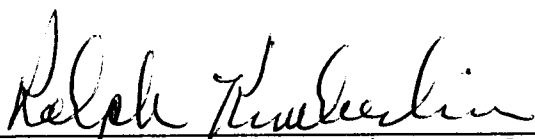


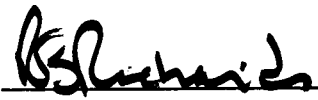
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IMPROVING THE U.S. NAVY SH-3 SEA KING HELICOPTER
TO COUNTER THE SUBMARINE THREAT IN THE 1990'S

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

John D. Rudzis

August 1989

ABSTRACT

The defense budget proposed for FY-90 reflects a reduction in the funding levels enjoyed by the Armed Services from 1981 through 1988. In light of these developments, it is useful to assess the needs of each branch of the Armed Services to determine which new weapons procurement programs are vital to their individual arenas. In one such arena the U.S. Navy has requested procurement of 175 SH-60F CV-Helicopters over the next ten years to replace the aging SH-3 Sea King as its primary aircraft carrier based Anti-Submarine Warfare (ASW) asset. While somewhat antiquated, the SH-3 may still be capable of adequate ASW protection for many years to come. This treatise will discuss deficiencies discovered during flight test of the most current version of the helicopter, the SH-3H, and recommend corrections which will allow the aircraft to continue to be an asset to the Navy throughout the 1990's. Flying qualities and performance testing of the SH-3H was conducted during four flights totalling 6.0 hours at the Naval Air Test Center, Patuxent River, Maryland in May 1987. Throughout the testing the aircraft was configured for a typical ASW mission. The most serious deficiencies

discovered were the inadequate out of ground effect (OGE) and in ground effect (IGE) hover performance of the helicopter under sea level, hot day conditions. These deficiencies preclude deployment of the helicopter under other than the most optimum conditions of low density altitude and/or low gross weight, otherwise depriving the Carrier Battle Group of much needed ASW protection. These deficiencies should be corrected as soon as possible. Four significant deficiencies were discovered which should be corrected as soon as practicable and numerous annoying deficiencies were identified which should be avoided in future designs. The current model SH-3H demonstrated very limited potential to provide adequate ASW coverage to the Carrier Battle Group primarily due to the hover performance deficiencies. Correction of these deficiencies through an engine upgrade program would make the SH-3H a viable ASW asset through the 1990's and into the next century.

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I. INTRODUCTION

BACKGROUND

In January 1989 a new Presidential Administration under the leadership of President George Bush took office to lead the United States into the 1990's. The first budget proposed by the Bush administration for FY-90 allows for a zero growth in defense spending when adjusted for inflation as compared to the FY-89 defense budget.¹ This reflects a reduction in the funding levels enjoyed by the Armed Services from 1981 to 1988 under President Ronald Reagan. In light of these events, it is useful to assess the needs of each of the Armed Services to determine which of the new weapons procurement programs are vital to their individual arenas. One such arena is the Anti-Submarine Warfare (ASW) programs for the U.S. Navy and more specifically the Aircraft Carrier (CV) Battle Group inner zone ASW defense which is the primary responsibility of the Navy ASW helicopter. The Navy is planning procurement, over the next ten years, of 175 Sikorsky SH-60F CV-Helicopters² to replace the aging SH-3 Sea King which has been in service performing this mission since 1965. While somewhat antiquated the most current model, the SH-3H, may still be capable of providing adequate ASW protection to the CV for many more years with moderate improvements using existing technology. To remain capable of effectively

countering the existing submarine threat the aircraft must be equipped to provide the following capabilities:

1. Detect and track submarines using passive and active sonar equipment and a magnetic anomaly detection (MAD) device;

2. Transmit target information to other aircraft and ships via radio data link;

3. Launch multiple weapons each of which is individually capable of destroying enemy submarines.

As a secondary mission the aircraft should have day and night search and rescue capability.³

PURPOSE

The purpose of this treatise is to present deficiencies discovered during flight testing of the U.S. Navy Model SH-3H Sea King ASW helicopter and recommend improvements which will allow the aircraft to continue as a viable ASW asset through the 1990's.

DESCRIPTION OF TEST AIRCRAFT

The SH-3H Sea King helicopter, manufactured by Sikorsky Aircraft Corporation, a subsidiary of United Technologies, Inc., is a dual piloted single main rotor helicopter designed for ship- and shore-based operations to detect, identify, track, and destroy enemy submarines. Additionally, it is designed to provide day and night SAR capability. The aircraft is powered by two T-58GE-10 turboshaft engines each

of which has an uninstalled power rating of 1400 SHP. The maximum gross weight is 21,000 lb and the longitudinal C.G. range is from 258.0 in. to 276.0 in. aft of station zero. The aircraft carries 826 usable gallons of JP-5 fuel in three fuselage tanks, and has provisions to carry two 110 gallon external drop tanks for extended operations. The landing gear consists of two retractable main landing wheels mounted on the outrigger-type sponsons and a non-retractable castering tail wheel mounted to the boat hull shaped fuselage. Minimum mission crew consists of two pilots and two sensor operators. The aircraft is equipped with a passive and active sonar system incorporating a submersible sonar transducer, a streamable MAD device, 12 sonobuoy launch chutes, and 24 smoke launchers. The aircraft is capable of launching two MK46 torpedoes, depth bombs or iron bombs from the forward external stores stations and can carry chaff dispensers on the two aft external stores stations. The cockpit incorporates dual conventional helicopter flight controls. The flight control system is an irreversible system, powered hydraulically by the primary and auxiliary hydraulic power supply systems. Mechanical mixing is designed to provide for collective to pitch coupling (longitudinal) and collective to yaw coupling. The helicopter is equipped with a foldable five-bladed fully articulated main rotor with an effective offset of 3% and a five-bladed tail rotor for anti-torque and directional

control. Automatic Stabilization Equipment (ASE) is incorporated into the longitudinal, lateral, directional and vertical axes. The ASE is designed to retain selected aircraft attitudes and altitudes during cruise, to enable automatic transition from forward flight to a hover and to maintain a stabilized hover during sonar search operations. The ASE system processes signals from gyros, accelerometers, doppler, the radar altimeter, and sonar cable sensors. ASE corrections are then input to the flight controls through the auxiliary hydraulic servos. A detailed description of the SH-3H is presented in reference 5. A three-view drawing of the SH-3H is presented in appendix C, figure 1, and a detailed description of the flight controls is presented in appendix D. The test aircraft, BUNO 149728, was representative of production models for the purpose of this evaluation.

SCOPE OF TESTS

Flying qualities and performance were evaluated during four flights totalling 6.0 hr. at Patuxent River, Maryland in May 1987. An additional 3.0 hr. was dedicated for ground evaluation. All tests were conducted within the normal flight envelope of the aircraft, as presented in reference 5. Test configurations consisted of one practice MK46 torpedo mounted to the left forward external store station. Test loadings consisted of two pilots and one sensor operator. Aircraft fuel load consisted of full internal fuel at engine

start. Longitudinal C.G. position was at 269.0 ± 2.0 in. during all flights. The SH-3H flying qualities were evaluated against the general specification requirements of reference 4. Time and funding constraints limited the scope of this evaluation to an investigation of basic flying qualities and performance characteristics with respect to the aircraft mission. A complete evaluation encompassing many months would be necessary to ascertain the viability extending the life of this aircraft sufficiently to conduct this mission into the next century. A complete summary of tests and test conditions is presented in appendix F, table I.

