct the transient droop problem of the AH-64A, an engine modification would be necessary.

Currently, the U.S. Navy is installing T700-401C engines in the SH-60 aircraft and among the improvements to the new engine is an increased acceleration HMU. As presented in reference 13, the increased acceleration HMU is designed to supply 15% higher fuel flow during engine acceleration relative to the current T700 HMU. The engines in the AH-64A helicopters could also be installed with the new HMUs to reduce transient droop.

Rotor Brake

According to the information provided in reference 14, there are three techniques for increasing the efficiency of a braking system: (a) increase hydraulic pressure, (b) increase the diameter of the brake disc, or (c) increase the size of the brake pucks. Of course, the efficiency of a

helicopter rotor brake system could also be increased through the addition of another rotor brake quill. The rotor brake assembly of the AH-64A is mounted to the aft portion of the transmission on the accessory drive section, as depicted in Appendix C, Figure 9, and is powered by the utility hydraulic system. When the pilot's rotor brake control switch is set to the BRAKE position, hydraulic pressure is supplied to the rotor brake actuator at 312 to 363 psi. A pressure relief valve is incorporated in the system and is designed to prevent rotor brake pressure from exceeding 600 psi. Once the rotor is stopped, the rotor brake control switch may be set to lock. In the lock position, the solenoid valves and the pressure relief valve in the utility hydraulic manifold will deenergize, thereby providing maximum utility hydraulic pressure (3000 psi) to the rotor brake assembly. To improve the effectiveness of the AH-64A rotor brake system, a small increase in the size of the rotor brake disc would be possible with only slight modification of the accessory drive section. However, any significant size increase and certainly the addition of a second rotor brake assembly, would require a major redesign of the transmission and a lengthy requalification effort. The most viable alternative for improving the efficiency of the rotor brake system would be to increase the size of the rotor brake disc and rotor brake actuator to the maximum extent possible, within the current size constraints of the transmission accessory drive

section. Additionally, the hydraulic pressure supplied to the rotor brake actuator when the control switch is in the BRAKE position should be increased to approximately 500-750 psi. The pressure relief valve limit should also be increased from 600 to 1000 psi.

PART II DEFICIENCIES

General

Part II deficiencies are those problems which do not substantially reduce the ability of the aircraft to accomplish its mission but the correction of which will result in significant improvement in effectiveness, maintainability, or safety. These Part II deficiencies should be corrected as soon as practicable and are listed below:

1. the extremely high lateral control sensitivity;
2. the pilot induced oscillations incurred during slope landings;
3. the excessive torque transients during left rolling maneuvers;
4. the low maximum level flight airspeed;
5. the limited forward field of view during confined area approaches and NOE quickstops;
6. the requirement to release the collective lever to engage the hover hold system;
7. the poor location of the TRIM REL switch on the pilot's cyclic grip resulting in stick jump;

8. the reduced accessibility of the engine fire pull handles, caused by glareshield obstructions;

9. the frequent collective inputs required during IGE/OGE hover operations;

10. the delay in activation of the APU fire bottle switch caused by obstruction of the fire pull handle.

Lateral Control Sensitivity and Slope Landing PIOs

The extremely high lateral control sensitivity deficiency and the PIOs incurred during slope landings deficiency will be addressed together since the slope landing PIOs are apparently a result of the first problem. As presented in reference 15, the initial roll acceleration is not only dependent on the size of the control input but also the control sensitivity. The control sensitivity is defined as the angular acceleration per unit of control input. Additionally, the steady state roll rate achieved for a given size control input is dependent upon the ratio of the sensitivity to the angular rate damping. For a given size step input, the effect on the resulting roll rate of increasing the sensitivity and the rate damping, while holding the other variable constant is shown in Figure 1. For the AH-64A helicopter, this sensitivity to damping ratio is modified by the CAS and SAS functions of the automatic flight control system.

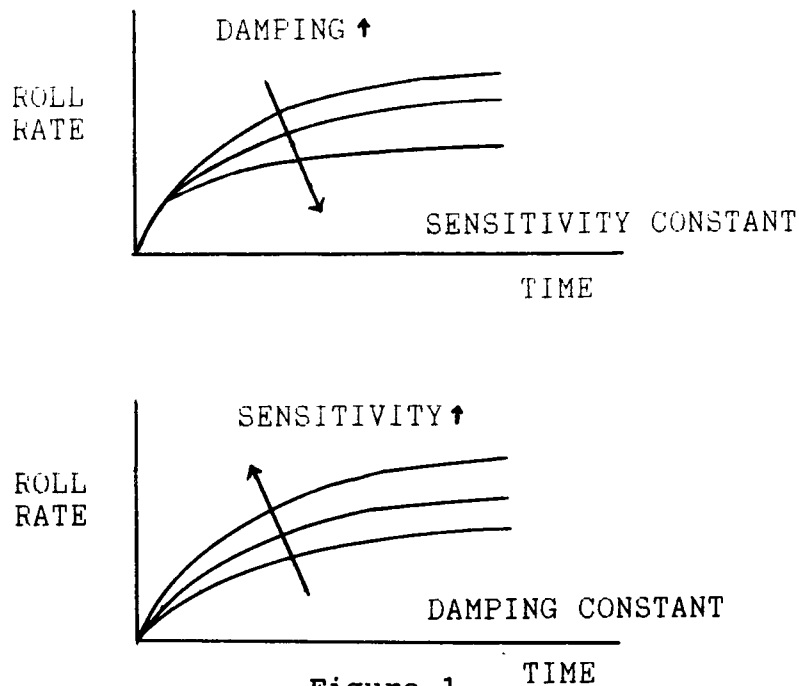


Figure 1

SENSITIVITY AND DAMPING EFFECT ON ROLL RATE

As discussed in Appendix D, the CAS enables the SAS to recognize control inputs from uncommanded airframe oscillations to ensure that the control inputs will not be damped out. Also, the CAS is gain scheduled with airspeed to provide less aircraft response per cyclic input at higher airspeeds. These two deficiencies can very easily be corrected by reducing the gain of the CAS. Currently, the roll rate developed for a one inch left step input is approximately 15 to 20 deg/sec. However, a one inch aft step input only produces a 10 deg/sec pitch rate. To reduce the lateral control sensitivity and corresponding PIO tendency, it is recommended that the CAS be gain scheduled to provide a roll rate of approximately 10 deg/sec for a one inch lateral step input.

Left Rolling Maneuver Torque Transients

When a left cyclic input is made, the pitch on the advancing blades increases and there is a corresponding decrease in the pitch on the retreating blades which will result in a left roll of the aircraft. The normal response of the main rotor to an increase in pitch on the advancing blades would be a reduction in main rotor RPM and a corresponding decrease in the engine free power turbine speed (N_p for T700 engines). The engine governing system, sensing the impending loss of engine and main rotor speed, responds with a momentary engine power increase. This characteristic alone is not a problem, but when the pilot attempts to maneuver with a transmission torque setting above 90%, the engine power increase during left rolls could easily result in a transmission overtorque. The optimum solution would be to install a torque limiting system to prevent the transmission torque from exceeding specified limits. Unfortunately, there has not been a system developed which can adequately respond to the rapid torque transients that result from the left rolling engine power surges. It is hoped that such a system can be developed for the next generation military helicopters, however, the immediate solution to the problem for the AH-64A is to improve the transmission to withstand the torque transients without damage. According to reference 16, transmission advancements can come from either specific component improvements

or a redesign of the transmission as a system. Reference 16 proposed specific component improvements which could have application to the AH-64A transmission. A component breakdown of the transmission is presented in Appendix C, Figure 10. One improvement could be to use high contact ratio planetary gear teeth to prevent damage by shifting of the tooth contact pattern during high load conditions. In addition, improvements can be achieved in transmission gear alignment and power capacity through use of two piece straddle mounted components such as the planetary or bevel gears. It is recommended that the transmission of the AH-64A be modified to withstand torque values to 125% for a five second duration.

Low Maximum Level Flight Airspeed

Although a power available increase is possible, through use of the transmission improvements discussed in the previous paragraph, the maximum level flight airspeed increase would be small if the shape of the power required curve remains the same. As depicted in Appendix F, Figure 48, the slope of the power required curve is extremely steep. For example, without a parasite drag decrease, a power available increase of 400 SHP would be required to realize a 10 knot maximum level flight airspeed increase. The primary speed increase could result from a reduction in the parasite drag of the aircraft. Since the parasite power increases based on forward speed cubed, parasite drag is the major contributor

to the steepness of the power required curve at high speed. There are a number of ways to reduce the parasite drag of the AH-64A helicopter. For example, a drag decrease could be achieved from a return to the rounded canopy panels with which the Apache had been originally designed. The U.S. Army responding to a concern for visual acquisition caused by sun glint off of rounded canopys, decided that the AH-64A would have to be redesigned with flat canopy panels, even though an increase in parasite drag would result (reference 17). A drag decrease could also occur from a change in the aircraft's pitch attitude during level flight. The nose down attitude during high speed level flight could be decreased in two manners. One option would be to shift the aircraft's CG aft. According to reference 18, a change in CG from fuselage station 202 to 206 caused a drag decrease of 3.1 ft^2 equivalent flat plate area. This CG shift could be accomplished by moving some of the avionics equipment from the forward compartments ("saddle bags") to the tail boom. An additional benefit of moving components from the "saddle bags" aft would be that the size of these extremely draggy protrusions could be reduced. However, the aft CG shift would have to be conservative to avoid hover attitude and longitudinal flying quality problems. The nose down attitude of the aircraft during high speed level flight could also be reduced by rescheduling the horizontal stabilator position with airspeed. Other ideas for parasite drag reduction

would include main rotor hub fairings, as depicted in Appendix C, Figure 11, and fairings for the avionics compartment to main fuselage junctions. It must also be stressed that parasite power is not the only contribution to the steepness of the power required curve. Profile power can also lead to a very steep curve when compressibility or blade stall, or both are encountered. Some designers have proposed the use of wings to increase speed which will assist by unloading the rotor to help eliminate the retreating blade stall and corresponding drag rise. However, a wing will generally only be useful in a compound helicopter (receives propulsion from sources other than the main rotor), as explained by R.W. Prouty in reference 19,

"For a helicopter that must be designed for high speed level flight, adding a wing just for this purpose is usually a bad idea. Besides producing an unwanted download in a hover, the wing may actually increase the retreating tip angle of attack leading to premature blade stall. This happens because the partially unloaded rotor must be tilted further forward to produce the required propulsive force that now must compensate for the drag of the wing in addition to the drag of the basic helicopter. This requires increased collective pitch to overcome the inflow and increased cyclic pitch to keep the rotors trimmed. In many applications, these two effects will increase the retreating tip angle of attack more than the decrease made by unloading the rotor."

The maximum level flight airspeed of the AH-64A helicopter should be increased through a combination of a power available increase, from transmission improvements, and

through use of the parasite drag reduction techniques presented above.

Limited Forward Field of View

Utilizing the rounded canopy panels recommended for the maximum level flight speed increase should also assist in correcting the field of view deficiency. No drawings of the Apache rounded canopy were available; however, judging from the difference between the Cobra flat plate and rounded canopy panels, the rounded canopy support frames would be smaller. In order to further correct the pilot's field of view problem, an instrument panel modification would be required. Unfortunately, just lowering the panel is not an option because insufficient leg clearance would result. An expensive and time consuming redesign of the instrument panel would be necessary which would probably preclude this modification, at least in the near term. In the absence of a further improvement in the pilot's field of view, an increase in the copilot's field of view would assist in correcting the deficiency. If the new rounded canopy is selected for construction and installation, at the same time, the copilot's instrument panel and/or glareshield should be lowered as much as possible.

Hover Hold Engagement

The requirement to release the collective lever to engage the hover hold system can be corrected very

simply. The ATTD/HOVER HOLD switch should be moved from its current position on the DASE panel to a position aft of the stabilator control panel on the collective. The recommended position is depicted in Appendix C, Figure 12.

Trim Release Switch Location

The poor location of the TRIM REL switch on the pilot's cyclic grip is another deficiency that can be corrected very easily. It is recommended that the trim release switch location be moved to a position mid way down the cyclic grip where the DASE release switch was previously located. The DASE release switch should then be moved to the bottom of the cyclic grip as shown in Appendix C, Figure 13. The trim off function of the TRIM REL switch should be moved to the DASE panel where the ATTD/HOVER HOLD switch used to be. These new positions are more consistent with the standard trim switch locations on military helicopters.

Engine Fire Pull Handles

The optimum solution would be to move the fire handles and agent discharge switch from the current location (depicted in Appendix C, Figure 4) to a more central and accessible location on the instrument panel. However, this modification would require a redesign of the instrument panel and the severity of this deficiency does not warrant such a complex and expensive change. The most logical and cost

effective manner to correct this problem would be to slightly modify the instrument panel glareshield to provide an additional 1/2 to 1 inch of clearance from the engine fire handles and the glareshield obstructions.

Hover Collective Inputs

It is recommended that an altitude hold system be installed in the AH-64A helicopter to eliminate the pilot collective inputs required during IGE/OGE hover operations. Particularly in the anti-tank mission, the aircraft will be required to operate in a hover for prolonged periods and an altitude hold system will greatly reduce the pilot's workload. The selected system should incorporate the following features:

1. the capability for either barometric or radar altitude hold;
 2. an electro-hydraulic collective actuator utilizing a time-velocity limited response;
 3. a collective trim system utilizing a collective trim trigger;
 4. an automatic disengage mode if the collective lever is manually moved, the collective trim trigger is pressed, or if the radar altimeter fails;
 5. a beeper altitude trim capability in the radar mode.
- Systems incorporating many if not all of these features are already military qualified and most U.S. Navy helicopters,

including the CH-46, SH-60, and SH-3, have similar systems installed.

APU Fire Bottle Selector Switch

The APU control panel is located in the pilot's right hand console as depicted in Appendix C, Figure 4. One option to correct the fire bottle selector switch blockage is to move the APU control panel to a more forward position in the right hand console. However, the APU panel frequency of use does not warrant moving the panel forward. To do this, other items that are used much more often would have to be moved aft into the relatively inaccessible location of the APU panel. Therefore, to alleviate the delay in activation of the APU fire bottle selector switch, the APU panel should be modified as shown in Appendix C, Figure 14. By moving the fire pull handle aft of the fire bottle selector switch, the handle obstruction to the switch will be corrected.

PART III DEFICIENCIES

As presented in Appendix A, Part III deficiencies are minor problems or those which appear too impractical or costly to correct in this model aircraft. The Part III deficiencies should be avoided in future designs and are listed below:

1. the location of the tail wheel lock switch outside the pilot's shoulder harness locked, fully arm stretched functional reach;

2. the fluctuation in airspeed encountered above 92% torque;
3. the small cyclic force and displacement cues to an off trim condition about a trim speed of 120 KIAS;
4. the limited sideforce cues to an out of balanced flight condition;
5. the frequent cyclic inputs required to maintain altitude during level accelerations;
6. the large break pedal inputs required for adequate braking action;
7. the moderate low frequency vertical vibration during flight above 120 KIAS, at bank angles greater than 45 degrees, and during translational lift;
8. the high force required to adjust the engine condition levers during ECU lockout operation.

SPECIFICATION CONFORMANCE

The flying qualities and performance of the AH-64A helicopter met the requirements of the Specification against which they were tested except:

1. MIL-STD-850B paragraph 8.2 in that unimpaired vision from the pilot's DEP of at least 25° down at 0° azimuth was not met by 16° (64%);
2. Detailed Specification paragraph 10.3.2.1.1 in that breakout force including friction for forward and aft cyclic exceeded 1.5 lbs. by 1.3 lbs. (87%) forward, 0.7 lbs (47%) aft and 0.5 lbs. (33%) left;

3. Detailed Specification paragraph 10.3.2.8 in that total system freeplay longitudinally exceeded 1/16 inch by 1/16 inch (100%);

4. Detailed Specification paragraph 10.3.3.2.2 in that the requirement for no stick jump was exceeded longitudinally by 1/4 inch and laterally by 1/8 inch;

5. Detailed Specification paragraph 10.3.2.2.1 in that undesirable discontinuities were present in the collective force gradients;

6. Detailed Specification paragraph 10.3.4.1 in that no forward displacement and push force were required for a 10 knot increase in airspeed from a 120 KIAS trim condition;

7. Detailed Specification paragraph 10.3.3.1.2 in that positive static longitudinal stability was not exhibited about a 0 KIAS trim point;

8. Detailed Specification performance section in that the 10 ft. IGE standard day hover ceiling of 21,500 ft. PA at the primary mission weight exceeded the requirement by 6,900 ft. (47%);

9. Detailed Specification performance section in that the OGE standard day hover ceiling of 18,200 ft. PA at the primary mission weight exceeded the requirement by 7,000 ft. (62%);

10. Detailed Specification performance section in that the vertical climb rate of 2020 FPM at the primary mission weight and standard day sea level condition failed the requirements by 340 FPM (14%);

FURTHER TESTING

Further flying qualities and performance testing is recommended in the following areas to fully identify the suitability of the AH-64A helicopter for the USMC attack mission:

1. to determine the source and magnitude of the aircraft's pitot static system errors;
2. to determine the ability of the hover augmentation system to counter the heading oscillation during a 20 kt crosswind from 225 to 270 degrees relative azimuth;
3. to determine the effects of the lateral control sensitivity and main landing gear on the aircraft's shipboard landing capability.

LIST OF REFERENCES

REFERENCES

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2. Military Specification, MIL-D-2322A, "Demonstration Requirements for Helicopters", dated 18 Mar 1971.
3. Military Specification, MIL-H-8501A, "Helicopter Flying and Ground Handling Qualities; General Requirements for", Sept 1961 with Amendment 1, dated 3 Apr 1962.
4. Military Specification, MIL-C-5011A, "Charts, Standard Aircraft Characteristics and Performance, Piloted Aircraft", dated 5 Nov 1951.
5. Military Specification, MIL-STD-250D, "Aircrew Station Controls and Displays for Rotary Wing Aircraft", dated 8 Jul 1966.
6. Military Specification, MIL-C-18244A, "Control and Stabilization Systems: Automatic Piloted Aircraft, General Specification for" dated 1 Dec 1962.
7. AH-64 Detailed Specification, AMC-SS-AAH-H1111A, Appendix I, "Flying and Ground Handling Qualities Specification", dated 15 Oct 1976.
8. USNTPS-FTM-102, "Helicopter Performance", dated 28 Jun 1968.
9. USNTPS-FTM-105, "Helicopter Stability and Control" (Preliminary), dated Nov 1983.
10. USNTPS TM 82-1 TP "Summary of Helicopter Performance Test Techniques", dated Feb 1984.
11. NASA TN D-5153 "The Use of Pilot Ratings in the Evaluation of Aircraft Handling Qualities", dated Apr 1969.
12. "Subjective Vibration Assessment Rating Scale", developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.
13. Rotary Wing Aircraft Test Directorate, Technical Report, RW-111R-88, dated Sept 1988.
14. Spotts, M. F., "Belts, Clutches, Brakes, and Chains", Design of Machine Elements, 5th edition, Chapter 6, 1978.

15. Miller, Robert, USNTPS Textbook, "Helicopter Stability and Control" (Draft not dated).
16. Bill, Robert C., "An Overview of Rotorcraft Propulsion Research at Lewis Research Center", Vertiflite May/June 1988, pp. 42.
17. Prouty, R.W., "Hide and Seek Basics to Helicopter Design", Rotor and Wing International, May 1988, pp. 20.
18. United States Army Aviation Engineering Flight Activity, Technical Report, Project No. 84-10, "First Article Preproduction Tests of the AH-64A Helicopter, Nov 1984, pp. 5.
19. Prouty, R.W., "Main Rotor Design", Helicopter Aerodynamics, 2nd Edition, 1984, pp. 92.

APPENDICES

DEFINITION OF DEFICIENCIES

Part I indicates a deficiency, the correction of which is necessary because it adversely affects:

- a. Airworthiness of the aircraft.
- b. The ability of the aircraft to accomplish its primary or secondary mission.
- c. The effectiveness of the crew as an essential subsystem.
- d. The safety of the crew or the integrity of an essential subsystem. In this regard, a real likelihood of injury or damage must exist. Remote possibilities or unlikely sequences of events shall not be used as a basis for safety items.

Part II indicates a deficiency of lesser severity than a Part I which does not substantially reduce the ability of the aircraft to accomplish its primary or secondary mission, but the correction of which will result in significant improvement in the effectiveness, maintainability, or safety of the aircraft.

Part III indicates a deficiency that appears too impractical or costly to correct in this model but which should be avoided in future designs. Included are violations of specifications for use by the contract negotiator in final settlement of the contract.

RATING SCALES

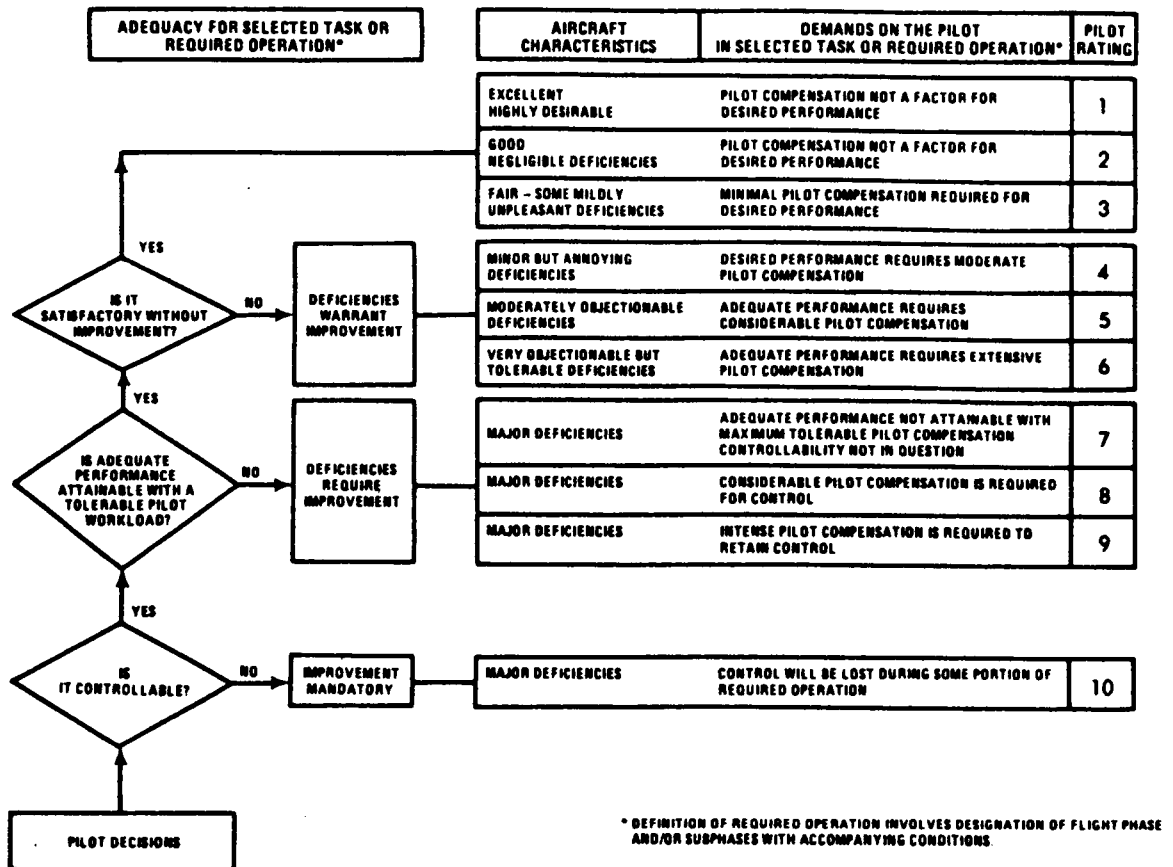


Figure 1

COOPER-HARPER HANDLING QUALITIES RATING SCALE

APPENDIX B

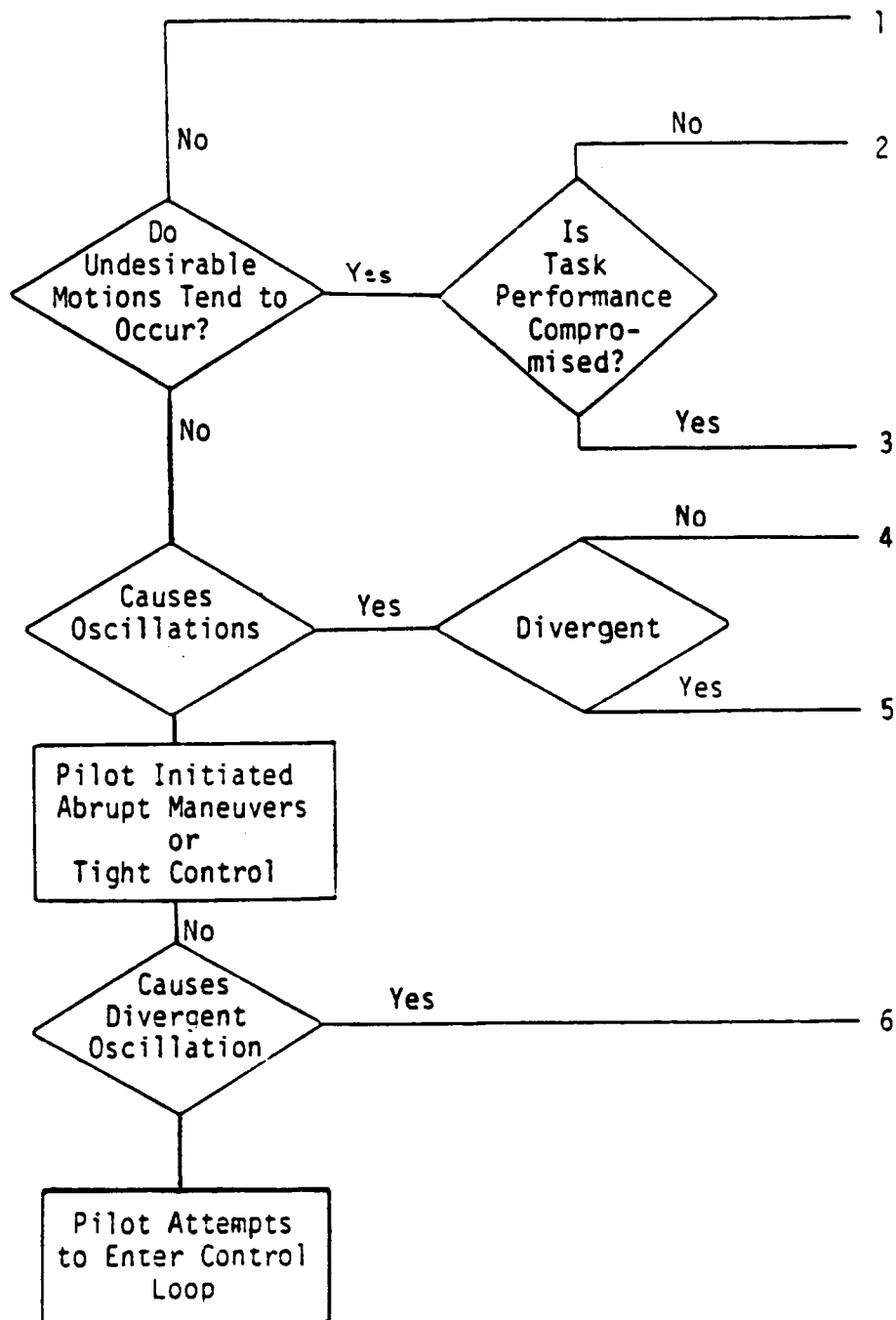


Figure 2

PILOT INDUCED OSCILLATION TENDENCY CLASSIFICATION

APPENDIX B

Table I
VIBRATION RATING SCALE

RATING	TERM USED	DESCRIPTION
0	No vibration	
1 2 3	Slight	Not apparent to experienced aircrew fully occupied by their tasks but noticeable if their attention is directed to it or if not otherwise occupied.
4 5 6	Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.
7 8 9	Severe	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.
10	Intolerable	Sole preoccupation of aircrew is to reduce vibration level.

ILLUSTRATIONS

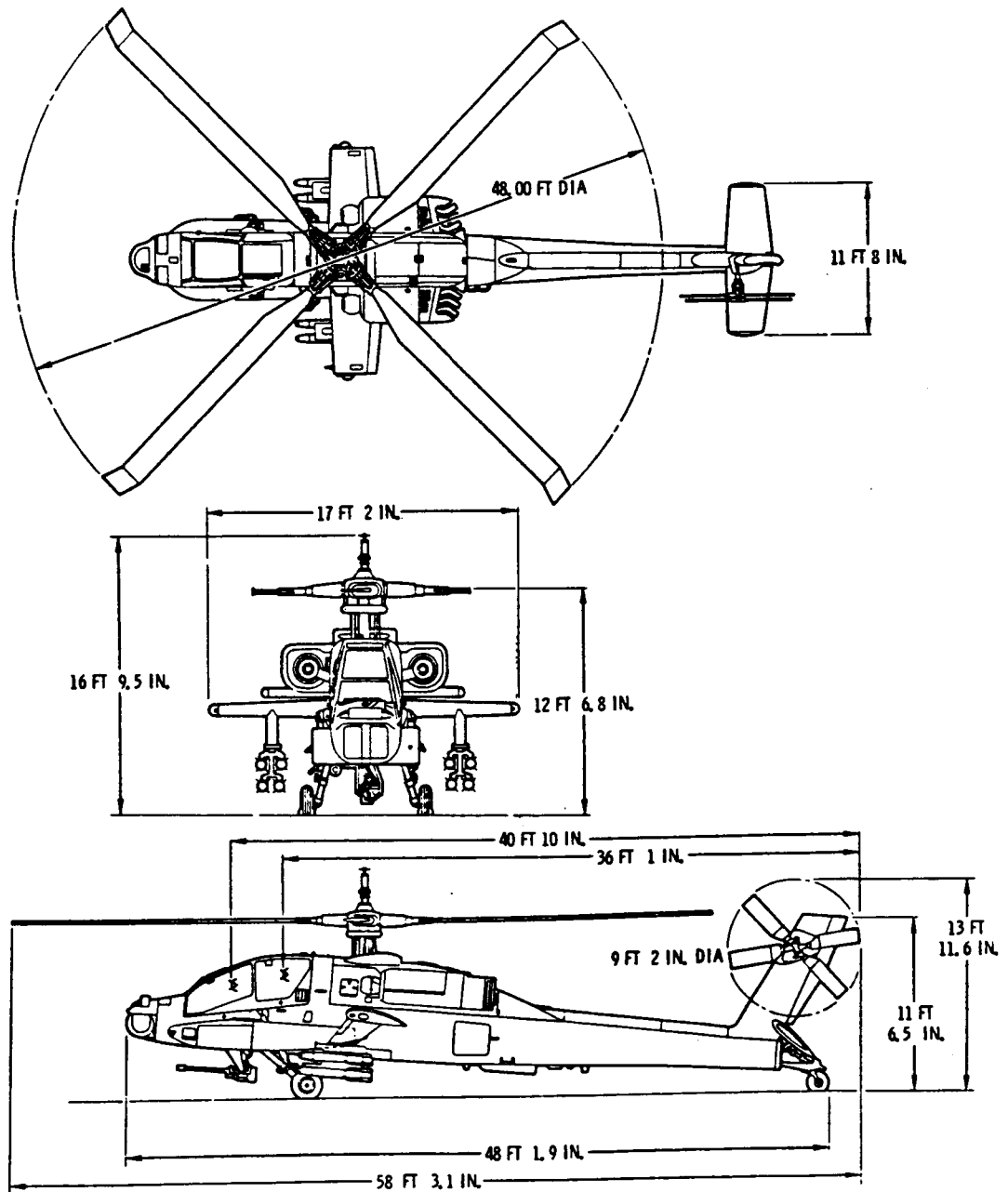


Figure 1

AH-64A APACHE HELICOPTER

APPENDIX C

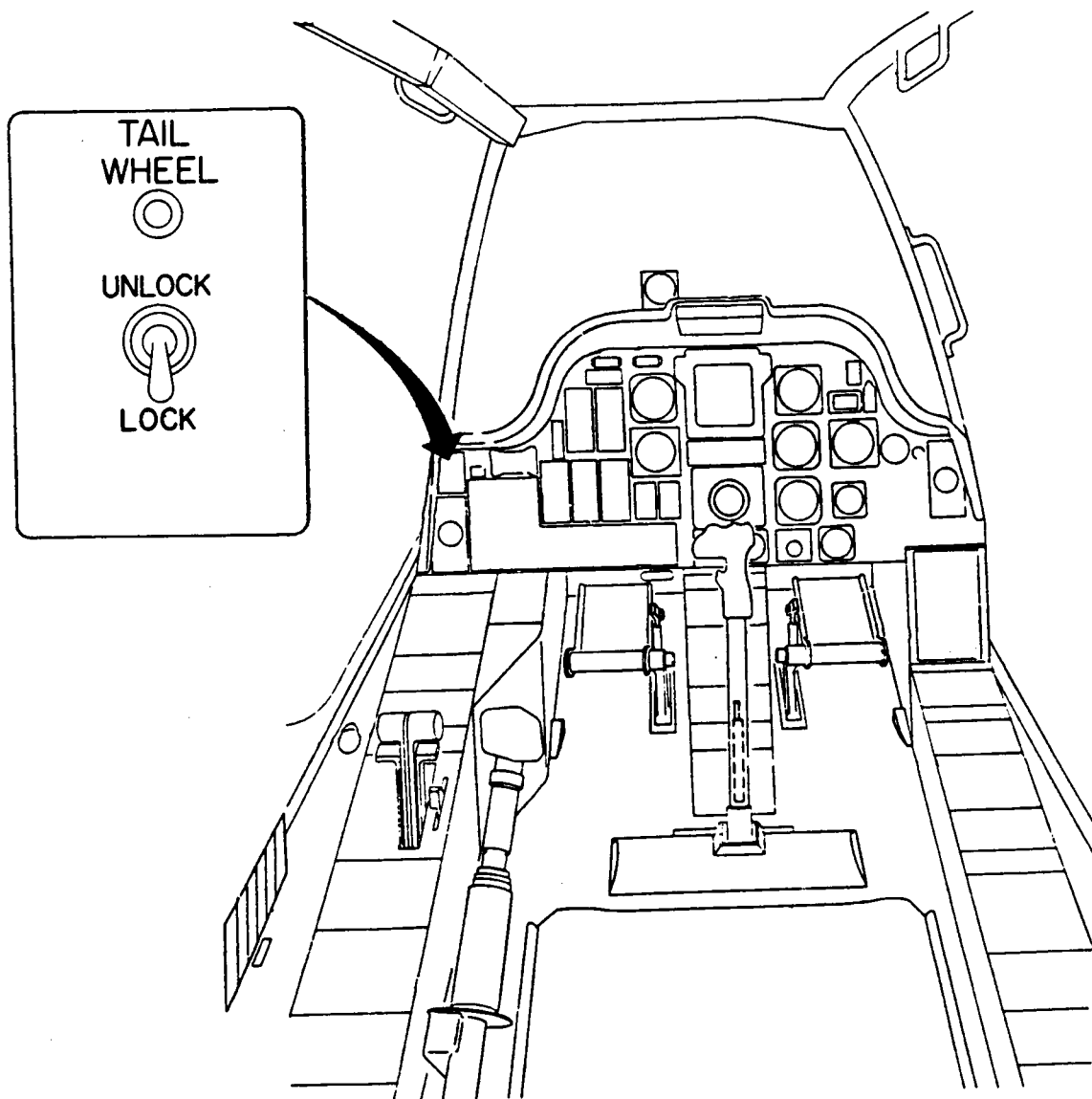


Figure 2

TAIL WHEEL LOCK SWITCH

APPENDIX C

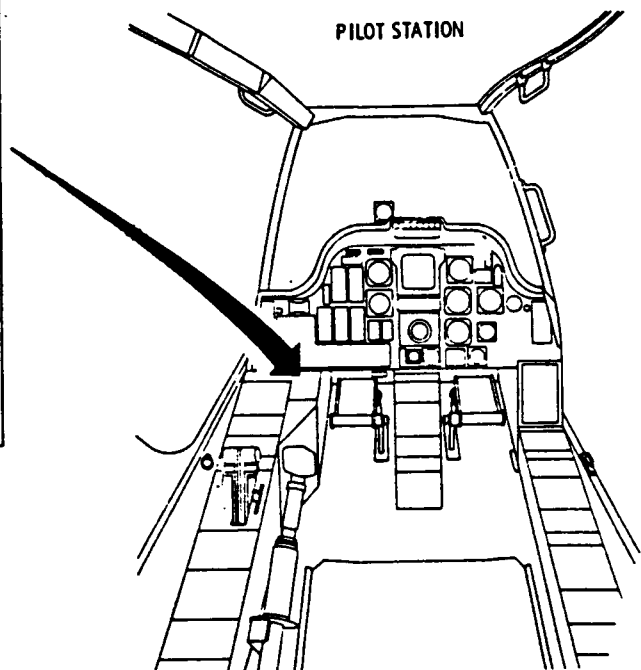
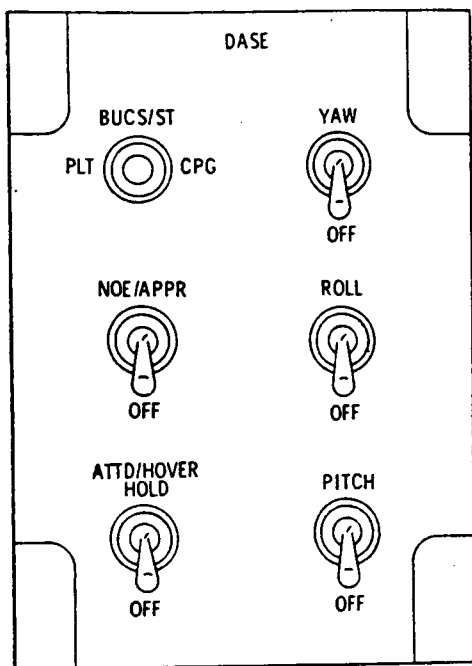