METHOD OF TESTS

Aircraft suitability for the mission was qualitatively evaluated during the representative mission tasks presented in appendix F, table I. Night/Instrument Meteorological Conditions (IMC) tasks were simulated using a cardboard helmet visor to reduce external visual cues. Hover, low airspeed, and forward flight flying qualities were evaluated using techniques and procedures presented in reference 3, controls fixed and/or free where applicable. Helicopter performance was evaluated in accordance with applicable techniques and procedures presented in references 7 and 8. Hover performance was evaluated using the free hover method in and out of ground effect utilizing the aircraft radar altimeter for measurement of hover height. Level

flight performance was evaluated using the W/ frontside and backside techniques at a reference weight of 20,800 lb. ASE influence on handling qualities was qualitatively evaluated using the applicable procedures of reference 9. ASE modes were qualitatively evaluated during forward flight, approach, hover, and landings. Cockpit control positions were measured using tape measures installed, as depicted in appendix C, figure 2. Control forces were measured using a hand-held force gauge. Control step inputs were accomplished using a control limit fixture. A portable accelerometer was placed adjacent to the glare-shield, as depicted in appendix C, figure 2, to measure normal accelerations. Ground speed, true airspeed, and wind velocity were measured using the production AN/ASN-123 Tactical Navigation computer which is part of the aircraft mission equipment. Tapeline altitude was measured using the radar altimeter. Rotor speed was measured using the aircraft production triple tachometer. All other parameters were measured on the aircraft production instruments. A pace truck was utilized during low airspeed testing to accurately measure airspeeds up to 60 knots. Data were recorded using kneeboard cards and a portable voice recorder. Altitudes during level flight performance testing were determined using "W/σ" programs developed for the hand-held HP-41 programmable calculator. Hover and level flight performance data were reduced using helicopter performance programs with the HP-9845 computer. Deficiency

classification is presented in appendix A. Handling Qualities Ratings (HQR), vibration assessment ratings (VAR), turbulence categories, and pilot-induced oscillation (PIO) ratings were assigned in accordance with the rating scales presented in appendix B.

MISSION PROFILE

The SH-3H helicopter is designed to operate from a U.S. Navy Attack Aircraft Carrier (CV), to provide ASW capability to protect the carrier battle group from hostile subsurface threats on the high seas. The primary mission of the aircraft is to deploy the active sonar system, from a 40 foot overwater hover during day and night all-weather conditions in sea states from 0 to 8 and winds up to 50 kt., to locate, classify and track submarines. The mission also involves deploying sonobuoys and streamed MAD for a passive search capability, and target destruction by launching torpedoes or depth charges. Additionally, the aircraft is designed to provide overwater SAR capability during day and night. The aircraft mission profile involves launching from the CV and transiting to the area of a suspected submarine contact. A typical mission profile is presented in table I. The aircraft would deploy sonobuoys and MAD and cruise from 500 to 3000 ft. while monitoring onboard sensors to localize the target. Once the submarine is located the aircraft would be required to transition to a 40 foot hover and submerge the

Table I
TYPICAL SH-3H MISSION PROFILE³

1. Take Off at mission GW from deck
2. Climb to 150 ft. at Maximum Continuous Power
3. Transit at Vcruise 60 nm.
4. Enter hover at 40 ft. and conduct active sonar operations for 15 mins.
5. Exit hover and transition to 150 ft./100 kt. and transit to new dip area 10 nm away.
6. Repeat 4 and 5 above 5 times.
7. Return to Ship 40 nm.
8. Carry out IFR Night Carrier Controlled Approach
9. Land with deck movement of ± 2 deg. pitch, ± 4 deg. roll, ± 15 ft. heave, with min. fuel of 450 lbs.

active sonar transducer below or between isothermal ocean layers to actively track the target. The aircraft would be required to maintain within 3,000 yards of the submarine for optimum performance of the sonar system. Tracking a submarine travelling at 40 kt. would require the aircraft to retrieve the sonar transducer, rapidly reposition five to seven nautical miles and lower the transducer to search depth. Every four to six min. this "dip cycle" would be repeated for as long as is feasible.

II. DISCUSSION AND RECOMMENDATIONS

COCKPIT EVALUATION

GENERAL

The cockpit of the helicopter was evaluated qualitatively on deck and in flight from the right seat, adjusted to the design eye position (DEP), during all tests under the conditions of appendix F, table I. A cockpit diagram is presented in appendix C, figure 3.

FIELD OF VIEW (FOV)

Forward Field of View (FOV) was adequate during all flight phases. During normal transitions to hover from 70 KIAS and 500 ft. above ground level (AGL), forward FOV of the 50 by 50 ft landing area during the approach allowed adequate cues for the pilot to position the aircraft over the spot in a 15 ft. hover. In forward flight, during right turns, 10-30 deg. angle of bank (AOB), the pilot's FOV in the direction of turn was slightly obstructed by the upper horizontal window frame between 10 and 30 deg. right and 20 to 30 deg. above DEP. However, the pilot was able to see around the obstruction using moderate lateral head movements. During

right turns to the ship at night, moderate movements of the head will be adequate to visually acquire external visual cues sufficient for the approach transition to the landing spot. The FOV from the cockpit is satisfactory for the mission.

Exterior Mirrors

The exterior cockpit rearview mirrors were evaluated during all tests. The rearview mirrors, mounted to the exterior of the cockpit above each pilot window and provided the pilot with a wide angle view of the fuselage aft of the cockpit. The mirror was useful for monitoring personnel who were on the fuselage during ground operations and for monitoring the progress of the blades during folding and spreading evolutions. In flight, the mirrors provided adequate FOV of the engine and main transmission compartments for crosschecks of malfunction indications. The pilot will be afforded the capability to quickly scan the corresponding engine compartment to check for secondary indications of an illuminated engine fire warning light. The external wide angle cockpit rear view mirrors was an enhancing characteristic which should be incorporated into other aircraft.

CONTROLS

Landing Gear Handle

The landing gear actuating lever was evaluated for accessibility from the right pilot seat during takeoffs and landings. The gear lever location is depicted in appendix C, figure 3, item 27. The 35 in. functional reach required to operate the gear lever from the right seat exceeded that of the 98th percentile aviator. While the co-pilot was flying the aircraft, operation of the gear handle from the left seat necessitated transferring of controls to the right seat pilot. An alternate means was for the right seat pilot to unlock his shoulder harness and stretch across the center console to reach the gear handle. Neither method was desirable during critical flight phases such as takeoffs and landings, therefore the responsibility for raising or lowering the landing gear handle rests with the co-pilot requiring the pilot to be on the controls when there was a requirement to operate the landing gear. Takeoffs and landings at the ship will be limited to being a right seat function preventing the co-pilot from gaining and maintaining proficiency in flight operations around the ship. It is recommended that the landing gear handle be moved to allow easy access by both pilots.

Tail Wheel Locking and Unlocking

Operation of the tail wheel locking handle was evaluated for functional utility during ground taxi. The locking handle, depicted in appendix C, figure 3, item 19, is located slightly forward and below the pilot's collective lever. To unlock the tail wheel necessitated transferring control of the collective to the co-pilot and reaching over the top of the collective pushing the handle down and aligning the downlock with a detent. Locking the tail wheel was essentially the reverse; pushing the downlock release on the handle and pulling the handle up. The locking handle was only accessible to the pilot in the right seat. During shore-based taxi, operating the tail wheel locking handle will unnecessarily distract the pilot requiring a transfer of controls to the co-pilot. It is recommended that the tail wheel locking system be redesigned to allow for easy operation by either pilot.

Engine Controls

Ease of adjusting the engine/rotor speed inflight was evaluated during all tests. The engine speed control levers are depicted in appendix C, figure 3, item 3, and in figure 1.

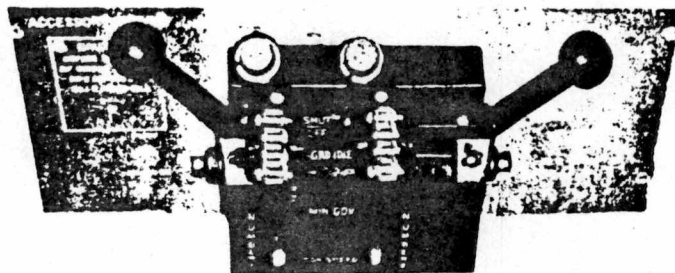


Figure 1

ENGINE SPEED LEVERS

Inflight engine speed adjustment was accomplished by the pilot not on the controls, reaching up to the engine speed levers, and manually moving the levers to set engine speed on the triple tachometer. Both speed levers had approximately 1/4 in. deadband in which the speed lever could be moved up to 1/4 in. in either direction without a corresponding change in engine speed. Combined with the rotor governing characteristics of the engines, maintaining desired engine and rotor speed was annoying in that the pilot could not adjust rotor speed without releasing the collective, normally relying on the co-pilot for engine trimming. While positioning in a coupled hover to track a localized submarine, the co-pilot will be distracted from his mission duties by the necessity to trim the engines for the pilot, increasing the submarine's chances of evasion. Automatic engine controls which maintain rotor speed within one to two percent should be incorporated.

DISPLAYS

Hover Indicator

The hover indicator D-mode was evaluated for adequacy of drift and altitude cues during coupled approaches and hover over the runway. A description of the D-mode symbology is presented in appendix F, table VI. The hover indicator is depicted in appendix D, figure 10. Each horizontal and vertical calibration mark represented 10 knots of longitudinal or lateral velocity, measured by the aircraft doppler navigation system. The horizontal or vertical bars would displace from center in a direction opposite to the measured drift. The indicator was designed to cue the pilot to "fly" the center of the indicator to the needles. During simulated mission approaches to a coupled hover using the runway border as a reference, the pilot was able to align the aircraft with the border and maintain alignment by holding the vertical bar superimposed over the center of the indicator. In a hover, with cyclic coupler disengaged, the pilot was able to hold the aircraft over the spot ± 3 ft. by keeping the needles centered with the cyclic. The pilot will have adequate cues from the symbology on the hover indicator to maintain position over the desired sonar dip point during ASW operations. Within the scope of this test, the doppler symbology of the hover indicator is satisfactory for the CV-ASW mission.

CONTROL SYSTEM CHARACTERISTICS

GENERAL

The flight control system characteristics were quantitatively evaluated on the ground with the rotor and engines stopped, ASE and Trim ON. External hydraulic pressure was provided to pressurize the primary and auxiliary hydraulic systems to 1400 psi and external electrical power was provided to power the aircraft electrical systems. Control measurement locations are depicted in appendix C, figure 4. Tabular results of the tests, recorded using tape measures, a hand-held force gauge and a stopwatch, are presented in appendix F, table II. Control envelope data are presented in appendix G, figures 1 and 2. Data were verified on deck with the rotors and engines operating at 100% and qualitatively evaluated in flight during all tests.

LONGITUDINAL

Longitudinal force versus displacement data are presented in appendix G, figure 3. Freeplay was negligible in flight. Control centering was absolute throughout the test envelope. The moderately high breakout forces combined with the strong stick gradient provided good centering characteristics and also encouraged the use of stick trim during forward flight. However, during high gain hovering tasks, holding position ± 5 ft. was annoying and fatiguing, requiring $\pm 1/4$ to $1/2$ in. longitudinal cyclic inputs against the breakout forces every 4-5 sec. (HQR-4). Attaining the same

performance with the cyclic trim release button depressed was less fatiguing requiring occasional $\pm 1/8$ in. longitudinal cyclic inputs every 5-10 sec. (HQR-3). Stick jump, tested in forward flight at 100 knots indicated airspeed (KIAS) using a 1 in. displacement against trim, produced negligible aircraft response. Stick dynamics, tested at 100 KIAS using a $1/2$ in. displacement against trim, was heavily damped with small overshoots of the cyclic and generated negligible pitch response in the aircraft.

LATERAL

Lateral force versus displacement data are presented in appendix G, figure 4. The 0.2 in. total system freeplay appeared to contribute to the lateral response which appeared to lag behind large rapid lateral cyclic inputs. During high gain hovering tasks, lateral position (± 5 ft) was easier to maintain with the cyclic trim release button depressed for the same reasons described in the preceding paragraph. However, the light control forces with cyclic trim disengaged, combined with the lagging lateral response, produced a mild ($\pm 1-2$ deg. AOB) pilot induced oscillation (PIO) which damped out when the pilot held the cyclic stationary (PIO-4). Stick jump, evaluated in flight at 100 KIAS using 1 in. displacement against trim produced a slight roll fluctuation to the left and a negligible response when tested to the right. Stick dynamics, using a $1/2$ in.

displacement was heavily damped with two small overshoots of the cyclic and a negligible aircraft response.

DIRECTIONAL

Directional control viscous damping data are presented in appendix G, figure 5. Positive control centering was indicated during small inputs ($\pm 1/2$ in.). Larger inputs were sufficient to overcome control system friction and displaced the pedals to a new position. The high viscous damping characteristics of the pedals resisted pedal displacement rates such that a 20 lb. pedal force displaced the pedals at a rate of .05 in./sec and a 100 lb. force moved the pedals at only 2.80 in./sec. The ASE generated additional pedal forces by driving the pedals to oppose aircraft yaw rates. The pilot could override these pedal forces, albeit at higher control forces. Maintaining balanced flight during forward flight turns was fatiguing because the pilot was required to hold a pedal force directly proportional to the turn rate (HQR-4). In hover, pedal turns using the directional pedals were difficult because the higher relative yaw rates induced a higher damping force to the pedals. Directional control predictability was poor because the pilot was required to overcome the pedal damping force as he displaced the pedals plus the yaw rate damping force induced by the ASE. Each damping force was dependent upon a different set of conditions and would vary causing the pilot to overdrive or underdrive the pedals. The

directional control characteristics also failed the specification paragraph 3.3.11, of reference 3 because the force gradient was not linear. The high pedal forces and inadequate directional trim cues were one of the most objectionable characteristics of the aircraft and will be addressed later in this section.

VERTICAL

The collective adjustable friction device provided the pilot with adequate proprioceptive cues for collective control. Normal operations with the coupler or altitude retention systems were performed with the friction off requiring the pilot to support the weight of the collective lever unless the altitude retention system was engaged. When the Barometric Altitude Hold (BAR ALT HOLD) feature was engaged, the pilot could override the system by pushing down against the spring gradient force of the collective open loop spring, and when the coupler was engaged, the pilot would override the radar altitude hold by pulling up against the collective open loop spring force gradient. Displacing the collective approximately 1 in. against the spring gradient and disengaging the appropriate system (BAR ALT HOLD or COUPLER) caused the collective to jump approximately 1/2 to 1 in. causing a sharp vertical jolt in the airframe.

MIXING

Mechanical collective to yaw mixing is discussed in appendix D. Data are presented in appendix G, figure 6. Data show the variation of tail rotor blade pitch with collective position at a constant pedal position. The largest variation is at the mid (50%) pedal position where tail rotor blade pitch increased 35% as the collective was raised from 20% to 100%. Collective position would vary pedal position as discussed in appendix D, paragraph 9. The effect was irreversible in that pedal position would not influence collective position while the auxiliary servo was pressurized. Reversibility is theoretically possible without aux servo pressure, however, during flight with the auxiliary servo secured, pedal movement did not produce noticeable collective movement.