Figure 3

ATTITUDE/HOVER HOLD SWITCH

APPENDIX C

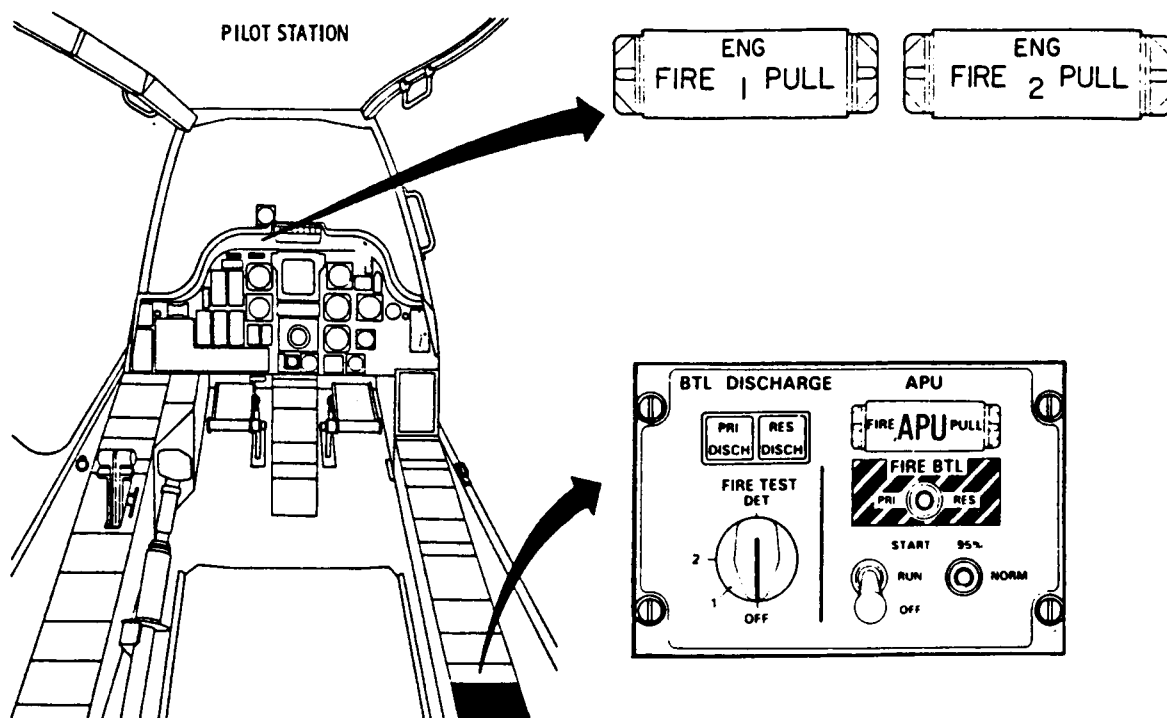


Figure 4
FIRE EXTINGUISHING SYSTEM CONTROLS

APPENDIX C

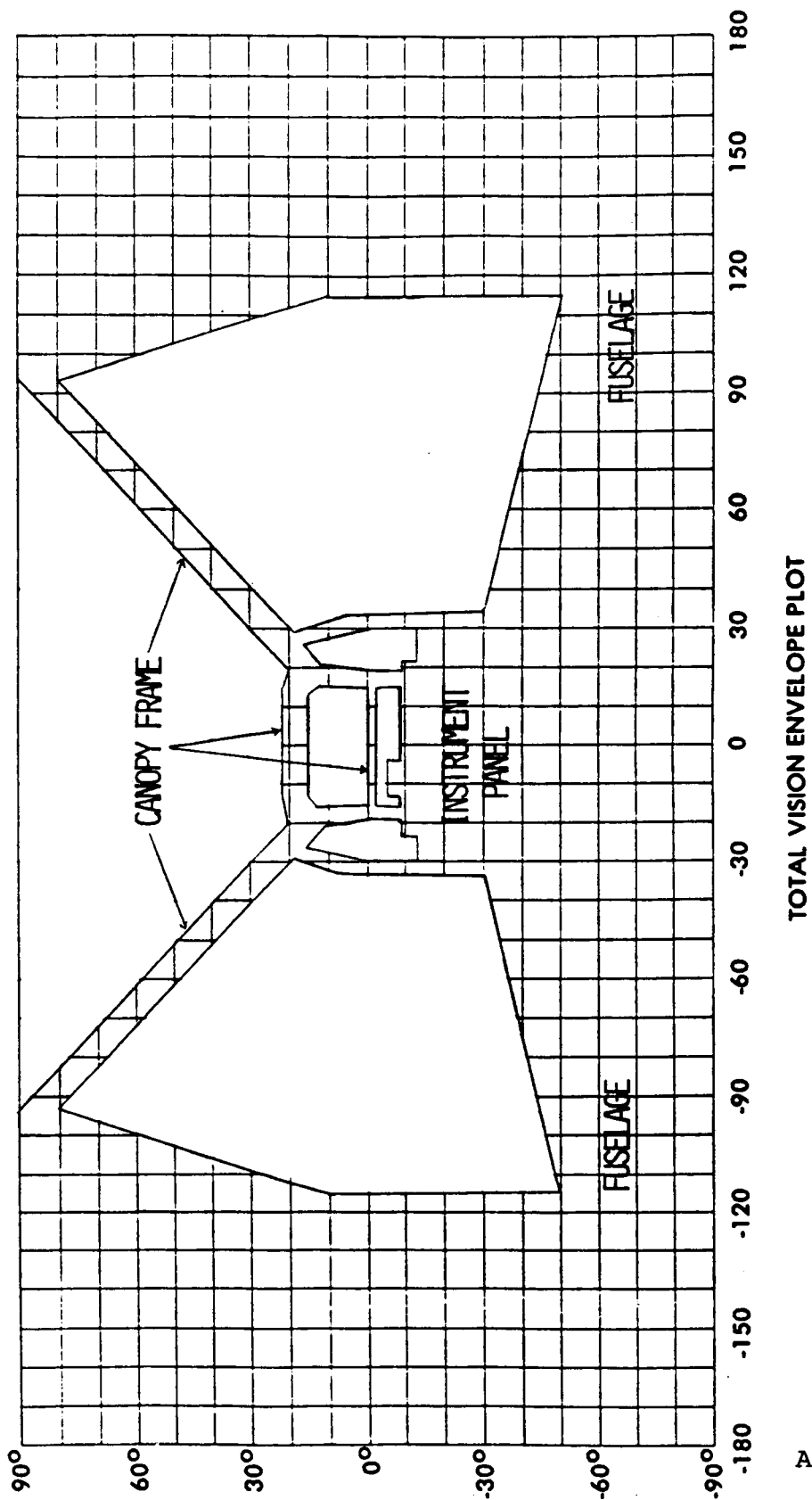


Figure 5
PILOT'S COCKPIT FIELD OF VIEW

APPENDIX C

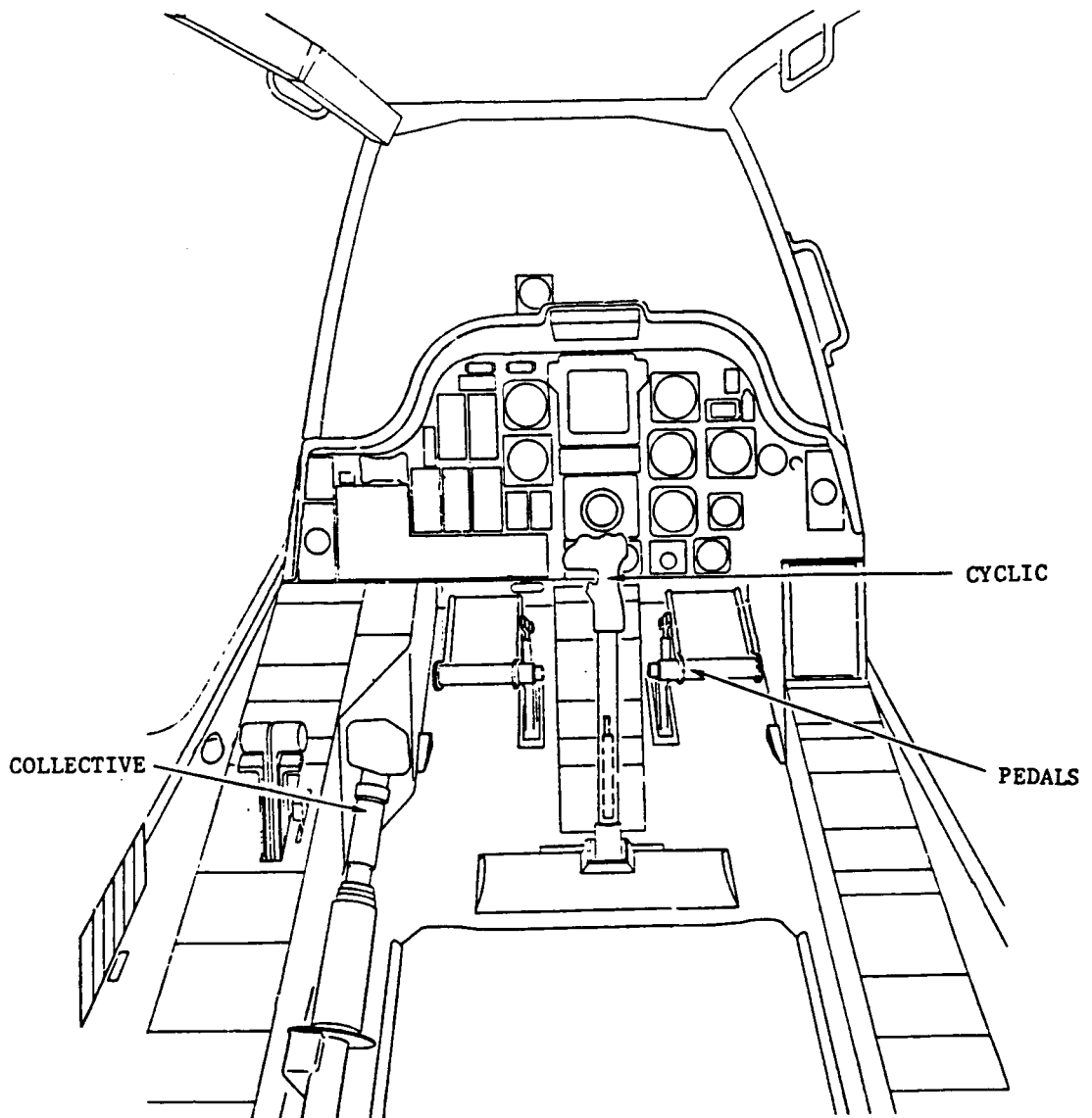


Figure 6
CONTROL FORCE MEASUREMENT POINTS

APPENDIX C

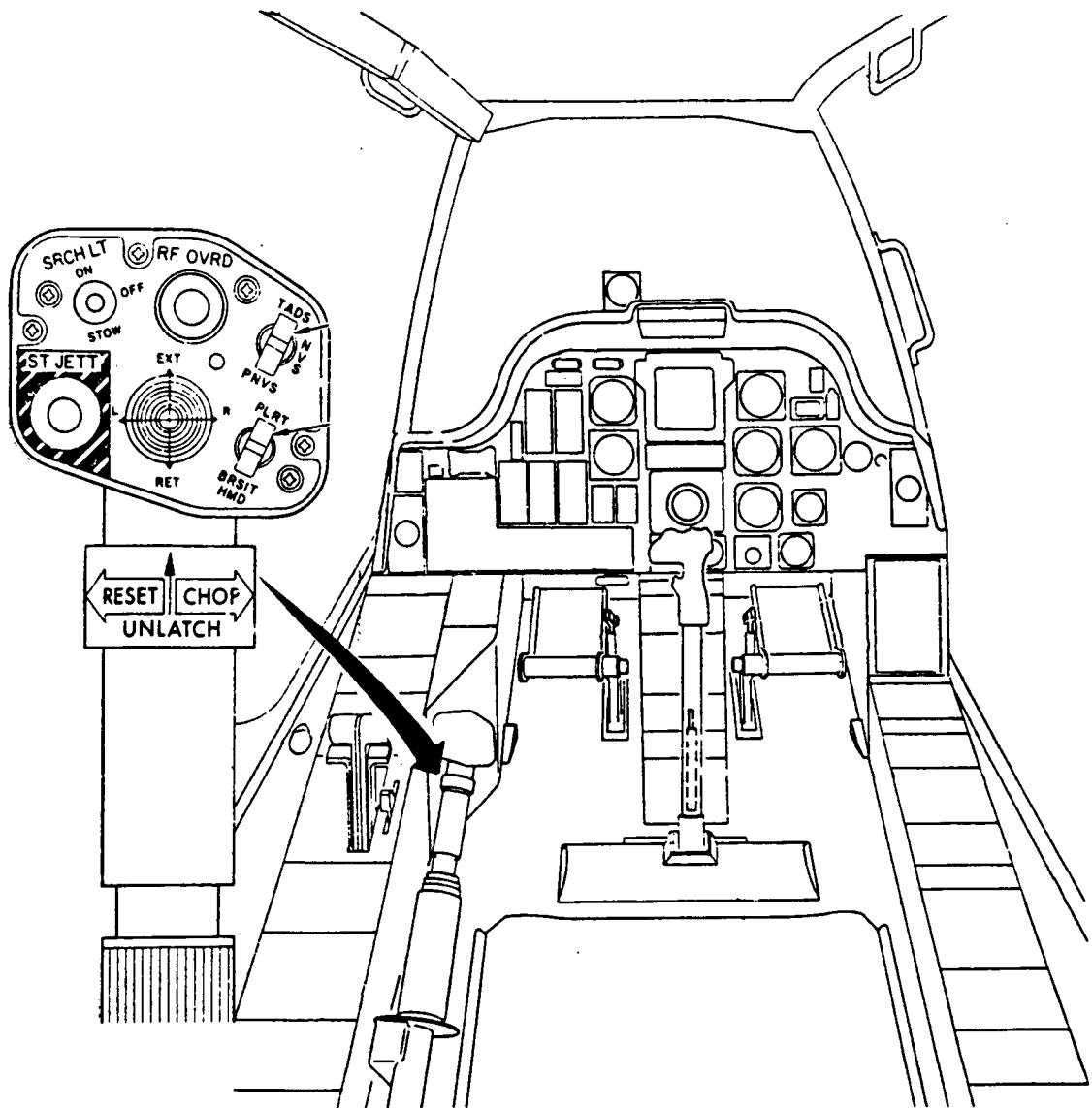


Figure 7

ENGINE CHOP COLLAR

APPENDIX C

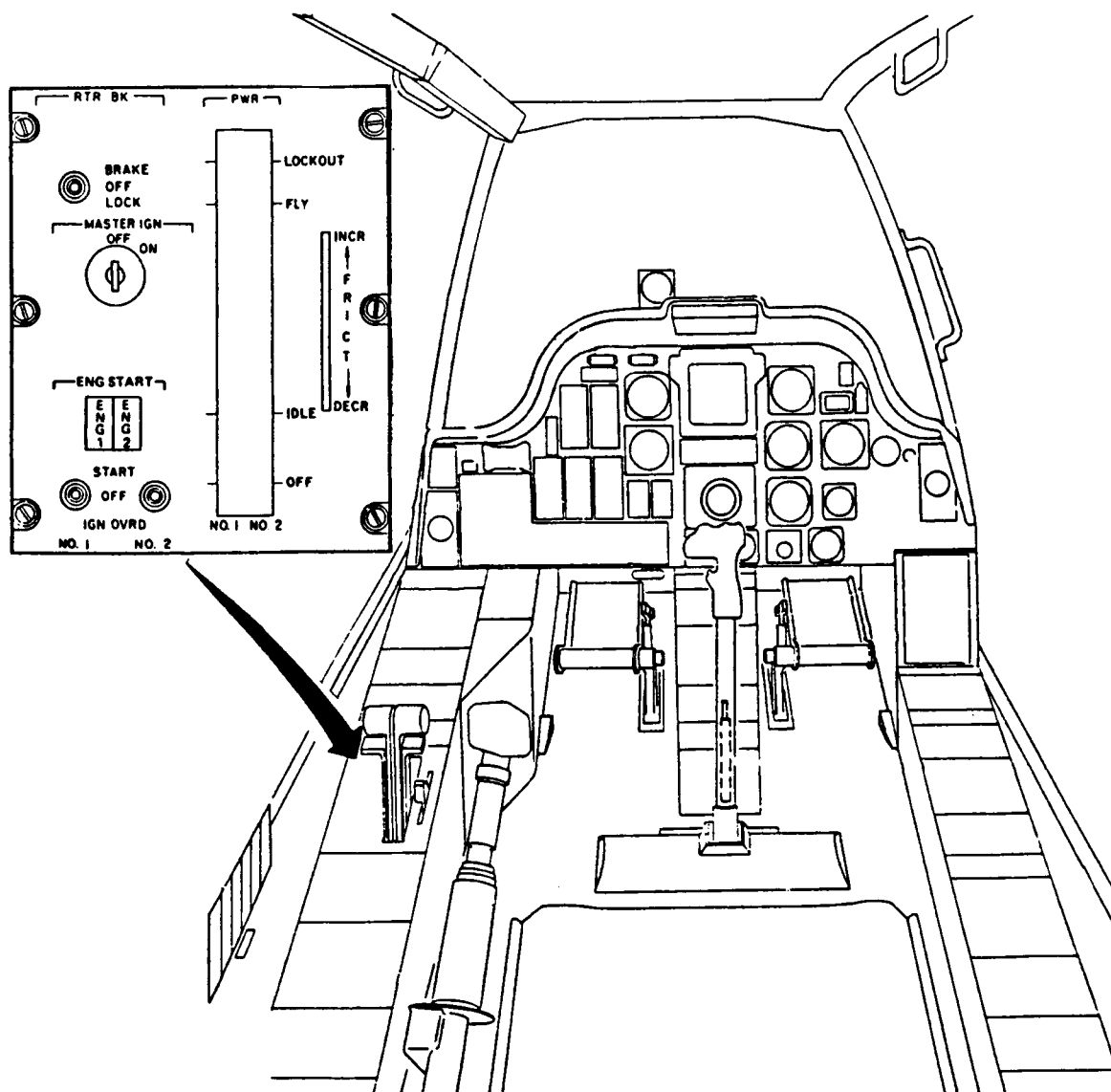


Figure 8
POWER LEVER QUADRANT

APPENDIX C

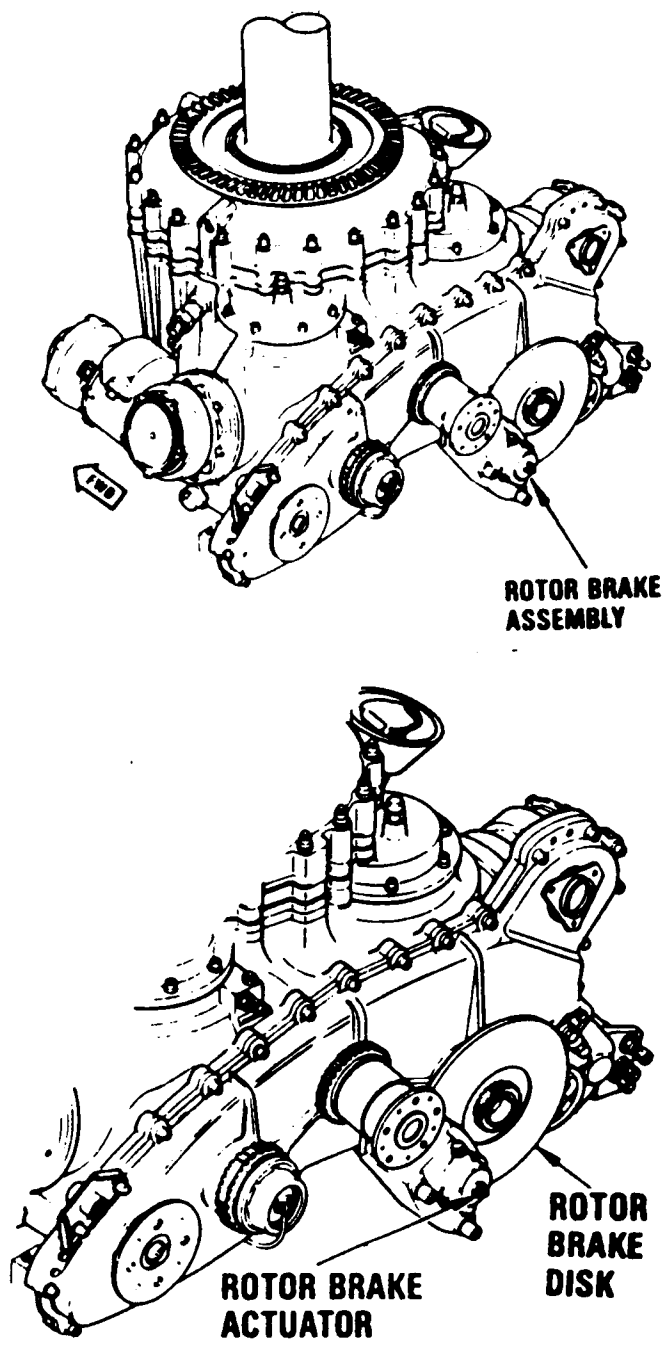


Figure 9
ROTOR BRAKE COMPONENTS AND LOCATION

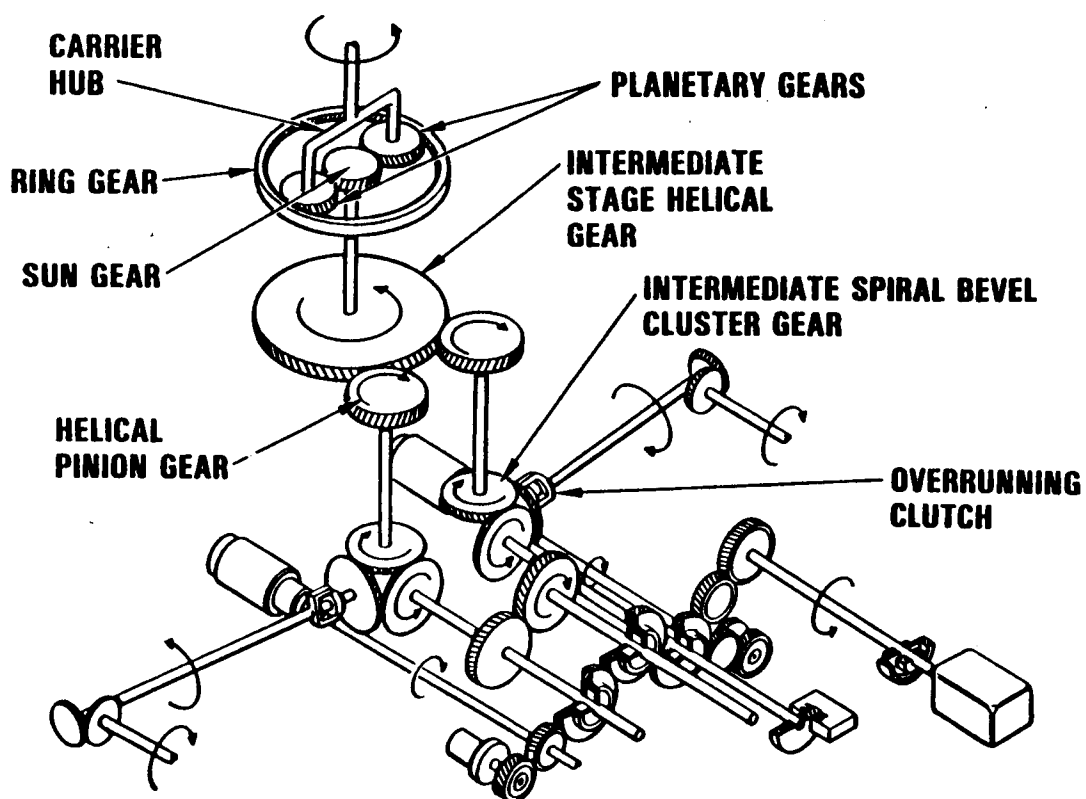


Figure 10
TRANSMISSION COMPONENTS

APPENDIX C

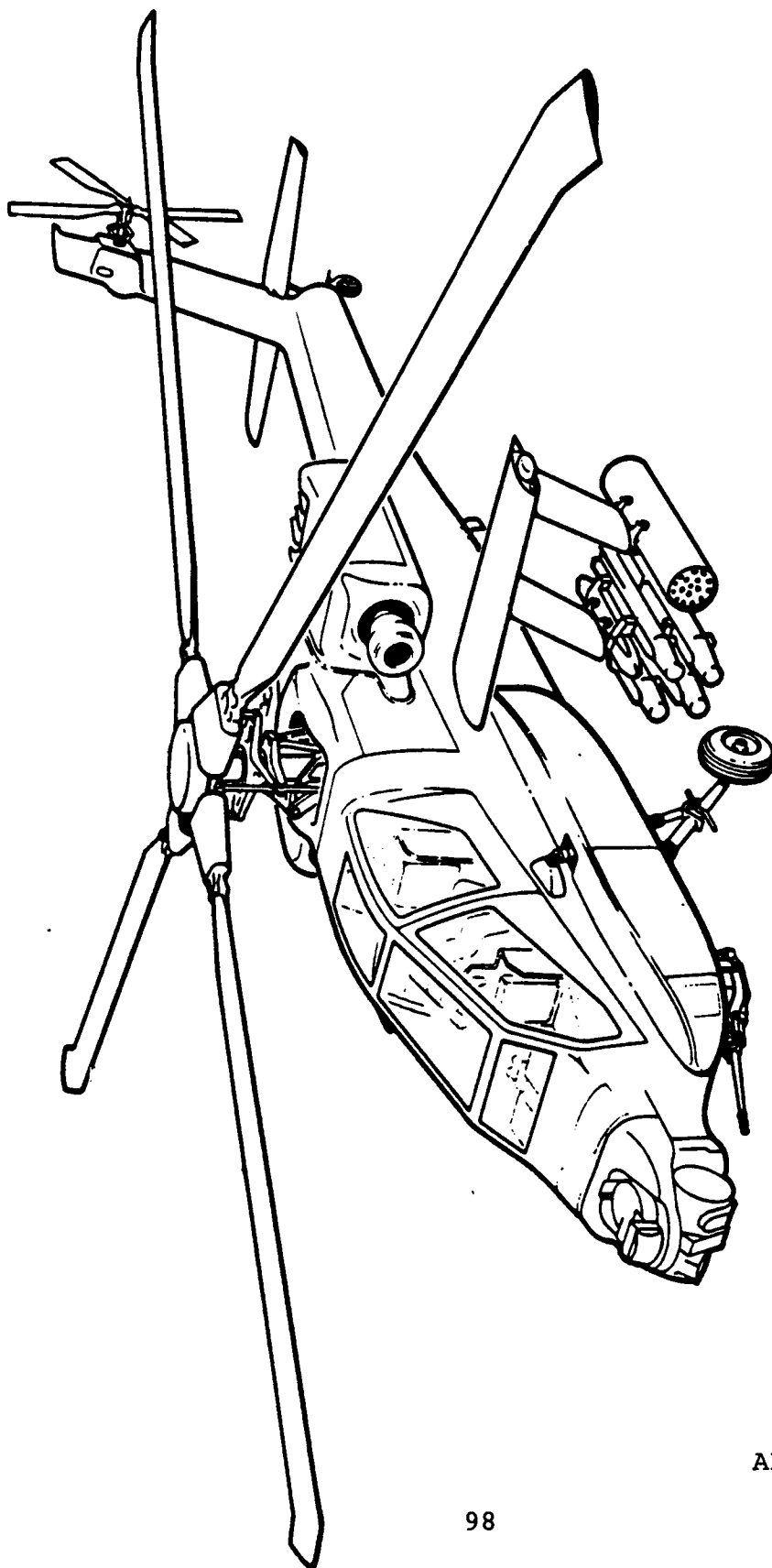


Figure 11
AH-64A WITH MAIN ROTOR HUB FAIRINGS

APPENDIX C

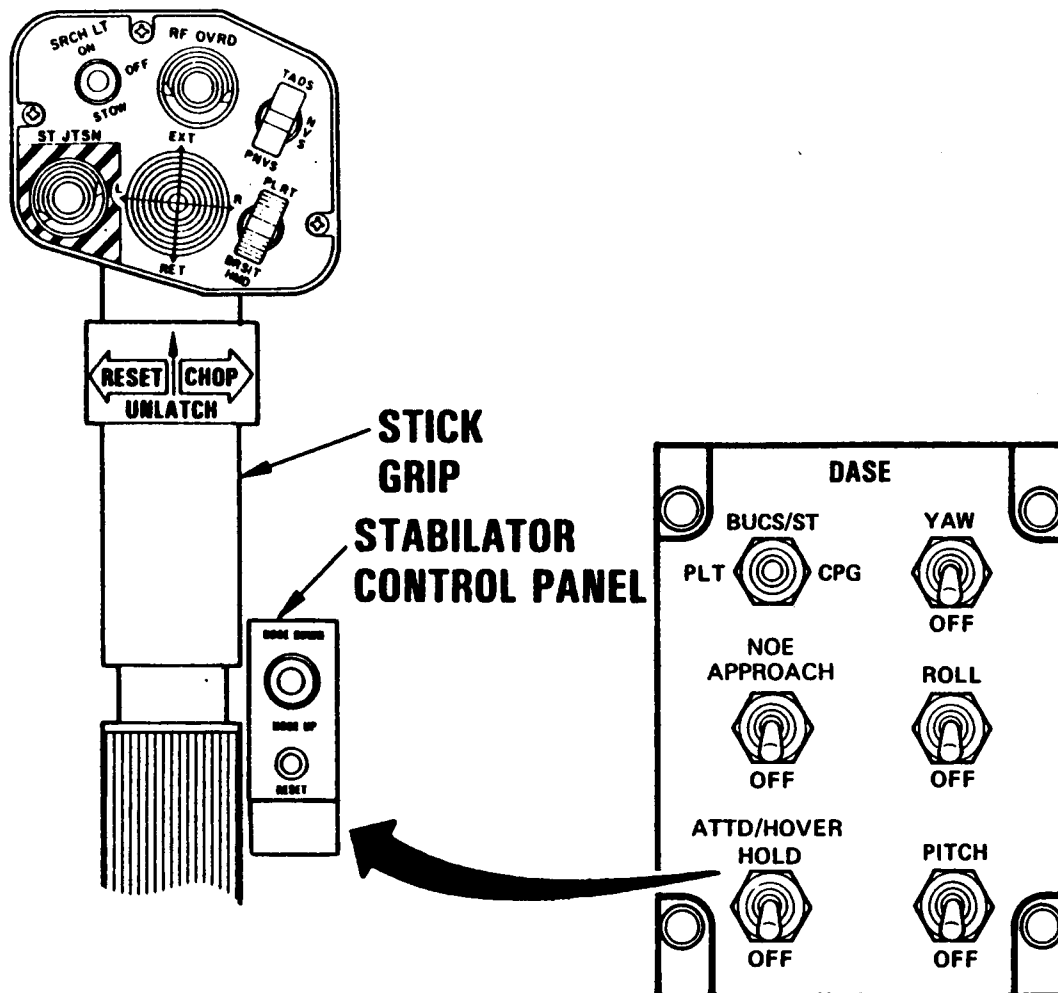


Figure 12

RECOMMENDED ATTD/HOVER HOLD SWITCH LOCATION

APPENDIX C

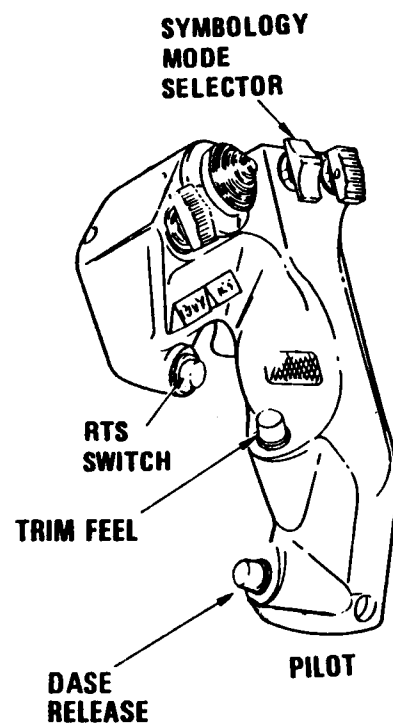
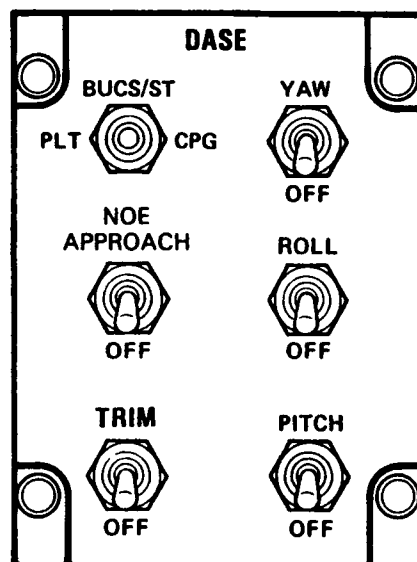


Figure 13
RECOMMENDED TRIM SWITCH LOCATIONS

APPENDIX C

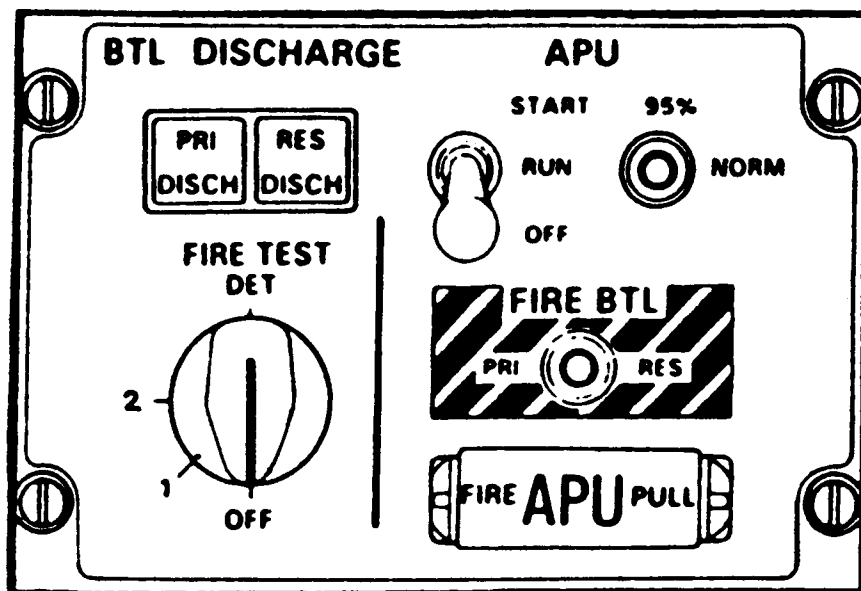


Figure 14
MODIFIED APU PANEL

FLIGHT CONTROL DESCRIPTION

INTRODUCTION

1. The AH-64A helicopter is equipped with a dual hydraulically boosted, irreversible flight control system. The system is controlled by dual conventional flight controls installed in the tandem cockpits. The aircraft incorporates digital automatic stabilization equipment (DASE) which is designed to improve the helicopter stability and control characteristics. The DASE includes a stability augmentation system (SAS), a command augmentation system (CAS), an attitude hold system, and a backup control system (BUCS). The aircraft also incorporates a programmable horizontal stabilator. The flight control system incorporates two significant survivability features. The entire system is designed to be ballistically tolerant to a 7.62 mm round. Additionally, a backup control system can provide an emergency fly by wire capability in the event of jammed or severed flight controls.

HYDRAULIC SYSTEM

GENERAL

2. Aircraft hydraulic power is provided by two separate systems, the Primary and the Utility hydraulic systems. Both systems are pressurized to 3000 PSI by constant pressure variable delivery pumps mounted on the accessory drive case of the main transmission. Each system provides pressure to the four flight control hydraulic servoactuators and incorporates manifolds, heat exchangers, and hand pumps. The function of the manifold is to store, filter, supply, and regulate the flow of hydraulic fluid to each system. Fluid level indicators are provided to allow preflight visual inspection. Additionally, reservoir low-level indicating switches and pressure sensing switches are incorporated to inform the pilot of system problems. The heat exchangers function to automatically cool hydraulic fluid returning to the manifolds from the servoactuators. Hand pumps are installed next to the Ground Servicing Equipment (GSE) panels to provide a method for ground crews to fill the manifold reservoirs. The utility hand pump serves an additional function as a method of charging the utility accumulator fluid pressure.

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PRIMARY SYSTEM

3. The primary hydraulic system, shown schematically in Enclosure 1, Figure 1, provides hydraulic power to the primary side of all four servoactuators (paragraph 5). The SAS, CAS and BUCS provisions of each actuator are provided by the primary system. The system components, except for the servoactuators and the ground service panel, are installed on the left side of the main transmission. The primary system components are depicted in Enclosure 1, Figure 2. The ground service panel is mounted on the right rear side of the transmission deck and can be used to service and bleed the primary system. A complete system checkout can also be performed by connecting a ground power unit to the panel.

UTILITY SYSTEM

4. The utility system is the backup power for the control servoactuators. Therefore, no SAS, CAS or BUCS provisions are provided for the utility side of the servoactuators. The utility system also provides power to the following components: rotor brake, 30 mm turret drive, ammunition handling system, APU start, tail wheel unlock, external stores elevation, and the emergency hydraulic system. The utility system components are depicted in Enclosure 1, Figure 3, and the system is shown schematically in Enclosure 1, Figure 4. Demands on the utility system are much greater than those on the primary system, therefore, the manifold is much larger. The utility manifold reservoir capacity is 1.3 gallons of hydraulic fluid as compared to 1 pint for the primary manifold. The utility hydraulic accumulator, depicted in Enclosure 1, Figure 5, supplies peak flow demands such as engine start, dampens fluid pressure surges, and provides limited hydraulic power for emergency flight control operation. The accumulator is a moveable piston design and is gas operated by nitrogen gas supplied at 1650 PSI from a storage bottle. The emergency hydraulic system uses components of the utility system to store 3000 PSI pressure for emergency use of the flight controls. The hydraulic power available is limited to one 180 degree turn and four full collective stick applications. The emergency system is controlled by the EMERG HYD switch on the pilot's instrument panel and the copilot/gunner's power lever quadrant.

SERVOACTUATORS

5. The hydraulic servoactuators provide the power boost necessary to move the main and tail rotor swashplates

(paragraphs 13 and 14). Longitudinal, lateral, collective, and directional servoactuators are provided. A servoactuator schematic is presented in Enclosure 1, Figure 6. The actuators are powered by hydraulic pressure entering the inlet pressure port through the filter. From the filter the pressure then passes to the SAS or the BUCS solenoid. When the mechanical input is made, the manual servo valve is moved in the controlled direction which allows the hydraulic fluid under pressure to flow into the left or right side of the piston. If the SAS system is energized, the SAS solenoid will be open which will provide hydraulic pressure to the Electro-Hydraulic (EH) valve. The EH valve will be positioned by SAS commands from the DASE computer (paragraph 15). Hydraulic pressure from the valve will be transmitted to either end of the stability augmentation actuator sleeve. Movement of this sleeve will cause the actuator primary piston to move in response to DASE computer commands. Feedback of the SAS actuator position is provided to the DASE computer by the SAS Linear Variable Differential Transducer (LVDT). BUCS (paragraph 19) servoactuator operation is similar to SAS operation except hydraulic pressure is routed to the BUCS plunger, which locks the manual servo valve in the neutral position, and to the EH valve. The BUCS also has 100 percent actuator authority vice 20 percent for the SAS. Only the primary sides of the servoactuators have EH valves that allow the SAS and BUCS to provide inputs. The collective servoactuator does not incorporate SAS provisions.

FLIGHT CONTROLS

CYCLIC CONTROLS

6. The cyclic provides longitudinal and lateral control of the helicopter through push-pull rods, bellcranks, and hydraulic servoactuators to the main rotor. Cyclic movement in any direction tilts the tip path plane of the main rotor by cyclically changing the pitch of each blade as it rotates, thereby causing helicopter motion in the desired direction. Cockpit cyclic controls consist of pilot and copilot/gunner cyclic sticks which are mechanically coupled as depicted in Enclosure 1, Figure 7. The copilot/gunner cyclic stick can be folded down, as also shown in Enclosure 1, Figure 7, while employing the aircraft weapons system. The stick remains functional in this position. Each cyclic stick grip incorporates numerous switches that are shown in Enclosure 1, Figure 8. Fore and aft cyclic inputs are transmitted from the sticks into the longitudinal control linkage. The longitudinal linkage is routed along the right side of the forward fuselage section and consists of pushrods and bellcranks.

APPENDIX D

The motion is transmitted to the hydraulic servoactuator (paragraph 5), then through the mixer assembly (paragraph 9) to the swashplate (paragraph 13). Tilt of the swashplate results in a corresponding tilt of the main rotor. Lateral cyclic movement is transmitted to the main rotor in the same manner, however, the lateral stick linkage is routed along the left side of the fuselage. The entire control linkage system is shown schematically in Enclosure 1, Figure 9. Linear variable differential transducers (LVDT's) are connected to both cyclic sticks. These transducers measure the amount of stick travel and provide inputs to the DASE computer. These inputs are used for pitch and roll SAS and CAS (paragraphs 16 and 17) solutions and for BUCS (paragraph 19) control. The linkages at both crewstations also incorporate longitudinal and lateral shear pin actuated decoupling systems (SPADS). The SPADS are designed to provide the pilot or copilot/gunner the capability to break controls out of a jammed condition. BUCS will engage when a SPADS pin has been sheared.

COLLECTIVE CONTROL

7. The collective control system provides vertical control of the helicopter by simultaneously changing the pitch of the main rotor blades. Pulling up on the collective lever will result in an increase in pitch of the main rotor blades and an upward vertical motion of the helicopter. Collective control inputs are transmitted to the main rotor in a manner similar to the cyclic control as depicted in Enclosure 1, Figure 9. Pushrods, bellcranks and a hydraulic servoactuator are utilized. The collective also incorporates an LVDT to provide position information to the DASE computer. An engine chop device, installed on each collective stick grip, is designed to permit both engines to be reduced to idle without moving the engine power levers. Both collective levers also include friction controls which can be adjusted to prevent the sticks from creeping.

DIRECTIONAL CONTROL

8. The directional control system provides directional control of the helicopter by varying the pitch of the tail rotor blades to move the aircraft about the yaw axis. The directional control system linkage is depicted in Enclosure 1, Figure 10. Pedal inputs are transmitted through a series of control tubes and bellcranks along the left side of the helicopter until aft of the main rotor. The control tubes are then directed toward the aircraft centerline and down

APPENDIX D

the tail boom to the directional servoactuator mounted on the tail rotor gearbox. In order to transmit control inputs to the tail rotor blades, a swashplate assembly is used in a manner similar to the main rotor swashplate assembly. The tail rotor swashplate assembly is depicted in Enclosure 1, Figure 11 and the main rotor swashplate is discussed in paragraph 13. A directional control LVDT provides pedal position data to the DASE computer for the yaw SAS, CAS, BUCS, and heading hold systems. Both sets of pedals are adjustable using the pedal adjust lever and by applying foot pressure to the pedals. The main landing gear brakes are also controlled by the directional pedals and are activated by pressing on the upper portion of the pedals.

MIXER ASSEMBLY

9. Mechanical mixing of inputs to the main rotor is accomplished by the mixer assembly depicted in Enclosure 1, Figure 12. The mixer assembly is mounted on the stationary mast. After the inputs from the collective, lateral, and longitudinal servoactuators are mixed, the output is transmitted to the main rotor stationary swashplate. The mixer is connected to the stationary swashplate by a torque link which is designed to provide longitudinal control and prevent stationary swashplate rotation. Two lateral links also connect the mixer to the stationary swashplate and are designed to provide lateral control. Both the torque link and the lateral links provide vertical control of the swashplate.

TRIM

10. The cyclic stick and pedal controls each incorporate a trim feel system consisting of a double acting spring bungee and a magnetic brake. The magnetic brakes and spring assemblies are connected to the control linkage below the pilot's station. Movement of the cyclic or directional controls with the trim system on will cause the spring assemblies to compress and provide feel to the controls. The trim system is designed to return the control to the trimmed position once the control pressure is released. Trim is accomplished by using either the TRIM REL button on the copilot/gunners cyclic grip or by the TRIM REL switch on the pilot's cyclic grip. Pressing the button or moving the switch up releases the magnetic brake which will allow the spring assemblies to move to the new control position. Releasing the button or switch will reengage the magnetic brakes and provide trim feel at the new control position. The trim system is designed to be operable throughout the full cyclic and pedal

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control envelope. The pilot may disengage the trim system entirely by placing the TRIM REL switch in the full down position. The trim system must be engaged for the attitude hold (paragraph 18) capability of the DASE to be functional.

MAIN ROTOR SYSTEM

MAIN ROTOR HEAD

11. The main rotor head, depicted in Enclosure 1, Figure 13, is a fully articulated system containing four blades. Each of the blades are allowed to flap, feather, lead, and lag independently. The hub assembly is constructed of steel and aluminum and is designed to support and drive the main rotor blades. The hub is driven by the main rotor drive shaft, however, the hub rotates about the static mast as depicted in Enclosure 1, Figure 14. The hub incorporates grease lubricated and sealed roller bearings that are designed to transfer the hub loads to the static mast. The hub assembly is secured to the static mast by a lock nut and lock ring. A plunger assembly is provided to indicate proper installation of the locking nut end ring. Centrifugal loads from the main rotor blades to the hub are transmitted by the strap assemblies which also provide the blades with a flapping and feathering capability. The strap assemblies pass through the pitch housings, which are depicted in Enclosure 1, Figure 15. Swashplate (paragraph 13) movements are transmitted to the pitch housings by pitch links. Elastomeric feathering bearings installed on the pitch housings allow for both feathering and flapping motions. Mechanical droop stops are also incorporated and are designed to limit blade droop to seven degrees. Lead and lag blade motions are accomplished through lead/lag links attached to the outboard end of the pitch housings. The lead/lag motions are controlled by the main rotor damper assemblies consisting of two side plates bonded together by an elastomer material.

MAIN ROTOR BLADES

12. The aircraft incorporates four main rotor blades which are constructed of steel, fiberglass, titanium and nomex honeycomb. The main rotor airfoil is an HH-02 with a transition to a NACA 64A006 at the tip. Main rotor blade construction is depicted in Enclosure 1, Figure 16. The primary load carrying portion of the blade is the four cell stainless steel structural box. Each cell has fiberglass filament tubes bonded to the inner surface which are designed to retard crack propagation. The aft portion of the blade

APPENDIX D

is constructed of nomex honeycomb filler with a layered fiberglass skin. The trailing edge strip, which runs the full span of the blade, is constructed of stainless steel. Each blade is constant chord with the final seven percent of the blade span swept aft 20 degrees. Removable leading edge tip caps, tip weights, and leading edge heater blankets for deicing are incorporated. Titanium blade retention fittings are provided for attachment to the lead/lag links. The blades may be manually folded by removing one of the blades retaining bolts which can be accomplished without the use of tools.

SWASHPLATE ASSEMBLY

13. The swashplate assembly consists of a rotating and a stationary swashplate as depicted in Enclosure 1, Figure 19. The assembly is mounted on a teflon spherical slider bearing which allows the swashplate to tilt in any direction responding to cyclic control inputs. Vertical motion of the swashplate assembly is also provided for collective control inputs. The rotating swashplate, mounted above the stationary swashplate, is supported by a double row of ball bearings. The rotating swashplate is connected to the main rotor hub by the two scissor assemblies. Control inputs are received at the stationary swashplate from the hydraulic servoactuators (paragraph 5), through the mixer assembly (paragraph 9). Tilt or vertical motion of the stationary swashplate is then transmitted to the rotating swashplate. Four pitch links connect the rotating swashplate with the blade pitch horn assemblies (paragraph 11) to transmit control motion inputs to the main rotor blades.

TAIL ROTOR SYSTEM

14. The tail rotor system, depicted in Enclosure 1, Figure 17, is designed to provide anti-torque action and directional control for the helicopter. The tail rotor system is a dual, semi-rigid, teetering design. The tail rotor assembly is attached to, and driven by, the tail rotor gearbox output shaft which passes through the static mast. The four tail rotor blades are mounted to the tail rotor hub. The hub assembly is supported by ball bearing sets in the static mast. The blades are mounted to the hub in a manner such that the angle formed by the intersection of the closest blades is 55 degrees. Blade pitch change is accomplished by pitch links which connect the rotating swashplate and the blade pitch horns. Blade pitch movement is made about the pitch change bearings. The tail rotor blades are constructed

of a stainless steel forward spar plus aluminum center and aft spars as shown in Enclosure 1, Figure 18. Unidirectional fiberglass liners, bonded to the inner surface of the two forward spars, are designed to retard crack propagation. The tail rotor blades also incorporate electrical deicing blankets.

DIGITAL AUTOMATIC STABILIZATION SYSTEM

GENERAL

15. The digital automatic stabilization equipment (DASE) is designed to increase the stability of the helicopter as a weapons delivery platform. The maneuverability of the aircraft is also enhanced by the DASE. Subsystems of the DASE include the following: stability augmentation system (SAS), command augmentation system (CAS), attitude hold, back up control system (BUCS), and the stabilator system. The heart of the automatic stabilization equipment is the DASE computer which sends control inputs to the primary side of the longitudinal, lateral, and directional flight control servoactuators. The command signals from the DASE computer are based upon inputs to the computer from various sources. The primary source is the heading and attitude reference set (HARS), which is a self-contained inertial platform that provides pitch, roll, and yaw velocity and acceleration information. Another source to the DASE computer is the air data sensor system (ADSS), which provides true airspeed and sideslip information. The DASE computer also relies on position transducer (LVDT) information from the flight controls and the hydraulic servoactuators. The DASE computer is powered by both 28V DC and 115V AC electrical power. The automatic stabilization equipment is controlled by the ASE panel depicted in Enclosure 1, Figure 20. The panel includes five magnetic switches and a BUC/TST switch whose functions will be discussed in paragraph 19.

STABILITY AUGMENTATION SYSTEM

16. The SAS is designed to reduce pilot workload by providing rate damping. The rate damping is utilized in the pitch, roll, and yaw axes to reduce airframe movement caused by turbulence or weapon recoil. The SAS is also designed to provide turn coordination above 60 KTAS and attitude hold (paragraph 18), if engaged. The SAS is controlled by the pilot's ASE control panel. Three single throw magnetic switches labeled PITCH, ROLL, and YAW allow for SAS single channel engagement or disengagement. A DASE release button

APPENDIX D

at the base of each cyclic grip will disengage all three channels simultaneously. SAS actuator inputs in the hydraulic servoactuators are controlled by the DASE computer. The DASE computer is programmed to automatically disengage any channel that deviates from a built-in monitor by more than 35%. If that occurs, a SAS caution light will illuminate and the associated switch or switches on the ASE panel will automatically move to the off position. The SAS actuators are limited to ± 10 percent movement in each axis. The SAS operates independently of the flight control linkage and therefore, its damping action is not fed back through the flight controls.

COMMAND AUGMENTATION SYSTEM

17. The command augmentation system is designed to prevent the aircraft from being sluggish in response to the control inputs required for maneuvering. The CAS enables the SAS to recognize control inputs from the pilot or copilot/gunner to ensure that these inputs will not be damped out. When a control input is initiated the LVDT sends a signal to the DASE computer proportional to the control movement. The computer is designed to recognize this as a legitimate control input and control the SAS damping action. The CAS is gained scheduled above 50 KTAS. This gain programming with airspeed is utilized when less aircraft response per cyclic input is desired at higher airspeeds. The gain scheduling will allow limited SAS action. A multipurpose squat switch on the left main landing gear disables CAS during ground taxi to avoid overcontrol. SAS and CAS system operation is shown schematically in Enclosure 1, Figure 21.

ATTITUDE HOLD

18. The attitude hold system is designed to provide a limited hands-off flying capability in cruise flight. The system is provided in only the pitch and roll axes and requires that pitch and roll SAS be engaged and functional. The attitude hold has limited authority which is half the $\pm 10\%$ damping limits of the SAS. In addition to SAS engagement, two other requirements for proper attitude hold function are that the trim system is on and the airspeed of the helicopter must be 60 KTAS or greater. The DASE computer processes attitude information from the HARS. Command signals are sent to the SAS actuators to change main and tail rotor pitch angles to maintain the desired aircraft attitude. The attitude hold function is monitored by the DASE computer which can automatically disengage the hold if a malfunction

is detected. Attitude hold and CAS are not compatible functions, however, attitude hold may be overridden by control movement at any time. When the cyclic stick is moved, the ATTD/HOVER HOLD switch will remain on but attitude hold operations will stop until the cyclic is repositioned and the TRIM REL button or switch is activated. The hover augmentation system (HAS) is a subsystem of attitude hold and is designed to reduce pilot workload during hover and flight below 15 knots ground speed. HAS is also controlled by the ATTD/HOVER HOLD switch on the ASE panel. The pitch, roll, and yaw SAS must also be engaged for proper HAS function. The HAS will only operate below 50 KTAS and 15 knots ground speed and the system relies on lateral and longitudinal accelerations signals to the DASE computer from the HARS. Heading hold is provided automatically by the HAS and uses heading references from the HARS and control inputs by the yaw SAS. It is not possible to pass directly from attitude hold to hover augmentation by decelerating through 50 KTAS. The ATTD/HOVER HOLD switch will release and must be re-engaged once ground speed is less than 15 knots. An attitude hold and hover augmentation block diagram is presented in Enclosure 1, Figure 22.

BACK-UP CONTROL SYSTEM

19. The back-up control system (BUCS) is designed to provide emergency fly-by-wire control of the helicopter in the event of jammed or severed mechanical flight controls. Direct electrical control of the flight control hydraulic servo-actuators is accomplished by utilizing control position LVDT's. The LVDT's send control position information to the DASE computer which provides output signals to the servo-actuators. The SAS actuator in the servos, which normally has limited authority, is used to transmit the BUCS control inputs to the rotor systems. With BUCS engaged, the SAS actuator has 100% authority over the hydraulic servoactuator, however, normal SAS damping action is not provided. The SAS actuators are only installed in the primary side of the servoactuators; therefore, if the primary hydraulic system fails, no BUCS capability is provided. BUCS is automatically placed in a standby mode whenever electrical power is applied to the DASE computer, with the exception of the collective channel which only goes to standby when any of the SAS channels are engaged. BUCS is designed to activate whenever the DASE computer senses that a mechanical control linkage has been severed. The computer determines the severed condition by sensing a mistrack between the pilot and the copilot/gunner's flight controls. BUCS is also designed to engage in the event of a flight control jam. If a control

becomes jammed, the pilot can forcibly shear the shear pin actuated decoupler (SPAD) installed on each control which will signal BUCS to engage. A BUCS engage switch is installed on the copilot/gunner's collective which is designed to allow the copilot to take control of the BUCS if the pilot's system does not properly control the servoactuators due to damage. A BUCS/TST switch on the DASE control panel, depicted in Enclosure 1, Figure 20, permits the pilot to perform a go/no-go check of the BUCS system. The self-test can only be accomplished when the aircraft is on the ground (landing gear squat switch depressed) and the rotor brake is locked.

STABILATOR SYSTEM

20. The stabilator system is designed to enhance the handling qualities and to improve the forward field of view of the helicopter through use of a variable angle of incidence horizontal stabilator. The stabilator is attached to the lower aft side of the vertical stabilizer as depicted in Enclosure 1, Figure 23. The stabilator is designed to be positioned automatically by two stabilator control units (SCU's). In addition to the automatic mode, the stabilator system also incorporates a manual and an NOE/approach mode. When operating in the automatic mode the SCU's receive input signals of airspeed, pitch rate and collective stick position. The SCU's determine the optimum stabilator position based on the input signals and send position commands to the stabilator actuator. The stabilator actuator is driven by two independent DC motors. The stabilator range of motion in the automatic mode is 5 degrees trailing edge up to 25 degrees trailing edge down. Stabilator system operation is shown schematically in Enclosure 1, Figure 24. Stabilator system operation requires both AC and DC power for all modes. The horizontal stabilator can also be controlled manually by the pilot. In manual mode the stabilator range of motion is 10 degrees trailing edge up to 45 degrees trailing edge down. Stabilator manual control is accomplished by use of the stabilator manual control switch, installed on each collective stick, and depicted in Enclosure 1, Figure 23. The manual mode of operation is designed for use below 80 KTAS. When accelerating through 80 KTAS an automatic switchover to the normal automatic mode will occur. If the automatic switchover malfunctions, the pilot will receive a caution panel warning and an audio warning tone. The pilot can control the switchover function through use of the auto operation/audio tone reset button which is also installed on each collective stick. The final mode of stabilator operation is the NOE/approach mode. This mode is designed

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to position the stabilator to 25 degrees trailing edge down whenever the aircraft's airspeed is below 80 KTAS. The NOE/approach mode is controlled by the NOE/APRCH switch located on the DASE panel as depicted in Enclosure 1, Figure 23. The switch may be actuated at airspeeds above 80 KTAS and upon decelerating through 80 KTAS the stabilator will program to the proper position. During accelerations through 80 KTAS, an automatic switchover to the normal automatic mode will occur. A stabilator position indicator is incorporated on both instrument panels to provide both pilots with a visual indication of the stabilator angle of incidence.

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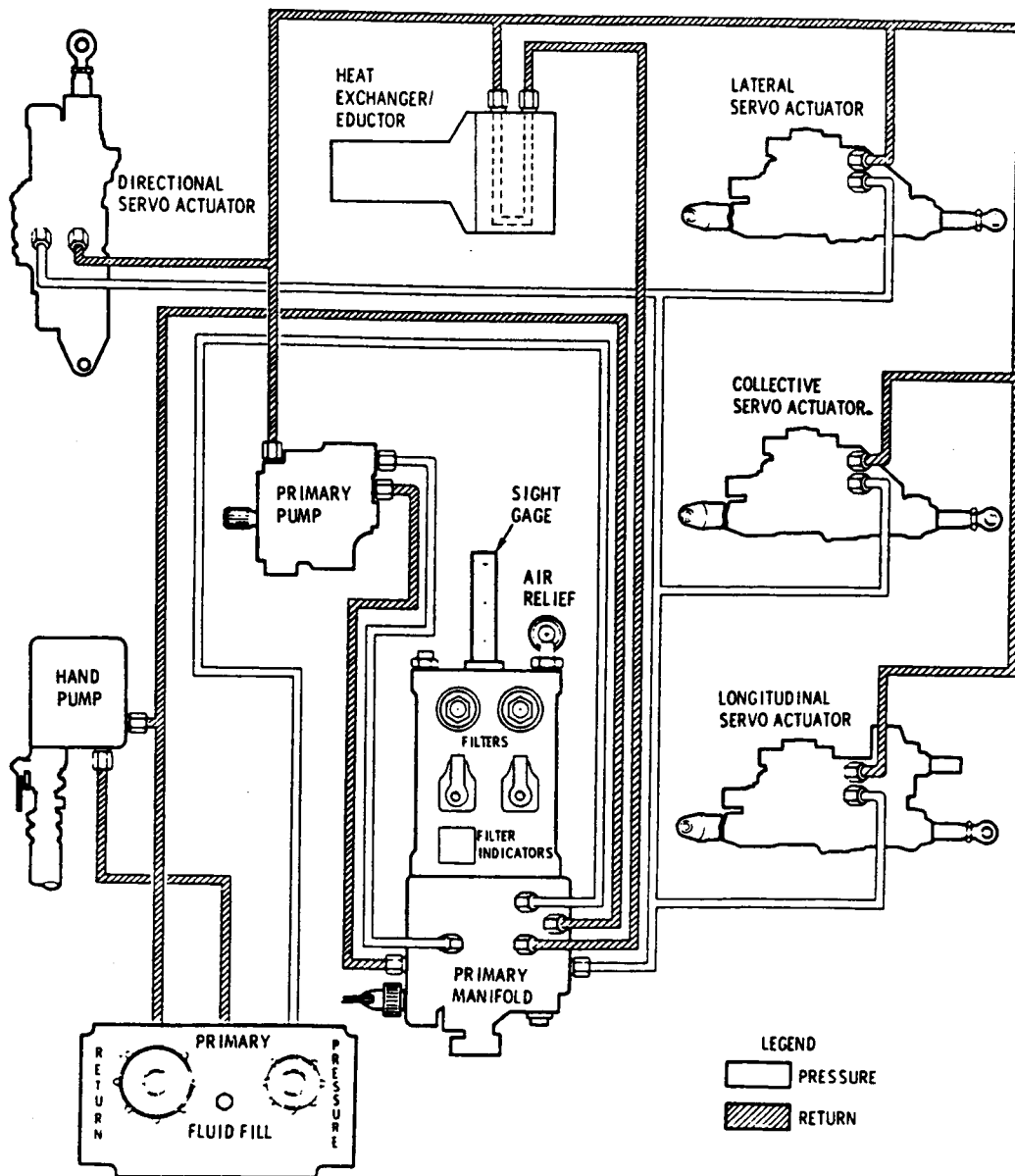


Figure 1
PRIMARY HYDRAULIC SYSTEM

ENCLOSURE 1
APPENDIX D

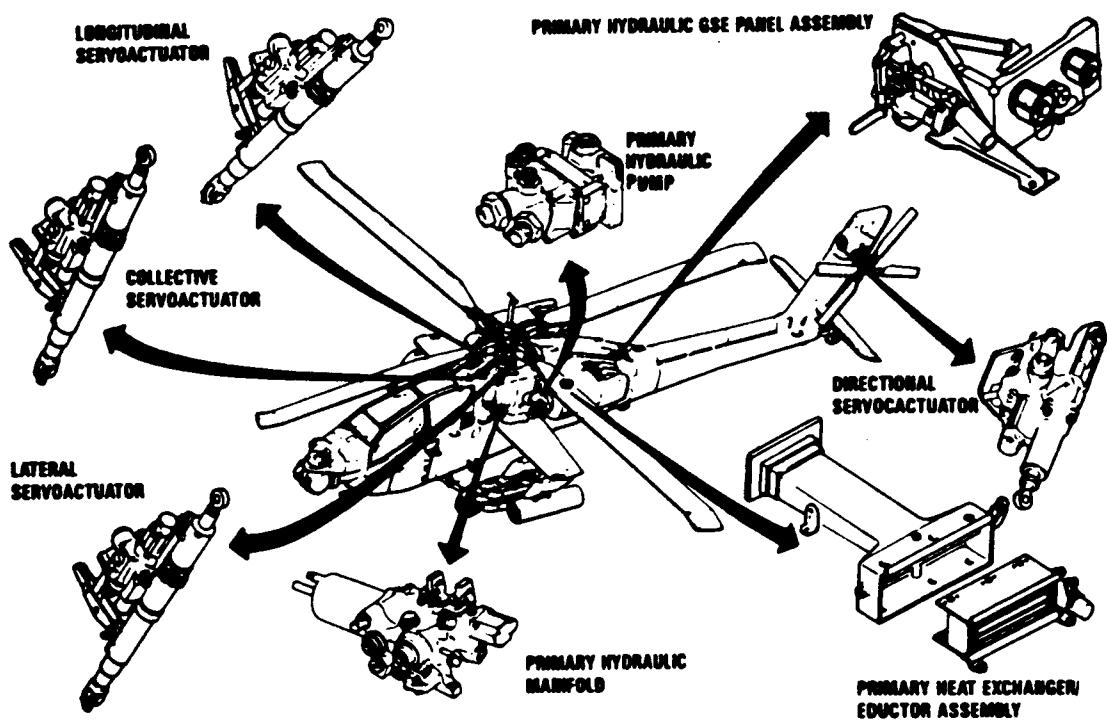


Figure 2
PRIMARY HYDRAULIC SYSTEM MAJOR COMPONENTS

ENCLOSURE 1
APPENDIX D

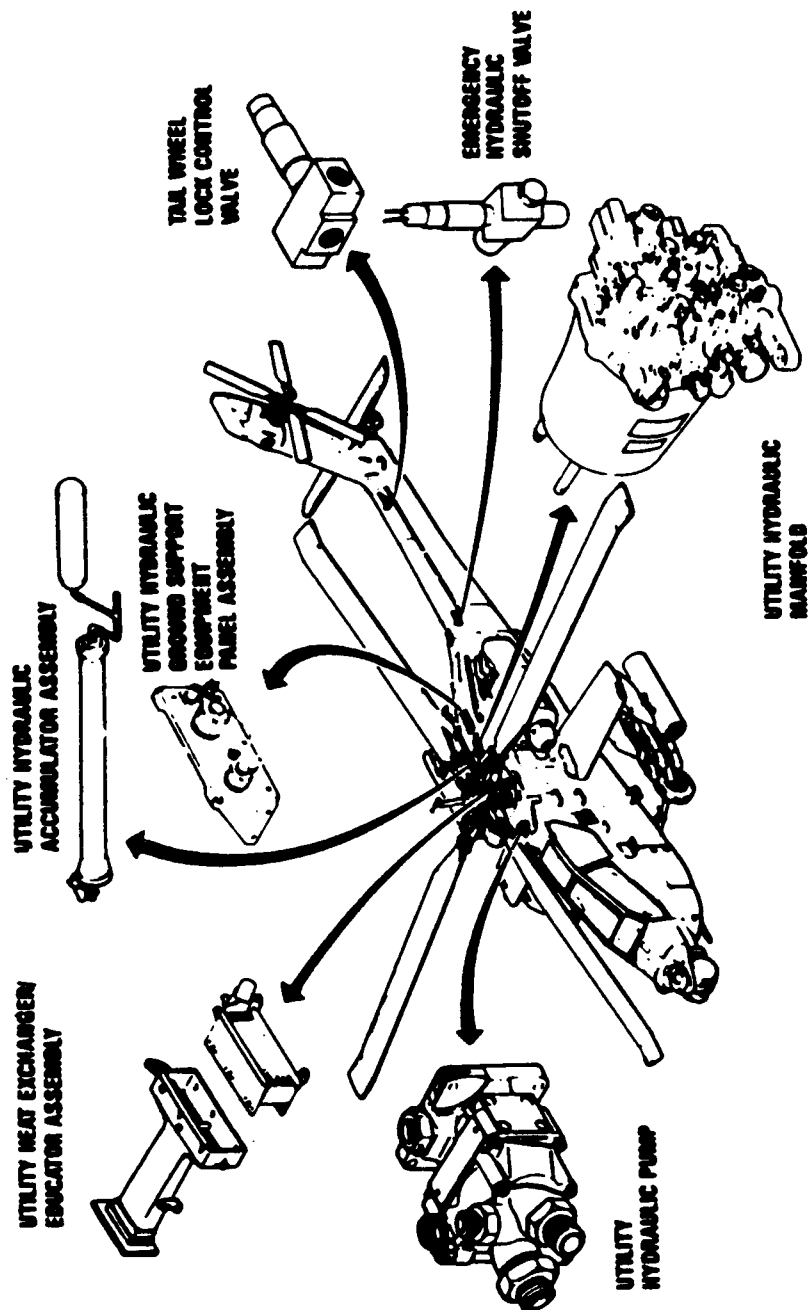


Figure 3
UTILITY HYDRAULIC SYSTEM MAJOR COMPONENTS

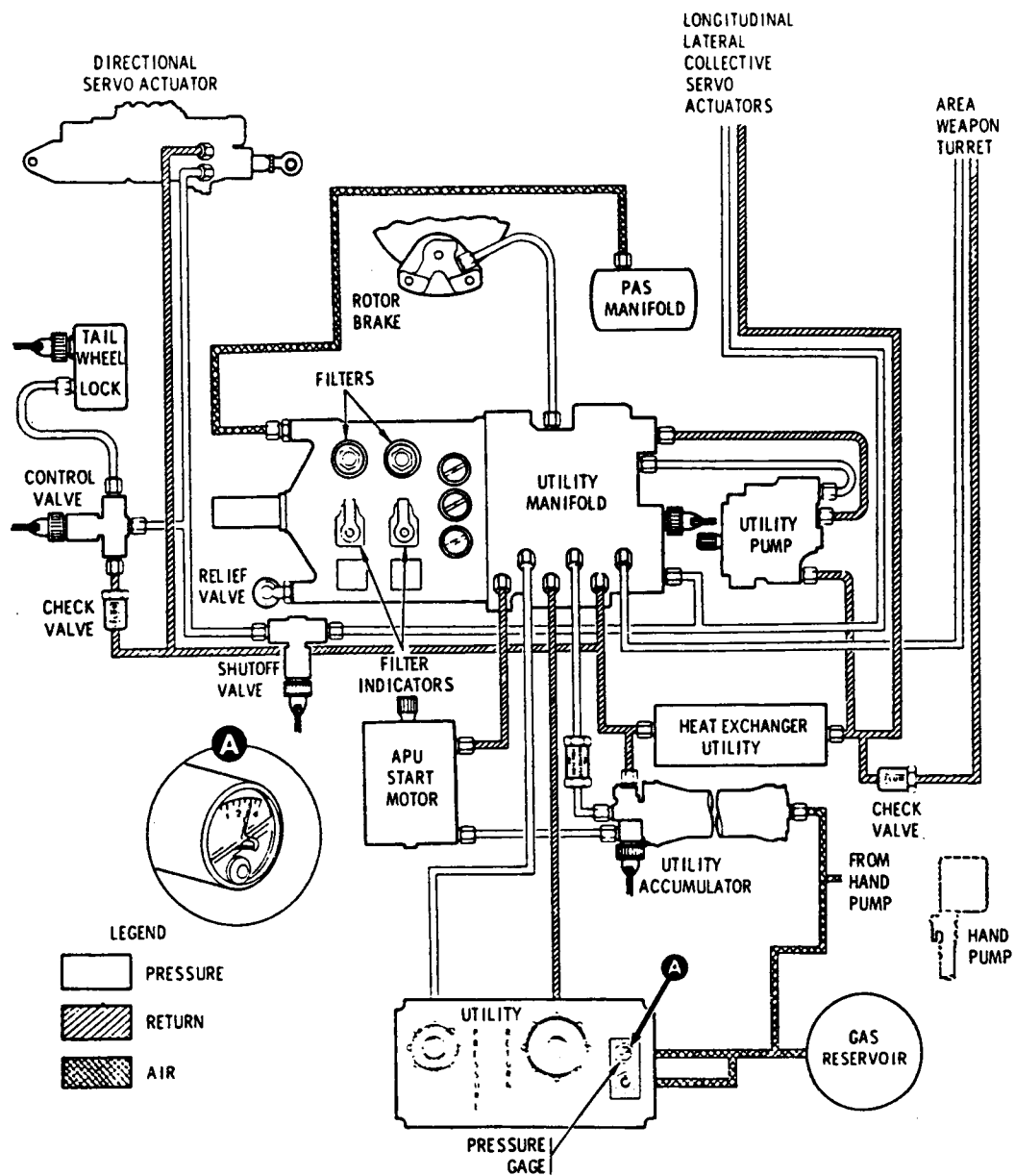


Figure 4
UTILITY HYDRAULIC SYSTEM

ENCLOSURE 1
APPENDIX D

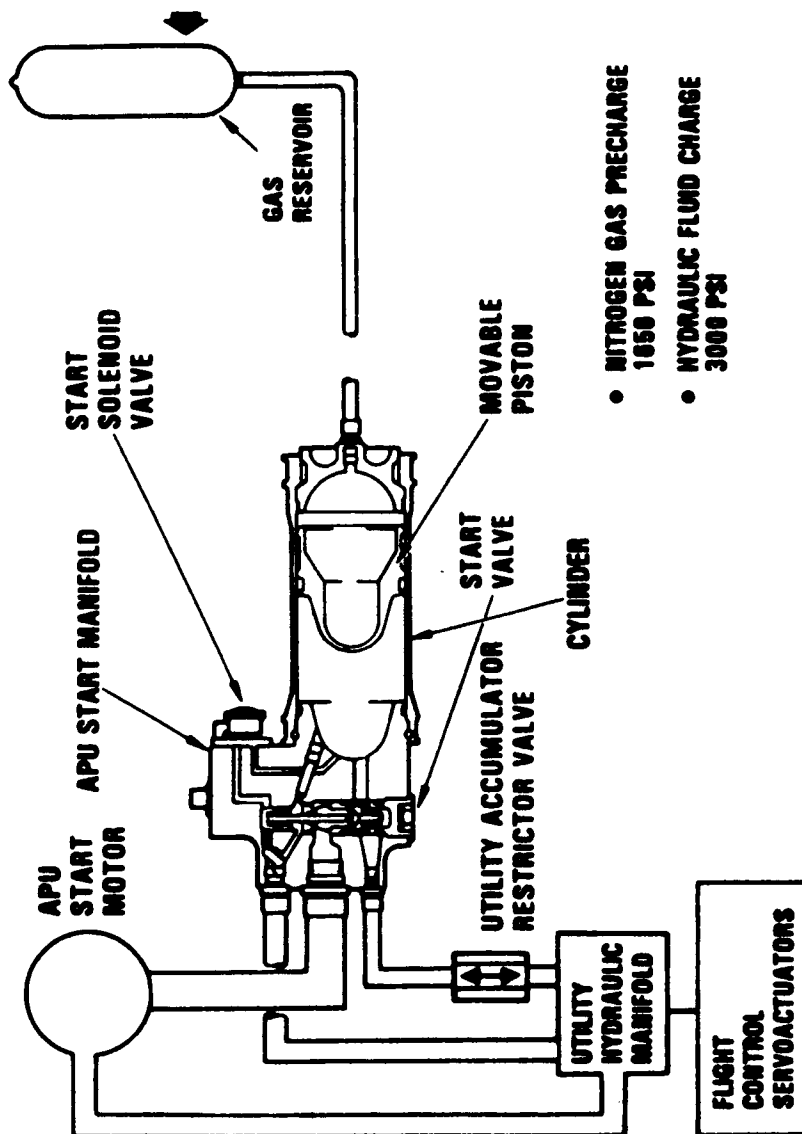


Figure 5
UTILITY HYDRAULIC ACCUMULATOR

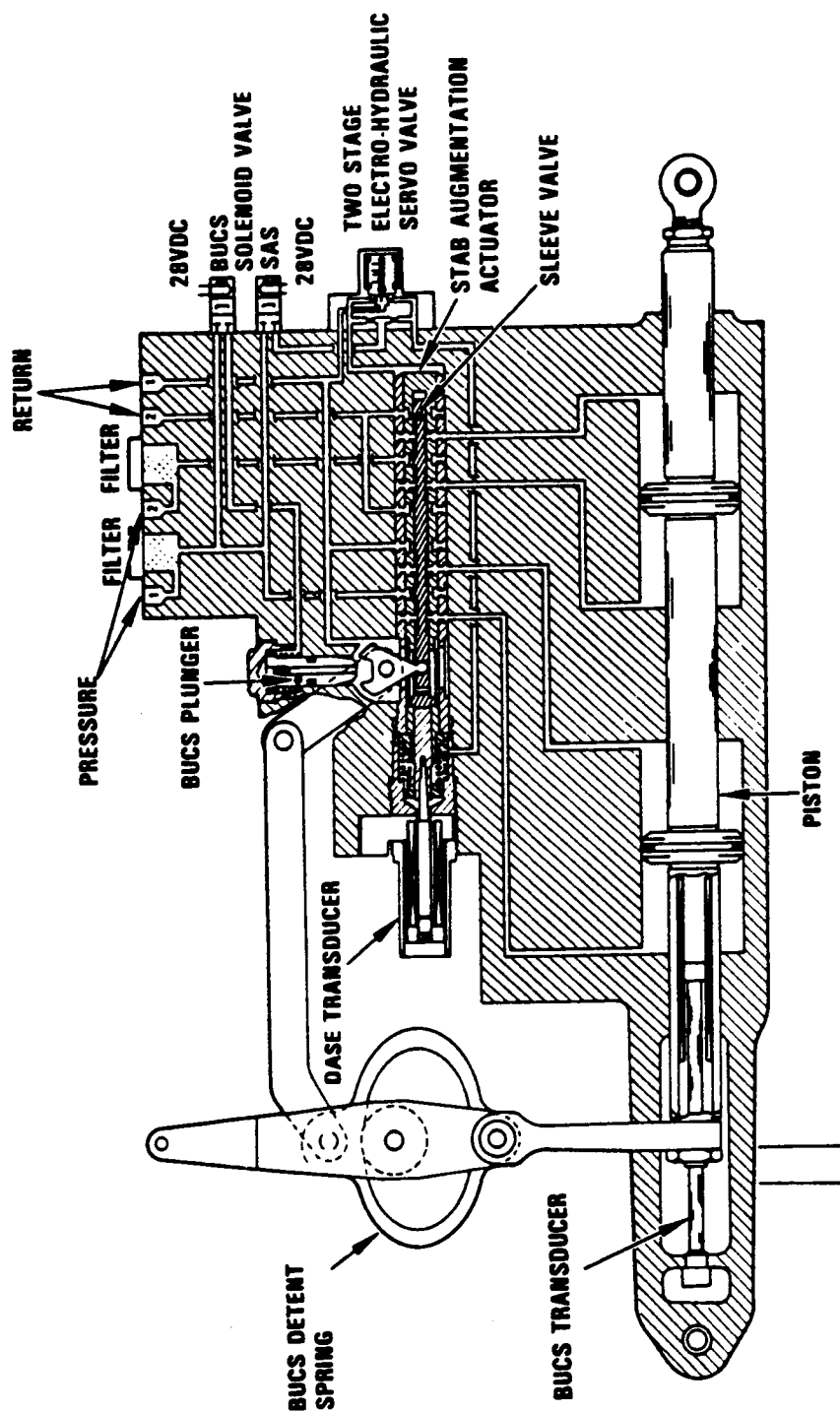


Figure 6
HYDRAULIC SERVOACTUATOR

ENCLOSURE 1
APPENDIX D

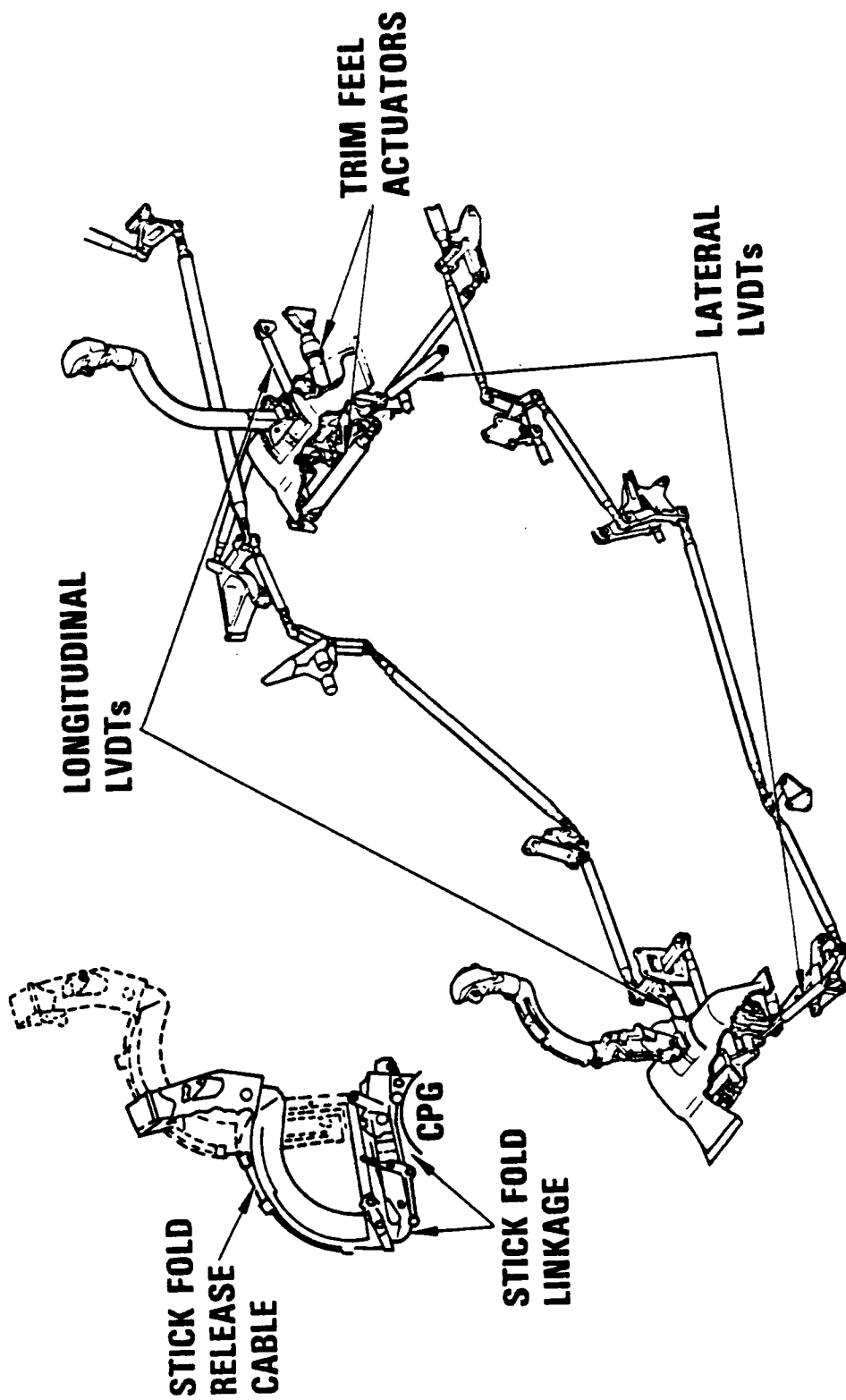


Figure 7
CYCLIC CONTROLS

ENCLOSURE 1
APPENDIX D

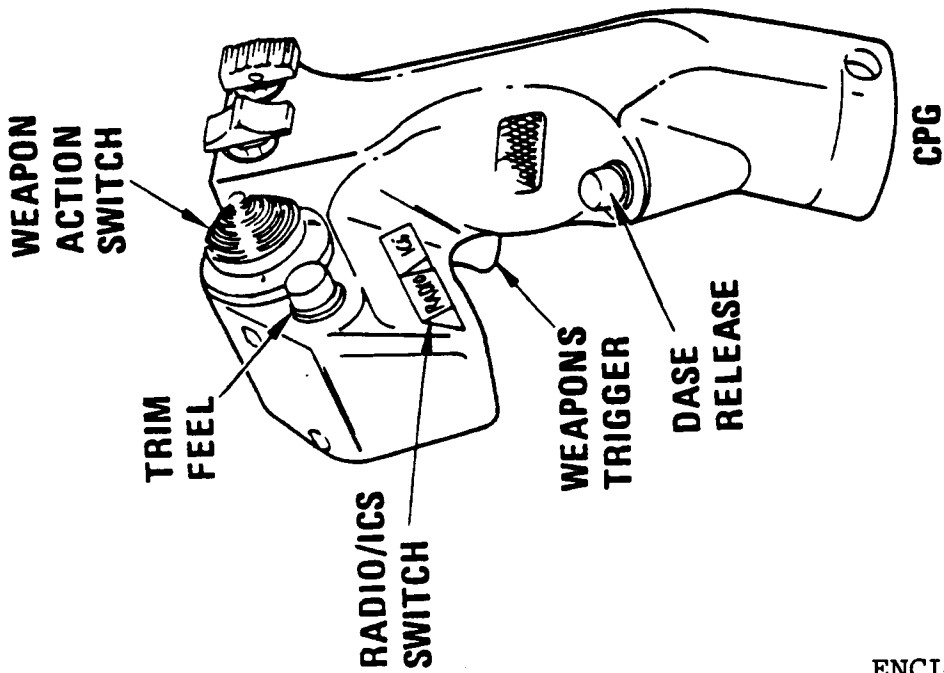
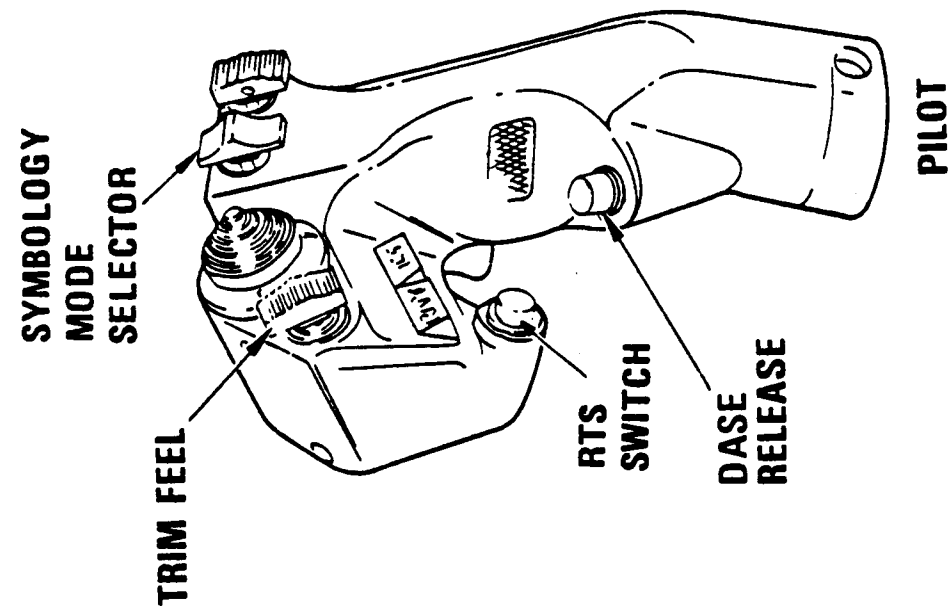