TRIM

The longitudinal and lateral control trim system was evaluated during all tests. A description of the trim system is presented in appendix D. The beeper trim authority was sufficient to include the entire cyclic control envelope. Initial engagement of the four-way trim switch produced an instantaneous jump of 1/32 in. in the direction of engagement followed by a steady state rate. The initial jump in the stick appeared as the onset of an excessive trim rate which caused the pilot to avoid engaging the beeper trim for more than 2-3 sec. at a time. The design of the four-position

switch prevented the pilot from trimming simultaneously in the longitudinal and lateral axis. The beeper trim was overly sensitive in roll in that setting a precise roll attitude (± 1 deg.) using beeper trim only was difficult as one quick engagement of the beeper caused the aircraft to overshoot the desired roll attitude by 2-3 deg., necessitating the use of the cyclic trim release to make minute attitude adjustments. The cyclic trim release was also adequate for quickly making large trim adjustments (± 5 deg.), however, making fine trim adjustments at times when precise attitude control was required; i.e., during an approach to a coupled hover, using the cyclic trim release was undesirable as the trim reference was disengaged in both axes (lateral and longitudinal) which also disengaged the autopilot features of the ASE.

LOW AIRSPEED FLYING QUALITIES

TRIMMED FLIGHT CONTROL POSITIONS

Trimmed flight control positions were evaluated at the 360 deg., 90 deg., 180 deg., 270 deg. azimuths from 0 to 20 kt. calibrated airspeed (KCAS) using a pace vehicle under the conditions of appendix F, table I. Graphical data are presented in appendix G, figures 7 and 8. Adequate control margins were exhibited in all axes. The minimum control margin (24%) encountered was in the longitudinal axis during left sideward and rearward flight at 20 kt. All control displacements generally appeared to be in the normal sense

during the test, however, data show that pedal position reversed direction by approximately 1/2 in. after 15 kt. sideward flight to both the left and right, a phenomenon which was not noticeable to the pilot during the test. The largest attitude changes occurred in roll during lateral flight varying from 8 deg left angle of bank to 5 deg right angle of bank between 20 kt. left lateral flight and 20 kt. right lateral flight. Once established on condition minimal control movement was required to maintain the parameters. The ASW pilot will be afforded adequate control margins to maintain position over the sonar while dipping in day visual meteorological conditions (VMC) in winds up to 20 kt. from any axis. The low airspeed trim flight control positions were satisfactory for the carrier ASW mission.

CRITICAL AZIMUTH

A critical azimuth determination was conducted at 20 KCAS using 45 deg. relative azimuth increments under the conditions of appendix F, table I. Data are presented in appendix G, figure 9. Adequate control margins were available in all axes throughout the test. The minimum control margin was 15% up collective remaining at the 135 deg. relative azimuth. Control forces could be trimmed to zero for all azimuths tested. FOV was adequate from the right seat between the 315 deg. and 135 deg. relative azimuth and from the left seat between the 225 deg. and 045 deg. relative azimuths. The right seat pilot had limited FOV of the

azimuths from 135 to 180 deg. relative and the left seat pilot had limited FOV of the 180 to 225 deg. relative azimuths through the use of small externally mounted wide angle rear view mirrors mounted above each pilot window. Moderate medium frequency vibrations were felt in all quadrants. The minimum levels occurred on the 360 deg. relative azimuths (VAR-2) and increased to a maximum (VAR-5) between the 045 deg. and 225 deg. relative azimuths due to an increase in intensity level. A moderate low frequency vibration was also experienced at the 315 deg. relative (VAR-5). A lightly damped roll oscillation (± 4 deg.) was encountered at the 135 and 180 deg. relative azimuths and was accompanied by a ± 4 deg. yaw oscillation at the 180 deg. azimuth. Attempting to counter the roll oscillation with $1/4$ to $1/2$ in. lateral cyclic inputs every 2-3 sec. tended to continue to drive the roll oscillation ± 5 deg every 2-3 sec. (PIO-4). By holding the cyclic fixed with the stick trim engaged reduced the amplitude of the roll oscillation ± 1 deg. every 1-2 sec. and damped out the yaw oscillation completely. The pilot will attempt to maintain the winds on the nose during transitions to a hover and during coupled and uncoupled hovering operations to minimize attitude excursions, requiring additional time to align the helicopter with the wind line for an approach, and increasing the chances of escape for the submarine which is being tracked. The lightly damped roll oscillations occurring at the 135 and 180 deg.

relative wind azimuth at 20 KCAS were annoying and could be corrected by incorporating modifications to the ASE. The critical azimuth was determined to occur from 135 deg. to 180 deg. relative due to the roll oscillation encountered.

STATIC STABILITY

Longitudinal Static Stability

Low airspeed longitudinal static stability characteristics were evaluated from 10 KCAS rearward to 10 KCAS forward from a trim point of 2 kt. forward under the conditions of appendix F, table I. Data are presented in appendix G, figure 10. Stick force versus airspeed data are presented in appendix G, figure 12B. Positive static stability was indicated by the requirement for increasing forward cyclic displacement and a push force for forward airspeeds and increasing aft cyclic displacement and a pull force for rearward airspeeds. Data indicate the presence of a 6 kt. trim speed band from 3 kt. rearward to 3 kt. forward of zero airspeed. Shallow cyclic force gradients outside the trim speed band averaging 0.08 lb./kt. were masked by high breakout forces of 1-1/2 lb. forward and 2 lb. aft. With the cyclic trim disengaged, weak cyclic displacement cues of approximately 0.1 in./kt. were available to the pilot outside the trim speed band. Pitch attitude remained nearly constant throughout the speed range tested, varying by only 3 deg. over 20 kt. During an uncoupled hover in gusty winds, the pilot will not be distracted from maintaining hover position

by large pitch attitude excursions. The low airspeed longitudinal static stability characteristics are satisfactory for the CV-ASW mission.

Lateral Static Stability

Low airspeed lateral static stability characteristics of the helicopter were evaluated in sideward flight up to 10 KCAS left and right from a trim point of 2 kt. right under the conditions of appendix F, table I. Data are presented in appendix G, figure 11, and force versus airspeed data are presented in appendix G, figure 12A. Positive lateral static stability was indicated by the requirement for left cyclic displacement/force for left lateral flight and right cyclic displacement/force for right lateral flight. Data indicate a lateral trim speed band of six kt., ± 3 kt. about the zero speed point. Pedal displacement, although not consistent with lateral speed, varied by only 0.9 in. over the range of speeds tested. Roll attitude gradients were generally neutral during the test and varied by only 3 deg. over the speed range tested. During a hover in gusty conditions, the pilot will not be distracted by variations in roll attitudes while he maintains position over the sonar cable. The low airspeed lateral static stability characteristics of the helicopter are satisfactory for the CV-ASW mission.

DYNAMIC STABILITY

Long Term Response

The low airspeed dynamic long term response characteristics of the helicopter were evaluated in a 500 ft. AGL uncoupled hover, controls free, using natural and artificial means of excitation under the conditions of appendix F, table I. The test was conducted in light winds (less than 6 kt.) and negligible turbulence. Setting controls free for a period of 20 sec. produced no excursions in pitch, roll, or yaw and the aircraft appeared to hold a relatively constant hover position. Artificial excitation was induced by raising the nose to 5 deg. above the trim attitude and quickly returning the stick to the trim position. The aircraft response appeared deadbeat in that the nose returned to the trimmed hover attitude almost immediately as the aircraft drifted aft and left at a slow constant rate for approximately 10 sec. until recovery was accomplished by the pilot. During sonar dipping operations utilizing manual hover, the pilot will not be distracted from cockpit duties by the requirement to suppress a long term dynamic response. The low airspeed dynamic long term response characteristics are satisfactory for the CV-ASW mission.