Figure 8
CYCLIC STICK GRIP

ENCLOSURE 1
APPENDIX D

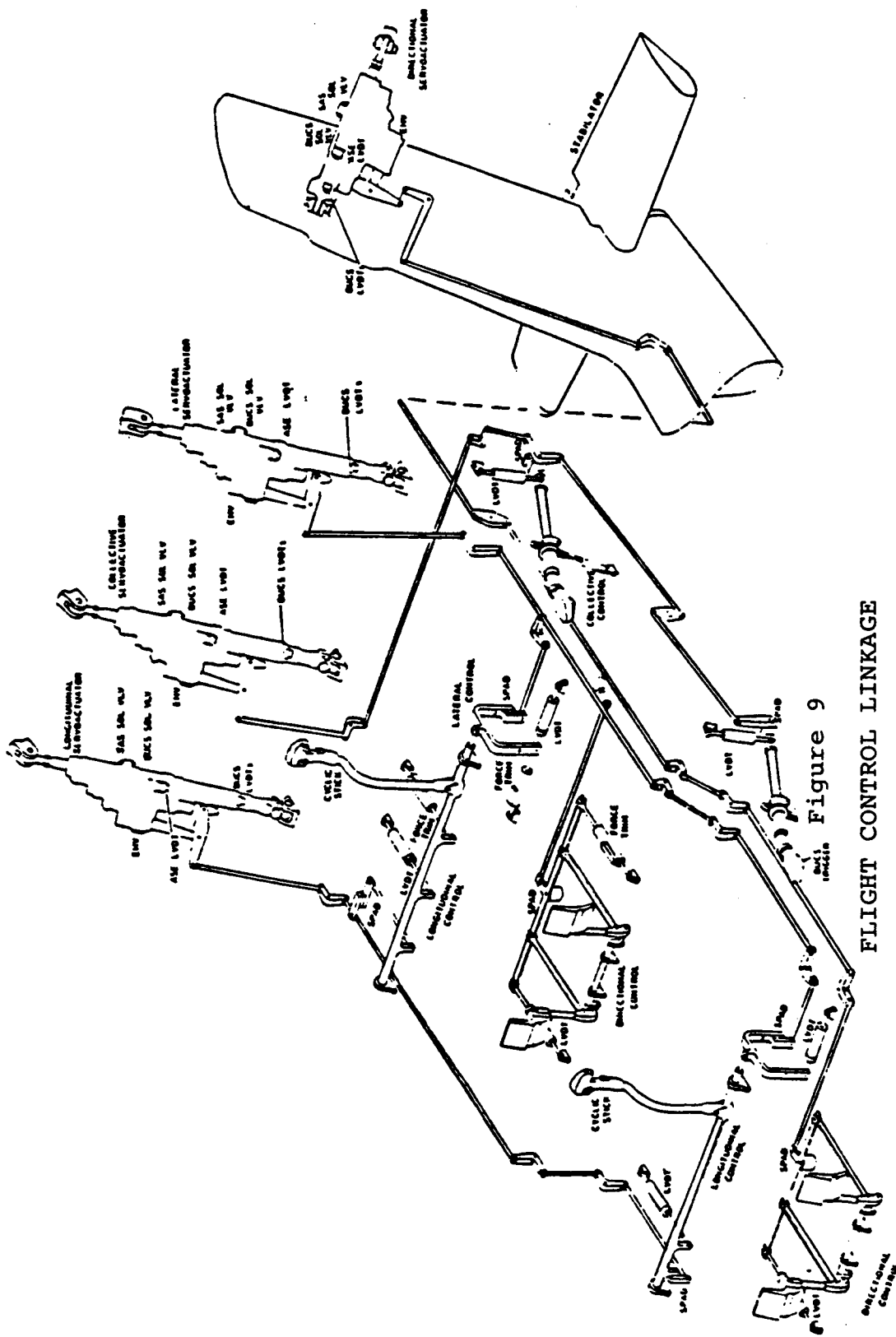


Figure 9

FLIGHT CONTROL LINKAGE

ENCLOSURE 1
APPENDIX D

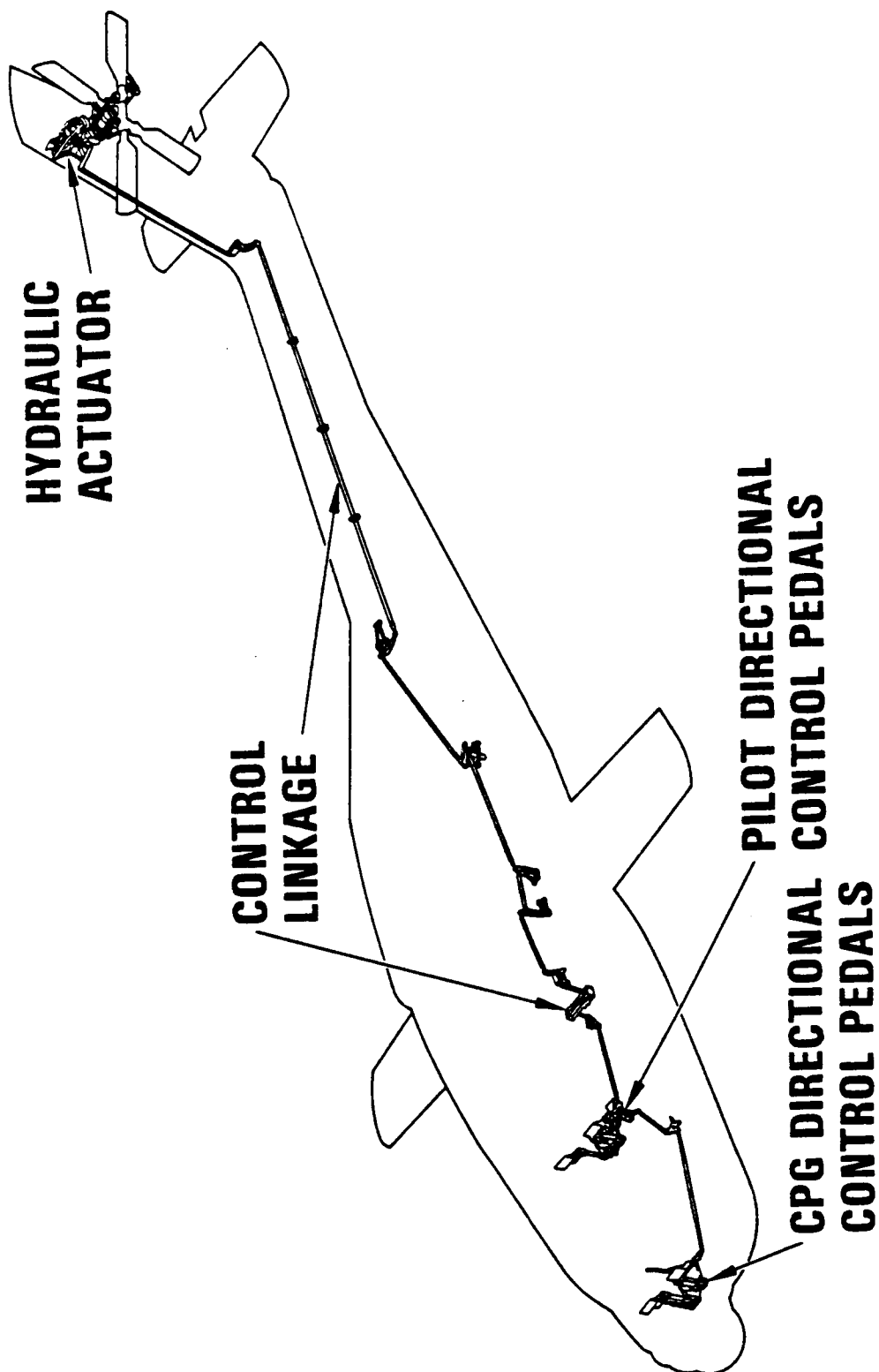


Figure 10

DIRECTIONAL CONTROL LINKAGE

ENCLOSURE 1
APPENDIX D

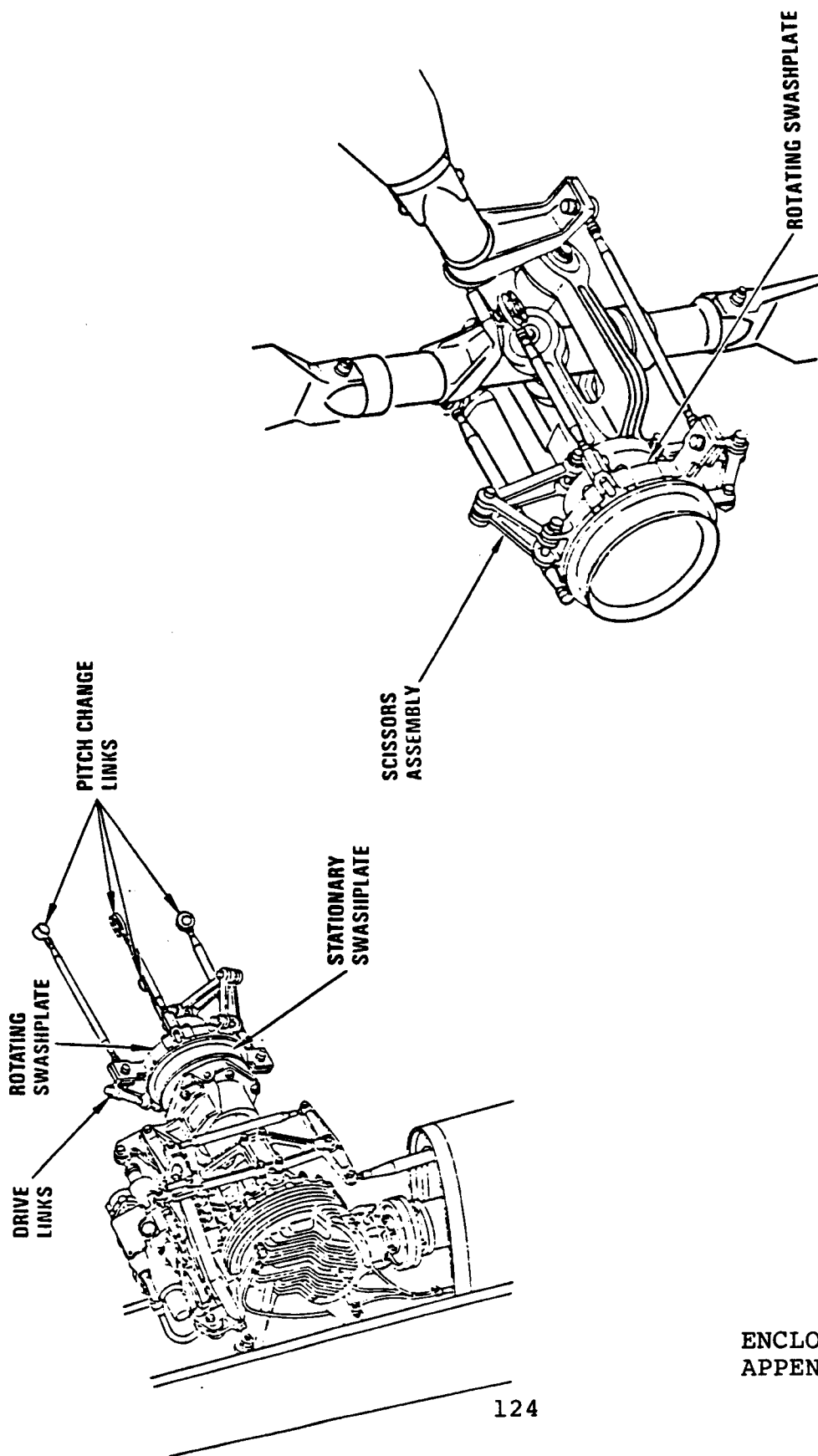


Figure 11
TAIL ROTOR HUB AND SWASHPLATE ASSEMBLY

ENCLOSURE 1
APPENDIX D

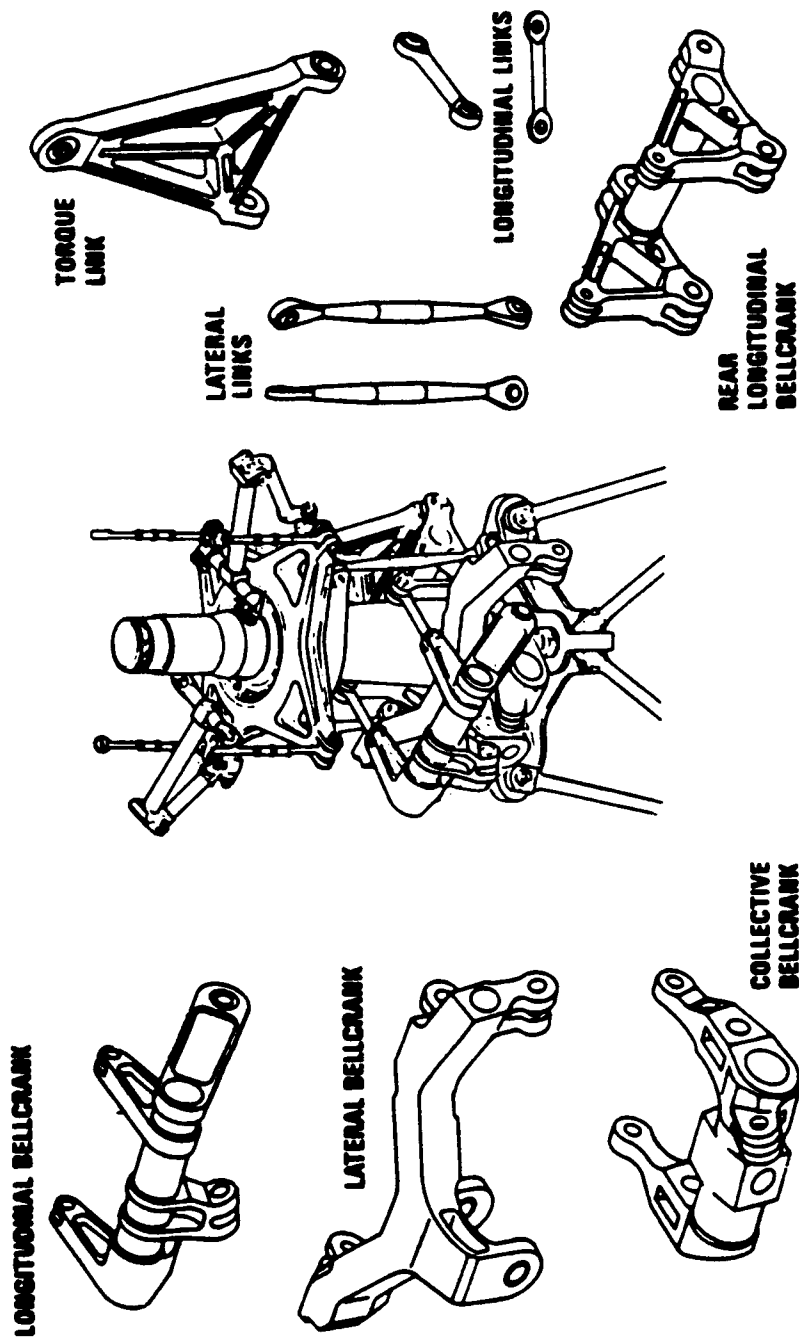


Figure 12
MIXER ASSEMBLY

ENCLOSURE 1
APPENDIX D

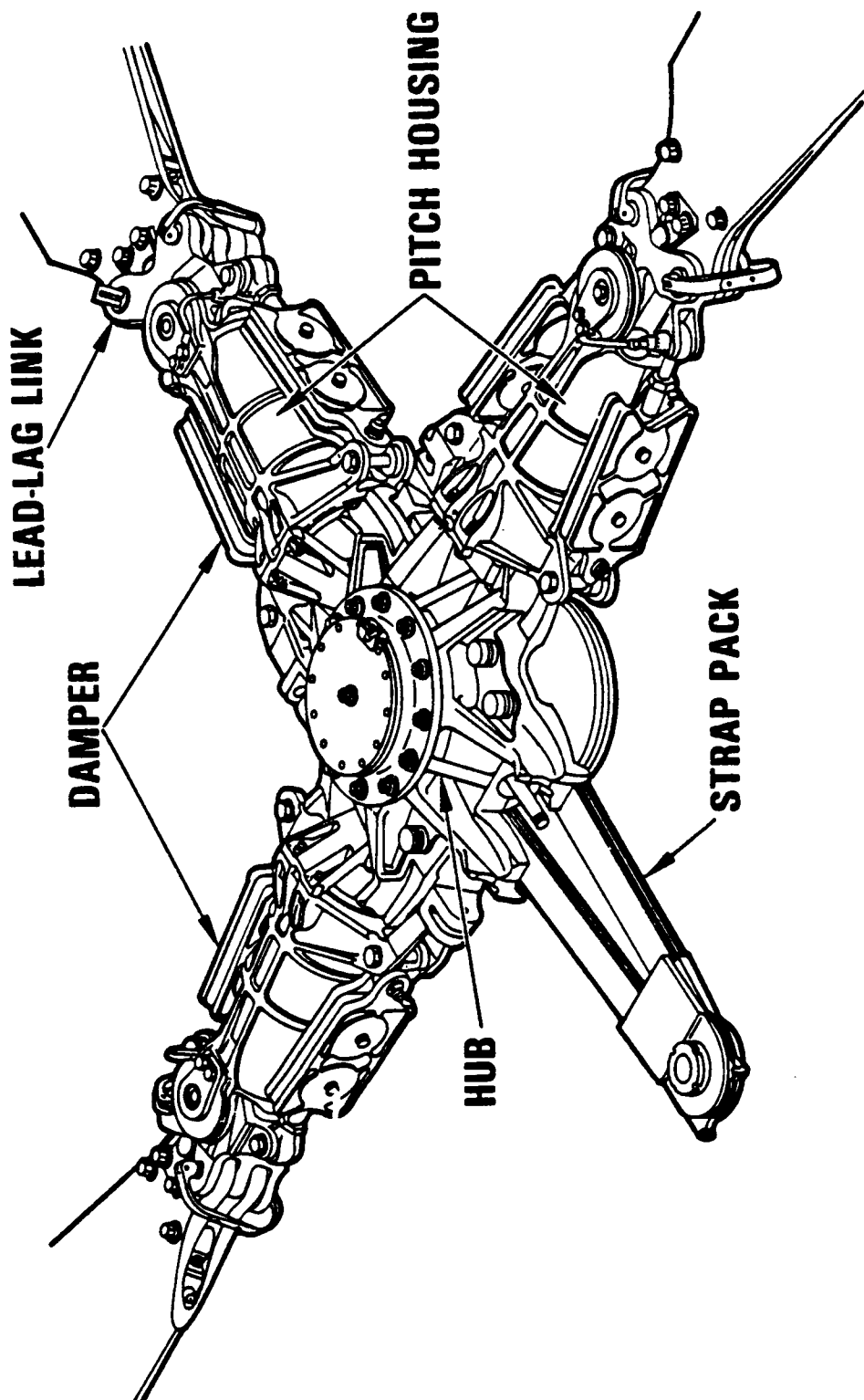


Figure 13

MAIN ROTOR HEAD

ENCLOSURE 1
APPENDIX D

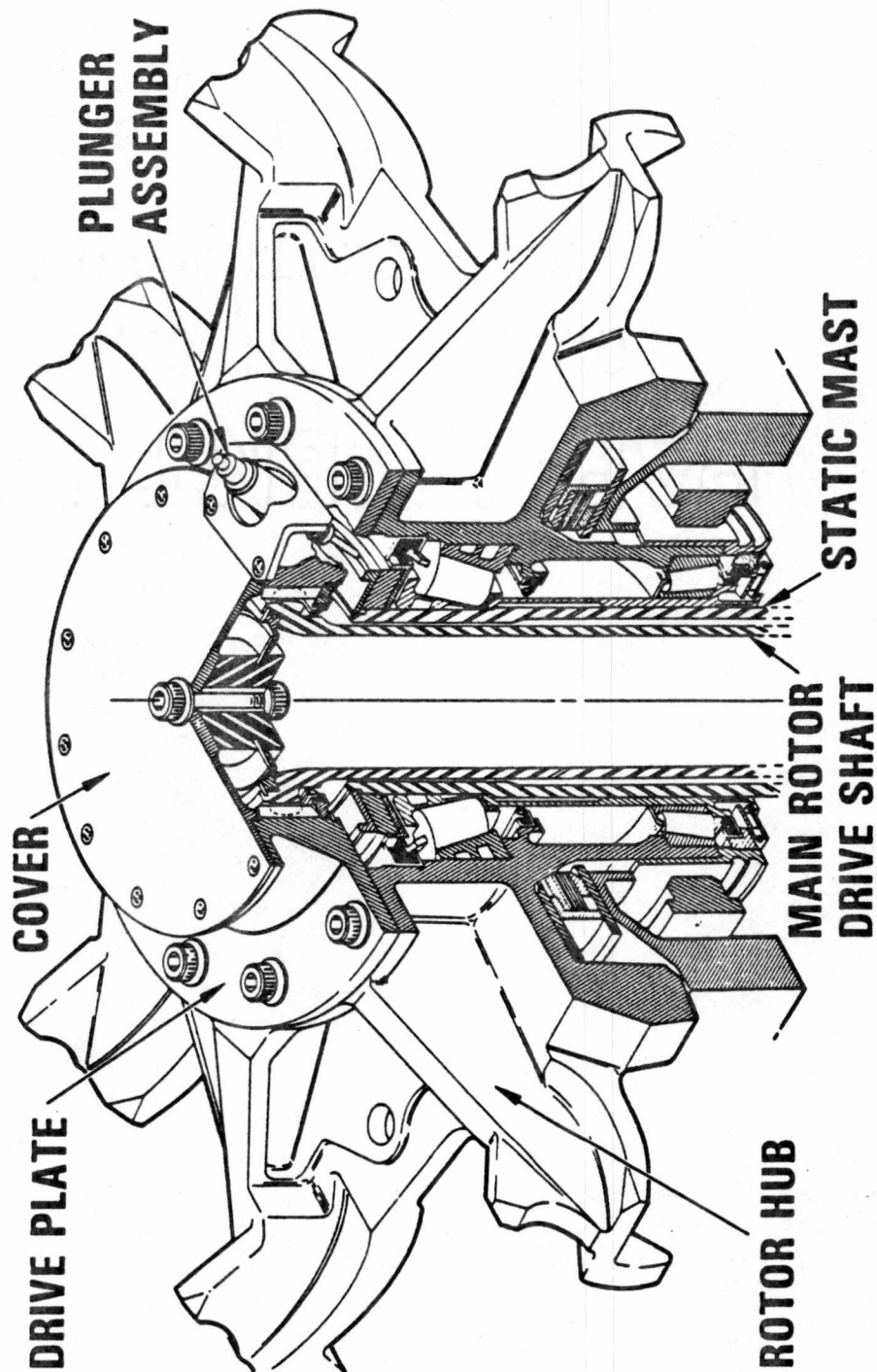


Figure 14
MAIN ROTOR HUB ASSEMBLY

ENCLOSURE 1
APPENDIX D

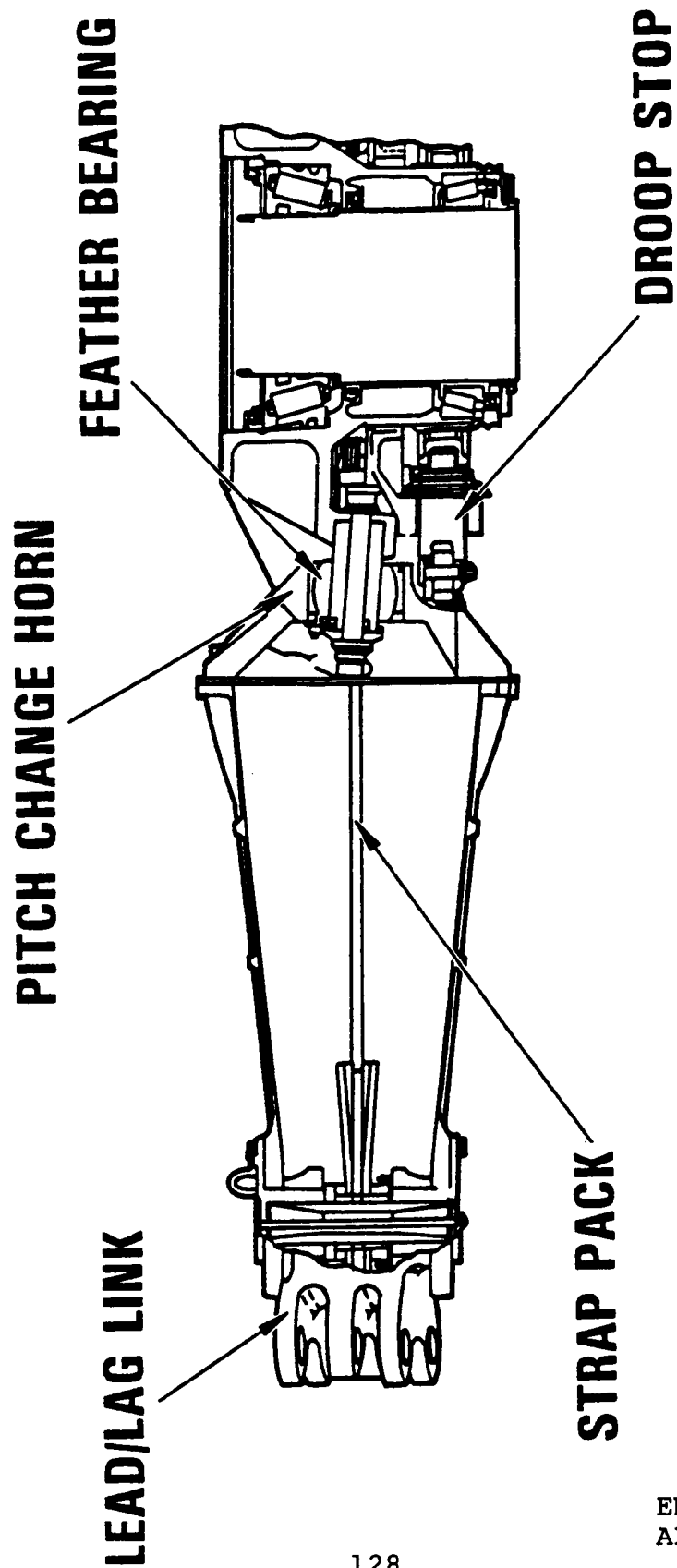


Figure 15
PITCH HOUSING

ENCLOSURE 1
APPENDIX D

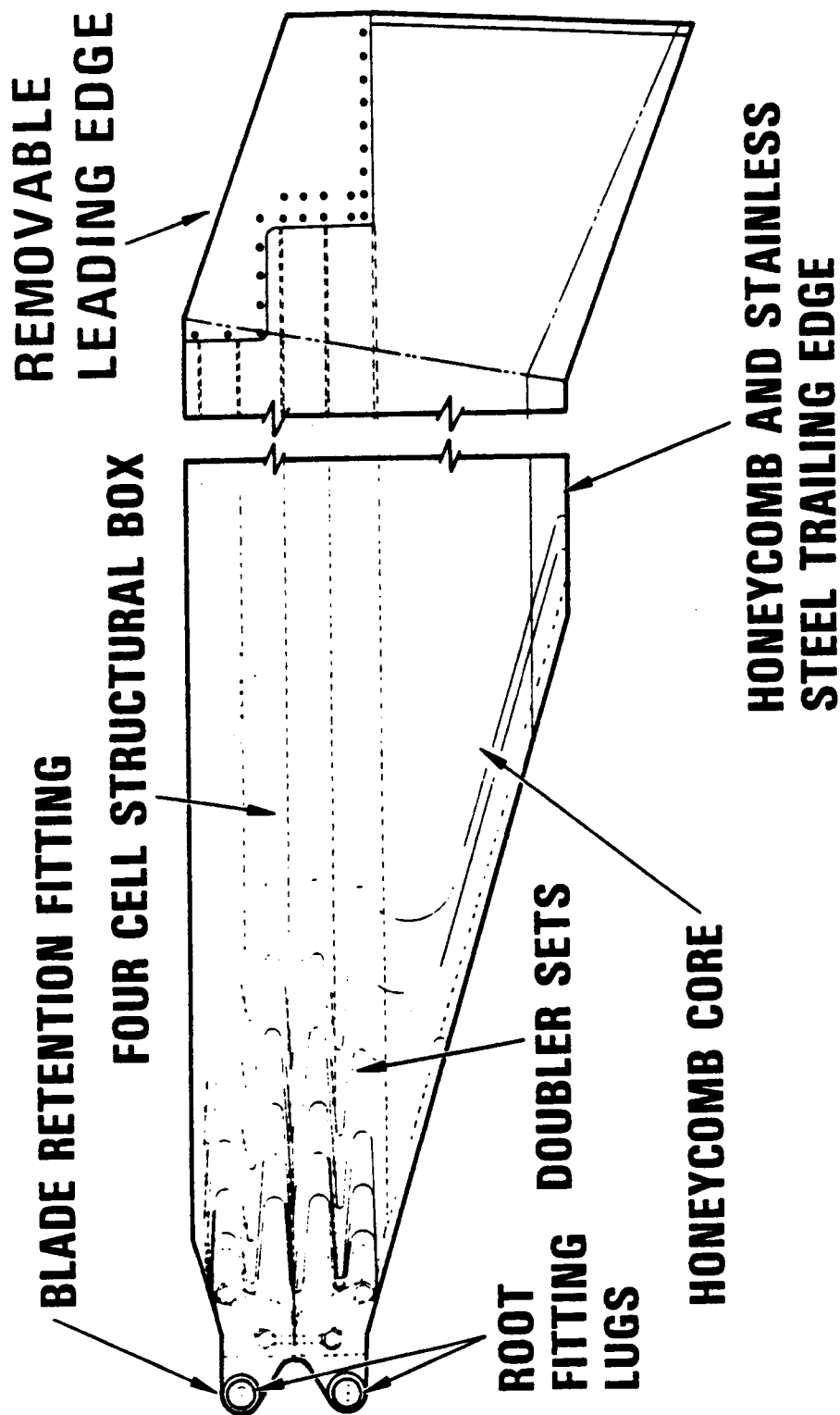


Figure 16
 MAIN ROTOR BLADE CONSTRUCTION

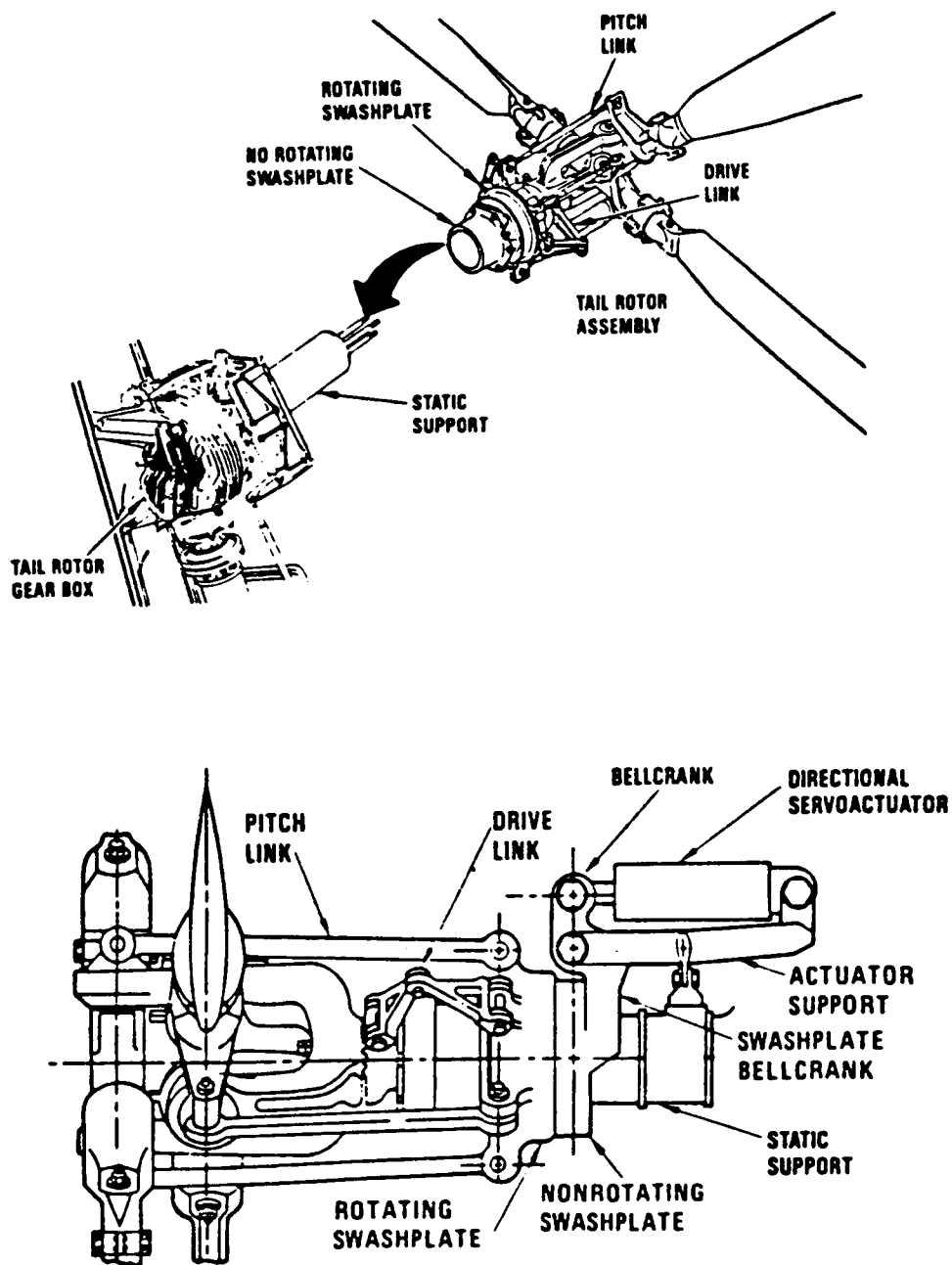


Figure 17
TAIL ROTOR ASSEMBLY

ENCLOSURE 1
APPENDIX D

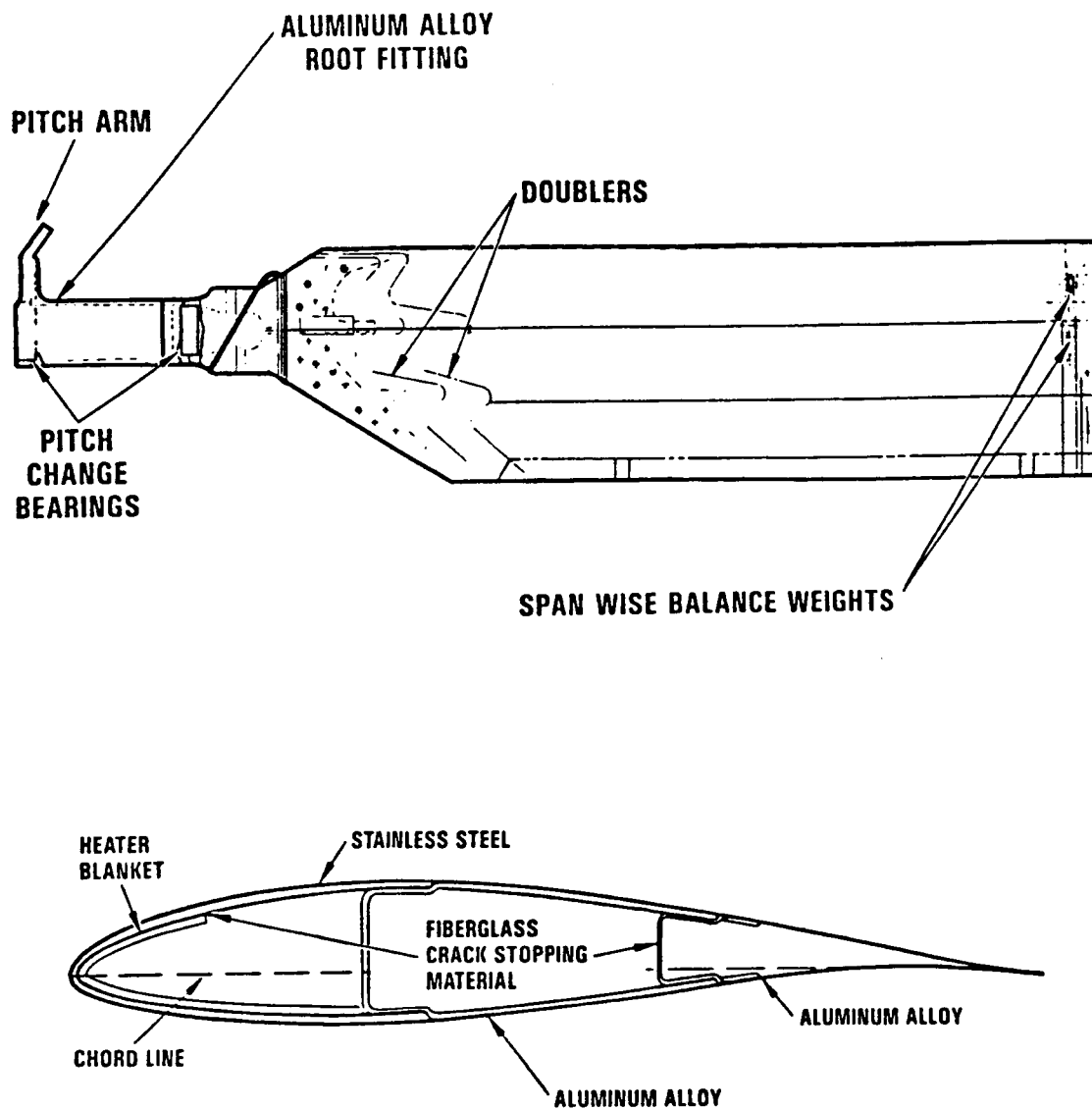


Figure 18
TAIL ROTOR BLADE CONSTRUCTION

ENCLOSURE 1
APPENDIX D

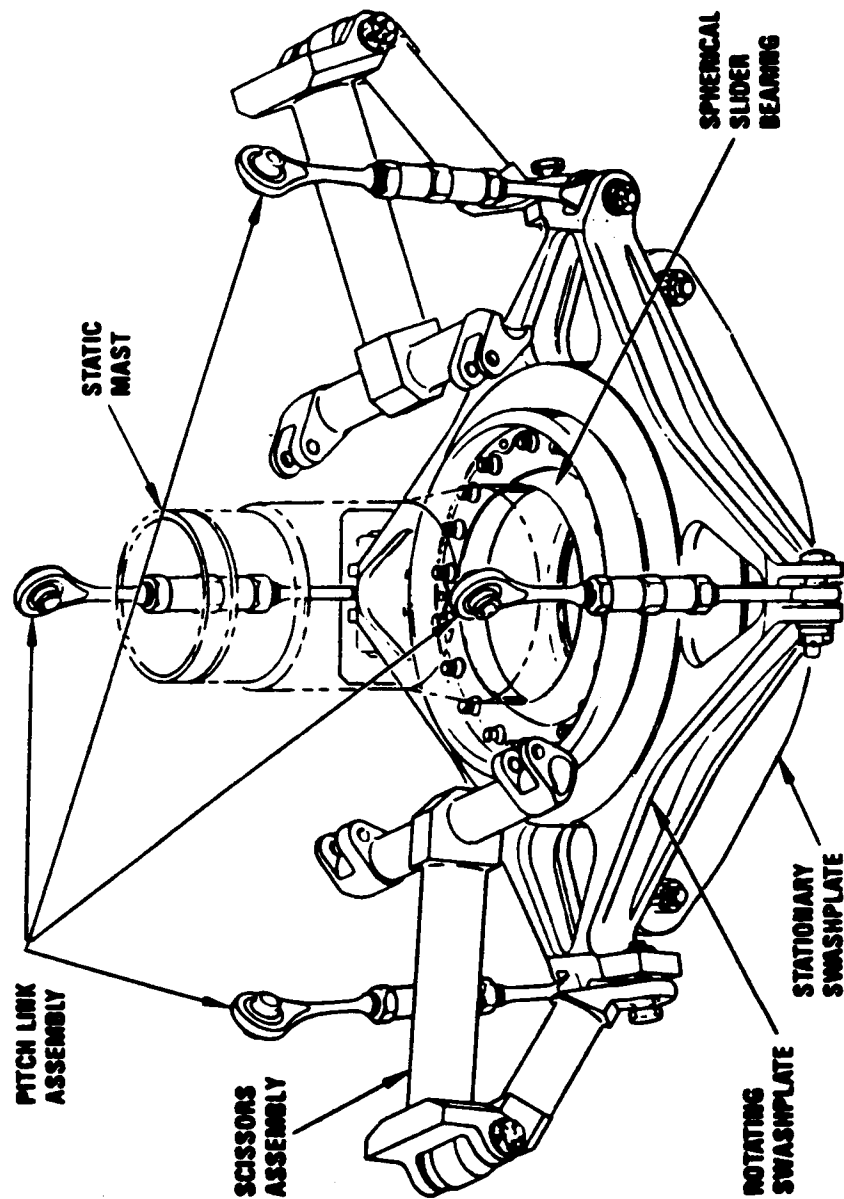


Figure 19
SWASHPLATE ASSEMBLY

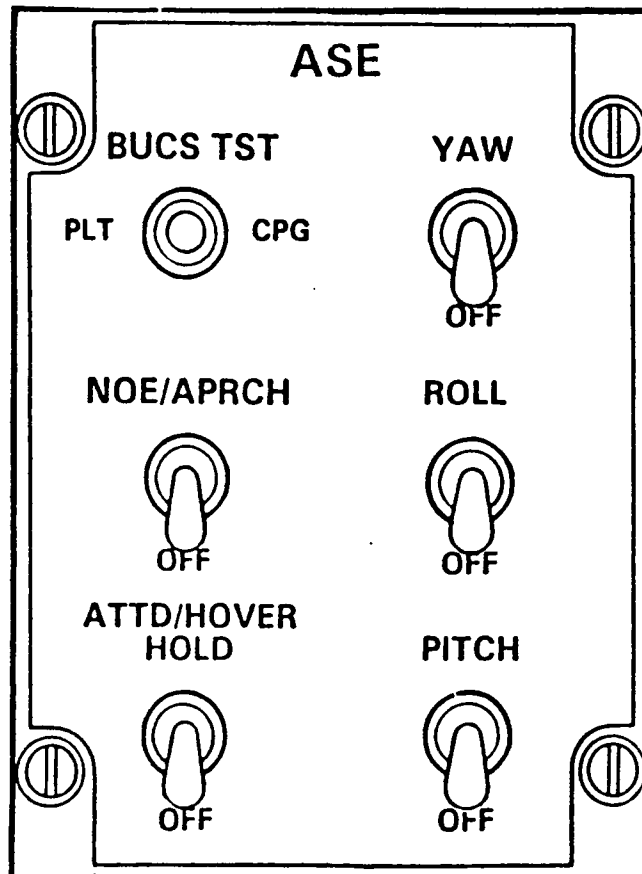


Figure 20
DASE CONTROL PANEL

ENCLOSURE 1
APPENDIX D

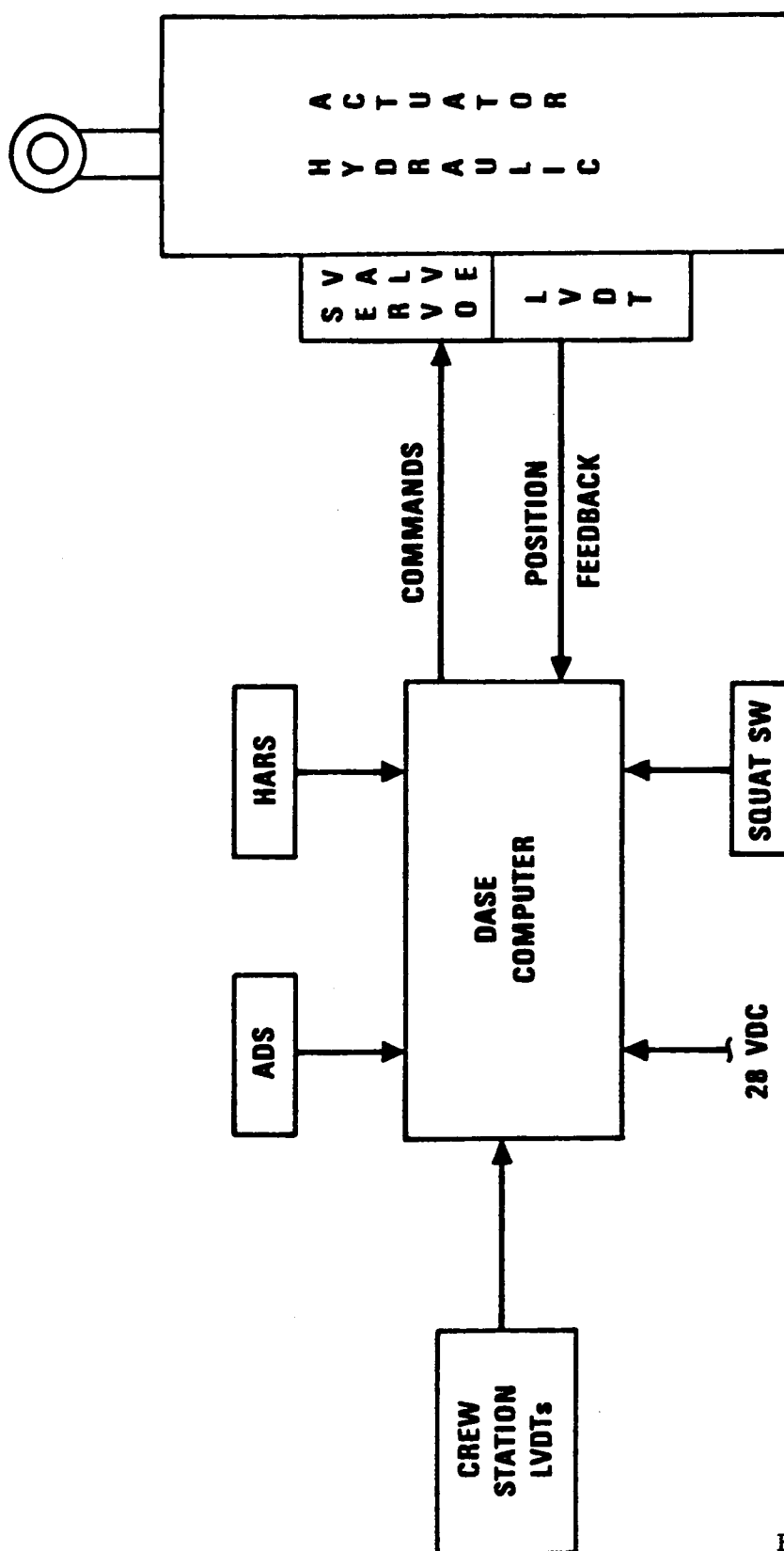


Figure 21
SAS AND CAS OPERATION

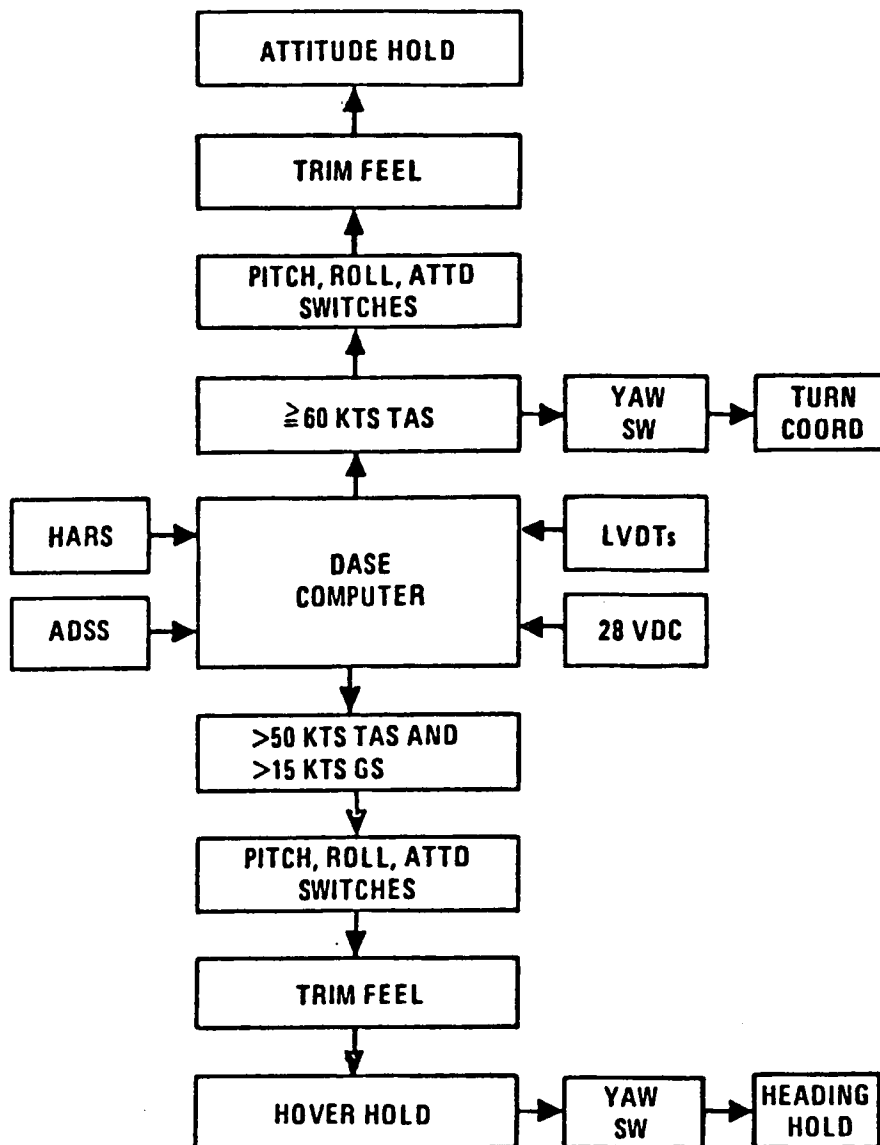


Figure 22

ATTITUDE HOLD AND HOVER AUGMENTATION

ENCLOSURE 1
APPENDIX D

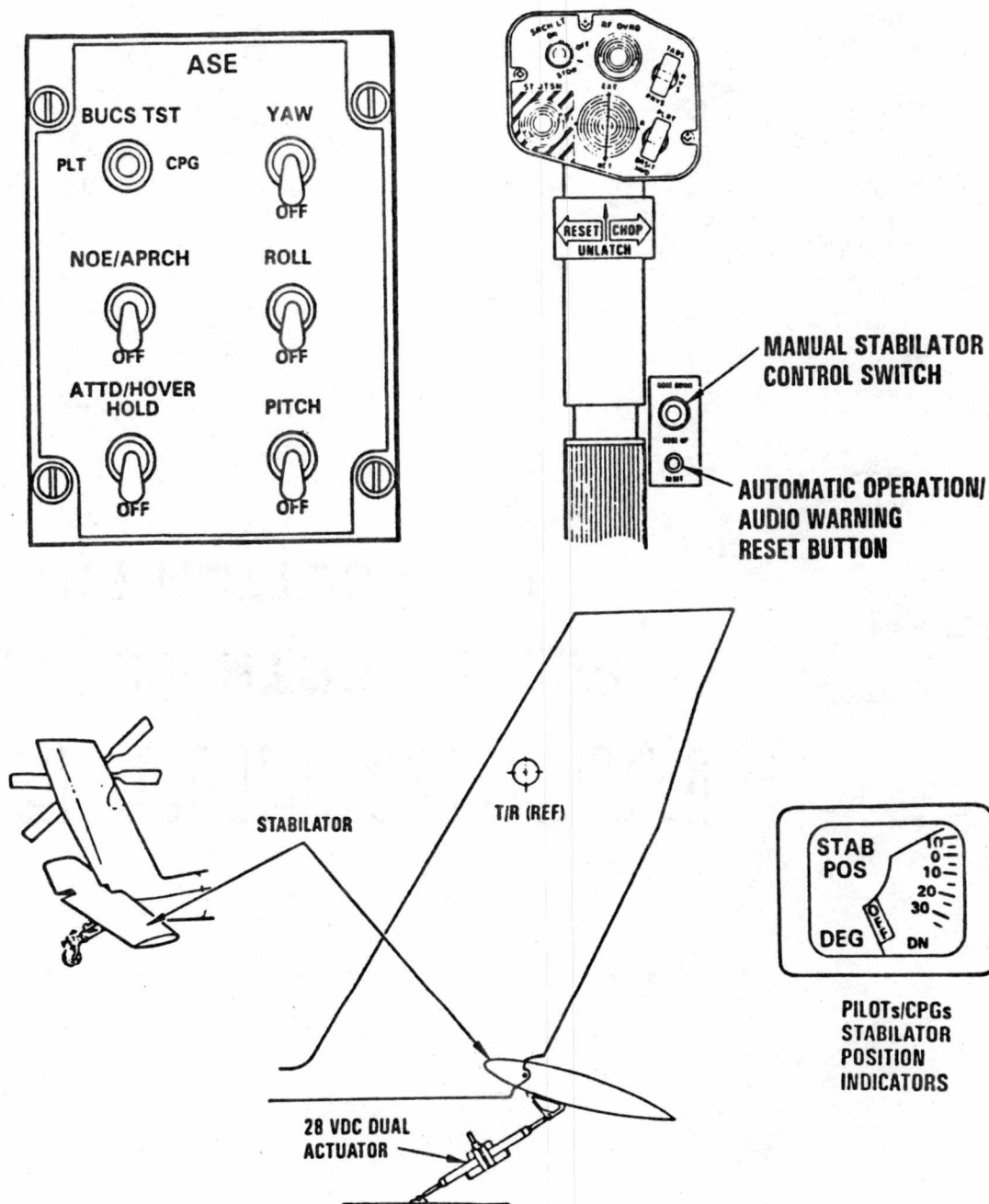


Figure 23
HORIZONTAL STABILIZER AND CONTROLS

ENCLOSURE 1
APPENDIX D

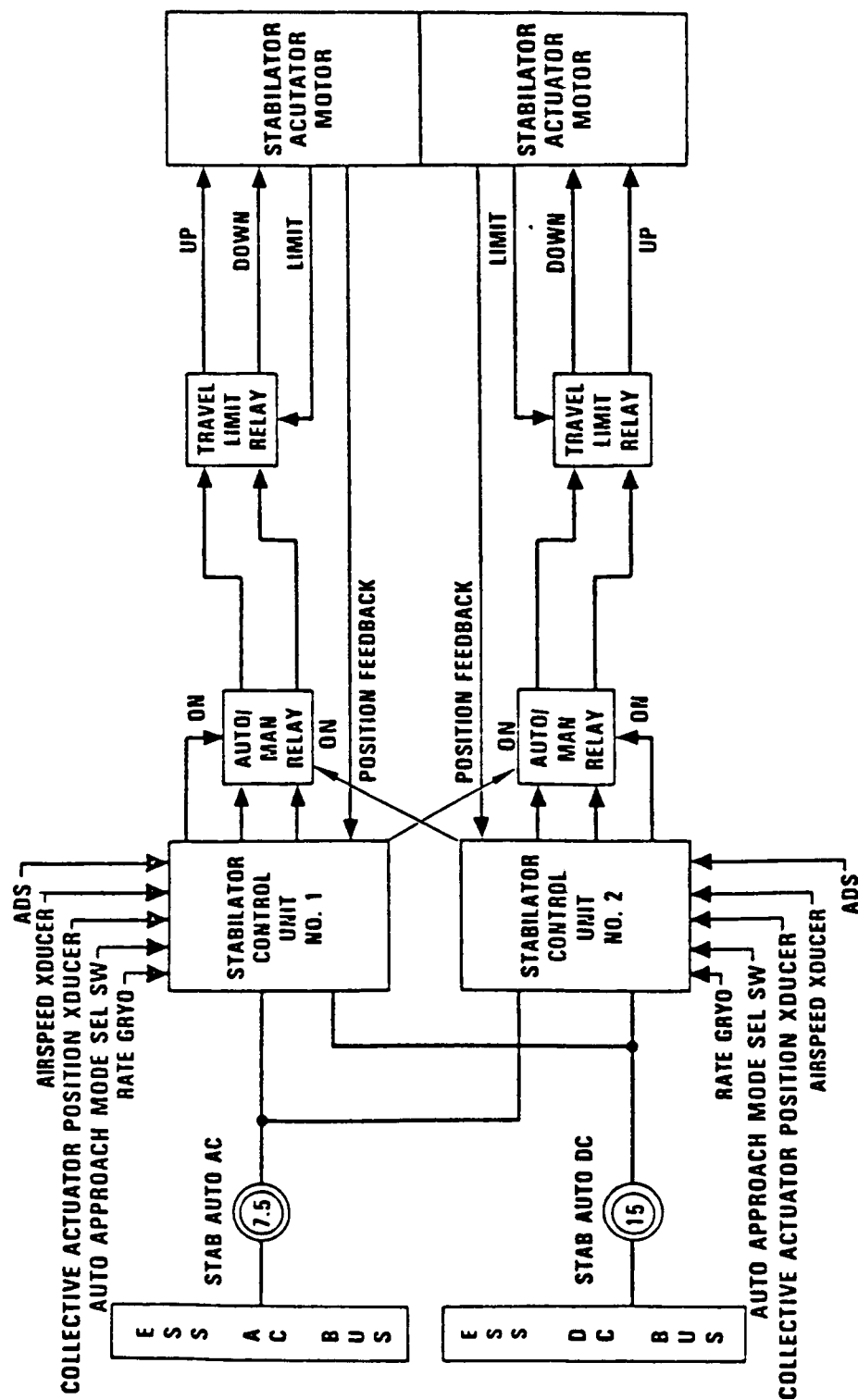


Figure 24
STABILATOR SYSTEM OPERATION

Table I

TESTS AND TEST CONDITIONS

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)	OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
GROUND EVALUATION	--	--	N/A	N/A	N/A	N/A	Cockpit & mechanical characteristics evaluation. External electrical and hydraulic power. SAS-On, ATTD HOLD-Off
QUALITATIVE EVALUATION	0	0	27	16,000	204.1	OFF/AUTO	Ground Taxi
MISSION	40-140	--	24	15,820	203.8	OFF/AUTO	Level Accel.
TASKS	120	--	24	15,700	203.8	ON/AUTO	Level Turns
	0-80	0-300	23	15,260	203.7	OFF/AUTO	Confined area landing
	0	0-10	23	15,200	203.7	OFF/AUTO	Slope landing.
	0	10 & 70	24	15,450	203.7	ON/AUTO	OGE & IGE Hover (HOVER HOLD-ON)
LEVEL FLIGHT PERFORMANCE AND LEVEL FLIGHT TFCP'S	40, 60, 80, 100, 120, Vh (Vne)	--	22	16,050	203.9	ON/AUTO	w/σ = 17,650 Vne point for TFCP in diving flight.

APPENDIX E

Table I (Continued)

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)		OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
IGE/OGE DETERMIN- ATION	0	10-70	30-100	30	16,170	203.9	ON/AUTO	10 ft. increments. Wind less than 5 kts.
HOVER PERFOR- MANCE	0	10-70	0-120	27	15,300	203.7	ON/AUTO	IGE & OGE free hover. Wind less than 5 kts.
COORDINATED TURNS TFCP'S	120	--	2000	26	15,700	203.8	ON/AUTO	15 & 30° RT & LT
STATIC LONGITU- DINAL STABILITY	120	--	2100 (Avg)	22	15,550	203.8	OFF/AUTO	± 2,6,10 & 20 kts. from trim Collective fixed
STATIC LATERAL- DIRECTIONAL STABILITY (STEADY HEADING SIDESLIPS)	120	--	2200 (Avg)	22	15,300	203.7	OFF/AUTO	±, ±, 1 ball width deflec- tion RT & LT
PEDAL ONLY TURNS	120	--	2300 (Avg)	21	15,150	203.7	OFF/AUTO	15 & 30° AOB RT & LT

APPENDIX E

Table I (Continued)

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)	OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
CYCLIC ONLY TURNS	120	-- 2300 (Avg)	21	15,100	203.7	OFF/AUTO	15 & 30° AOB RT & LT
MANEUVERING STABILITY STEADY TURNS PULLUPS AND PUSHOVERS	120	-- 2500 (Avg)	21	14,800	203.7	OFF/AUTO	Collective fixed 15,30,45 & 60° AOB RT & LT 1/2" increments from 1-2 in.
CYCLIC CONTROL RESPONSE	120	-- 2000 (Avg)	20	14,550	203.7	OFF/AUTO	Collective fixed 1/2" increments, 1-2 in. Fore & Aft, RT & LT
SPIRAL STABILITY	120	-- 2000 (Avg)	21	15,050	203.7	OFF & ON/ AUTO	15,30° AOB RT & LT
CLIMB & DESCENT PERFORMANCE	50,60, 70,80, 90	-- 2000- 4000	24	15,700	203.8	ON/AUTO	Max/Min Q. TFCP at 90 kts.
LOW AIRSPEED TRIMMED FLIGHT CONT. POSIT.	0-30	15 50	28	16,250	204.2	OFF & ON/ NOE	0.90,180,270° relative azimuth 10 kt. increments

APPENDIX E

Table I (Continued)

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)		OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
CRITICAL AZIMUTH	20	15	50	28	15,550	203.9	OFF/NOE	0-360° in 45° increments
LOW A/S LAT-DIR STATIC STABILITY	0-10	15	50	28	15,250	204.1	ON/NOE	0.90,180,270° relative azimuth A/S increments 3-4 kts. Collec- tive fixed.
LOW A/S MISSION TASKS	0-40	10-25	40	27	15,010	203.8	OFF & ON/ NOE	Turns on a spot, masking/unmask NOE quickstop. Qualitative evaluation
COUPLED DYNAMIC RESPONSE	0	500	525	26	15,220	203.7	OFF & ON/ AUTO	Controls fixed. Natural exci- tation.
VERTICAL CLIMB PERFOR- MANCE	0	50- 1550	100- 1600	28	14,600	203.7	ON/AUTO	3 incremental Q increases to max Q
LONG TERM DYNAMIC RESPONSE	120	--	2000 (Avg)	21	14,540	203.7	OFF & ON/ AUTO	Natural excita- tion 10 kt. decel. Controls fixed.

APPENDIX E

Table I (Continued)

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)	OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
SHORT TERM DYNAMIC RESPONSE	120	-- 2000	21	14,600	203.7	OFF/AUTO	Longitudinal input at A/C natural frequency. Qual. Eva. ±5° pitching task.
LATERAL- DIRECTIONAL OSCILLATION	120	-- 2000	21	14,480	203.7	OFF & ON/ AUTO	Natural excita- tion, pedal & lat. Cyclic doublet, SHSS release.
GUST RESPONSE	120	500 550	22	14,400	203.7	OFF & ON/ AUTO	1 inch collective pulse.
TRANSIENT DROOP	120	-- 2000	23	15,400	203.8	ON/AUTO	Min. pwr to 80% Q in incremen- tally decreasing times from 22 to 5 sec.
DEGRADED MODES	100	-- 2000	24	15,000	203.7	OFF/AUTO	SAS off Qual. Eval.
MISSION TASKS	40- 140	-- 2000 (Avg.)	23	15,200	203.8	OFF/AUTO	EVM: Climbing turns, wingovers, roll reversals

APPENDIX E

Table I (Continued)

TEST	AIRSPEED (KIAS)	ALTITUDE (FT AGL) (FT PA)	OAT (°C)	AVG GROSS WEIGHT (LB.)	AVG CG (IN.)	ATTD/HOLD STABILATOR	REMARKS
AUTOROTA- TIONS	80-0	1000-	--	22	14,600	203.7	OFF/AUTO Qual. Eval.

NOTES: (1) Rotor RPM 100% during all tests except Ground Eval.
 (2) Loading - 2 pilots, full fuel (2444 lbs.), 500 rds. 30 mm, 38 2.75" rks, 8 Heli-
 fire missiles (inert ordnance).
 (3) Configuration - engine bleed air - off, SAS - pitch, roll and yaw engaged
 except for SAS off eval.

Table II
EVALUATION PILOT
ANTHROPOMETRIC MEASUREMENTS

MEASUREMENT	IN/LB	PERCENTILE ^a
Weight	140.00	5th
Height	71.75	70th
Sitting Eye Height	31.82	60th
Trunk Height	24.07	60th
Sitting Height	35.43	25th
Buttock-Knee Length	23.93	45th
Functional Reach	31.59	55th

^aPercentiles based on the 1981 Naval and Marine Corps Aviation Anthropometric Survey of 20 November 1981. Systems Engineering Test Directorate (SETD) Report Number SY-121R-81.

Table III

FLIGHT CONTROL MECHANICAL CHARACTERISTICS

PARAMETER	CONTROL	MEASURED VALUE	AH-64A DETAILED SPECIFICATION AMC-SS-AAH-H10000A	
			Paragraph	Requirement Conformance
Total System Freeplay (in)	Longitudinal	± 1/8		Failed (100%)
	Lateral	± 1/16		Met
	Directional	± 1/16	10.3.2.8	Met
	Collective	± 1/16		Met
Trim System Freeplay (in)	Longitudinal	± 1/16	N/A	N/A
	Lateral	Negligible		
Control Travel (in)	Longitudinal	4 3/4		
	Lateral	7		
	Directional	3 3/4	N/A	N/A
	Collective	13		
Breakout Force Including Friction (lb)	Longitudinal	2.8 fwd		Failed (87%)
		2.2 aft		Failed (47%)
	Lateral	1.5 right		Met
		2.0 left		Failed (33%)
	Directional	3.0 right	10.3.2.1.1	Met
		4.0 left		Met
	Collective	5.8 up		Met
		4.0 down		Met
Average Force (lb)	Longitudinal	2.4 fwd		Met
		2.3 aft		Met
	Lateral	1.1 right	10.3.2.2	Met
		0.8 left		Met
			35-60% of the longitudinal gradient	

APPENDIX E

Table III (Continued)

PARAMETER	CONTROL	MEASURED VALUE	AH-64A DETAILED SPECIFICATION AMC-SS-AAH-H10000A		
			Paragraph	Requirement	Conformance
Average Force (lb)	Directional	4.2 right 4.3 left	10.3.2.2	2.0-10.0	Met Met
Limit Control Force (lb)	Longitudinal	10.5 fwd		20.0	Met
		8.5 aft			Met
	Lateral	6.2 right 5.5 left	10.3.3.2.3	10.0	Met Met
	Directional	13.0 right 15.0 left		50.0	Met Met
	Collective	13.0 up 9.0 down		20.0	Met Met
Trim Control	Longitudinal	0			
Displacement	Lateral	1/4	N/A	N/A	N/A
Band (in)	Directional	3/8			
Stick Jump ^d (in)	Longitudinal	1/4			Failed
	Lateral	1/8	10.3.3.2.2	None	Failed
Control Centering	Longitudinal	Positive			Met
	Lateral	Positive		Positive	Met
	Directional	Positive			Met
	Collective	None			Met
Control Dynamics ^e	Longitudinal	Mod. Damp.			
	Lateral	Hvy. Damp.	N/A	N/A	N/A
	Directional	Hvy. Damp.			
	Collective	None			

APPENDIX E

Table III (Continued)

PARAMETER	CONTROL	MEASURED VALUE	AH-64A DETAILED SPECIFICATION AMC-SS-AAH-H10000A	
			Paragraph	Requirement Conformance
Cyclic Coupling	Directional Collective	None None	10.3.2.9	None Met Met
Directional Coupling	Cyclic Collective	None None	10.3.2.9	None Met Met
Collective Coupling	Cyclic Directional	None None	10.3.2.9	Met Met Met

NOTES: aCollective down, pedals neutral.
bCollective down, cyclic centered.
cCyclic centered, pedals neutral.
dWith one inch cyclic deflection.
eAfter two inch cyclic deflection.

CURVES

MODEL AH-64A HELICOPTER

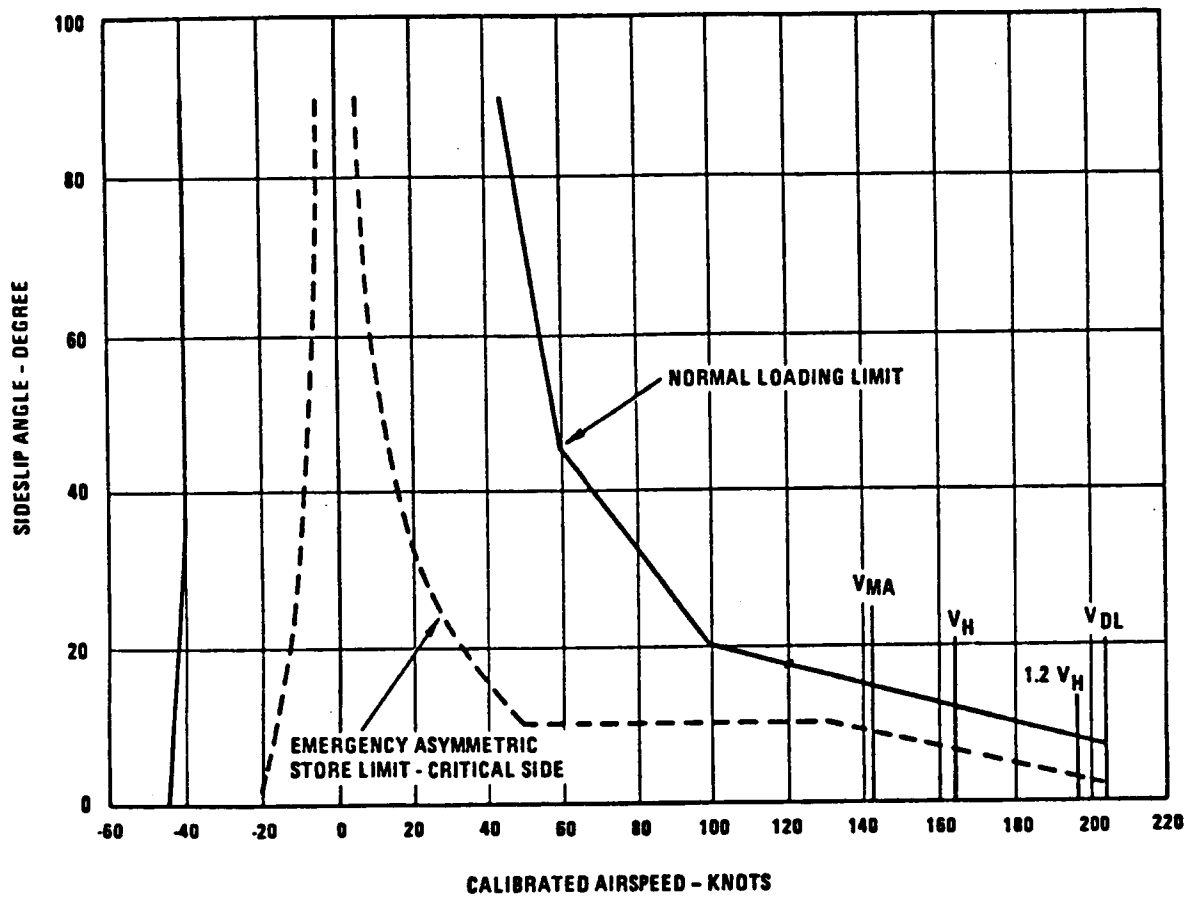


Figure 1
SIDESLIP LIMITS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE AND ROTORS: STOPPED
ELECTRICAL PWR: EXTERNAL
HYDRAULIC PRESSURE: EXTERNAL
SAS/TRIM: ON

LATERAL CONTROL: CENTERED
DIRECTIONAL PEDALS: NEUTRAL
COLLECTIVE: DOWN
ATTD/HOVER HOLD: OFF

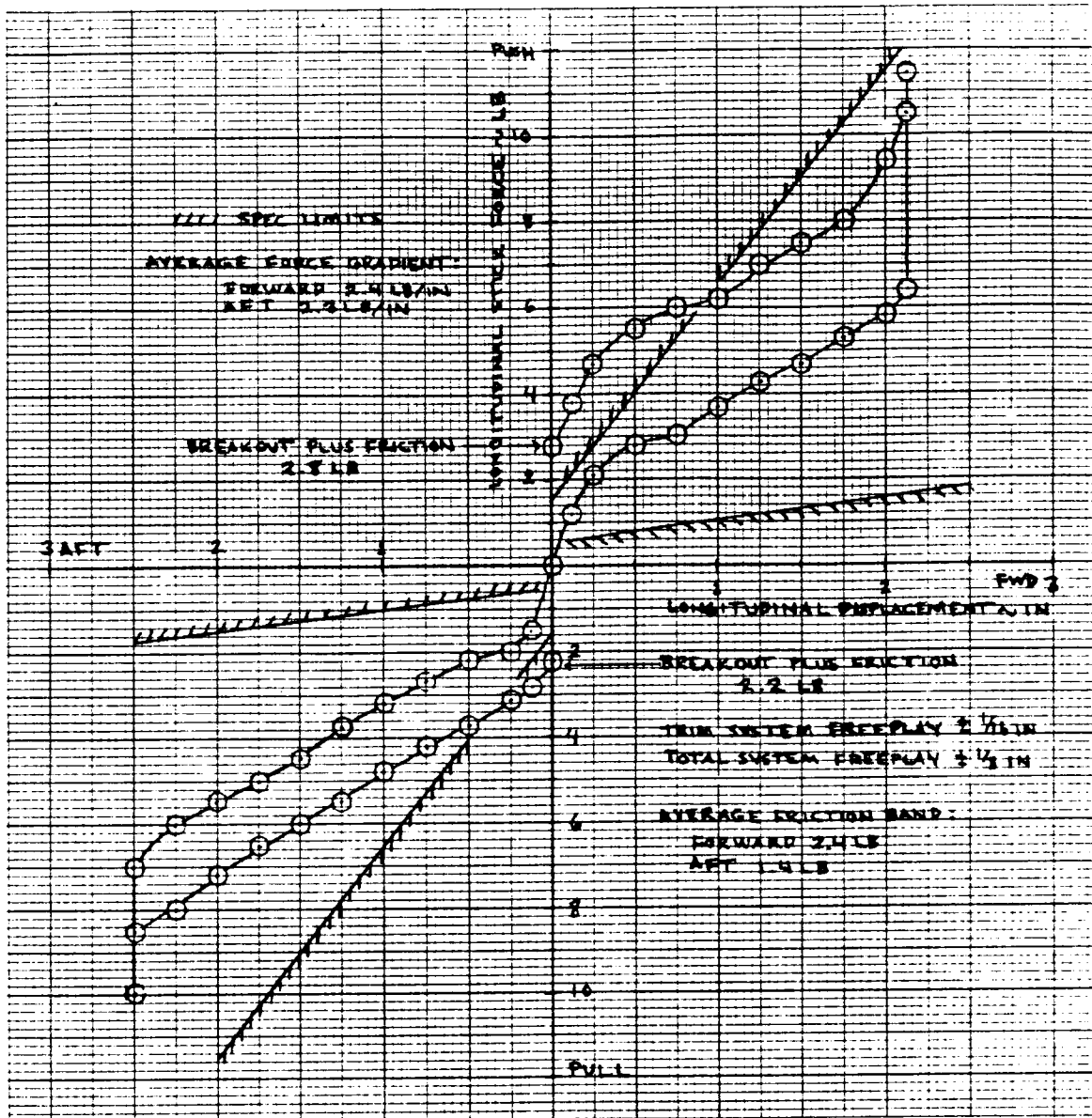


Figure 2

LONGITUDINAL CYCLIC CONTROL SYSTEM CHARACTERISTICS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE AND ROTORS: STOPPED
ELECTRICAL PWR: EXTERNAL
HYDRAULIC PRESSURE: EXTERNAL
SAS/TRIM: ON

LATERAL CONTROL: CENTERED
DIRECTIONAL PEDALS: NEUTRAL
COLLECTIVE: DOWN
ATTD/HOVER HOLD: OFF

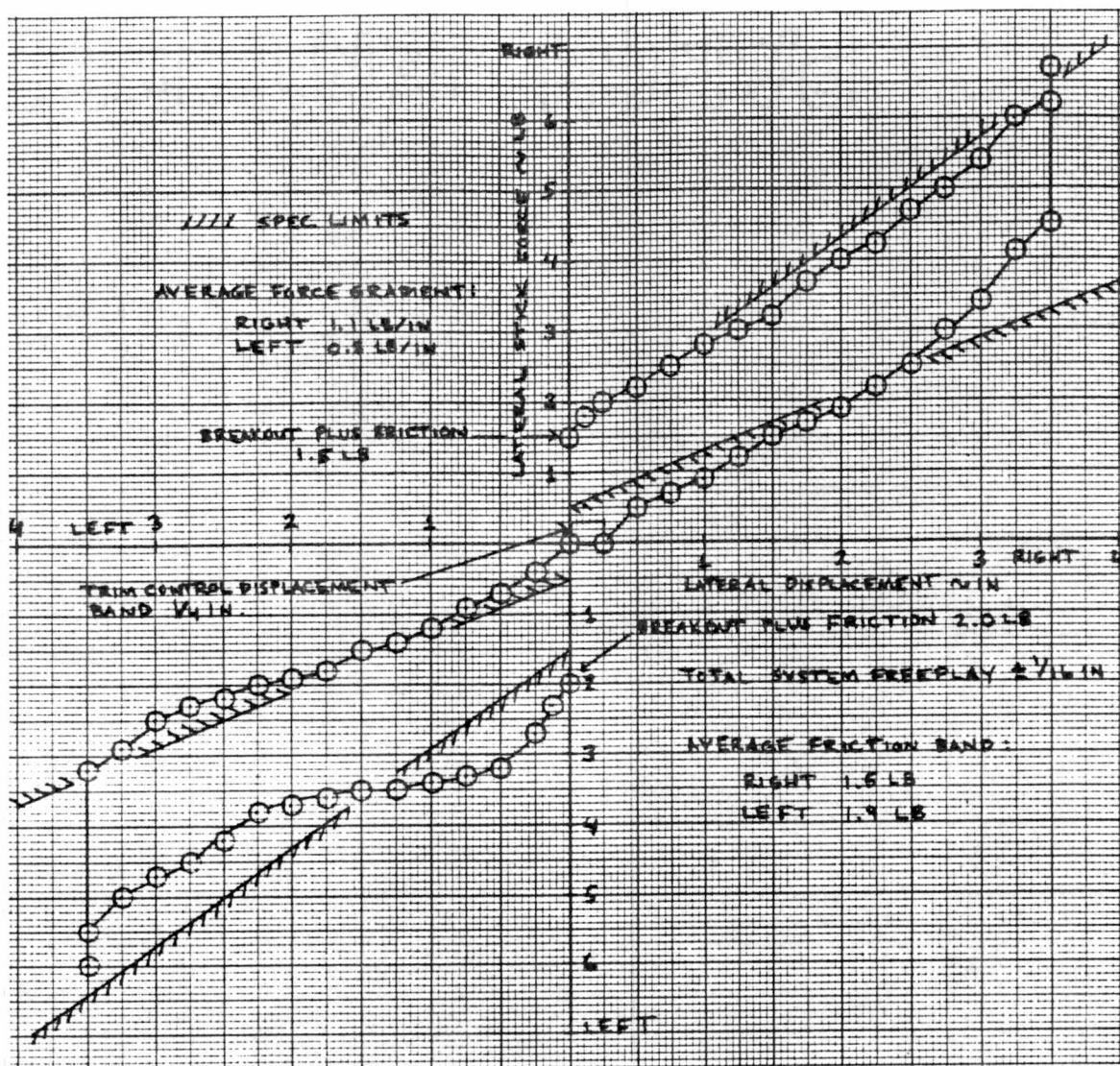


Figure 3
LATERAL CYCLIC CONTROL SYSTEM CHARACTERISTICS

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE AND ROTORS: STOPPED
ELECTRICAL PWR: EXTERNAL
HYDRAULIC PRESSURE: EXTERNAL
SAS/TRIM: ON

COLLECTIVE: FULL UP TO FULL
DOWN
DIRECTIONAL CONTROL: FULL
RIGHT TO FULL LEFT
ATTD/HOVER HOLD: OFF

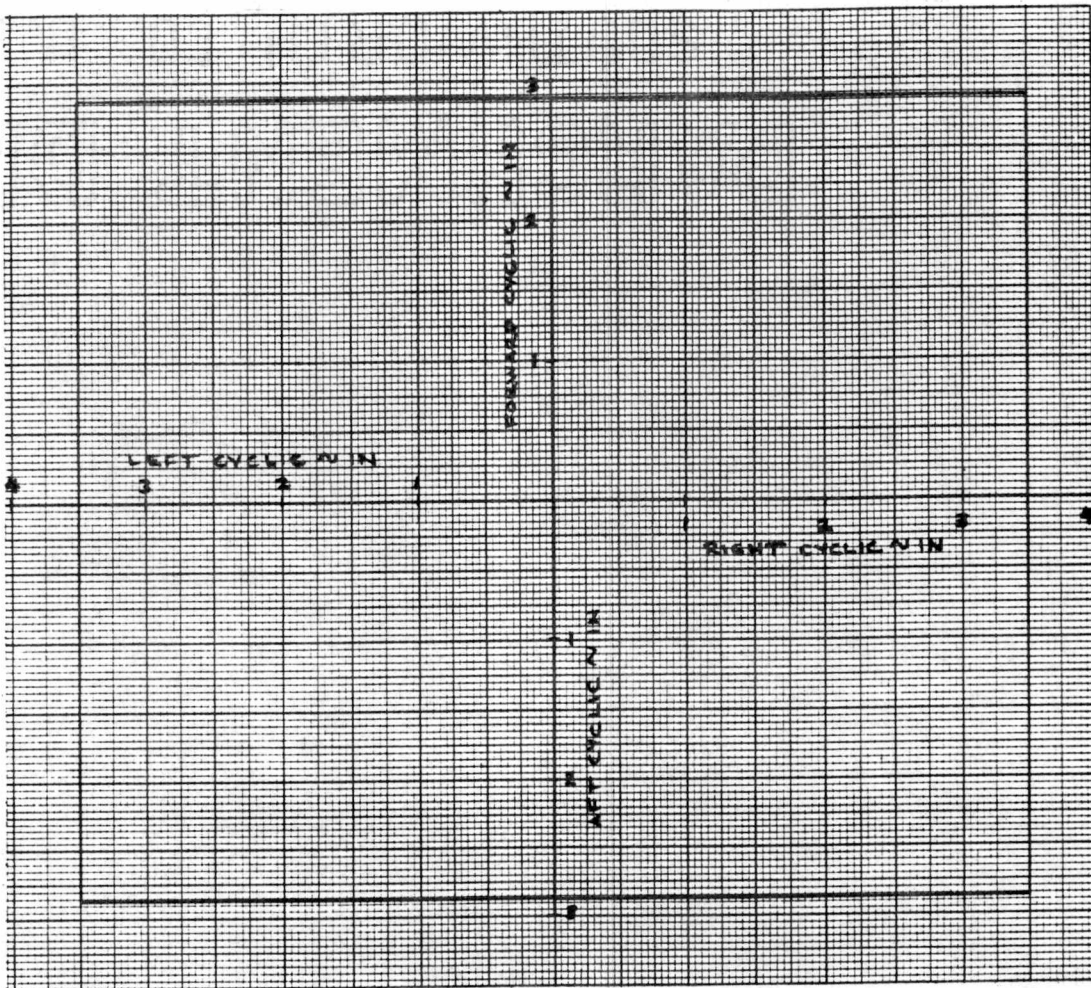


Figure 4

CYCLIC CONTROL ENVELOPE

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE AND ROTORS: STOPPED CYCLIC: CENTERED
ELECTRICAL PWR: EXTERNAL COLLECTIVE: DOWN
HYDRAULIC PRESSURE: EXTERNAL ATTD/HOVER HOLD: OFF
SAS/TRIM: ON

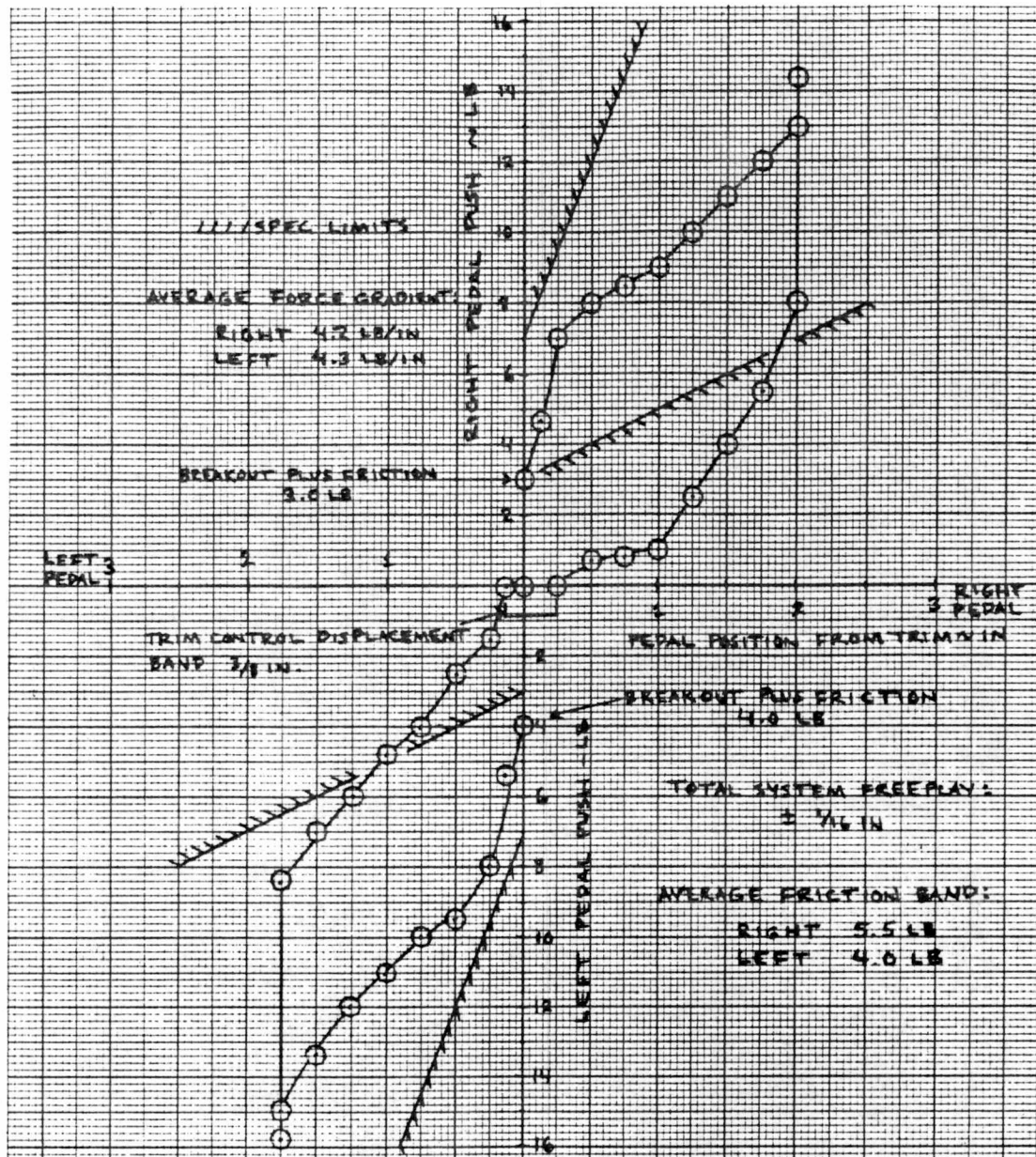


Figure 5
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE AND ROTORS: STOPPED CYCLIC: CENTERED
ELECTRICAL PWR: EXTERNAL PEDALS: NEUTRAL
HYDRAULIC PRESSURE: EXTERNAL ATTD/HOVER HOLD: SET
SAS/TRIM: ON

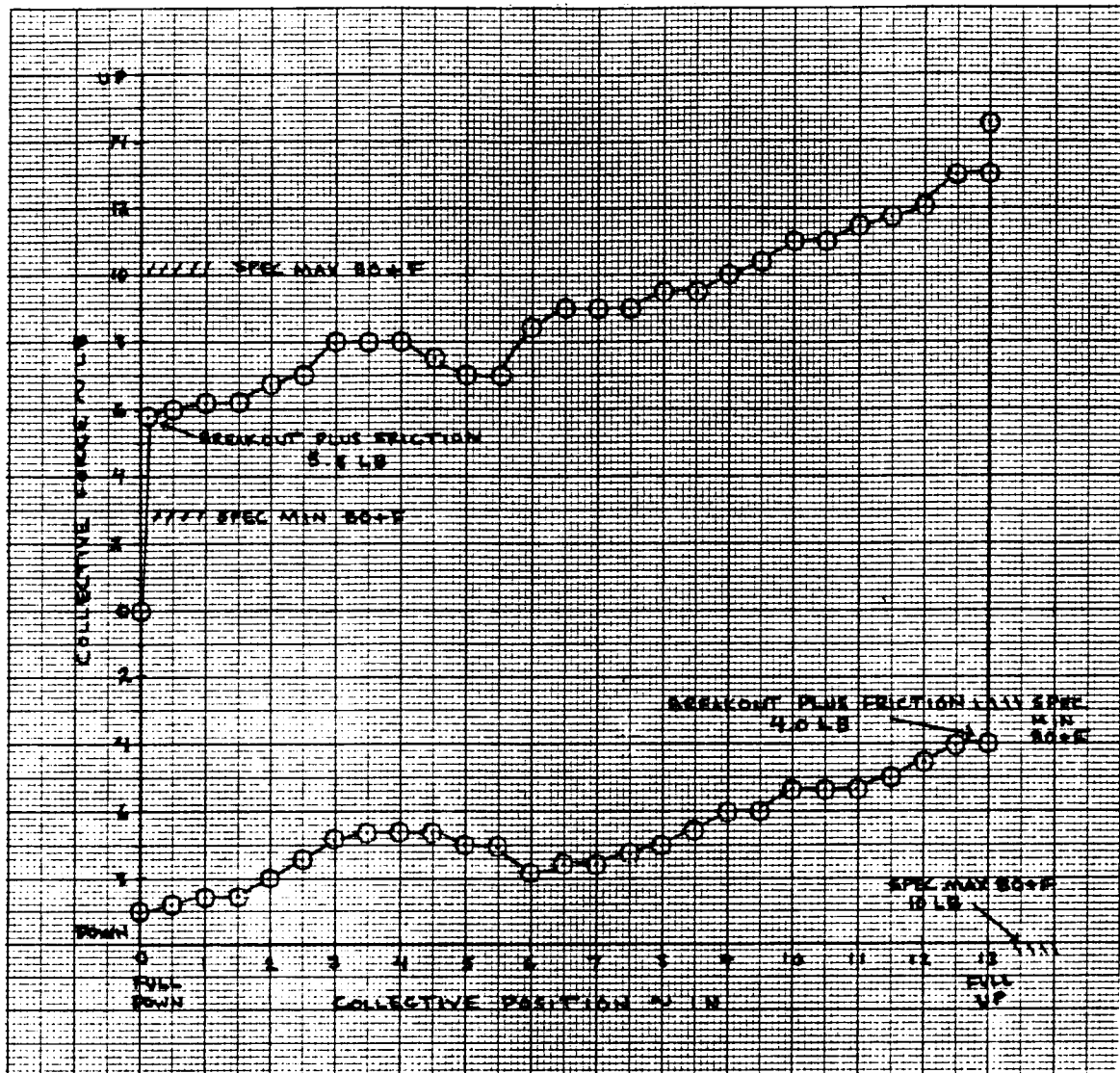


Figure 6
COLLECTIVE CONTROL SYSTEM CHARACTERISTICS

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%

AVG. GROSS WEIGHT: 16,050 LBS.
AVG. CG: 203.9 IN.
AVG. PA: 2100 FT.
OAT: 22°C
METHOD: TRIMMED FLIGHT CONTROLS

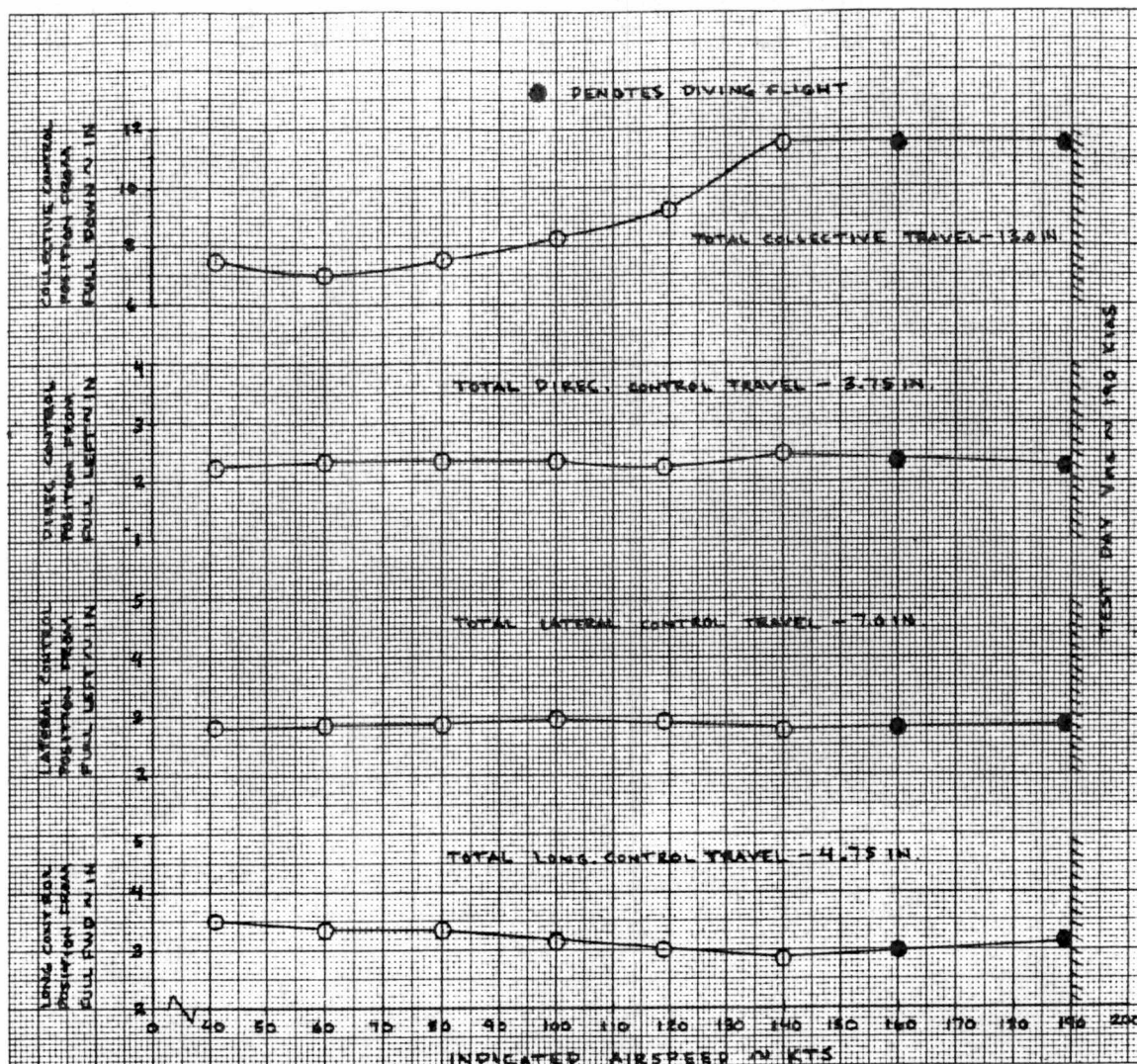


Figure 7

LEVEL FLIGHT TRIMMED FLIGHT CONTROL POSITIONS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR RPM: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 15,700 LBS.
AVG. CG: 203.9 IN.
AVG. PA: 2000 FT.
OAT: 26°C
METHOD: TRIMMED FLIGHT CONTROLS

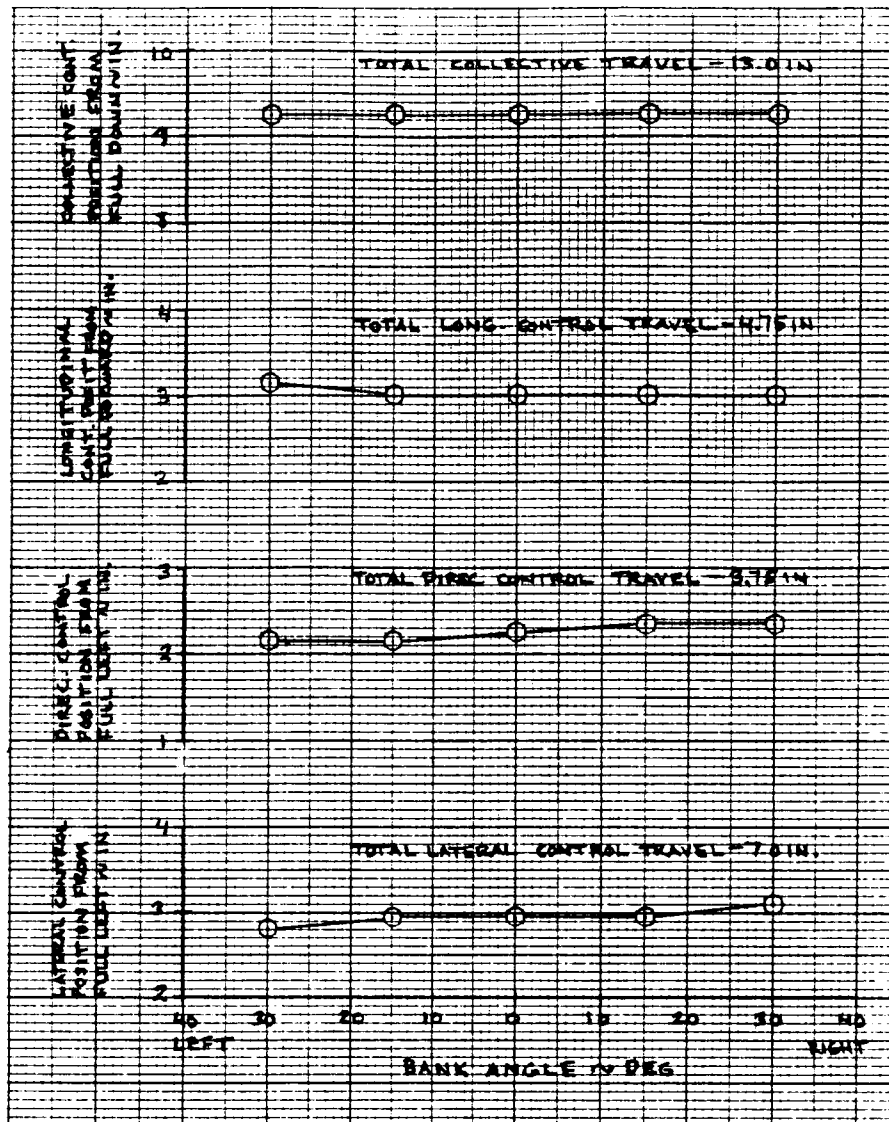


Figure 8

COORDINATED LEVEL TURNS TRIMMED FLIGHT CONTROL POSITIONS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%
AIRSPEED: 90 KIAS

AVG. GROSS WEIGHT: 15,700 LBS.
AVG. CG: 203.8 IN.
AVG. PA: 3000 FT.
OAT: 24°C
METHOD: TRIMMED FLIGHT CONTROLS

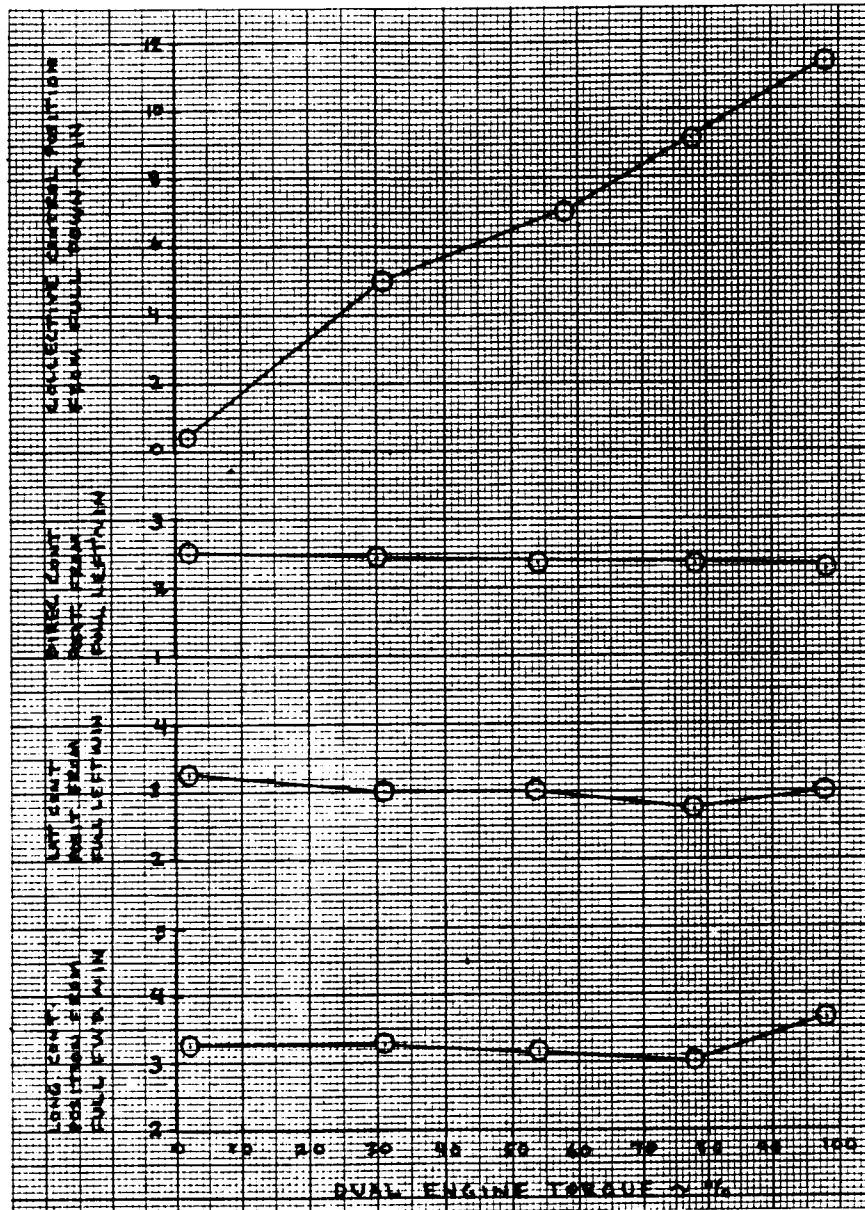


Figure 9
CLIMB AND DESCENT TRIMMED FLIGHT CONTROL POSITIONS
APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR RPM: 100%

AVG. GROSS WEIGHT: 15,550 LBS.
AVG. CG: 203.8 IN.
AVG. PA: 2100 FT.
OAT: 22°C
METHOD: COLLECTIVE FIXED

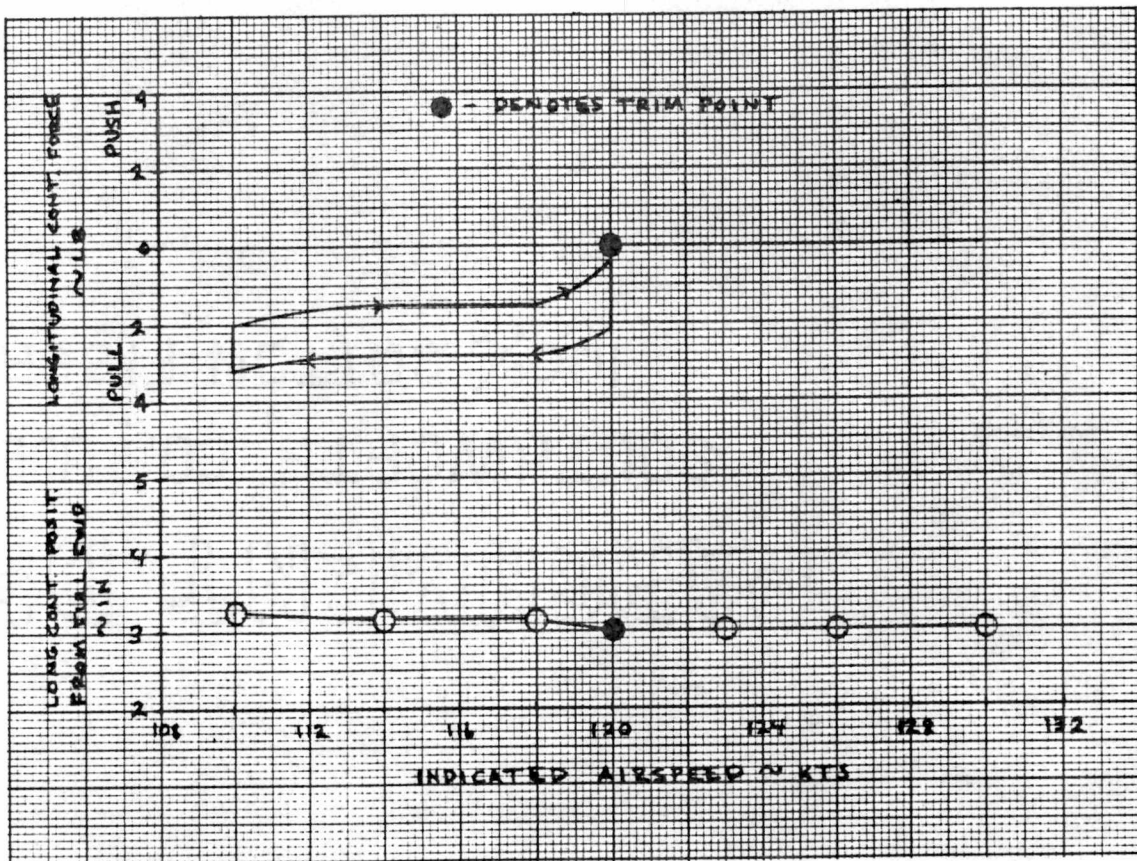


Figure 10
STATIC LONGITUDINAL STABILITY

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR SPEED: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 15,300 LBS.
AVG. CG: 203.7 IN.
AVG. PA: 2200 FT.
OAT: 22°C
METHOD: SHSS

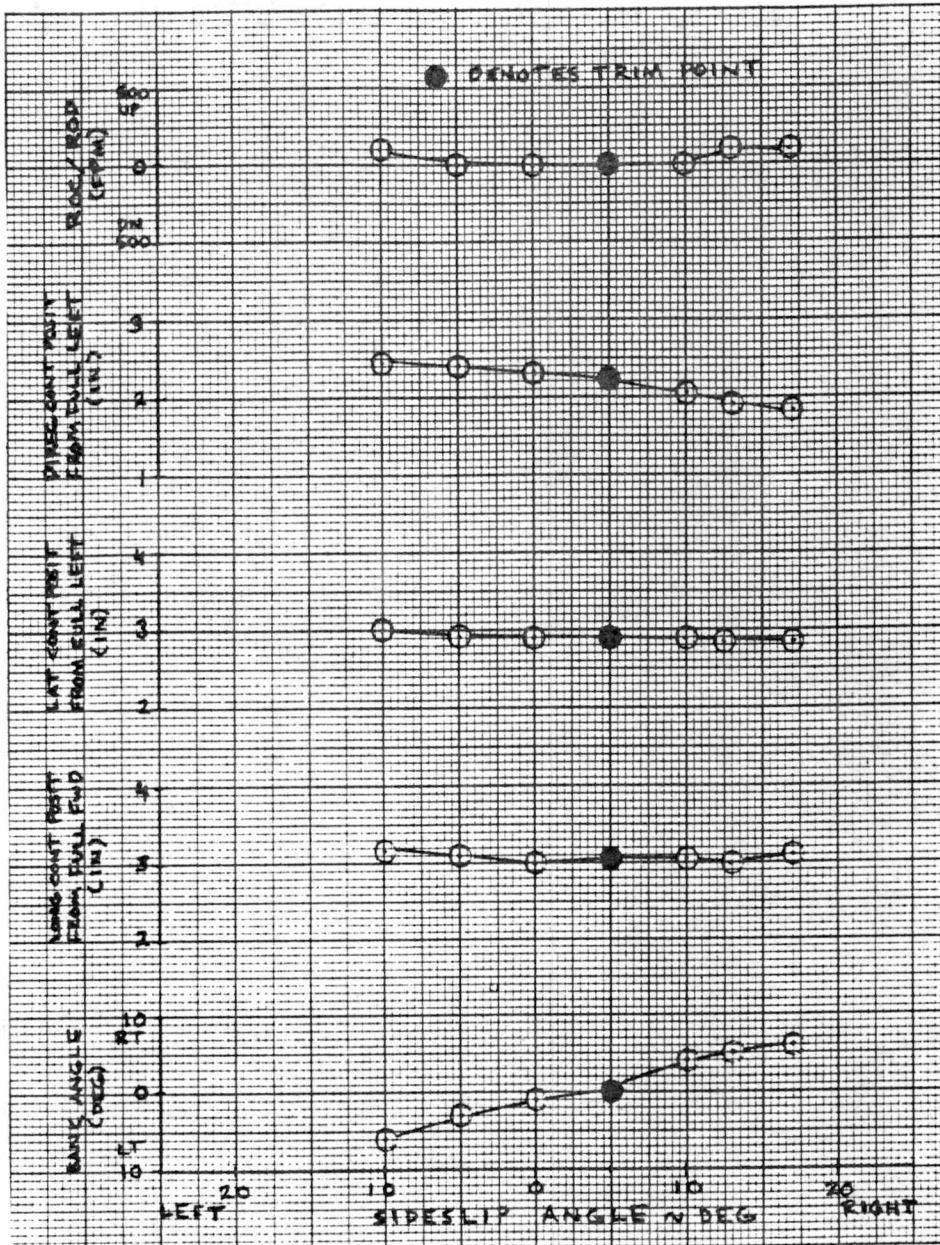


Figure 11

STATIC LATERAL-DIRECTIONAL STABILITY

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR SPEED: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 14,700 LBS.
AVG. CG: 207.3 IN.
AVG. PA: 3000 FT.
OAT: 20°C
METHOD: PULLUPS & PUSHOVERS

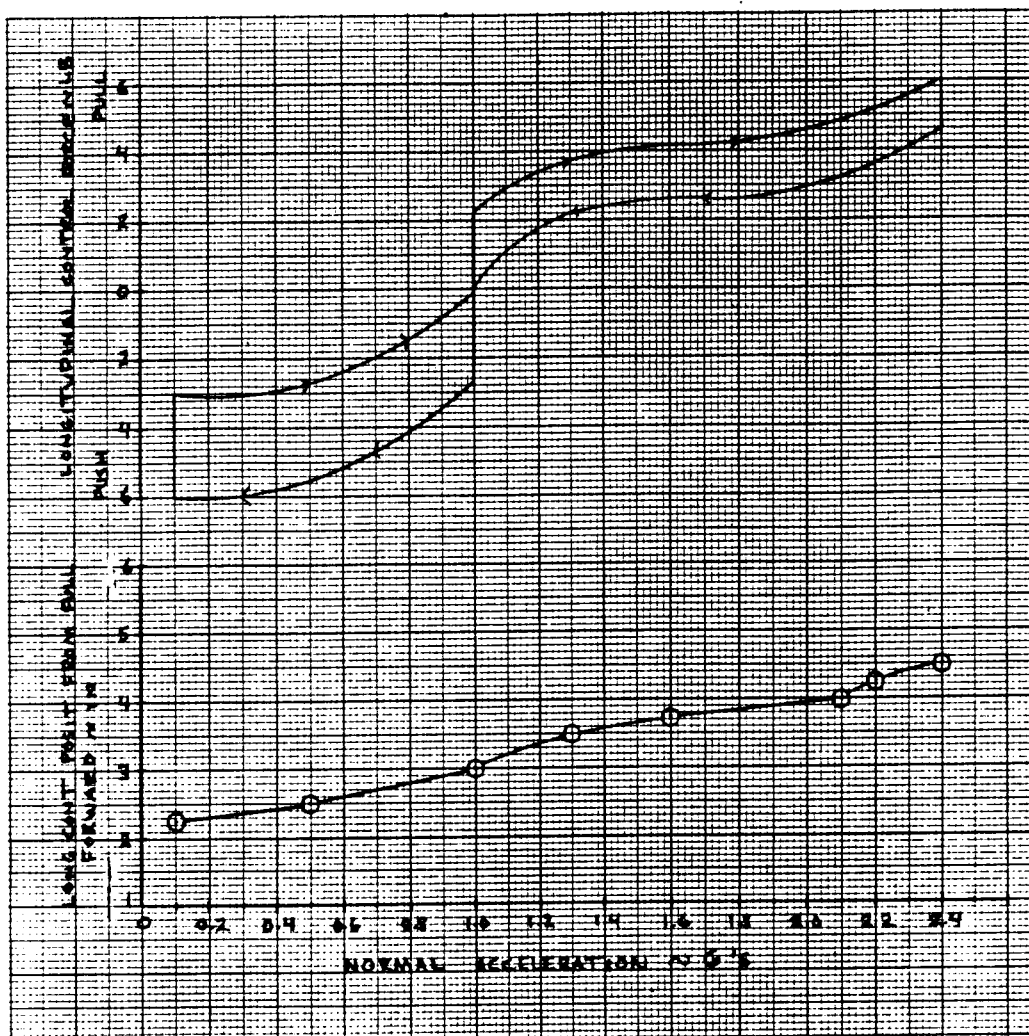


Figure 12
MANEUVERING STABILITY

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR RPM: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 14,900 LBS.
AVG. CG: 203.7 IN.
AVG. PA: 1700 FT.
OAT: 22°C
METHOD: STEADY TURNS