CONTROL RESPONSE

General

Low airspeed control response characteristics were qualitatively evaluated in the longitudinal, lateral, vertical, and directional axes during uncoupled hovering flight in winds of 6 kt. or less utilizing control step inputs and during mission tasks under the conditions of appendix F, table I.

Longitudinal

Aircraft response to fore and aft step inputs appeared sluggish and heavily damped. The response was characteristic of an attitude command system in that pitch rates decayed to zero, within 4-5 sec., for all inputs up to 1-3/4 in. aft and 2.0 in. forward. The "commanded" pitch attitudes achieved during the response from a 2 deg. nose up initial hovering attitude are listed below in table II.

Table II
STEADY STATE PITCH ATTITUDE RESPONSE

Forward		Aft	
Input (in.)	Pitch Attitude (deg. nose down)	Input (in.)	Pitch Attitude (deg. nose up)
0.5	0	.5	3
0.75	0	.75	5
1.0	-1	1	9
1.25	-4	1.25	15
1.50	-7	1.50	20
1.75	-10	--	--

An approximate 0.5 sec. lag noted during all inputs was followed by a rapid initial pitch rate that decayed quickly toward zero. Inputs of 1.0 in. and larger were characterized by a sharp vertical jerk felt in the cockpit. These response characteristics caused the pilot to slightly overdrive the aircraft in pitch while setting the flare attitude during an approach. During transitions below 30 KIAS, rapidly commanding 10 ± 2 deg. nose up pitch attitude in one sec. required 1 to 1-1/2 in. aft cyclic followed by 1/4 to 1/2 in. forward cyclic to stop the nose up pitch rate (HQR-3). The sluggish longitudinal control characteristics will cause helicopter pilot to over-rotate the helicopter in the flare during an approach to the ship necessitating extra cyclic and

collective inputs to maintain glide path. The sluggish low airspeed longitudinal control response characteristics are annoying and could be corrected by incorporating modifications to the ASE.

Lateral

Lateral control response characteristics were characterized by approximately 0.5 sec. delay and rapid initial roll accelerations and rates which began to decay almost immediately. The overall roll response to lateral step inputs left and right was sluggish with negligible coupling into the other axes. The combination of delay and heavily damped roll rates caused the pilot to slightly overdrive the aircraft in roll. Commanding small, ± 2 deg., bank angle changes using small, $\pm 1/4$ in., lateral cyclic inputs to maintain lineup during approach to a hover was annoying in that the aircraft overbanked ± 2 deg. beyond the desired value (PIO-4). The ASW pilot will be distracted by the pilot-induced roll oscillation from monitoring glide path and altitude during manual approaches causing him to overshoot his dip point significantly increasing the dip cycle time during submarine prosecution. The oscillatory lateral control characteristics for small trim adjustments during low airspeed and hover is annoying and could be corrected through modifications to the ASE.

Vertical

Aircraft response to collective step inputs was characterized by a jerky initial vertical acceleration in the corresponding direction coupled with a lateral acceleration which was felt in the cockpit and related to control coupling in the yaw axis. Large collective inputs also coupled slightly in pitch in that a large (greater than 1 in.) up collective step input produced a slight nose down acceleration which felt as a forward tilt in the cockpit, but with negligible pitch rates. Vertical response was heavily damped in that the aircraft appeared to reach steady state vertical climb and descent rates in less than 1 sec. The coupled yaw response was heavily damped as well in that yaw rates due to collective input decayed to zero (pedals fixed and pedal switches depressed) after an initial yaw excursion of 2-5 deg. Heading control was fatiguing due to the high pedal forces combined with moderate 1/2 to 1 in. pedal inputs to maintain heading ± 2 deg., while making moderate collective inputs (± 10 -20% torque) (HQR-6). During hover operations over the heaving deck of a destroyer in IMC conditions, the pilot will become fatigued while attempting to counter the collective to yaw coupling characteristics of the aircraft, degrading his ability to maintain precise hover control for longer than 5 to 10 min. The collective to yaw coupling characteristics during high gain hovering tasks is annoying. A stronger heading hold feature could be

incorporated into the ASE to correct this condition.

Directional

Directional response characteristics of the helicopter to pedal step inputs were characterized by slow yaw rates which increased consistently with the size of the input. The response appeared to be heavily damped with negligible delay and negligible coupling. High pedal forces made the response appear sluggish in that approximately 40 lb. of pedal force were required to generate an approximately 15 deg./sec. yaw rate from a 1.0 in. step input left and right. The lack of pedal control centering characteristics made it difficult to adequately predict the trim pedal position during pedal turns. The high pedal forces required to initiate a yaw rate in hover combined with lack of pedal control centering caused poor predictability in capturing desired headings during hovering pedal turns left and right. The sluggish response to pedal inputs will result in poor heading capture causing the pilot to overshoot his desired heading while attempting to make precise yaw adjustments with the pedals while engaged in a sonar dipping hover. Increasing the directional control sensitivity by changing the mechanical control gearing could partially alleviate the high control forces.

TRIMMABILITY

Ease of cyclic trimmability was evaluated qualitatively during all tests. Coupled approaches in the alternate mode

were accomplished using cyclic inputs through the use of beeper trim to establish and maintain 5-8 deg. nose up while the coupler controlled rate of descent. Departures from a coupled hover were accomplished using cyclic beeper trim to set and hold 5 deg. nose down attitude for acceleration while BAR ALT hold programmed the collective to establish a climb to the reference altitude. To maintain desired pitch attitude required 4-5 quick cyclic beeper trim engagements followed periodically by one quick engagement every 2-3 sec. (HQR-3). Longitudinal response to beeper inputs was sluggish in that 2-3 engagements were necessary to establish a perceptible change in pitch attitude. Holding the beeper switch engaged for 2 ± 1 sec. caused the nose to overshoot the desired pitch attitude by 4-5 deg. Aircraft response to quick lateral beeper trim engagements was characterized by a quick initial stick displacement and a lagged roll response of approximately 2 deg. about 0 ± 5 deg. angle of bank. Trimming a 2 deg. bank angle adjustment during a coupled approach to minimize lateral drift rates was annoying in that one quick engagement of the beeper in either direction would consistently result in a 1-2 deg. overshoot requiring the pilot to make infinitesimal trim adjustments by depressing the trim release to manually reposition the cyclic stick (HQR-5). During hover with 5-10 kt. winds, with the cyclic coupler disengaged, holding position within ± 5 ft. laterally and longitudinally with cyclic beeper trim was difficult due

to the sluggish, longitudinal trim response which required several quick beeper engagements to counter small fore-aft drift rates and over sensitive lateral trim response which prevented the pilot from nulling small lateral drift rates to zero. The mechanical characteristics of the beeper trim switch allowed trim adjustments to be input to only one axis (longitudinal or lateral) at a time. To maintain ± 1 deg. nose up pitch attitude while countering small lateral drift indications on the hover indicator during an alternate approach was difficult requiring the pilot to hold attitude using stick forces in one axis while trimming out stick forces with rapid beeper engagements in the other axis (HQR-6). While attempting to maintain optimum attitude for the approach to a hover in one axis, cyclic beeper characteristics will prevent the ASW pilot from quickly trimming out drift or pitch rates in the other axis causing the pilot to exceed pitch or roll tolerances and necessitating a waveoff. This will result in extra time to establish a sonar search hover, increasing a submarine's chances of evasion. The cyclic trimmability characteristics which allow only single axis trim inputs is annoying. A redesigned cyclic beep trim switch which allows for simultaneous two axis trimming should be installed.