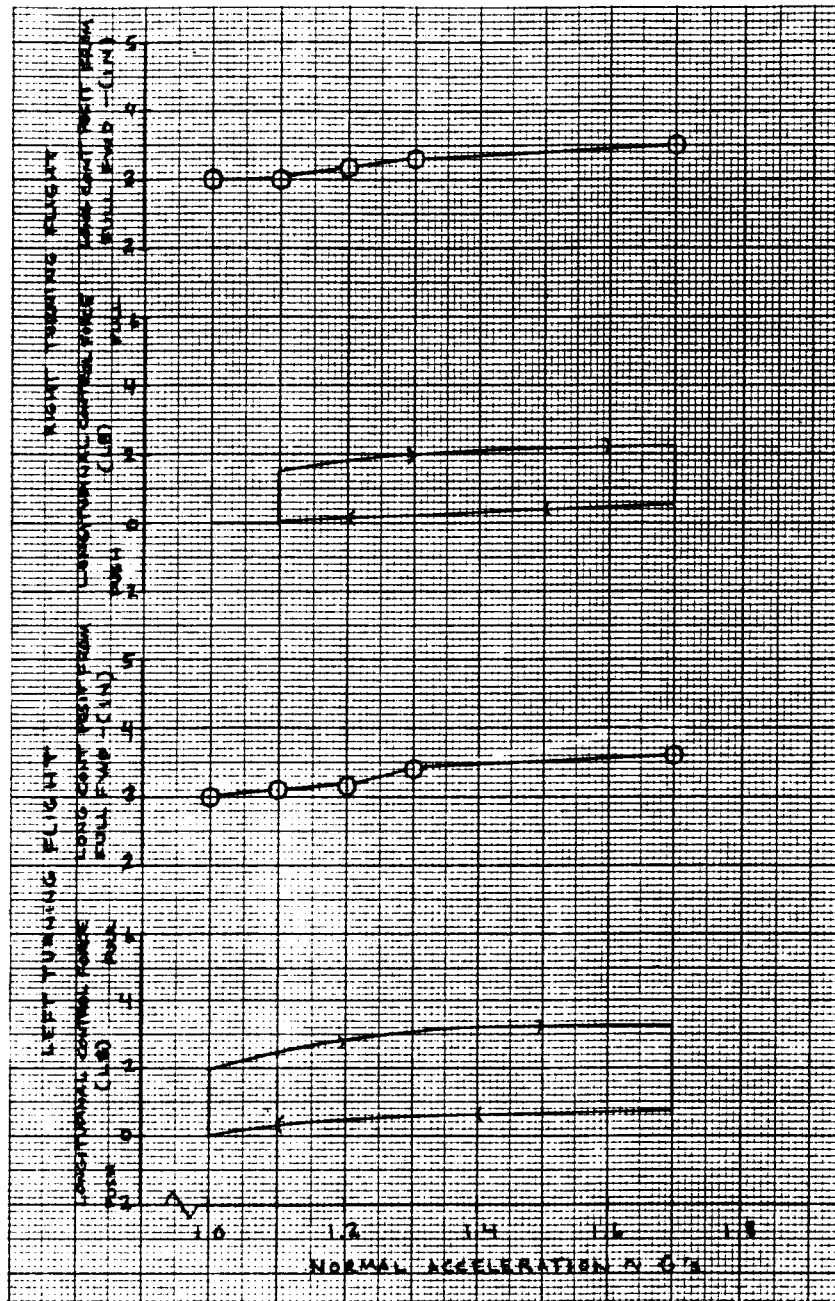


Figure 13

MANEUVERING STABILITY (STEADY TURNS)

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR RPM: 100%

GROSS WEIGHT: 14,540 LBS.
CG: 203.7 IN.
PA: 2000 FT.
OAT: 21°C
EXCITATION: NATURAL

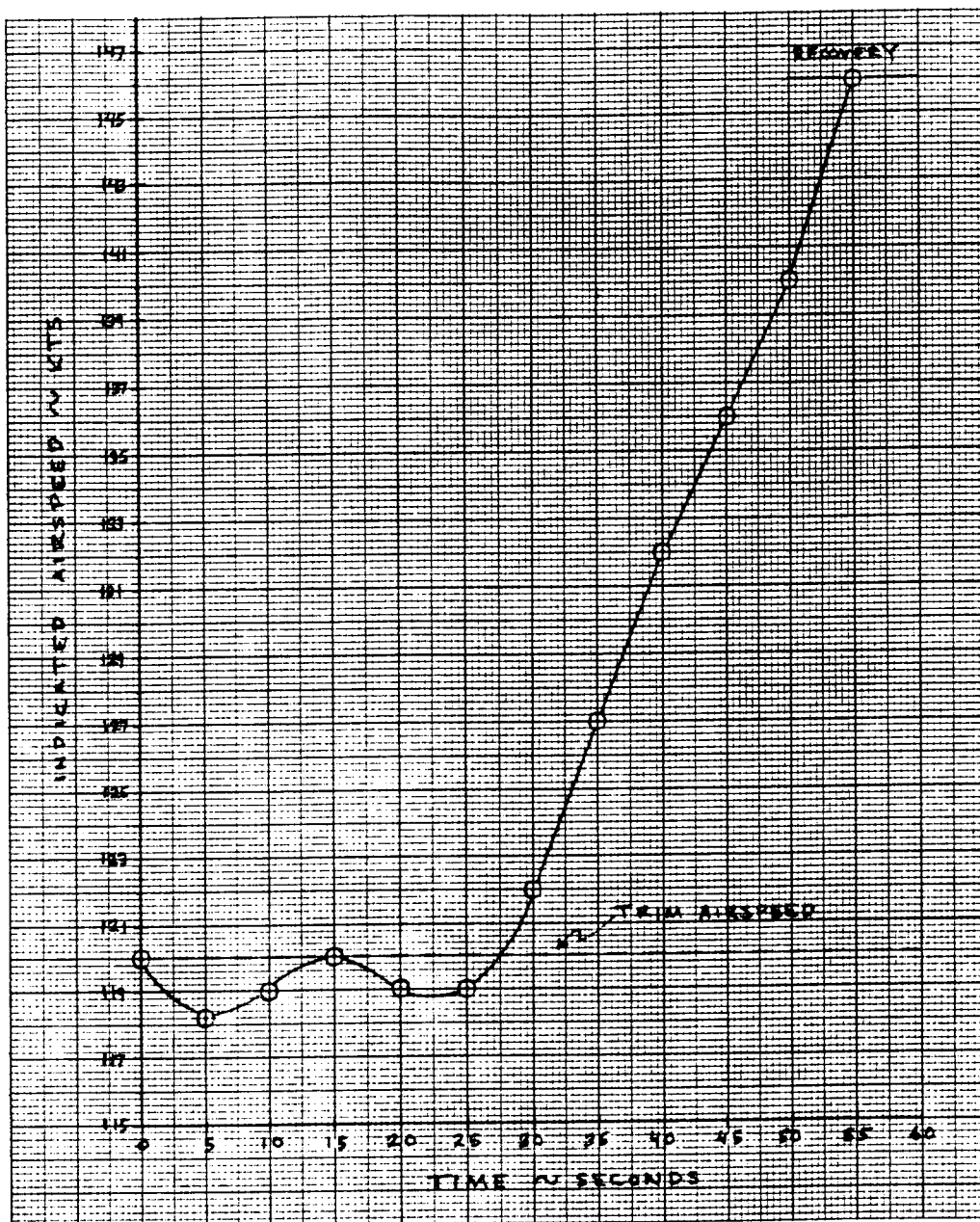


Figure 14
LONG TERM RESPONSE

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR SPEED: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 14,650 LBS.
AVG. CG: 203.7 IN.
AVG. PA: 2000 FT.
OAT: 20°C
METHOD: STEP INPUTS

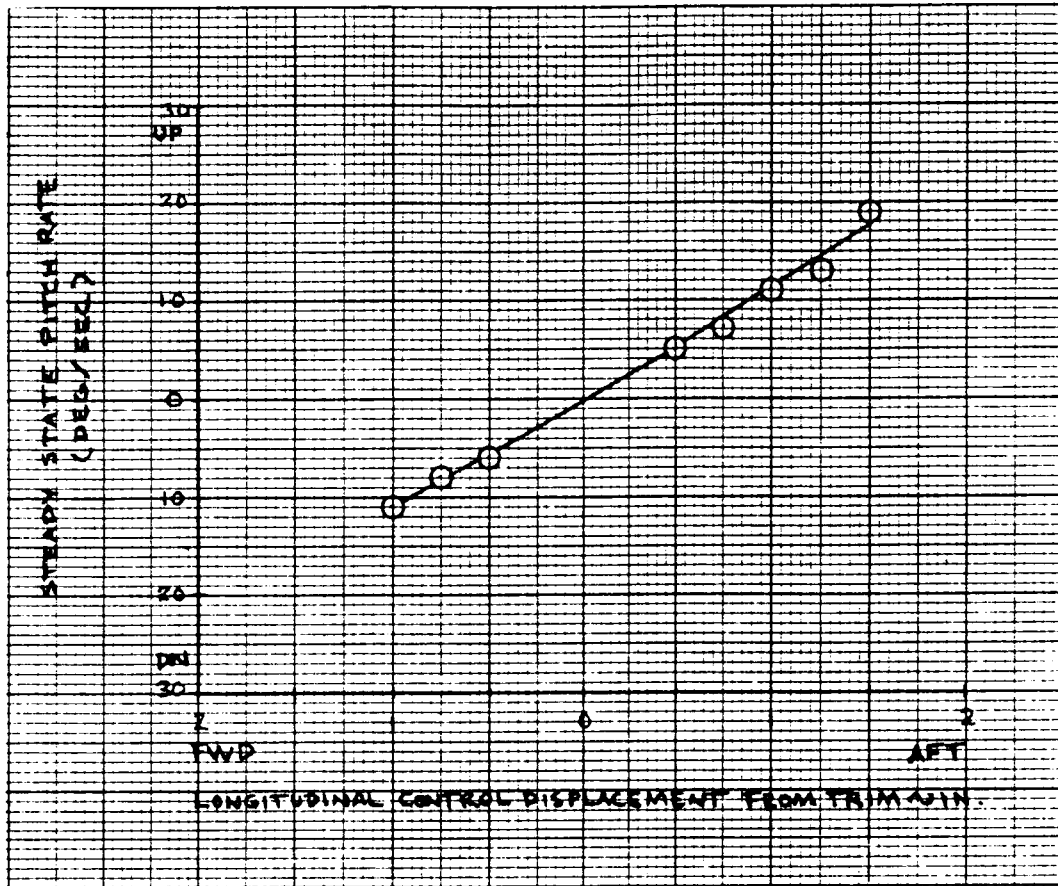


Figure 15

LONGITUDINAL CYCLIC CONTROL RESPONSE

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: OFF
ROTOR SPEED: 100%
AIRSPEED: 120 KIAS

AVG. GROSS WEIGHT: 14,450 LBS.
AVG. CG: 203.7 IN.
AVG. PA: 2000 FT.
OAT: 20°C
METHOD: STEP INPUTS

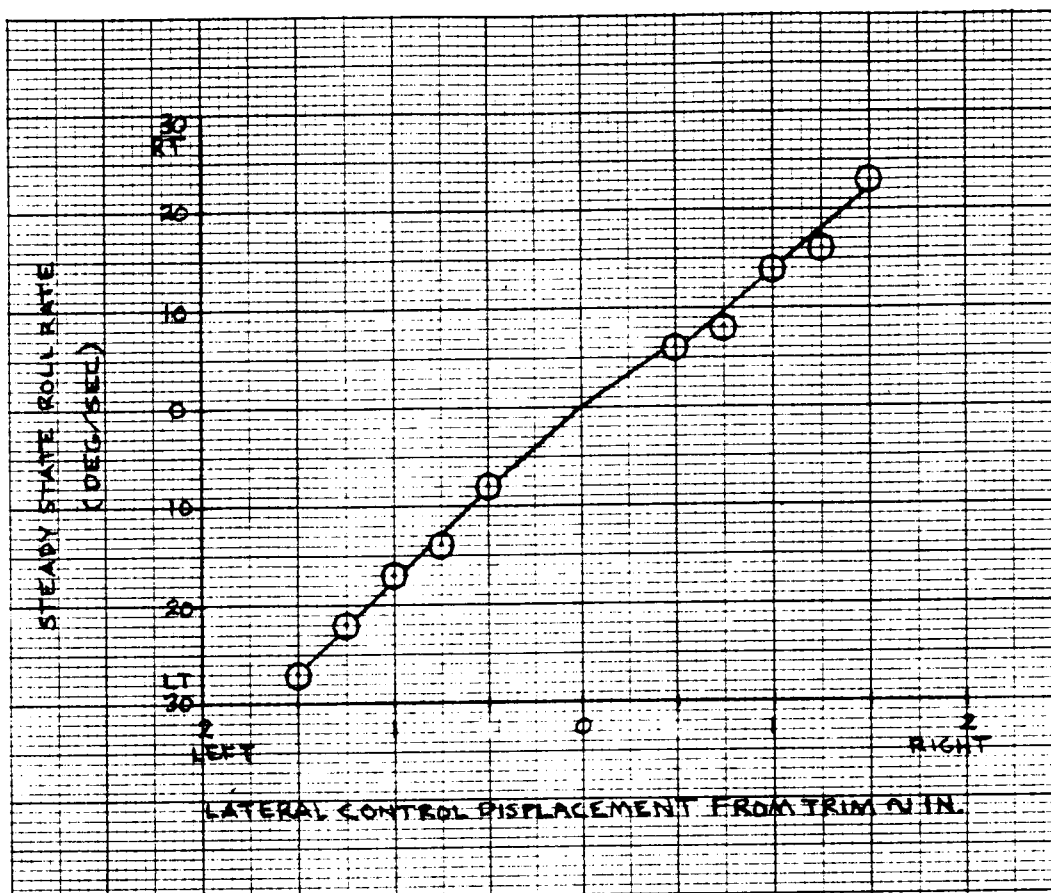


Figure 16

LATERAL CYCLIC CONTROL RESPONSE

MODEL AH-64A HELICOPTER

SN: 83-23788

ENGINE: TWO T700-GE-701

FUEL: JP-4

SAS/TRIM: ON

ATTD/HOVER HOLD: ON

ROTOR RPM: 100%

AVG. GROSS WEIGHT: 16,250 LBS.

AVG. CG: 204.2 IN.

PRESSURE ALTITUDE: 50 FT.

OAT: 28°C

METHOD: AIR DATA SENSOR
REFERENCE

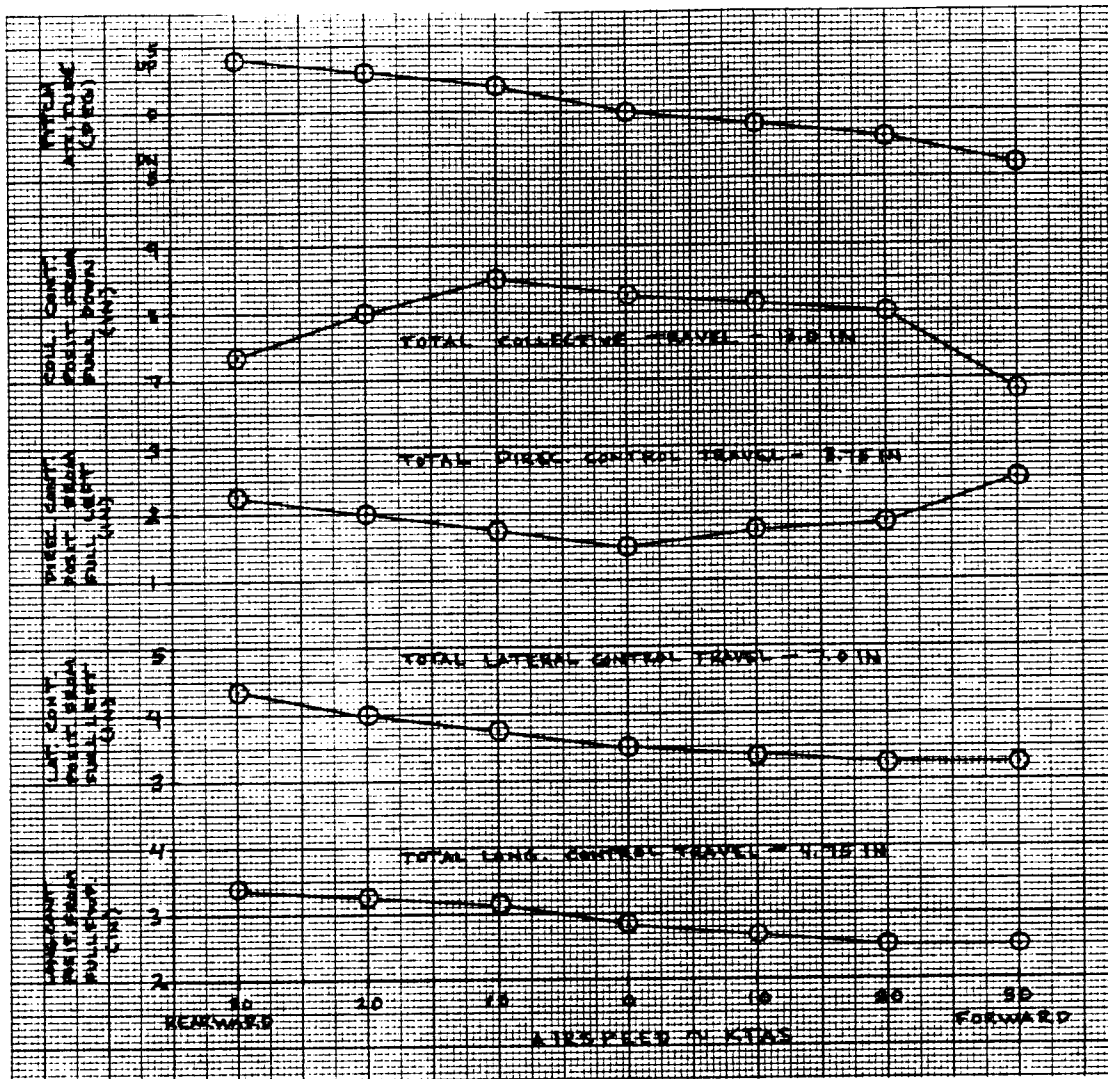


Figure 17

LOW AIRSPEED FORWARD AND REARWARD TRIMMED FLIGHT CONTROL POSITIONS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%

AVG. GROSS WEIGHT: 16,250 LBS.
AVG. CG: 204.2 IN.
PRESSURE ALTITUDE: 50 FT.
OAT: 28°C
METHOD: AIR DATA SENSOR
REFERENCE

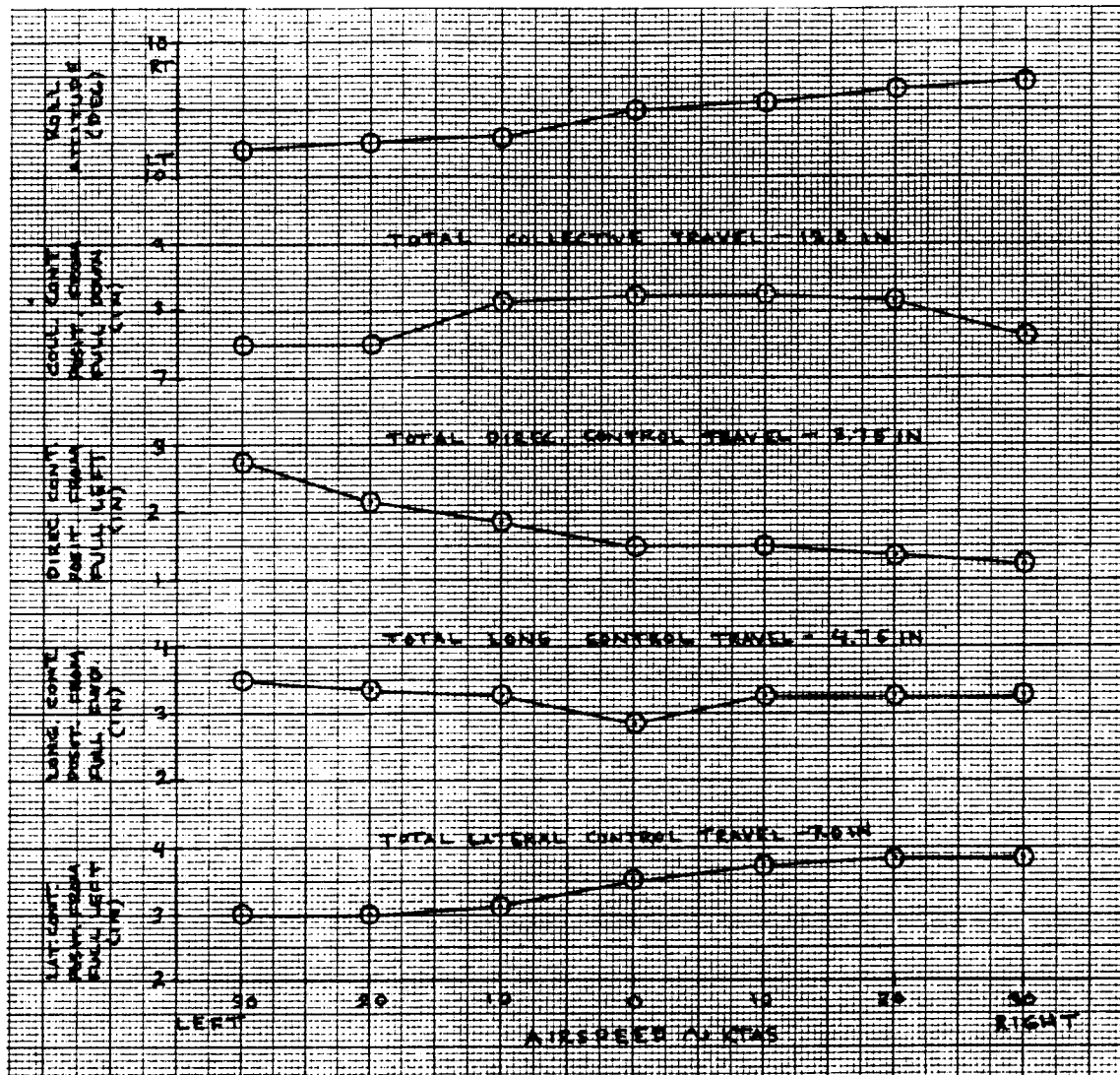


Figure 18

LOW AIRSPEED SIDEWARD TRIMMED FLIGHT CONTROL POSITIONS

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%
AIRSPEED: 20 KTAS

AVG. GROSS WEIGHT: 15,550 LBS.
AVG. CG: 203.9 IN.
PRESSURE ALTITUDE: 50 FT.
OAT: 28°C
METHOD: AIR DATA SENSOR
REFERENCE

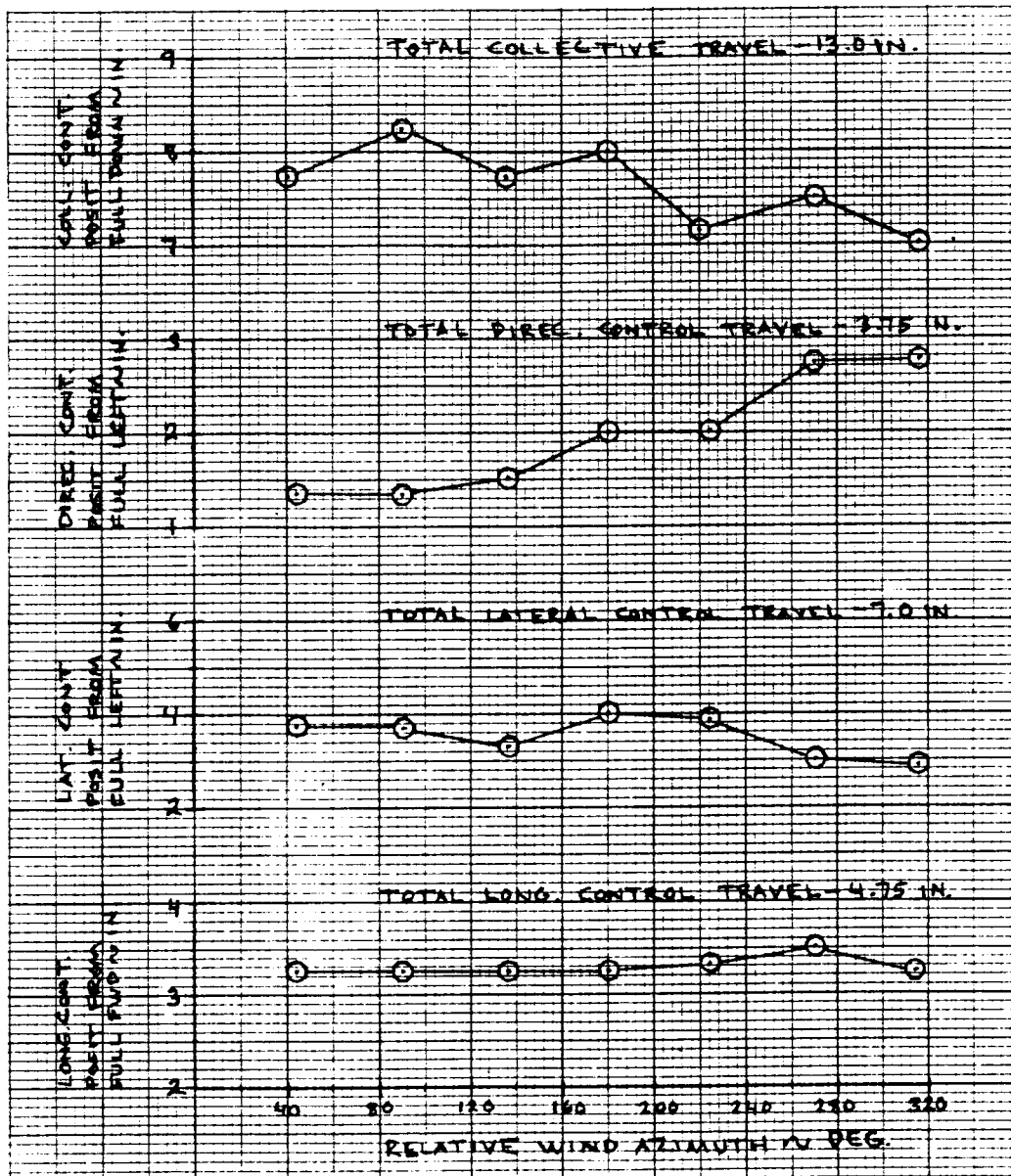


Figure 19

CRITICAL AZIMUTH DETERMINATION

SN: 83-23788

AIRSPEED: 20 KTAS

METHOD: AIR DATA SENSOR
REFERENCE



Figure 20

CRITICAL AZIMUTH

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%

AVG. GROSS WEIGHT: 15,250 LBS.
AVG. CG: 204.1 IN.
PRESSURE ALTITUDE: 50 FT.
OAT: 28°C
METHOD: AIR DATA SENSOR
REFERENCE, COLLECTIVE
FIXED

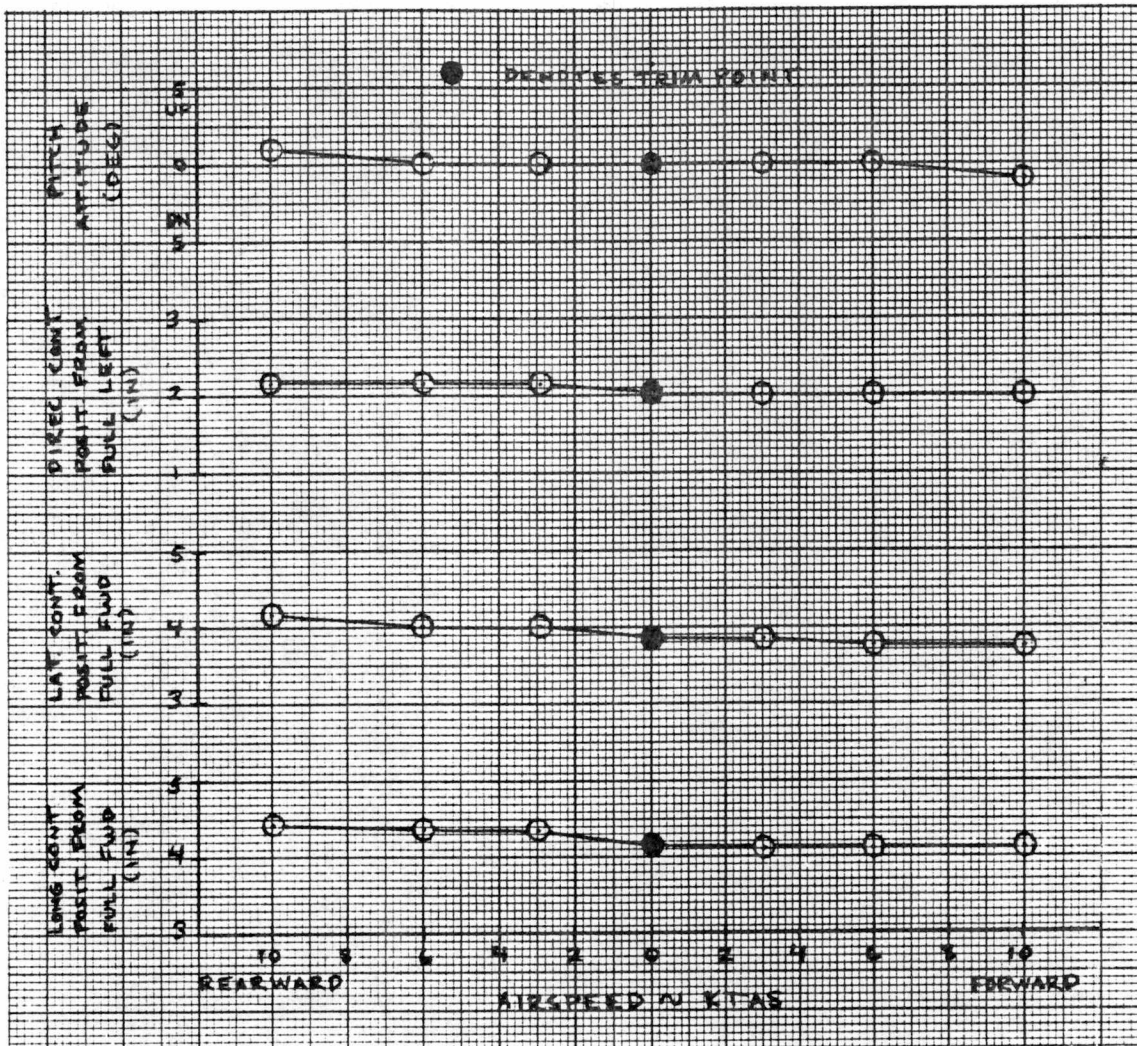


Figure 21
LOW AIRSPEED STATIC LONGITUDINAL STABILITY

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
SAS/TRIM: ON
ATTD/HOVER HOLD: ON
ROTOR RPM: 100%

AVG. GROSS WEIGHT: 15,250 LBS.
AVG. CG: 204.1 IN.
PRESSURE ALTITUDE: 50 FT.
OAT: 28°C
METHOD: AIR DATA SENSOR
REFERENCE, COLLECTIVE
FIXED

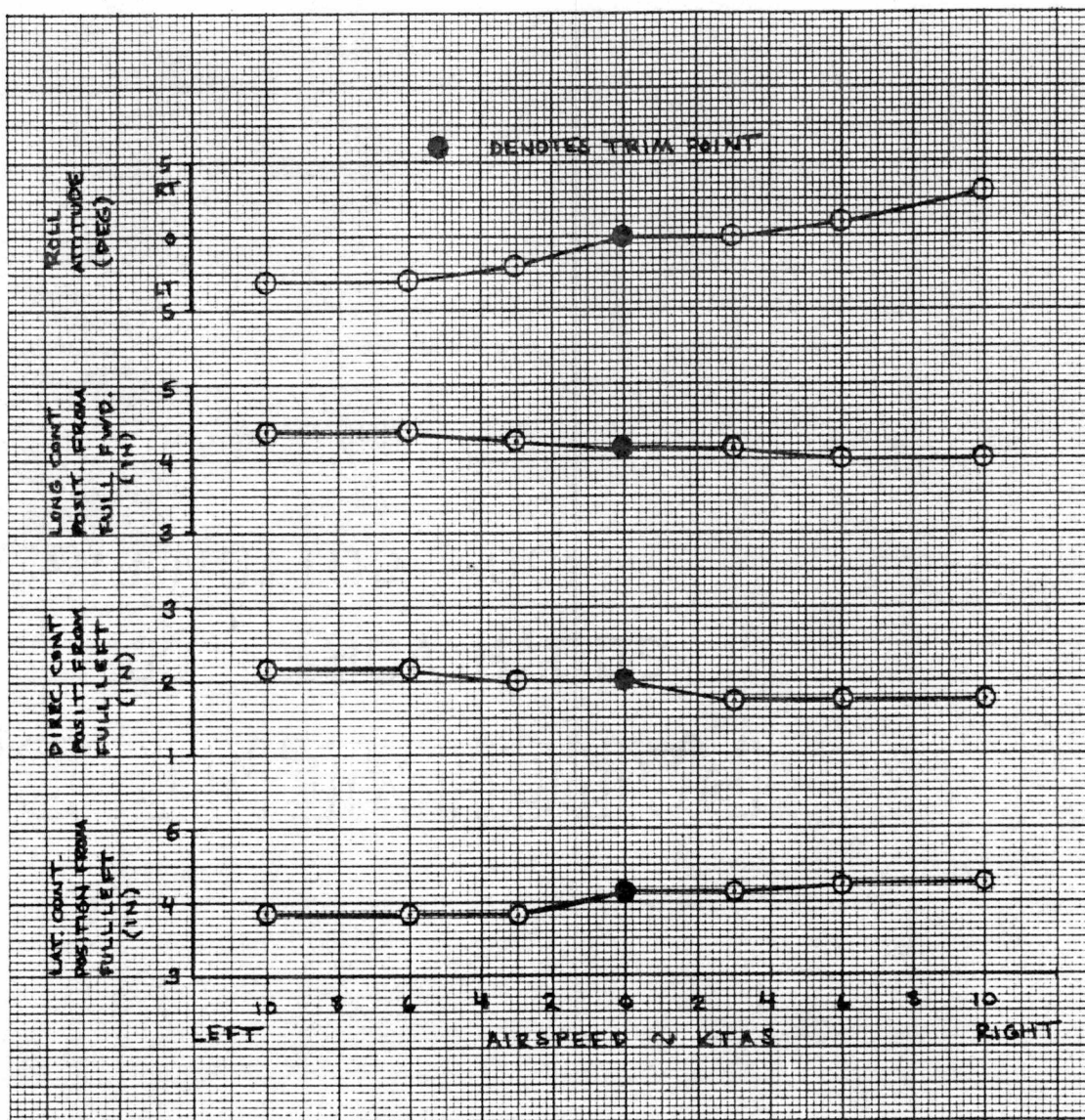


Figure 22
LOW AIRSPEED LATERAL-DIRECTIONAL STATIC STABILITY

MODEL AH-64A HELICOPTER

SN: 83-23788

ENGINE: TWO T700-GE-701

NO. 1 SER. NO: 374142

NO. 2 SER. NO: 374141

FUEL: JP-4

ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.

AVG CG: 203.7 IN.

PA: 20 to 2500 FT.

OAT: 21 to 27°C

ROTOR SPEED: 100%

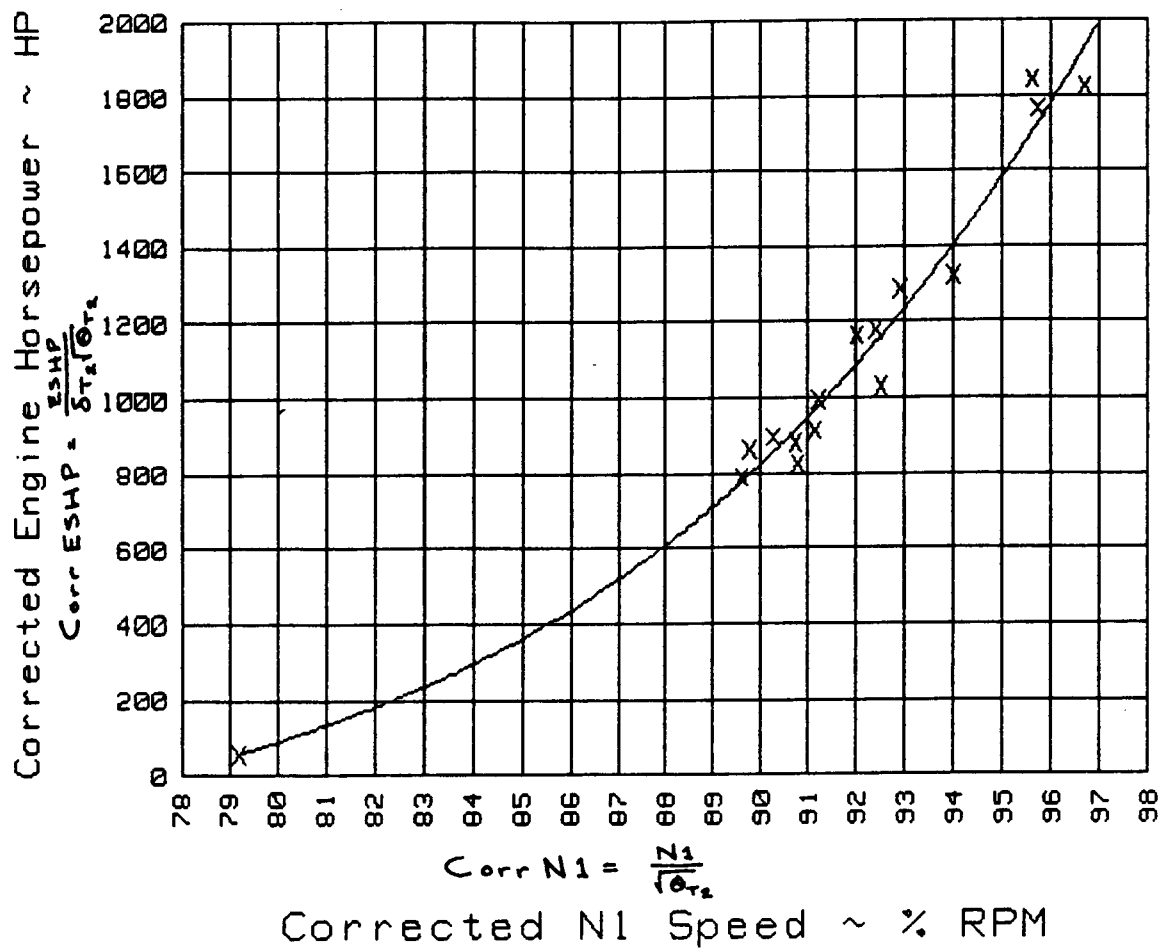


Figure 23

CORRECTED ENGINE SHAFT HORSEPOWER

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
NO. 1 SER. NO: 374142
NO. 2 SER. NO: 374141
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
PA: 20 to 2500 FT.
OAT: 21 to 27°C
ROTOR SPEED: 100%

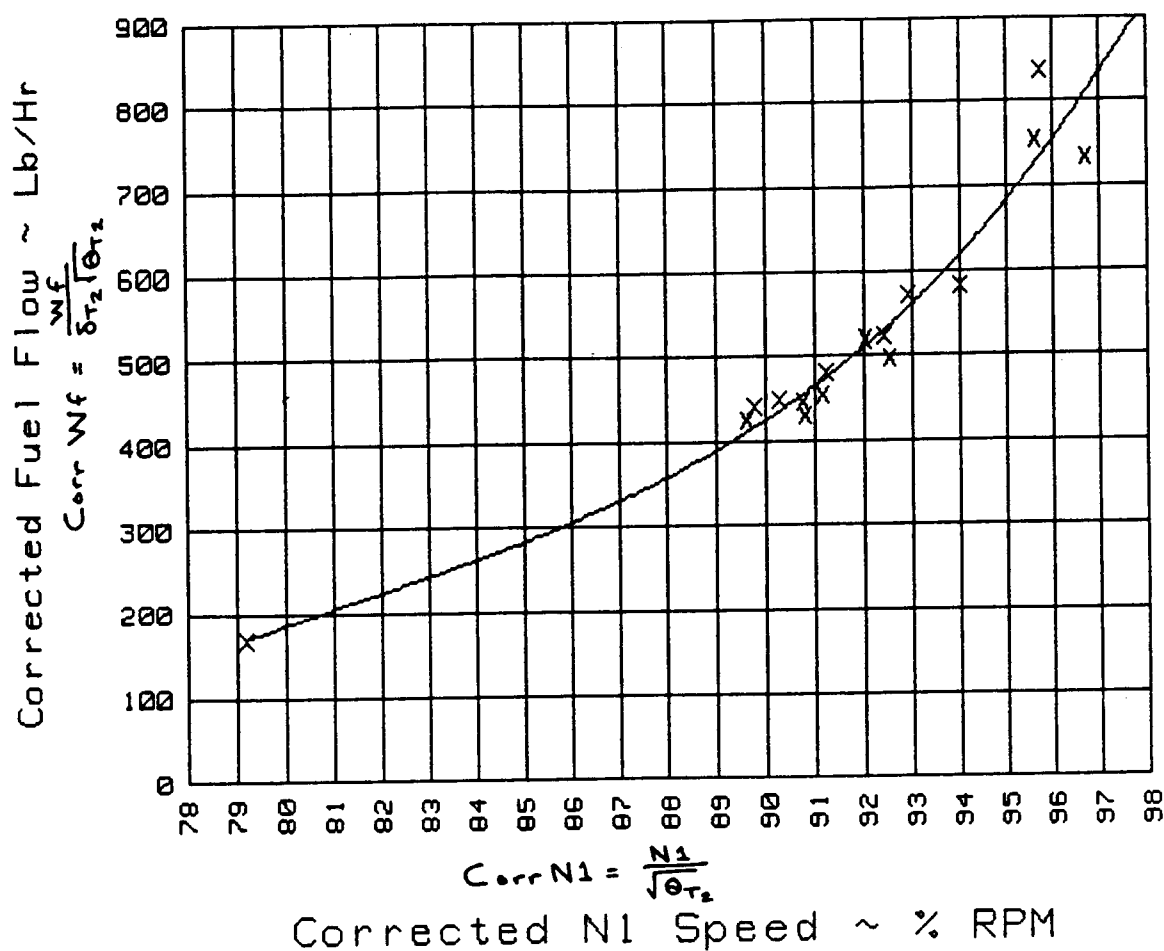


Figure 24

CORRECTED FUEL FLOW

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
NO. 1 SER. NO: 374142
NO. 2 SER. NO: 374141
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
PA: 20 to 2500 FT.
OAT: 21 to 27°C
ROTOR SPEED: 100%

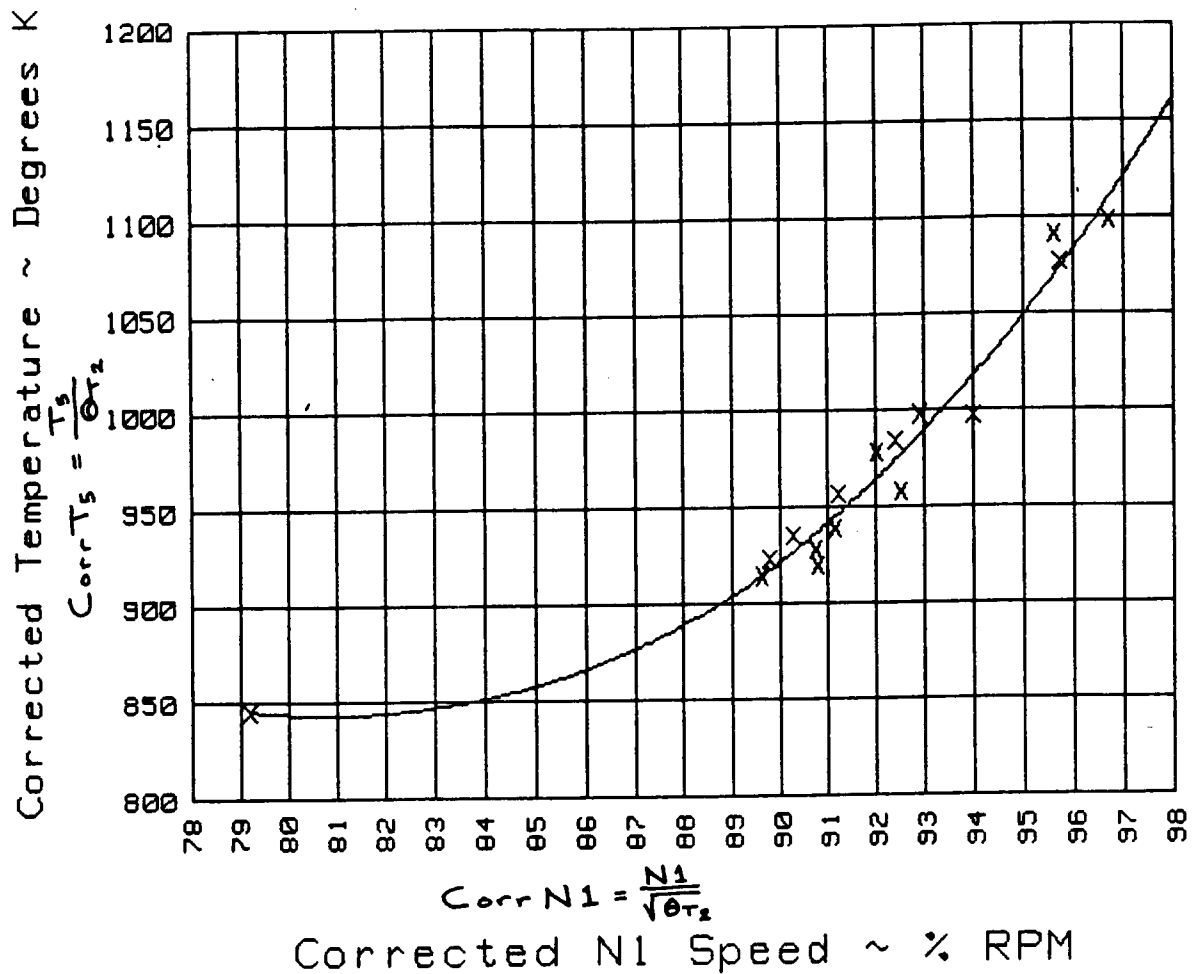


Figure 25

CORRECTED ENGINE TEMPERATURE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
NO. 1 SER. NO: 374142
NO. 2 SER. NO: 374141
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
PA: 20 to 2500 FT.
OAT: 21 to 27°C
ROTOR SPEED: 100%

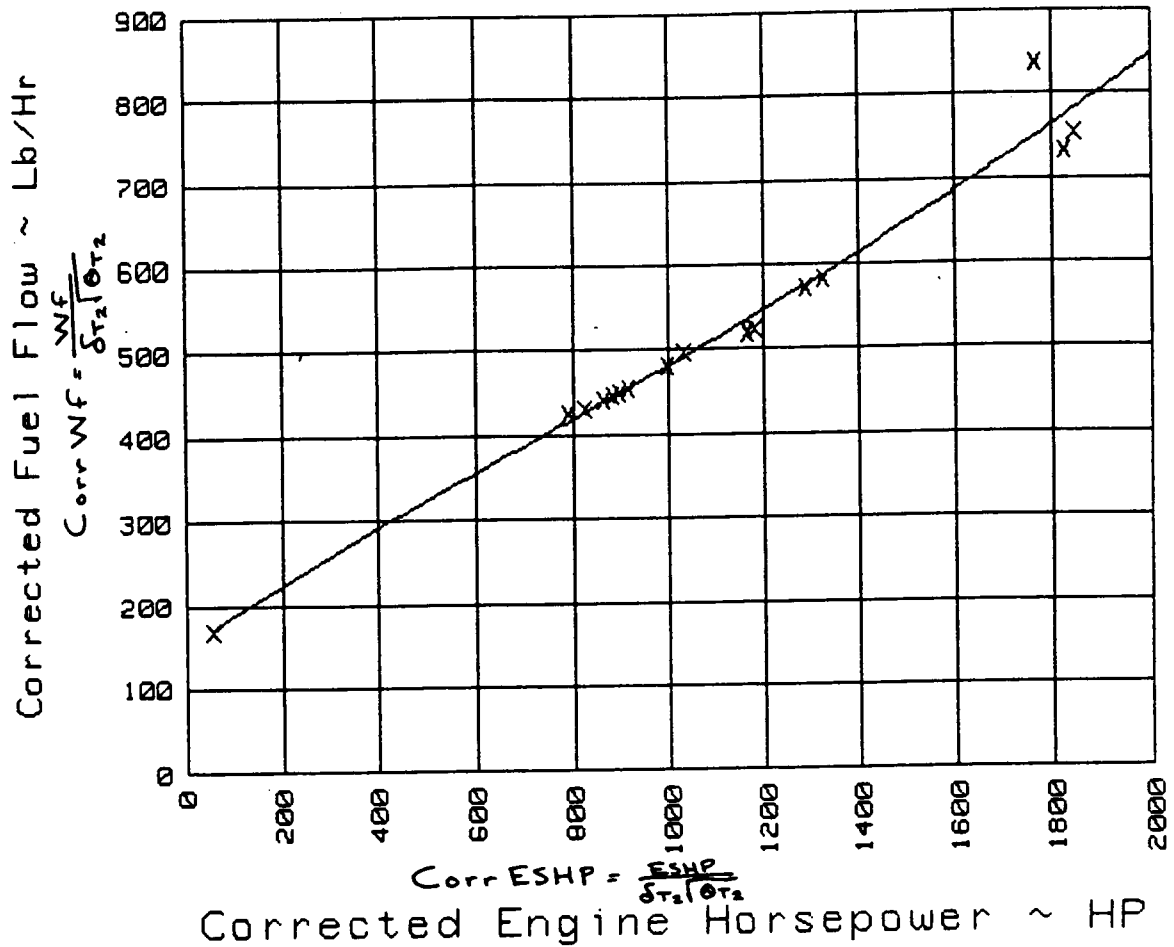


Figure 26
CORRECTED FUEL FLOW VS. CORRECTED ESHP

MODEL AH-64A HELICOPTER

SN: 83-23788

ENGINE: TWO T700-GE-701
NO. 1 SER. NO: 374142
NO. 2 SER. NO: 374141
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
PA: 20 to 2500 FT.
OAT: 21 to 27°C
ROTOR SPEED: 100%

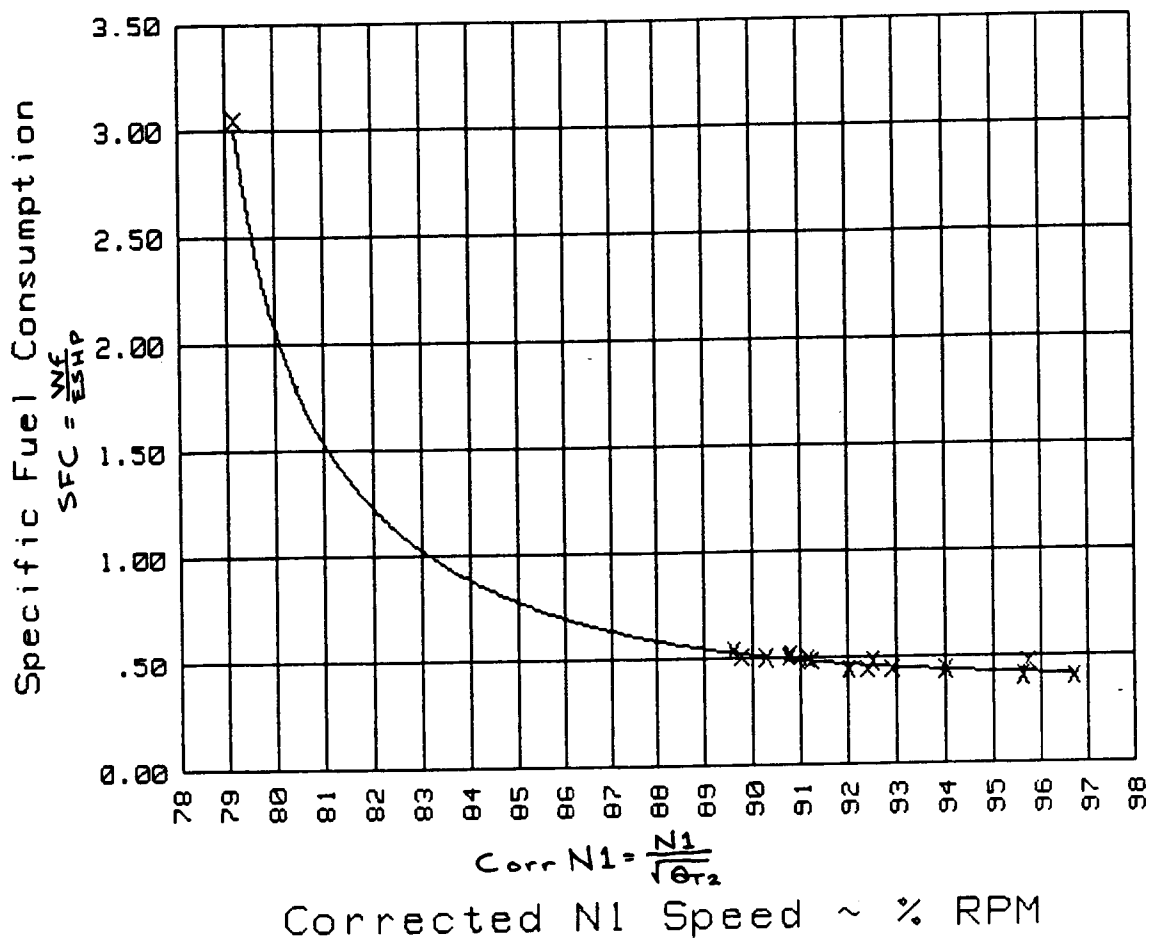


Figure 27
SPECIFIC FUEL CONSUMPTION

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
ROTOR SPEED: 100%

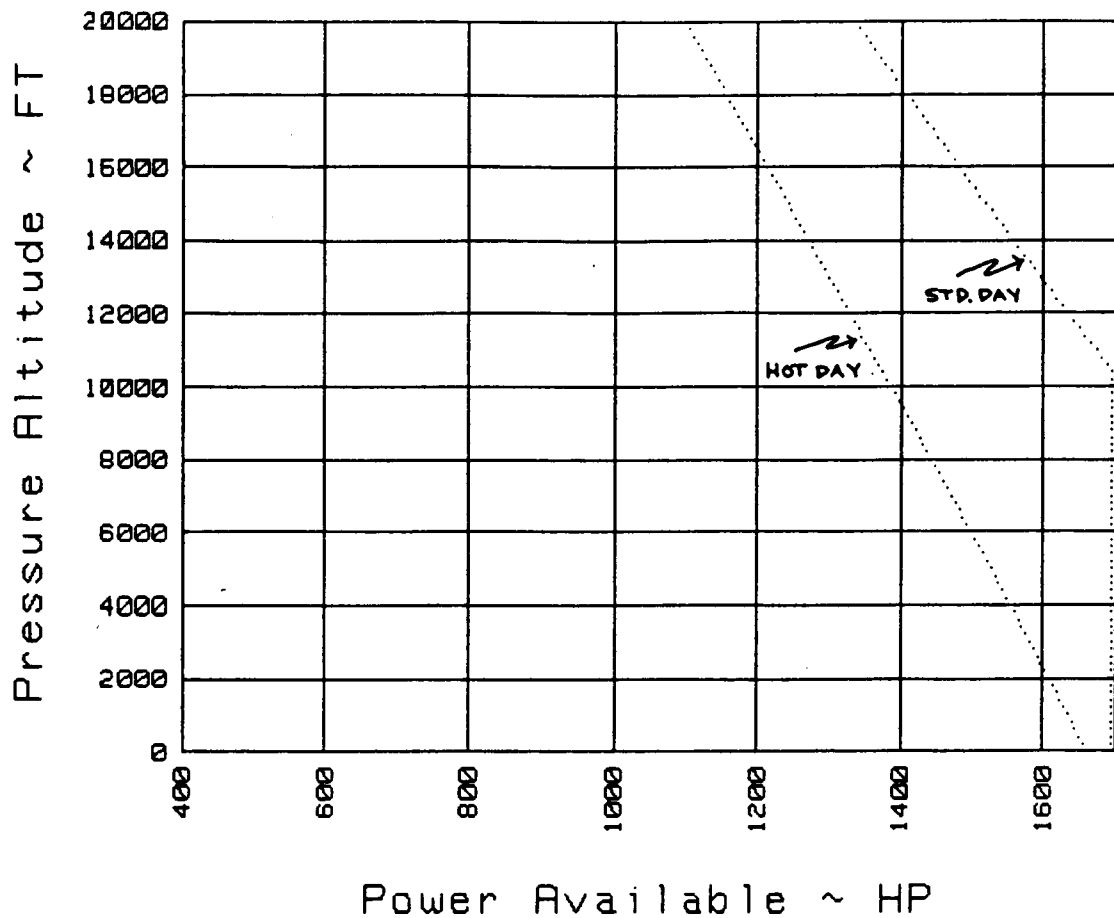


Figure 28

ENGINE POWER AVAILABLE MILITARY POWER (30 MIN)

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

AVG GW: 15,500 LB.
AVG CG: 203.7 IN.
ROTOR SPEED: 100%

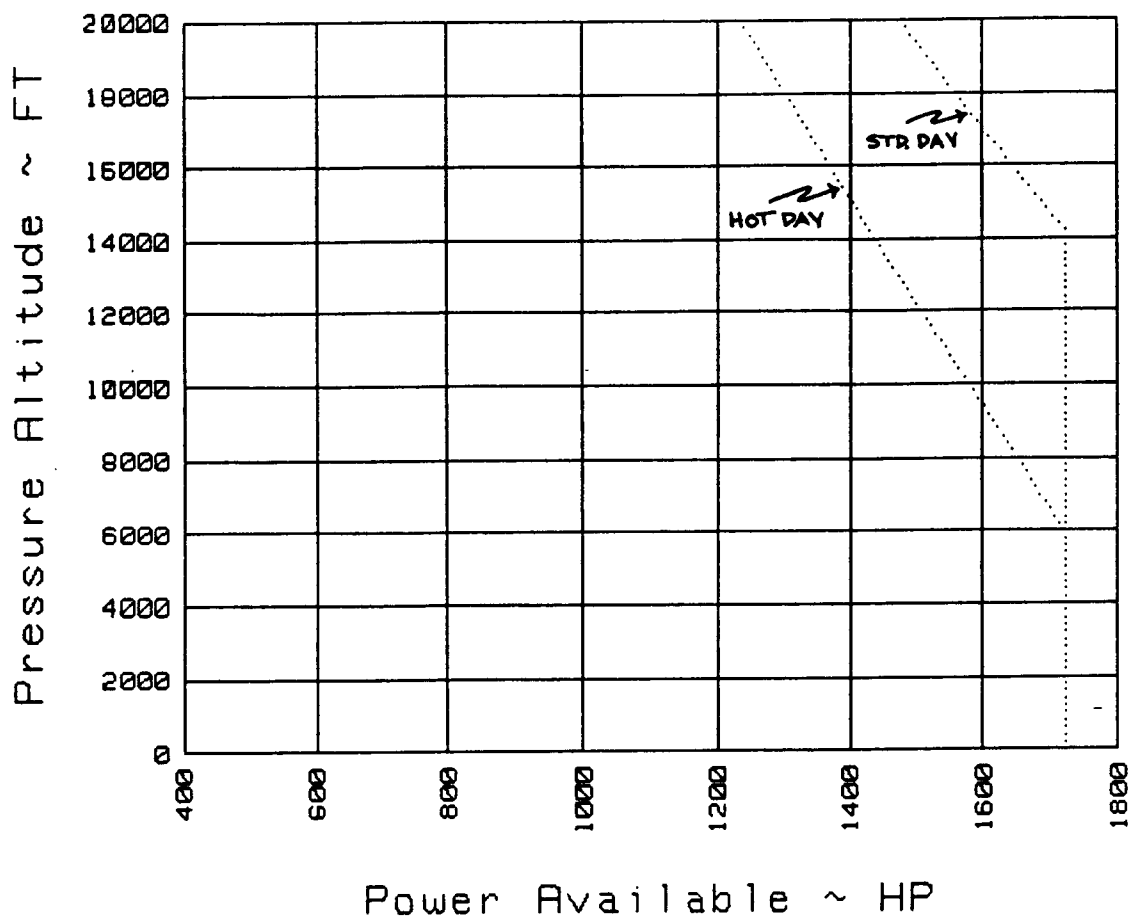


Figure 29

ENGINE POWER AVAILABLE CONTINGENCY POWER (2-1/2 MIN)

APPENDIX F

MODEL AH-64A HELICOPTER

ENGINE: TWO T700-GE-701	GROSS WEIGHT: 16,220 to 14,390 LB.
FUEL: JP-4	CG: 204.8 to 203.7 IN.
ENGINE BLEED AIR: OFF	PA: 0 to 60 FT.
TECHNIQUE: FREE HOVER	OAT: 25 to 28°C
ROTOR SPEED: 100%	

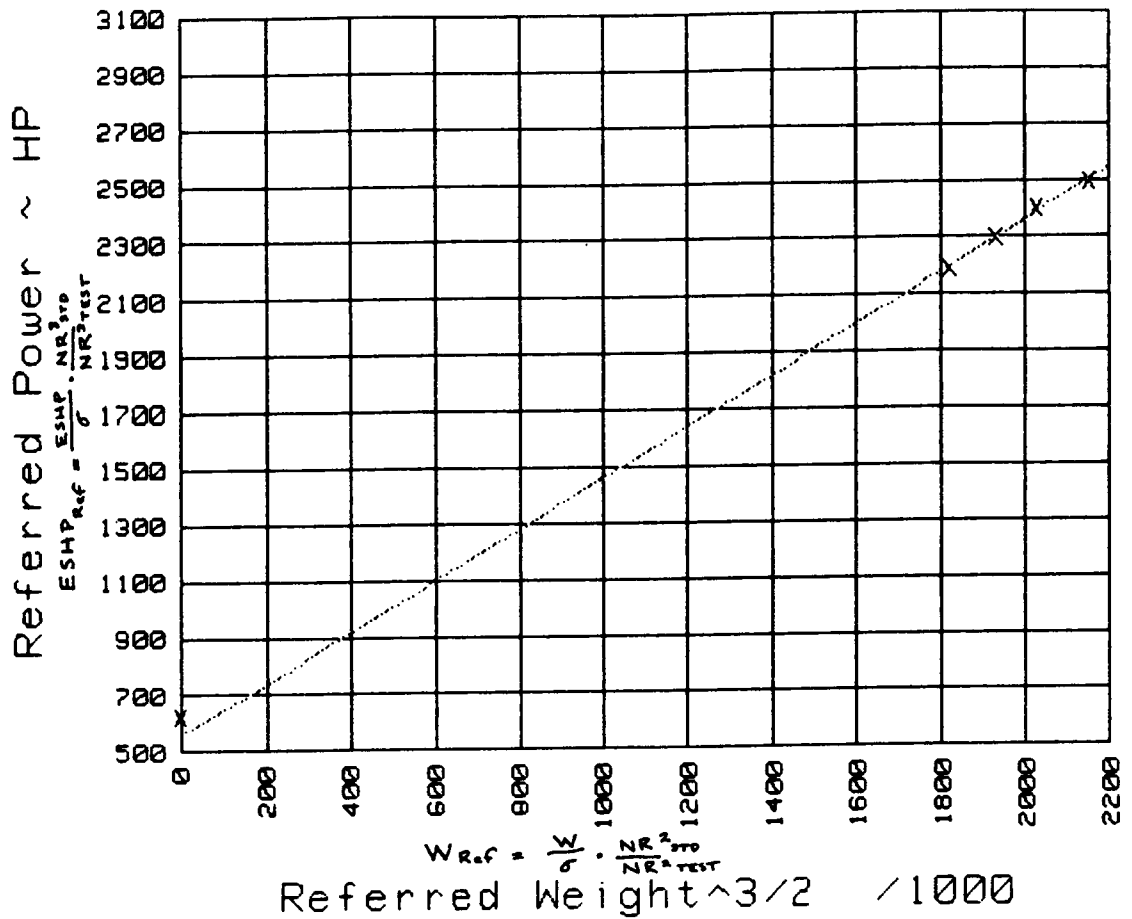


Figure 30

IGE GENERALIZED HOVER PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER

ENGINE: TWO T700-GE-701	GROSS WEIGHT: 16,220 to 14,430 LB.
FUEL: JP-4	CG: 204.8 to 203.7 IN.
ENGINE BLEED AIR: OFF	PA: 60 to 120 FT.
TECHNIQUE: FREE HOVER	OAT: 24 to 28°C
ROTOR SPEED: 100%	

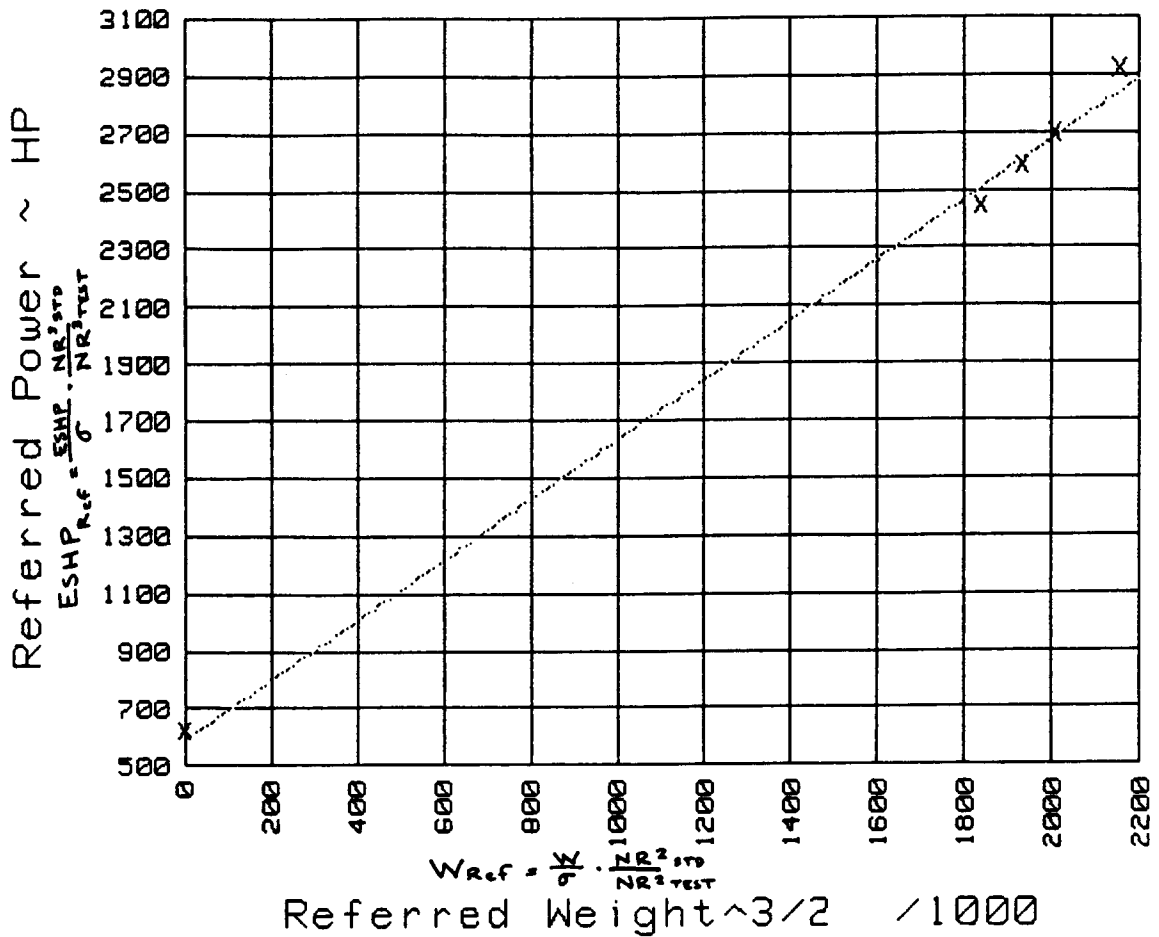


Figure 31

OGE GENERALIZED HOVER PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER

ENGINE: TWO T700-GE-701
 FUEL: JP-4
 ENGINE BLEED AIR: OFF
 TECHNIQUE: FREE HOVER
 ROTOR SPEED: 100%

GROSS WEIGHT: 16,220 to 14,390 LB.
 CG: 204.8 to 203.7 IN.
 PA: 0 to 60 FT.
 OAT: 25 to 28°C

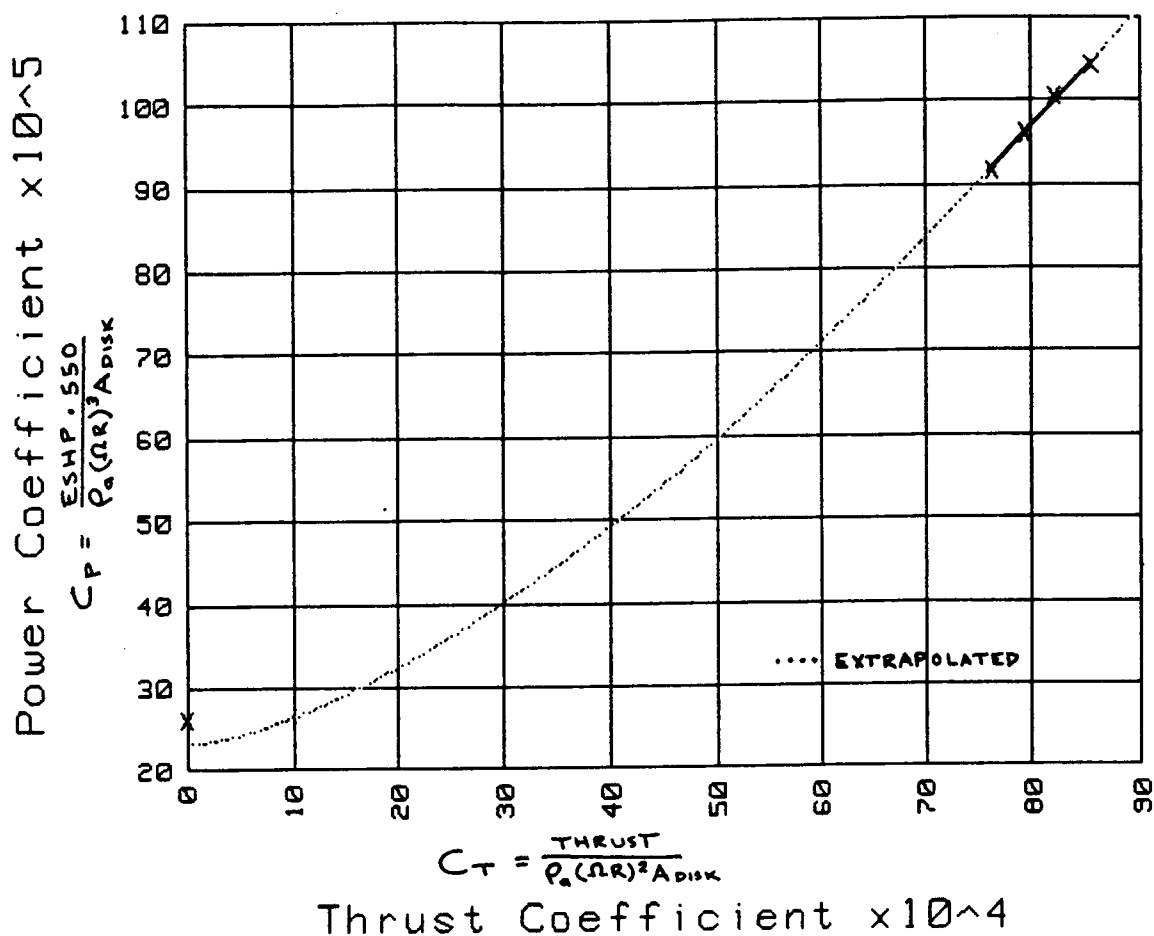


Figure 32

IGE NON-DIMENSIONAL HOVER PERFORMANCE

MODEL AH-64A HELICOPTER

ENGINE: TWO T700-GE-701	GROSS WEIGHT: 16,220 to 14,430 LB.
FUEL: JP-4	CG: 204.8 to 203.7 IN.
ENGINE BLEED AIR: OFF	PA: 0 to 60 FT.
TECHNIQUE: FREE HOVER	OAT: 25 to 28°C
ROTOR SPEED: 100%	

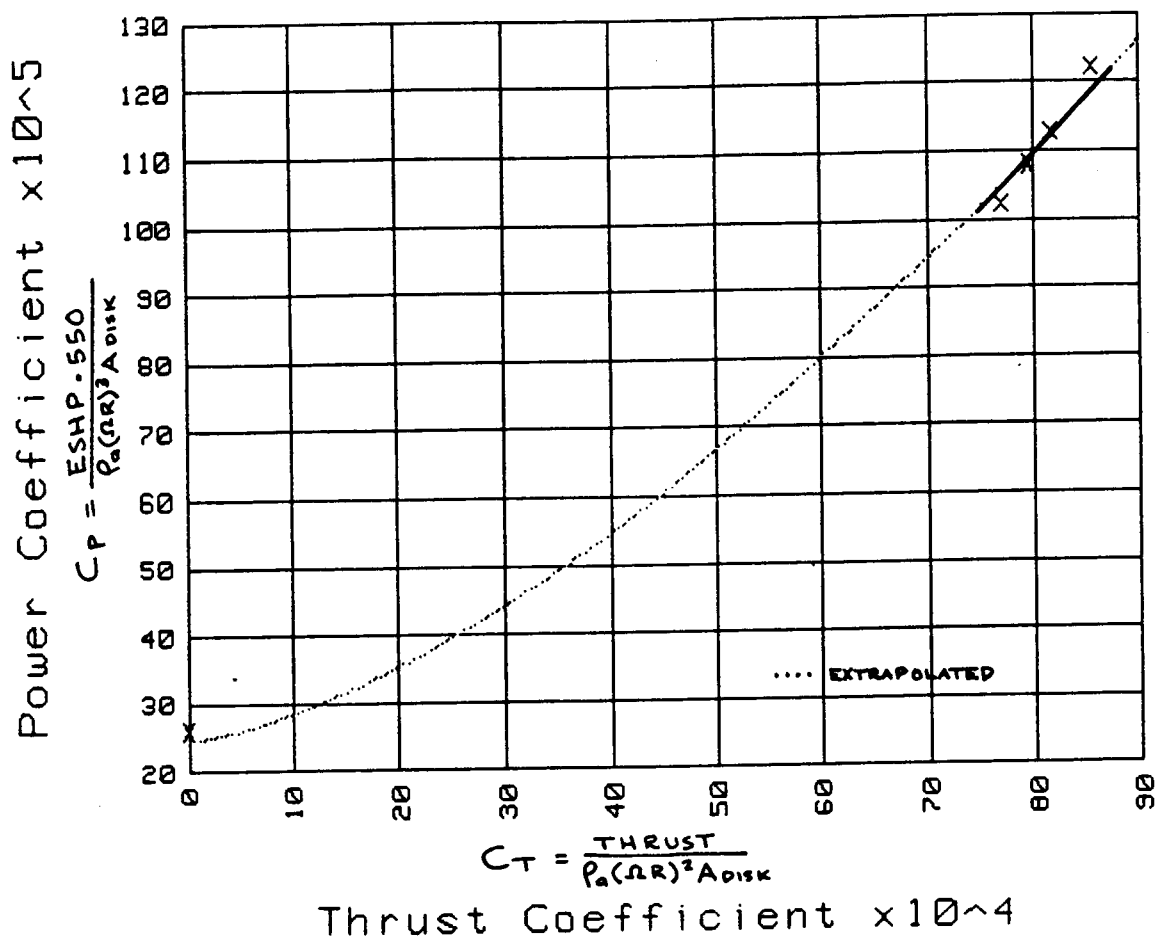


Figure 33

OGE NON-DIMENSIONAL HOVER PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 16,170 LB.
FUEL: JP-4 CG: 203.9 IN.
ENGINE BLEED AIR: OFF OAT: 30°F
ROTOR SPEED: 100% PRESSURE ALTITUDE: 30 to 100 FT.
TECHNIQUE: FREE HOVER

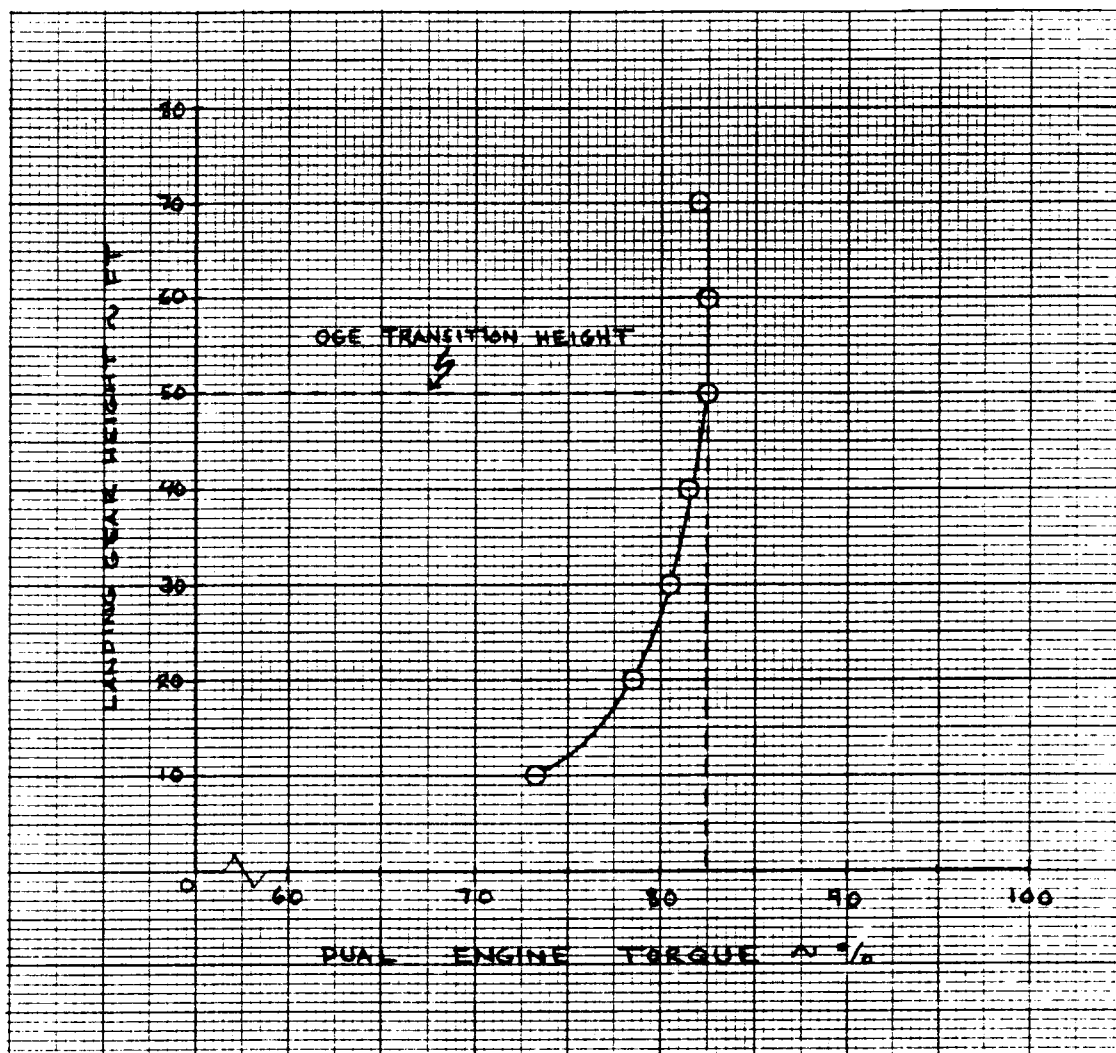


Figure 34
IGE/OGE TRANSITION DETERMINATION

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

ROTOR SPEED: 100%
OAT: STANDARD (15°C)

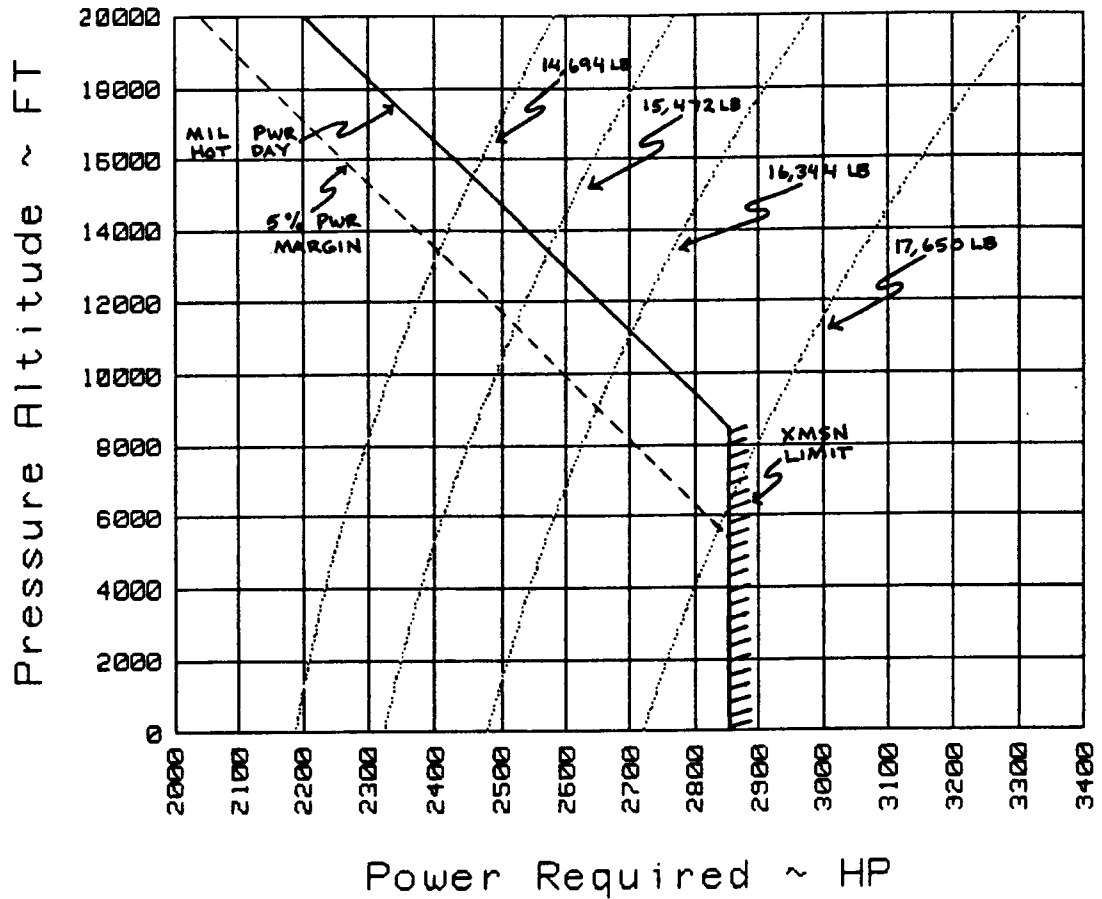


Figure 35

STANDARD DAY 10 FT. IGE HOVER POWER

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

ROTOR SPEED: 100%
OAT: 95°F

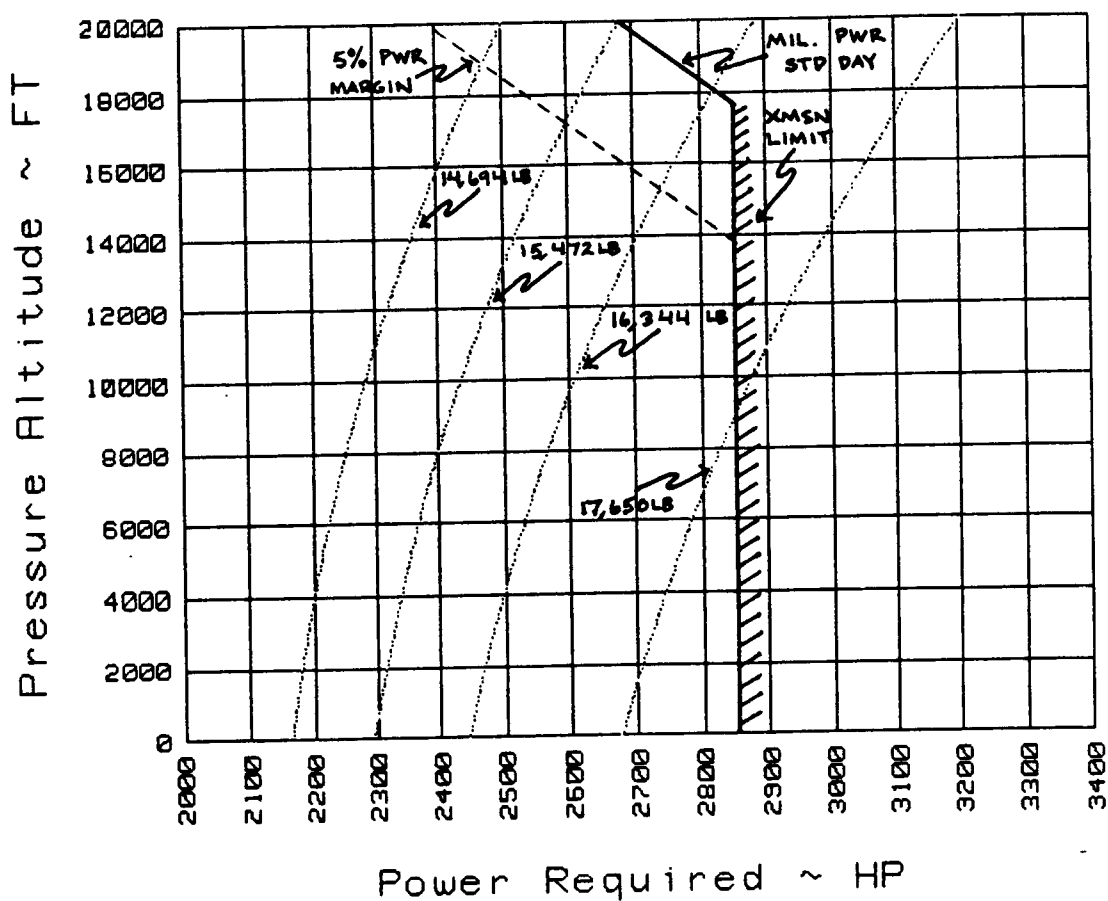


Figure 36

HOT DAY 10 FT. IGE HOVER POWER

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4

ENGINE BLEED AIR: OFF
ROTOR SPEED: 100%

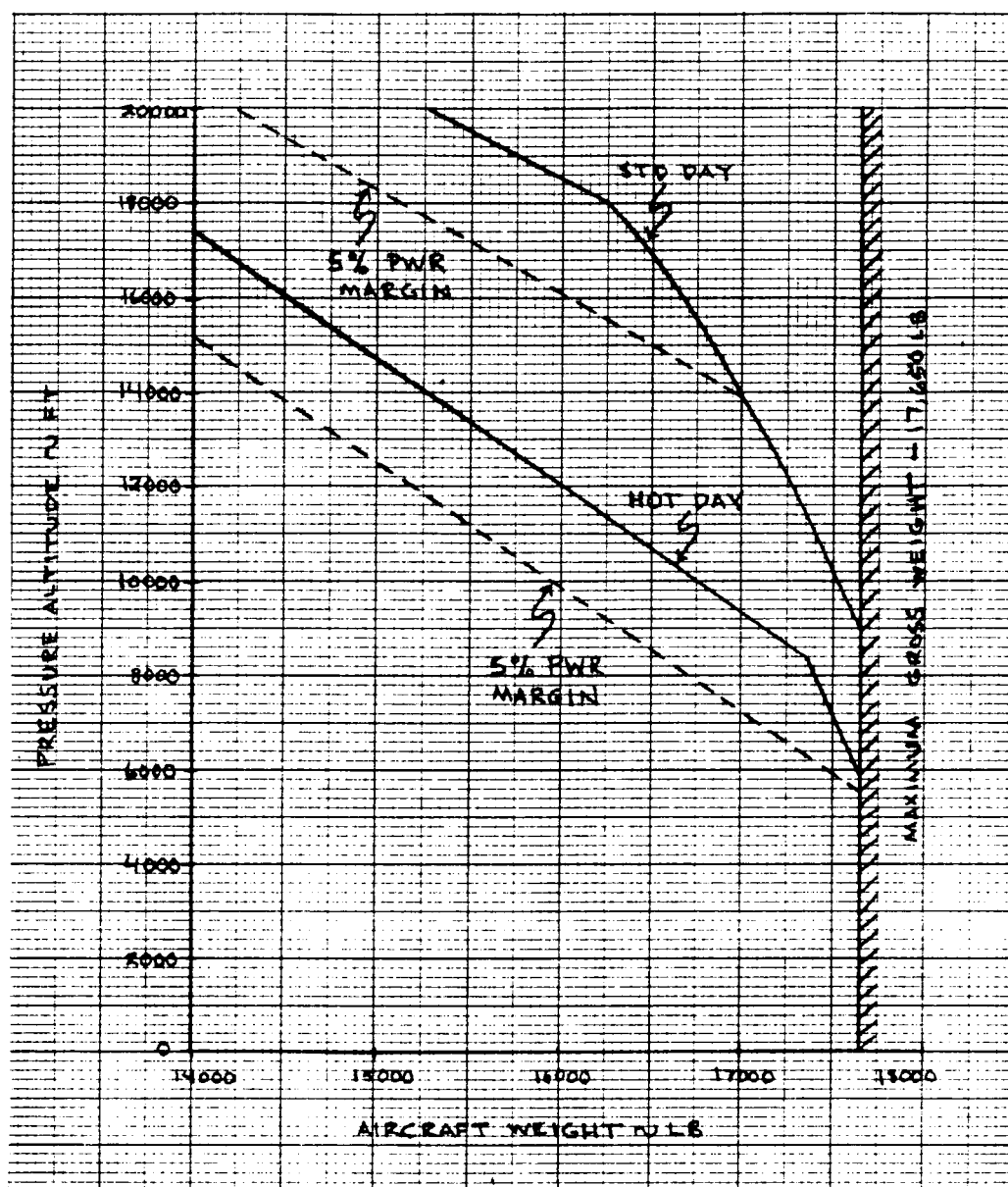


Figure 37

IGE HOVER CEILING

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

ROTOR SPEED: 100%
OAT: STANDARD (15°C)

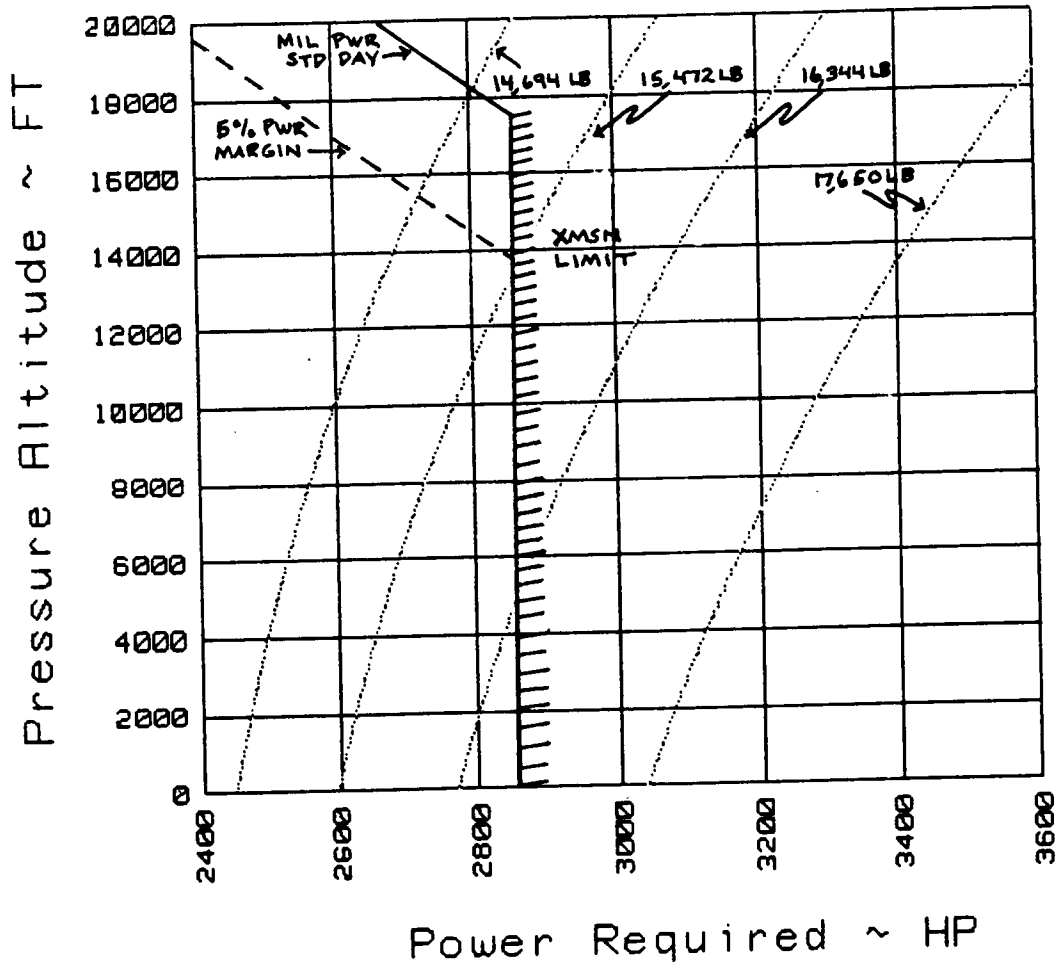


Figure 38

STANDARD DAY OGE HOVER POWER

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

ROTOR SPEED: 100%
OAT: 95°F

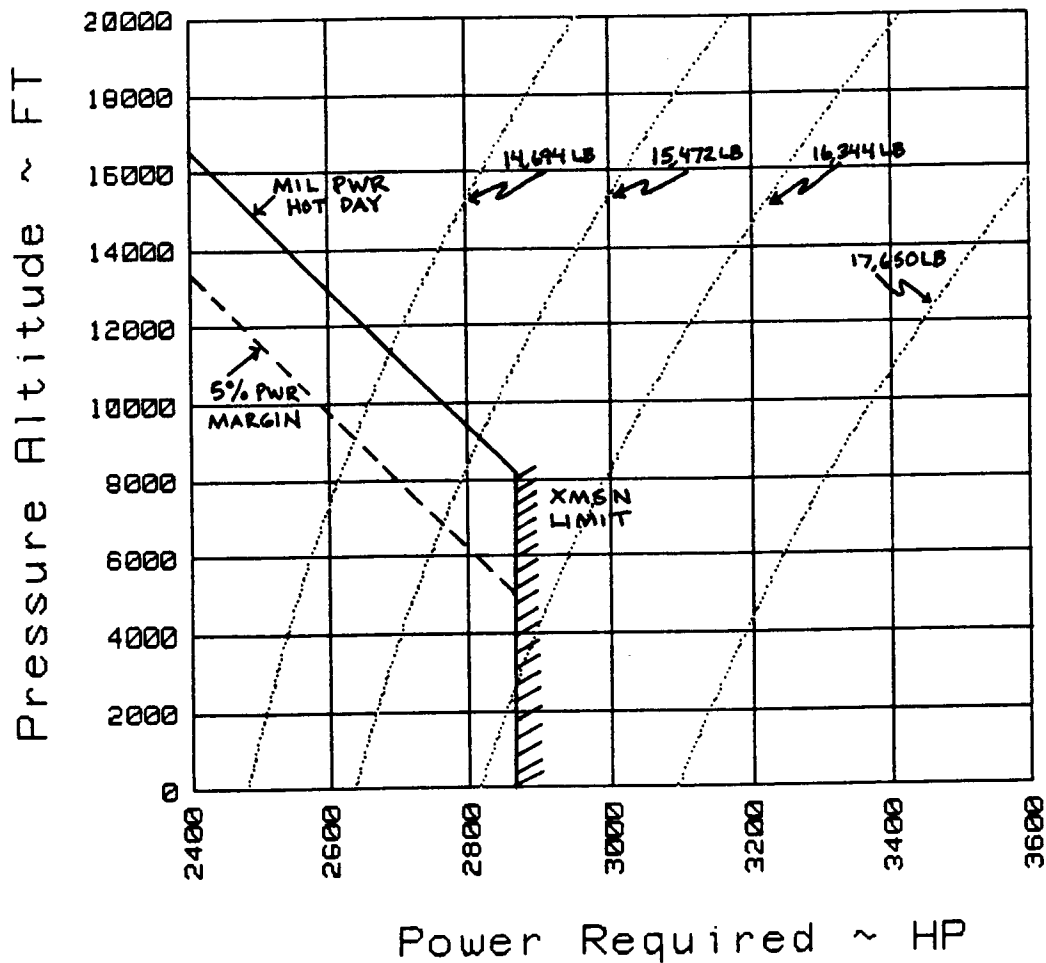


Figure 39

HOT DAY OGE HOVER POWER

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4

ENGINE BLEED AIR: OFF
ROTOR SPEED: 100%

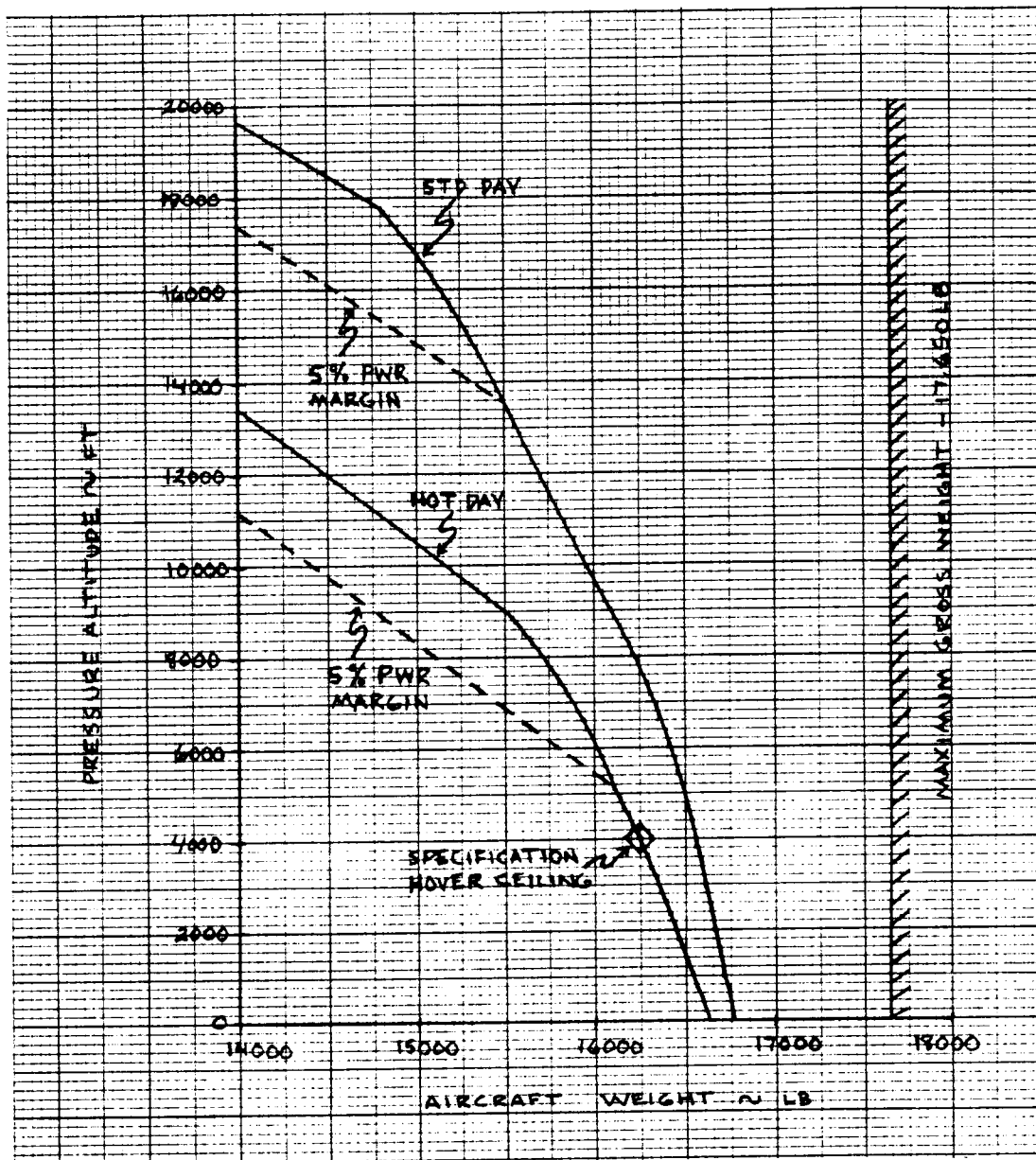


Figure 40

OGH HOVER CEILING

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 14,540 to 14,600 LB.
FUEL: JP-4 CG: 203.7 IN.
ENGINE BLEED AIR: OFF OAT: 28°C
TECHNIQUE: GROUND REF. PA: 100-1600 FT.
ROTOR SPEED: 100%