MISSION TASKS

Departure

Handling characteristics while performing an upwind departure from a 40 ft hover were qualitatively evaluated under the conditions of appendix F, table I. Transition to forward flight was accomplished by beeping forward cyclic to 5-8 deg. nose down pitch attitude to establish forward drift then disengaging the coupler system by depressing the collective coupler release switch and allowing the BAR ALT hold function to raise the collective to re-establish the aircraft at the initial altitude where the approach/hover coupler had originally been engaged. Establishing the 5 deg. nose down pitch attitude was difficult requiring the pilot to hold the cyclic beeper switch engaged for 2-3 sec in order to effectively override the coupler which was attempting to maintain a zero drift rate (HQR-4). Upon releasing the coupler, the pilot was required to reduce his beeper inputs to the cyclic to keep from overshooting the desired pitch attitude (5-8 deg. nose down), as the coupler was no longer opposing cyclic trim inputs. Simultaneously, upon releasing the coupler, the collective rose rapidly to climb the aircraft to the original reference altitude. The pilot was required to restrain the collective to limit power demands below the aircraft torque limits and control his flight path on climbout. Maintaining flight path control to climb and accelerate from a 40 ft. hover to 150 ft. and 70 KIAS was

difficult requiring the pilot to make several quick forward beeper engagements to continue accelerating while holding downward force on the collective and simultaneously monitoring attitude (pitch and roll), altitude, airspeed, and torque (HQR-6). The ASW helicopter pilot will be excessively tasked during transitions from a coupled hover to forward flight and will overshoot desired altitude and airspeed profiles requiring excessive time to retrim the aircraft for subsequent dips, which will increase the submarines' ability to evade the helicopter. The overtaking coupled hover departure characteristics of the helicopter requiring the pilot to closely monitor and control the pitch, roll, and collective axes during the transition to forward flight is a significant deficiency which should be corrected as soon as practicable. Coupled departure systems currently in use on several helicopters provide for a virtual hands off coupled departure from a hover. The doppler system can provide initial velocity and position cues to the coupler until an airspeed threshold is exceeded at which time an air data computer would take over to control rate of climb and acceleration of the helicopter. A necessary additional feature are engine limiting devices which prevent inadvertant over torque or over temp of the engines during automatic power application. An example is the coupled departure used by the MH-53E Sea Dragon helicopter which provides a rate of climb of 480 feet per minute and airspeed acceleration of

two knots per second to an altitude of 500 feet and 100 KIAS. An improved coupled departure system would greatly reduce cockpit workload, especially during night and low visibility dip to dip operations.

LOW AIRSPEED FLYING QUALITIES SUMMARY

The overall low airspeed flying qualities were a composite of those listed on pp. 20-35. Adequate control margins and static stability characteristics were indicated during all tests. The critical azimuth was determined to occur between the 135 and 180 deg. relative azimuths at 20 KCAS due to lightly damped roll oscillations. Choppy medium to low frequency vibrations encountered while transiting ETL and while hovering in 25-30 kt. headwinds were distracting and would reduce the aircrew mission effectiveness during sonar operation. Moderately high cyclic breakout forces and high pedal forces combined with heavily damped control response in longitudinal, lateral, and directional axes caused difficulty in precise control during high gain tasks. Collective to yaw coupling increased pilot workload significantly during high gain hovering tasks and prevented precise hover altitude control for more than 5-10 min. Automatic departure characteristics severely tax the pilot's ability to maintain precise control of the aircraft while transitioning to forward flight from a coupled hover interfering with his ability to effectively monitor aircraft performance and flight path during departures from a hover.

FORWARD FLIGHT FLYING QUALITIES

TRIMMED FLIGHT CONTROL POSITIONS

Forward flight trimmed flight control positions were evaluated against variations in airspeed (43 to 118 KIAS), torque (4 to 85%) and bank angle (0 to 45 deg. left/right) under the conditions of appendix F, table I. Data are presented in appendix G, figures 13 through 15. Control margins were adequate during all tests. The minimum margin measured was at 118 KIAS with 15% collective remaining from the upper stop. All control position variations were in the proper sense, however, lateral cyclic varied 1.2 in. left to right from 43 to 118 KIAS and the pedals varied from 1.5 in. right at 4% torque to 1.0 in. right at 85% torque. The cyclic and collective controls could be trimmed to zero force during all tests. Field of view (FOV) was adequate during all tests. The ASW helicopter pilot will have adequate control margins throughout the normal airspeed torque and angle of bank ranges in forward flight. The forward flight trimmed flight control positions are satisfactory for the CV-ASW mission.

FORWARD FLIGHT STATIC STABILITY

Longitudinal

Longitudinal static stability characteristics were evaluated at 90 ± 11 KIAS under the conditions of appendix F, table I. Data are presented in appendix G, figures 16 and 17. Positive static stability was indicated for airspeeds

slower than trim by the requirement for aft cyclic displacement and force with decreasing airspeed. A neutral static stability gradient was indicated by the constant stick displacement from 90 to 95 KIAS. Positive static stability was indicated from 95 to 101 KIAS by the requirement for increasing forward stick displacement and force with increasing airspeed. The average longitudinal static gradients are listed in table III.

Table III
AVERAGE LONGITUDINAL STATIC GRADIENTS

Direction	Displacement (in./kt.)	Force (lb./kt.)
Aft	.05	.04
Forward	.03	.03

The shallow gradients listed above were imperceptible to the pilot and masked by the high longitudinal breakout forces. A trim speed band was noted in flight in that the airspeed would vary ± 2 to 3 kt. at a constant fore-aft stick displacement. Maintaining 90 ± 1 KIAS was annoying in that the pilot was required to crosscheck the airspeed indicator excessively and small ($\pm 1/4$ in.) cyclic displacement corrections against trim resulted in a 2-3 kt. overshoot (HQR-5). The ASW pilot will have negligible cues to off-trim airspeed within 3 kt. of trim at 90 KIAS requiring him to

scan his airspeed indicator more often and distracting him from navigational and mission tasks in forward flight. An airspeed hold system should be incorporated into the ASE to maintain selected airspeeds above 60 knots.

Lateral-Directional

a. General. Forward flight lateral-directional static stability characteristics were qualitatively evaluated at 90 KIAS during steady heading sideslips, cyclic only turns and pedal only turns under the conditions of appendix A, table I. Inflight calibration of the balance ball with the aircraft compass resulted in a sideslip angle of approximately 5 deg. per width deflection of the balance ball. Indicated rates of descent were negligible at 2 ball width displacements left and right. The cockpit airspeed indicator on the upwind side indicated approximately 10 kt. greater than the airspeed indicator on the downwind side during sideslips at two ball widths.