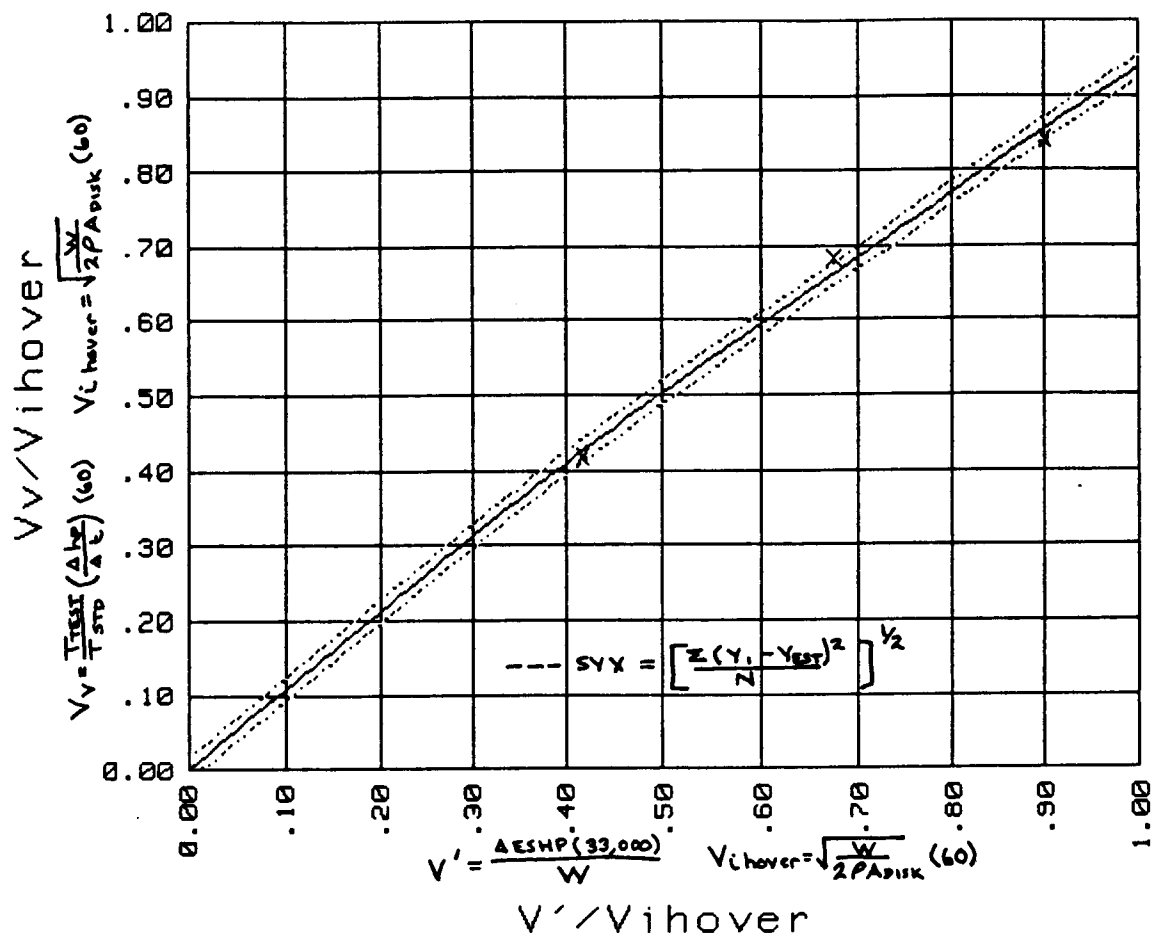


Figure 41
GENERALIZED VERTICAL CLIMB PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 14,540 to 14,600 LB.
FUEL: JP-4 CG: 203.7 IN.
ENGINE BLEED AIR: OFF OAT: 28°C
TECHNIQUE: GROUND REF. PA: 100-1600 FT.
ROTOR SPEED: 100%

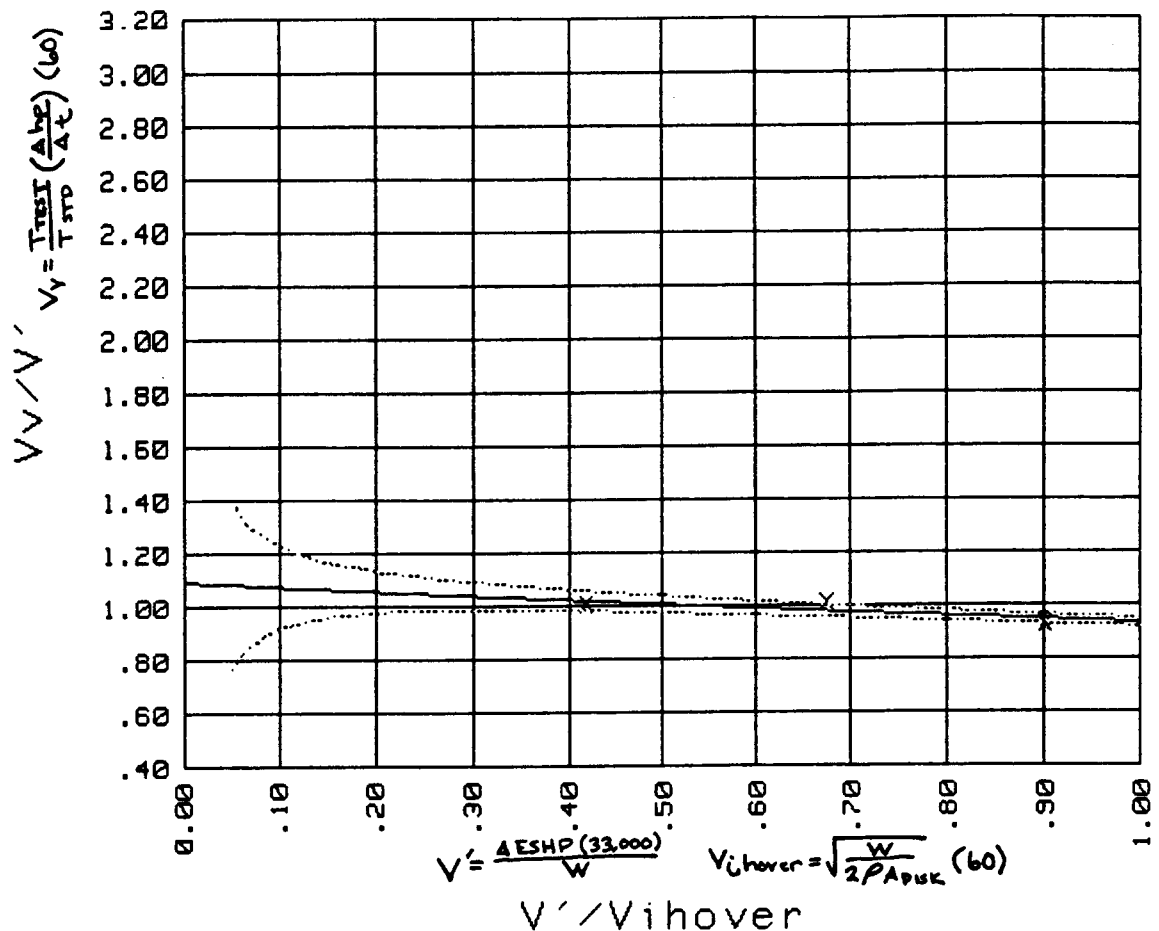


Figure 42
VERTICAL CLIMB CORRECTION FACTOR

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 14,694 to 17,650 LB.
FUEL: JP-4 ROTOR SPEED: 100%
ENGINE BLEED AIR: OFF STANDARD DAY CONDITIONS

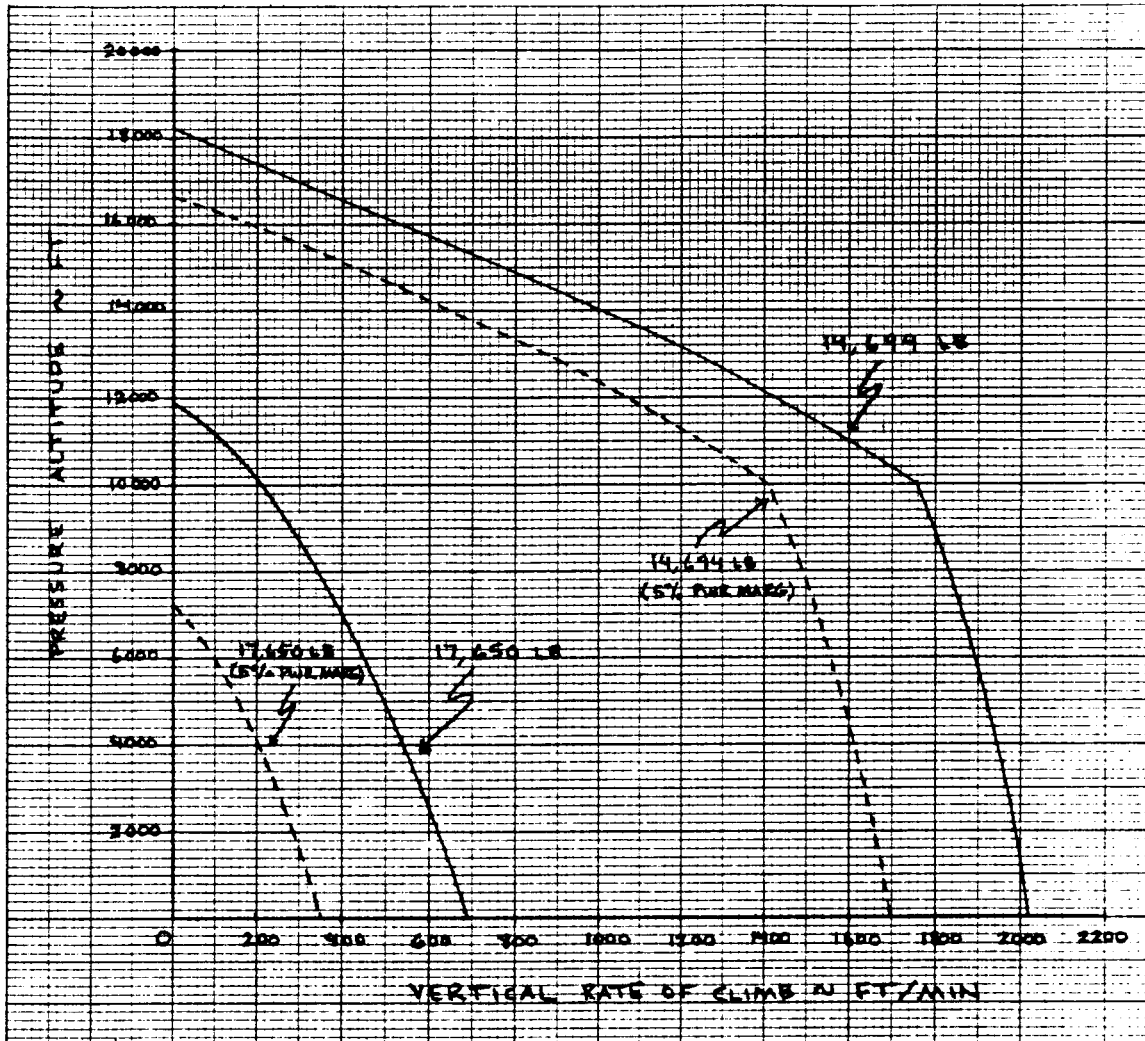


Figure 43
STANDARD DAY VERTICAL CLIMB PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 14,694 to 17,650 LB.
FUEL: JP-4 ROTOR SPEED: 100%
ENGINE BLEED AIR: OFF HOT DAY CONDITIONS

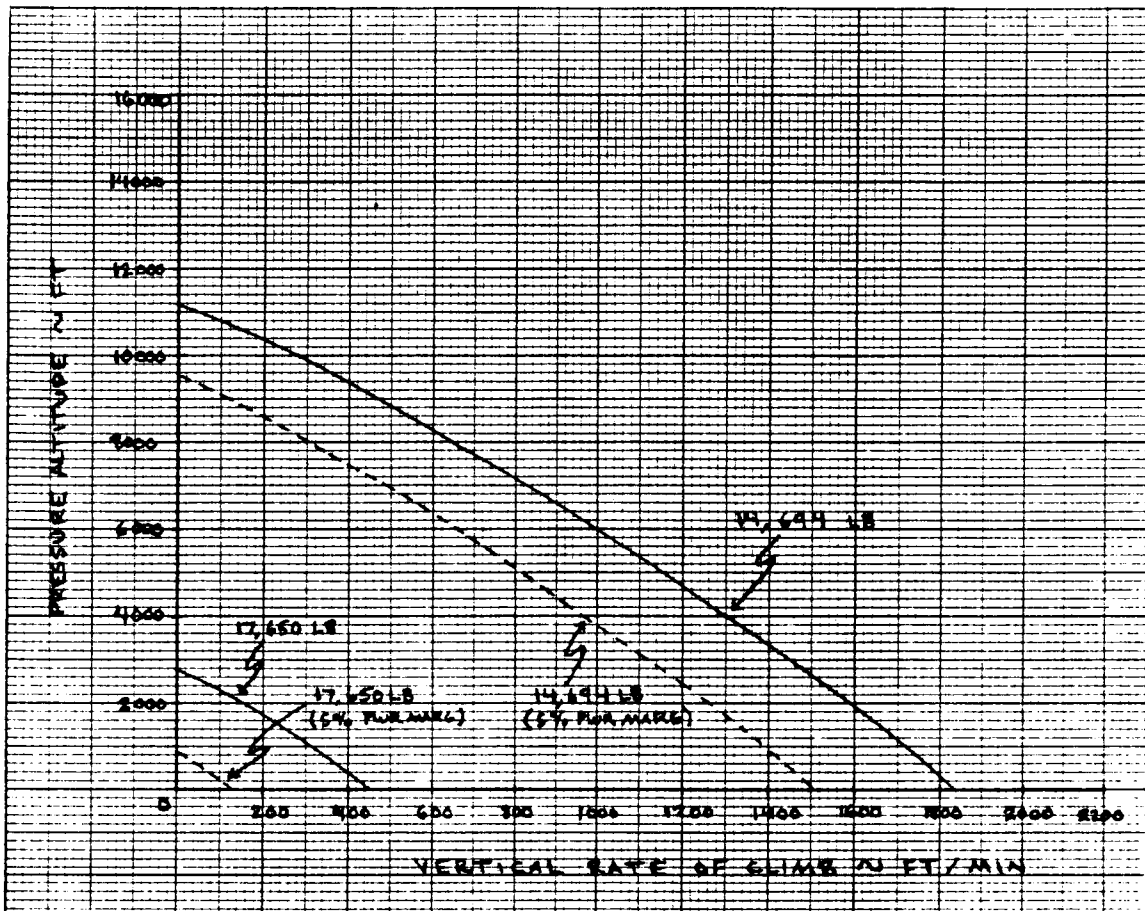


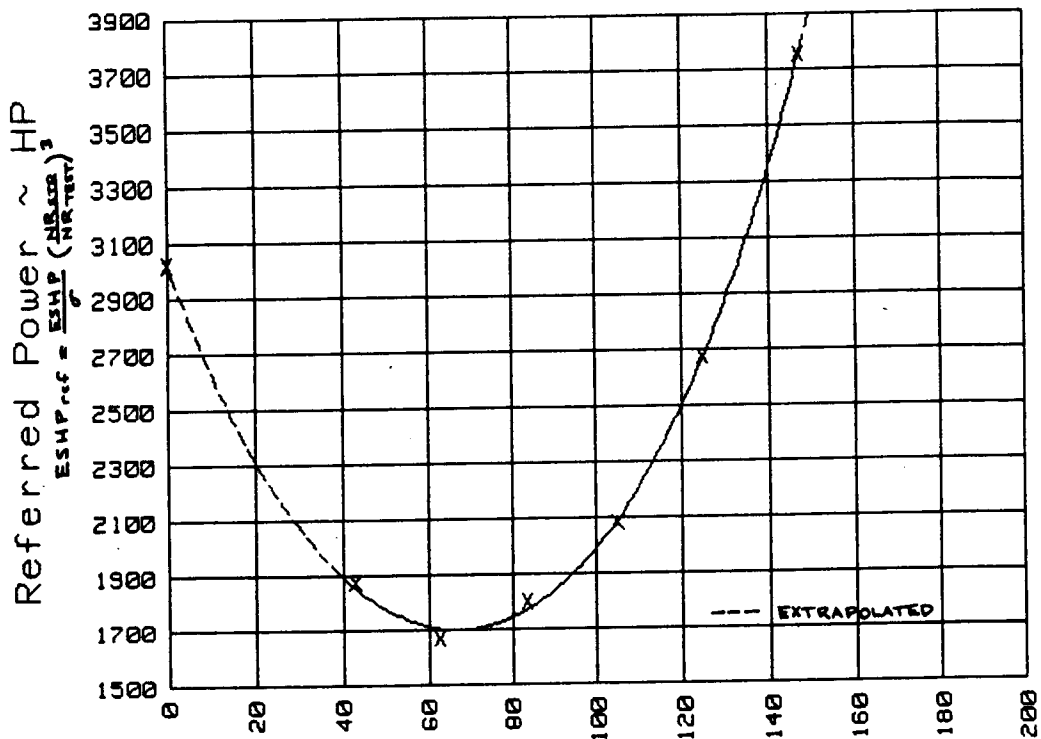
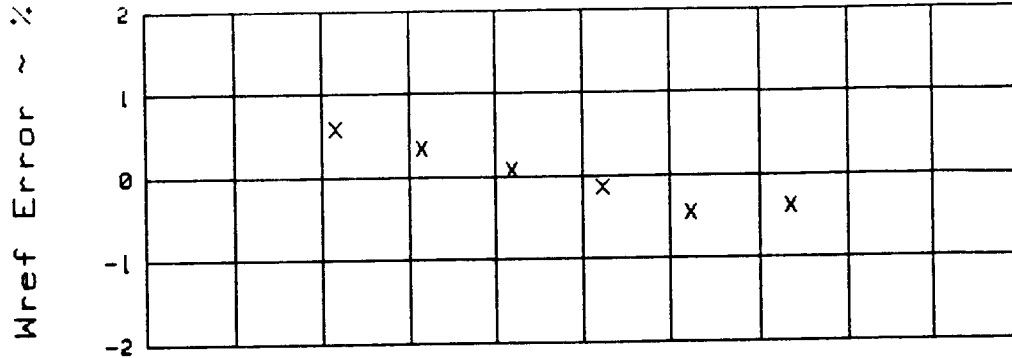
Figure 44

HOT DAY VERTICAL CLIMB PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 15,900 to 16,150 LB.
FUEL: JP-4 CG: 203.8 to 204.6 IN.
ENGINE BLEED AIR: OFF OAT: 22°F
TECHNIQUE: CONSTANT w/σ PA: 1860 to 2250 FT.
ROTOR SPEED: 100% REFERRED WEIGHT: 17,650 LB.



$V_{Tref} = V_T \left(\frac{P}{P_0} \right)^{0.5}$
Referred True Airspeed ~ KT

Figure 45

GENERALIZED LEVEL FLIGHT PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701	GROSS WEIGHT: 15,900 to 16,150 LB.
FUEL: JP-4	CG: 203.8 to 204.6 IN.
ENGINE BLEED AIR: OFF	OAT: 22°F
TECHNIQUE: CONSTANT w/σ	PA: 1860 to 2250 FT.
ROTOR SPEED: 100%	REFERRED WEIGHT: 17,650 LB.

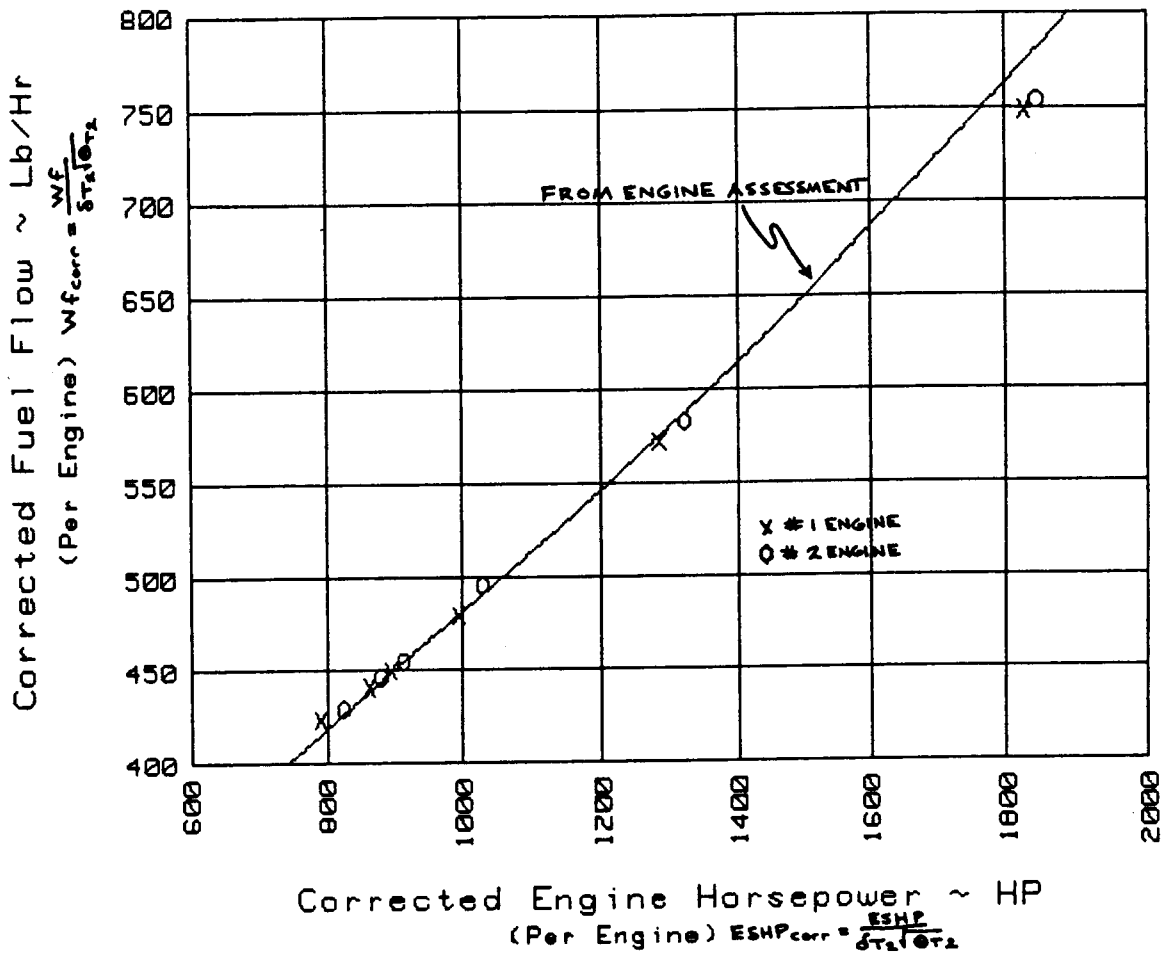


Figure 46
GENERALIZED LEVEL FLIGHT ENGINE PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701	GROSS WEIGHT: 15,900 to 16,150 LB.
FUEL: JP-4	CG: 203.8 to 204.6 IN.
ENGINE BLEED AIR: OFF	OAT: 22°F
TECHNIQUE: CONSTANT w/σ	PA: 1860 to 2250 FT.
ROTOR SPEED: 100%	REFERRED WEIGHT: 17,650 LB.

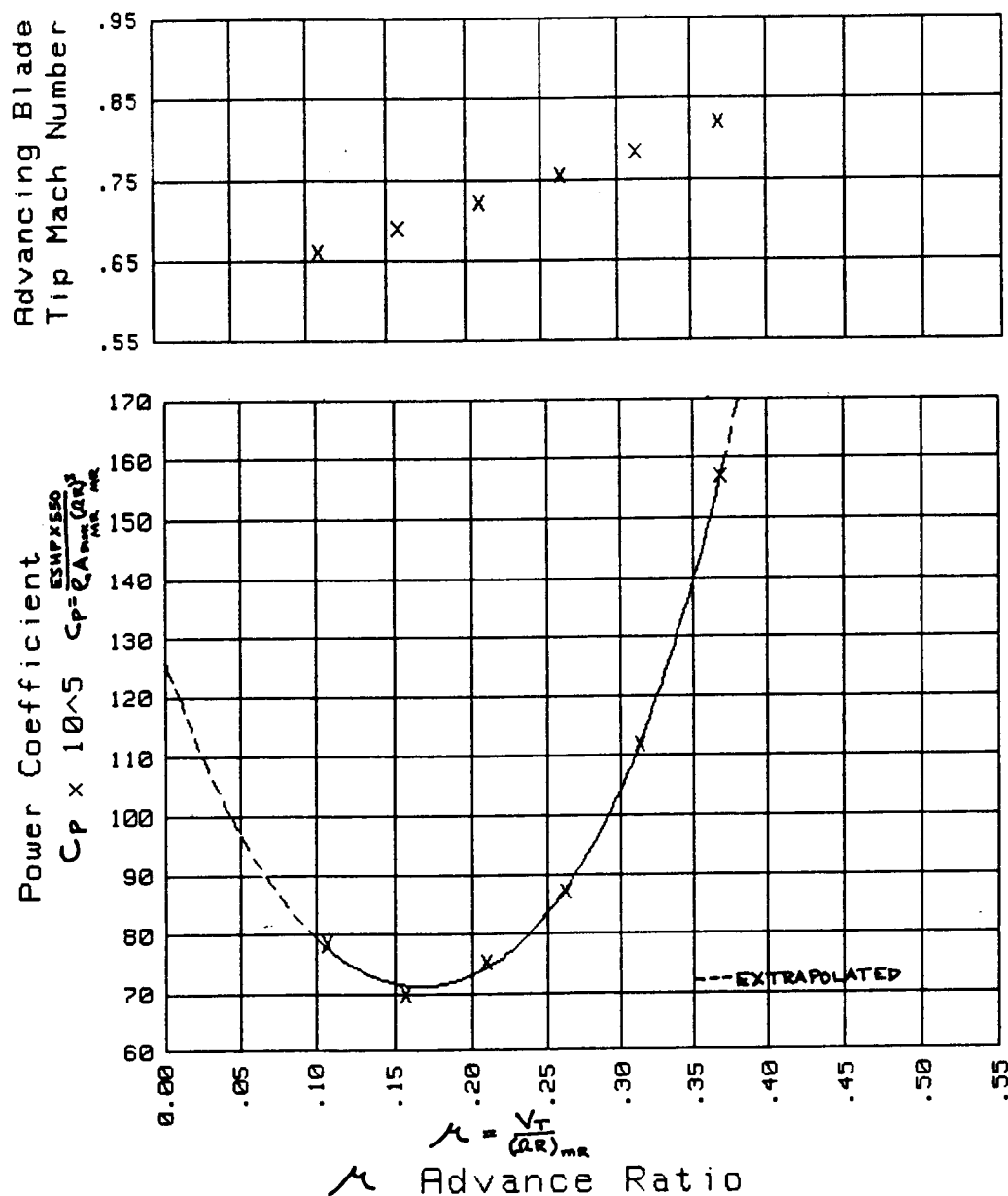


Figure 47

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701
FUEL: JP-4
ENGINE BLEED AIR: OFF

GROSS WEIGHT: 17,650 LB.
ROTOR SPEED: 100%
PRESSURE ALTITUDE: SEA LEVEL
OAT: STANDARD (15°C)

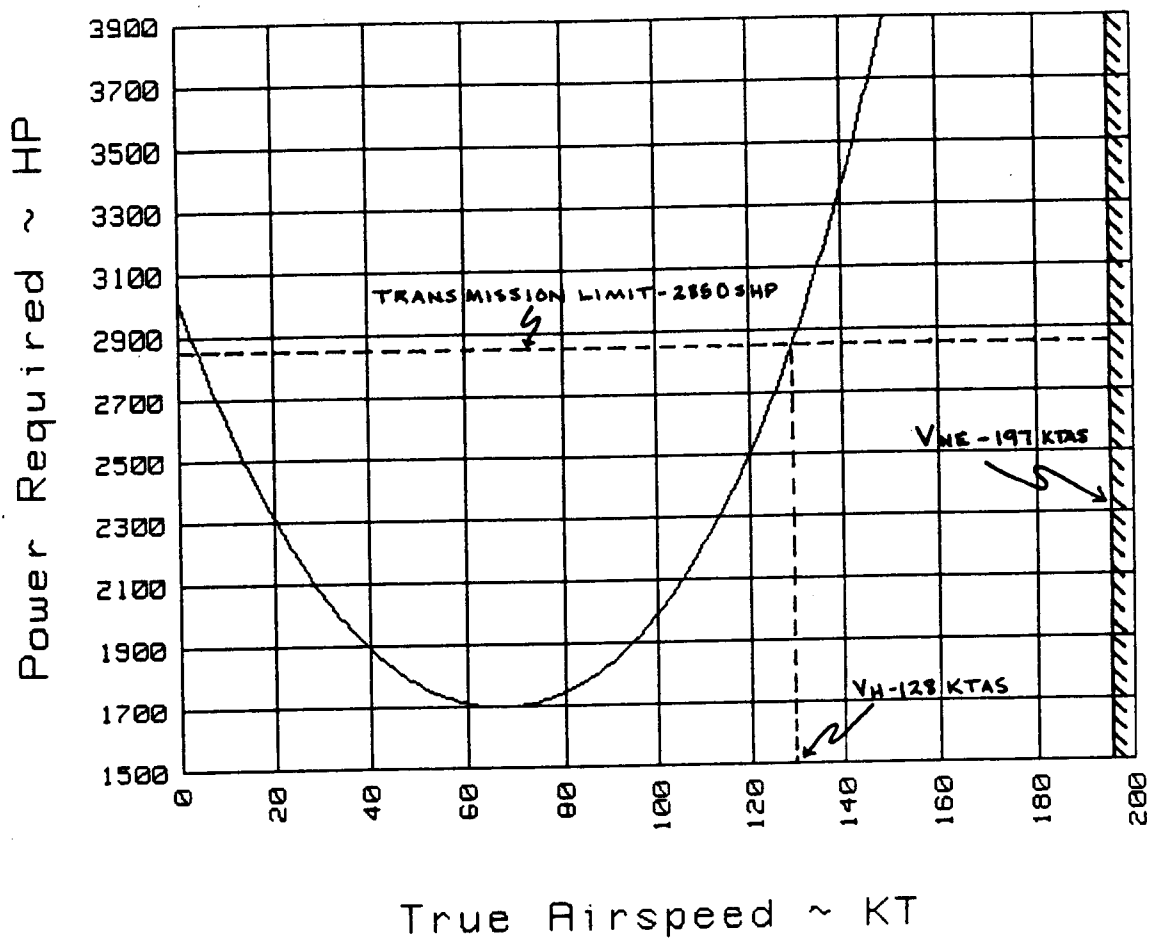


Figure 48
LEVEL FLIGHT PERFORMANCE

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 17,650 LB.
FUEL: JP-4 ROTOR SPEED: 100%
ENGINE BLEED AIR: OFF PRESSURE ALTITUDE: SEA LEVEL
OAT: STANDARD (15°C)

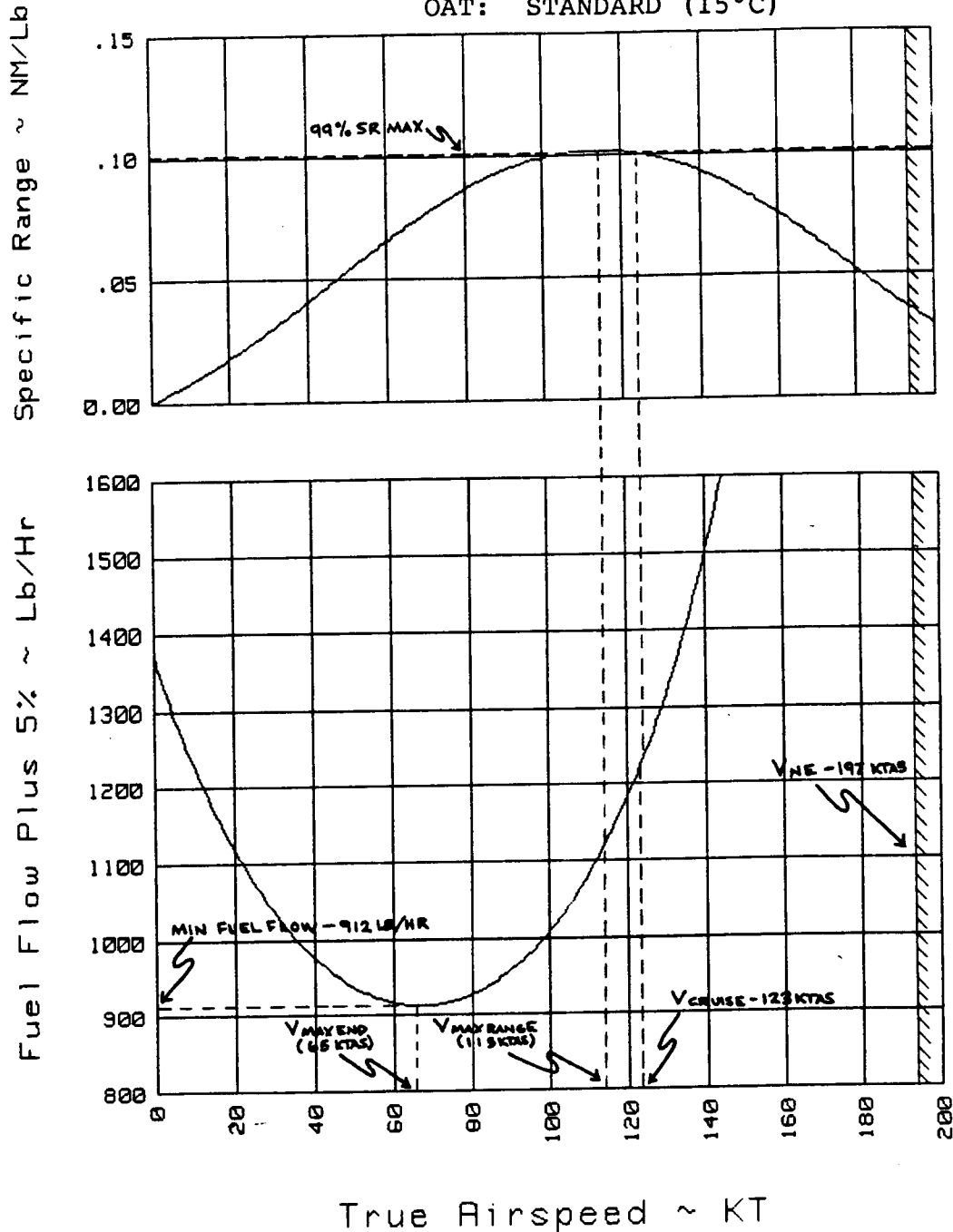


Figure 49

LEVEL FLIGHT PERFORMANCE (FUEL FLOW)

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 16,344 LB.
FUEL: JP-4 CG: 205.1 IN.
ENGINE BLEED AIR: OFF PA: 2500 FT.
ROTOR SPEED: 100% OAT: STANDARD
CONFIGURATION: 8 HELLFIRE MISSILES,
38 2.75" RKTS, 500 RDS. 30 mm

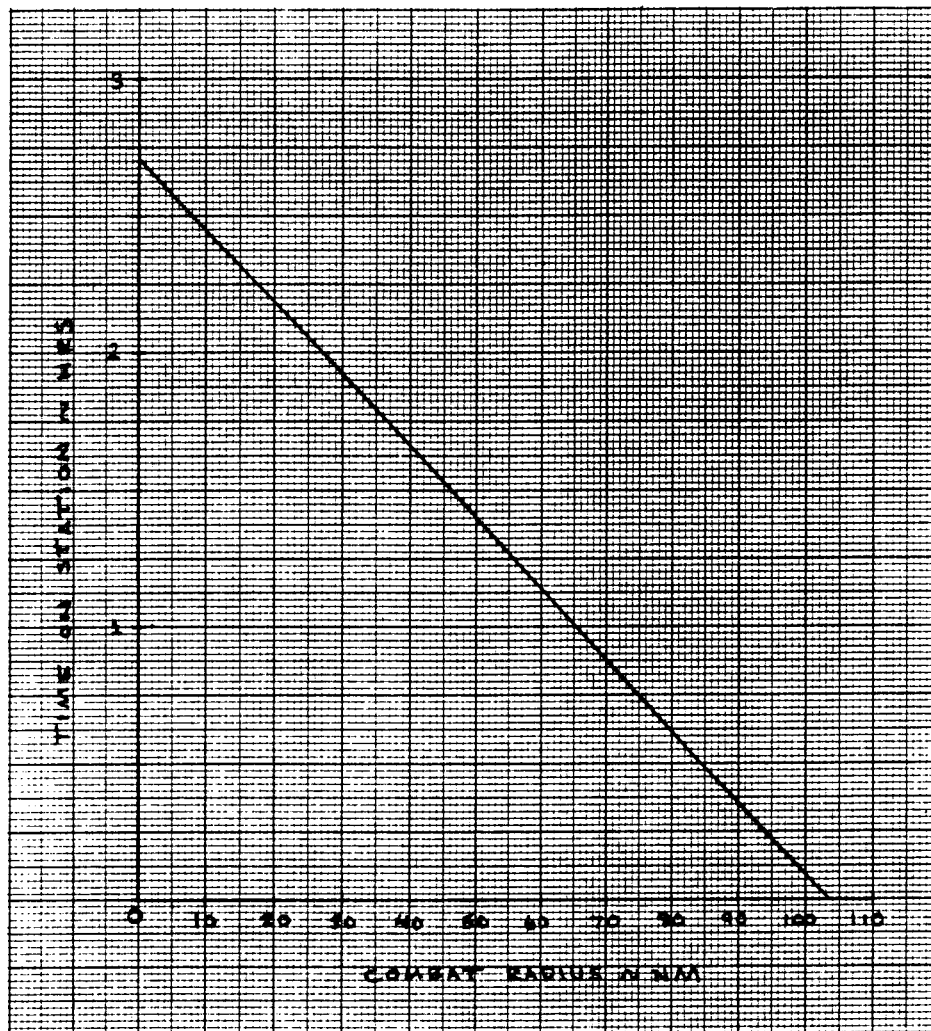


Figure 50
COMBAT RADIUS VS. TIME ON STATION

APPENDIX F

MODEL AH-64A HELICOPTER
SN: 83-23788

ENGINE: TWO T700-GE-701 GROSS WEIGHT: 15,700 LB.
FUEL: JP-4 CG: 203.8 IN.
ENGINE BLEED AIR: OFF PA: 2000 to 4000 FT.
TECHNIQUE: SAWTOOTH C & D OAT: 24°C
(MIN/MAX PWR) ROTOR SPEED: 100%
CONFIGURATION: 8 HELLFIRE MISSILES,
38 2.75" RKTS, 500 RDS. 30 mm

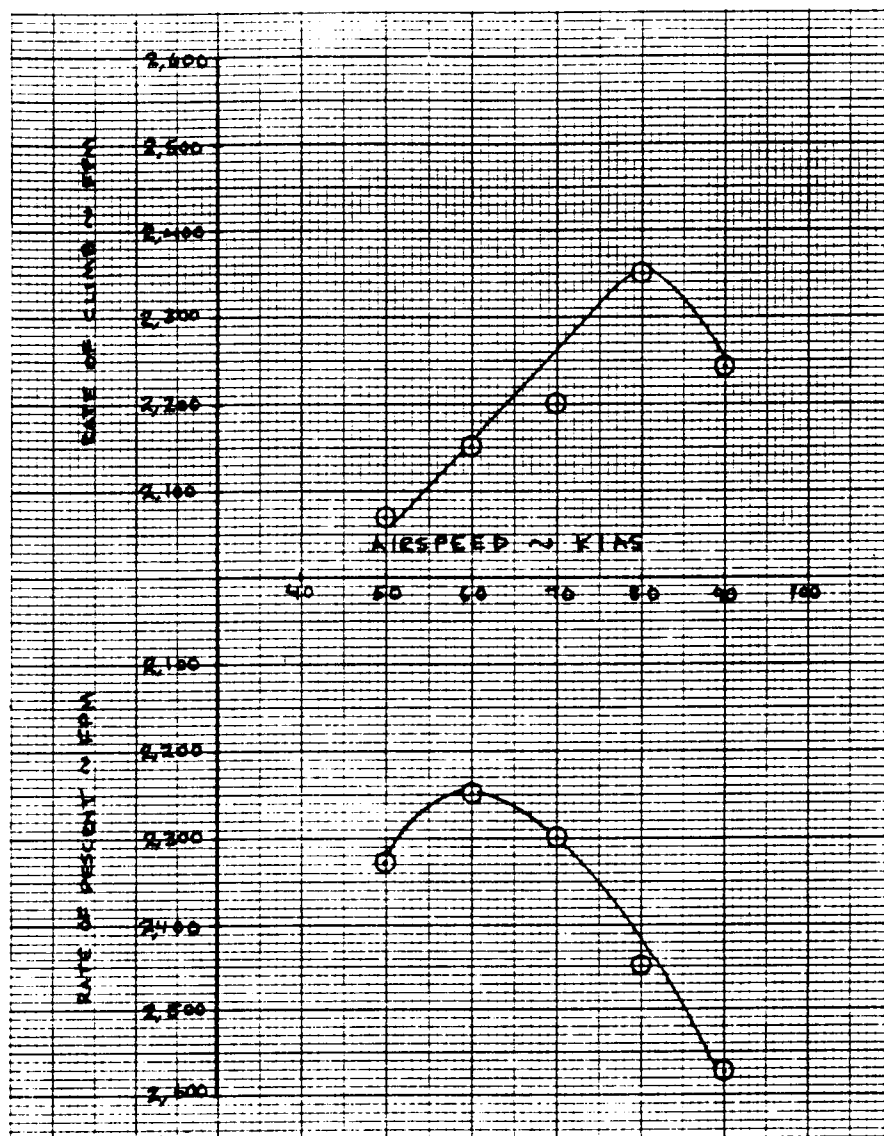


Figure 51

FORWARD FLIGHT CLIMB AND DESCENT PERFORMANCE

VITA

Captain Stephen Edward Joseph was born on December 23, 1955 in Kingsport, Tennessee. In 1973 he graduated from Dobyms-Bennett High School and entered Auburn University, Auburn, Alabama. Upon graduation from Auburn in 1977, he received a Bachelor of Science degree in Aviation Management and was commissioned a Second Lieutenant in the U.S. Marine Corps.

In 1979 he completed Naval Flight Training and reported to Marine Attack Helicopter Squadron 269 in Jacksonville, North Carolina to fly AH-1 Cobra helicopters. In 1984 Captain Joseph was selected to attend Naval Test Pilot School at Patuxent River, Maryland.

After graduation from Test Pilot School in December 1985, he was assigned to the Rotary Wing Aircraft Test Directorate where he currently serves as an AH-1 Project Pilot. In September 1987 he began pursuing a Master of Science degree in Aviation Systems from the University of Tennessee Space Institute. This thesis will complete the requirements of that degree.

The Author was placed on the Commodore's List of Distinction for his performance during Naval Flight Training, and was awarded the Navy Commendation Medal for service with HMA-269. Captain Joseph was also selected as the outstanding student of his Naval Test Pilot School class and is a member of the Society of Experimental Test Pilots.