b. Directional Stability. Positive directional stability was indicated by left pedal displacement from trim during right sideslips and right pedal displacement from trim during left sideslips. Weak directional stability was indicated in flight during cyclic only turns to 20 deg. bank angle in that the balance ball slipped out one width left during left turns and 1-1/2 widths right during right turns. Negligible proprioceptive cues were present to the pilot at sideslips of up to two ball width displacement left and

right. Rapid cyclic only turns to 20 deg. bank angle left and right yielded negligible adverse or proverse yaw, or yaw due to roll rate. High pedal forces combined with weak proprioceptive cues to an out-of-trim condition caused unpredictable directional trimmability during turns. The pilot was also required to hold a pedal force into the turn directly proportional to the angle of bank to maintain balanced flight. Maintaining a centered balance ball $\pm 1/2$ width deflection during left and right turns at 45 deg. bank angle even while concentrating solely upon the position of the balance ball could not be achieved (HQR-6). The ASW pilot will avoid aggressive turns with the aircraft while rapidly repositioning during sonar search operations, due to the difficulties of sustaining a coordinated turn at moderate bank angles (30-45 deg.). This will result in longer dip cycles, increasing a submarine's chances of evasion. The weak directional stability characteristics of the aircraft during turns at 90 KIAS is a significant deficiency and was the most objectionable handling qualities deficiency identified during the test. The high directional control forces were aggravated by the stability augmentation characteristics of the ASE which was designed to dampen yaw excursions by driving the pedals to oppose rates sensed by the ASE. The pilot could override the pedal movement albeit at higher control forces which were proportional to the yaw rate or turn rate of the aircraft. A commonly employed solution

would be to integrate a lateral accelerometer into the ASE which would sense an out of balance condition during a forward flight turn and program the pedals so as to eliminate the lateral acceleration. This circuit should be disengaged below 60 KIAS to allow for low airspeed maneuvering. Pedal microswitches could also be used to disengage the yaw damping feature of the ASE when the pilot is making pedal turns during hovering flight. The high static pedal forces are due to a mechanical viscous damper which is designed to reduce structural loads to the empennage by preventing rapid directional control inputs by the pilot. While the high static pedal forces are annoying, they serve to prevent structural damage and cyclical fatigue to the empennage and should be retained.

c. Effective Dihedral. Positive effective dihedral was indicated by the requirement for a right lateral cyclic force and displacement from trim during right sideslips and left lateral cyclic force and displacement from trim during left sideslips. Negligible roll rates were generated during pedal only turns left and right which deflected the balance ball out to two ball width displacements from center using both fast and slow pedal displacement. During level flight at 90 KIAS the ASW helicopter pilot will not have to correct for inadvertant aircraft rolls due to lateral gusts. The effective dihedral characteristics of the helicopter are satisfactory for the CV-ASW mission.

d. Sideforce Characteristics. Positive sideforce characteristics were indicated by the increasing right bank angle during right sideslips and increasing left bank angle with left sideslip. Sideforce characteristics were weak and not perceptible to the pilot until a change in bank angle of 5 deg was generated to maintain heading with the balance ball displaced 1-1/2 widths from trim left and right. The weak sideforce characteristics about trim will require the ASW pilot to continually monitor the balance ball, increasing pilot workload during night low altitude transits to the dip area. While satisfactory for the mission, the weak sideforce characteristics of the helicopter were annoying and should be avoided in future designs.

MANEUVERING STABILITY

Maneuvering stability was evaluated at 90 KIAS using steady collective fixed turns, symmetrical pullup/pushovers and qualitatively during simulated mission tasks under the conditions of appendix F, table I. Data are presented in appendix G, figure 18. Average maneuvering stability gradients during steady turns are presented in table IV.

Table IV

STEADY TURN MANEUVERING STABILITY GRADIENTS

Direction	Displacement (in./G)	Force (lb./G)
Left	0.8	0.8
Right	0.4	0.5

Positive maneuvering stability was indicated by the requirement for aft cyclic displacement and force for increasing G and forward cyclic displacement and force for decreasing G. Symmetrical pullup/pushover data were inconclusive in that the attitude command system of the ASE decayed pitch rates which prevented the pilot from attaining steady G levels. Steady turn gradient cues were shallow and masked by the 2 lb. longitudinal cyclic breakout force. Above 35 deg bank angle (1.22 G) the aircraft tended to dig in, manifested as a rise in G level at a constant AOB, and a rapid decay in airspeed. Maintaining 90 ± 3 KIAS during a 40-45 deg. angle of bank turn was difficult, requiring the pilot to focus his scan on the airspeed indicator, increase power and reduce aft cyclic displacement which generated a rate of descent of 300 to 700 FPM (HQR-6). The unpredictable maneuvering stability characteristics of the helicopter during forward flight turns above 45 deg. AOB will cause the ASW pilot to avoid aggressive maneuvering of the helicopter

during dip to dip sonar operations thus increasing dip cycle times which increase an enemy submarine's chances for evasion. The unpredictable maneuvering stability characteristics of the helicopter in forward flight at turns above 40 deg. was annoying. An artificial stick force per G system would give the pilot more positive cues during maneuvering flight.

FORWARD FLIGHT DYNAMIC STABILITY

Longitudinal

a. Long Period Response. The dynamic longitudinal long period response characteristics were evaluated, controls free, at 90 KIAS in light turbulence, as defined in appendix B, table III, using natural and artificial methods of excitation under the conditions of appendix F, table I. Releasing the controls at trim airspeed for a period of 20 seconds resulted in negligible changes in airspeed and pitch attitude. The pilot was unable to artificially excite the long term response in that reducing airspeed to 80 KIAS against trim and resetting the cyclic to the trim position resulted in deadbeat, aperiodic convergent response in which the airspeed slowly returned to 90 ± 2 KIAS in 40 sec. with negligible overshoots in airspeed or pitch attitude. During IMC flight, airspeed control will not be interfered with by the pilot having to suppress a longitudinal long period response. The heavily damped longitudinal long term response at 90 KIAS is satisfactory for the CV-ASW mission.

b. Short Period Response. The longitudinal short period response was evaluated, controls fixed, at 90 KIAS under the conditions of appendix F, table I. The short period response was excited by a 1 to 1-1/2 in. longitudinal cyclic doublet at a frequency of 1.0 cycles per sec. The nature of the response appeared heavily damped with one small overshoot of 1-2 deg. The heavily damped short period response will allow the ASW pilot to make precise flare attitude changes during manual approaches to the dip point. The heavily damped short period response characteristics of the helicopter are satisfactory for the CV-ASW mission.

Lateral Directional

a. Spiral Stability. Spiral stability characteristics of the helicopter were evaluated at 90 KIAS, controls free, under the conditions of appendix F, table I. Lateral cyclic was used to establish 20 deg. AOB left and right with the pedals fixed. Convergent spiral stability was indicated by the aircraft tendency to return to 0 ± 2 deg. AOB as the cyclic was returned to trim. Strong spiral stability was indicated in that the time to half amplitude was 2 sec. for left and right angles of bank. The strongly convergent spiral stability will allow the pilot to maintain trimmed angles of bank up to standard rate turns during turns in IMC conditions. The strongly convergent spiral stability characteristics of the helicopter are satisfactory for the CV-ASW mission.

b. Lateral Directional Oscillation (LDO). Lateral directional oscillations were evaluated at 90 KIAS, controls free, in light turbulence, as defined by appendix F, table I. Natural excitation failed to excite the LDO during 20 sec. of controls free flight. Artificial excitations using a 1 - 1-1/2 in. downward collective pulse and a 1/2 to 1 in. pedal doublet input at approximately 1/2 cycle per sec. failed to excite perceptible coupled yaw or roll oscillations, resulting in an essentially deadbeat LDO response. Rapidly turning through 90 deg. of heading change at 45 deg. AOB failed to excite perceptible yaw or roll oscillations while rapidly rolling in and out of the turns. The ASW pilot will not be required to suppress LDO modes while rolling into or out of turns during mission approaches to the sonar dip point. The heavily damped LDO characteristics are satisfactory for the CV-ASW mission.

CONTROL RESPONSE

Longitudinal

Longitudinal control response characteristics of the aircraft were qualitatively evaluated at 90 KIAS using fore and aft longitudinal control step inputs of up to 2.0 in. and during mission tasks. The responses were characteristic of an attitude command system in that pitch rates from step inputs of up to 1-1/2 in. aft and 2 in. forward decayed to zero in 2-4 sec. resulting only in a net change in pitch attitude. The responses to all step inputs appeared sluggish

and heavily damped with a slight (200-500 millisec) lag between the input and the initial pitch acceleration. Pitch rates appeared to washout toward zero immediately after the initial pitch acceleration except for the response to aft step inputs of greater than 1-1/2 in. which established a slow steady state pitch rate. The maximum pitch rate observed was approximately 10 deg./sec. passing 30 deg. nose up in response to a 2.0 in. aft step input. The response to a 2 in. forward step input washed out at 20 deg. nose down in 4 sec. Overall, the aircraft response was sluggish but predictable and consistent with the size of the step input. Commanding a 20 deg nose up pitch attitude to initiate an approach from 90 KIAS required a single 1 - 1-1/2 in. aft cyclic input (HQR-3). The ASW helicopter pilot will have predictable pitch attitude control during approaches to a dipping sonar hover allowing him to direct more attention to mission tasks. The forward flight longitudinal control response characteristics of the helicopter are satisfactory for the CV-ASW mission.

Lateral

Lateral control response characteristics were qualitatively evaluated in forward flight at 90 KIAS using lateral step inputs of up to 2.0 in. left and right. The aircraft response was characterized by a perceptible lag (approximately 500 milliseconds) followed by a rapid roll acceleration and initial roll rate which quickly decayed

toward zero. The roll response was characteristic of an attitude command control in that roll rates decayed to zero in 3-4 sec. The decay rate was inconsistent in that for step inputs of 1 in. left and right, the response appeared to decay quickly, accelerate and decay again over several cycles creating a ratchet roll response. The roll response to a 2 in. right input nulled at 23 deg. right AOB in 3-4 sec. and a 2 in. left response continued to roll slowly through 30 deg. left AOB at 3 sec. when recovery was accomplished. The sluggish response caused the pilot to overshoot desired bank angles during rapid rolls to high bank angles. Rapidly commanding 40 to 45 deg. right AOB using 2-3 in. of right cyclic displacement during an aggressive 270 deg. turn to the wind line, simulating a rapid reposition during a dip cycle, was unpredictable in that the roll response began to decay as the aircraft approached the desired bank angle and then accelerated briefly towards the 45 deg. AOB limit requiring an immediate opposite cyclic input of 2-3 in. to prevent the aircraft from exceeding 45 deg. AOB (HQR-6). During aggressive rolling maneuvers to set up for a dip while tracking a submarine, intense pilot attention will be required for roll control diverting his attention from mission tasks. The unpredictable oscillatory roll rate of the helicopter during rapid rolling maneuvers to high bank angles at 90 KIAS could be hazardous. The aircraft should be

restricted to 35 deg. AOB or less to enhance safety during low altitude maneuvering.

GUST RESPONSE

Gust response characteristics of the helicopter were qualitatively evaluated during flight through intermittent light turbulence, as defined in appendix B, table III, between 1000 and 500 ft. AGL at 90 KIAS under the conditions of appendix F, table I. Gust response characteristics were manifested as small vertical accelerations not overly perceptible to the pilot. Ride quality was slightly rougher in that small gust accelerations could be felt in the cockpit when not otherwise occupied with mission tasks. Attitude changes in the pitch, roll, and yaw axes were negligible during gusts and no tendency to excite any of the longitudinal or lateral-directional dynamic modes was noted. Airspeed varied $\pm 2-3$ kt. about trim during light gust accelerations and generally returned to trim in 5-10 sec. Flight path control in light turbulence during a 500 ± 50 fpm descent at 90 KIAS simulating a Ground Controlled Approach (GCA) was easy requiring small ($\pm 1/8$ in.) longitudinal cyclic inputs every 5-10 sec. and occasional small ($\pm 3-5\%$) torque changes with the collective every 10-15 sec. (HQR-3). The deadbeat gust response characteristics of the helicopter will permit the pilot to maintain adequate flight path control during a GCA in light turbulence. The gust response

characteristics of the helicopter are satisfactory for the CV-ASW mission.

TRIMMABILITY

The ease and effectiveness of trimming the helicopter in forward flight in the lateral and longitudinal axes was evaluated during all tests under the conditions of appendix F, table I. Trimming in the longitudinal and lateral axis was accomplished through the use of the cyclic trim release switch for manually repositioning the cyclic and with the four-way "coolie hat" type cyclic beeper switch which programs the cyclic at a constant rate laterally and longitudinally. The cyclic trim release was adequate for large trim changes during transitions in that the pilot could set pitch attitude and bank attitudes ± 3 deg. and airspeed ± 5 kts. Stick jump characteristics during a 1 in. cyclic displacement from trim at 100 KIAS were negligible in the longitudinal axis. In the lateral axis a light momentary oscillation was felt coupled to a 1/8 to 1/4 in. jump in the stick. Trim adjustments using the beeper trim were somewhat unpredictable. Longitudinal trimmability using the beeper trim for pitch attitude was adequate to within 1 deg., however, trimming airspeed to within 1 kt. of desired airspeed was difficult. Lateral trimmability was more sensitive in that it was difficult to make moderate (10-20 deg.) angle of bank adjustments using the beeper trim. Trimming from 30-10 deg. left angle of bank at 100 KIAS using approximately 1

sec. of beeper engagement resulted in the aircraft overshooting the desired bank angle and stabilizing at 10 deg. right angle of bank. Trimming from zero to 25 deg. right AOB at 100 KIAS was annoying in that the pilot engaged the beeper switch for 1-2 sec., then as the bank angle approached 20 deg., two quick beeper engagements to counter the roll were required as the bank angle overshot to 30 deg. (HQR-4). Excess pilot attention will be required to monitor bank angle, while using lateral beeper trim switch to avoid overshooting desired bank angles which will distract the pilot from other cockpit duties. The excessively sensitive lateral cyclic beeper trim in forward flight is annoying. The ASE roll channel output signal gains should be adjusted to reduce the over-sensitive trim characteristics.

FORWARD FLIGHT FLYING QUALITIES SUMMARY

The overall forward flight flying qualities of the SH-3H helicopter were the composite of those listed on pp. 36-50. Unpredictable lateral control response characteristics will distract the pilot, in order to prevent the aircraft from exceeding the 45 deg. AOB limit during rapid rolling maneuvers. The 4-6 kt. trim speed band at 90 KIAS in level flight prevented precise airspeed trimmability and the sensitive lateral cyclic beeper trim prevented precise AOB trimmability. High pedal forces and poor proceptive cues at up to 2 displacements of the trim ball, made directional trimmability annoying during turns. Moderate vibration

levels (VAR-5) were present during all forward flight regimes and were most severe at angles of bank above 40 deg. (VAR-7). Good speed and attitude retention was indicated during unaccelerated level cruise between 43 and 120 KIAS. The longitudinal and lateral directional modes tested were deadbeat and difficult to excite. Pilot workload during non-maneuvering flight will be low allowing him to concentrate on mission tasks, however the maneuvering flight characteristics will tax pilot ability to maintain precise flight path control leaving limited excess capability to handle adverse conditions or mission tasks.

PERFORMANCE

ENGINE ASSESSMENT

General

Assessment of engine characteristics was accomplished qualitatively and quantitatively under the conditions of appendix F, table I. Engine start, acceleration and shutdown were accomplished per the procedures listed in reference 2. Static droop characteristics were evaluated during vertical climbs. Transient droop was evaluated qualitatively in forward flight and varied a maximum of 2% with changes of up to 40% torque in 2 sec. Engine power available data, calculated from reference 2, are presented in appendix G, figure 19, for standard and hot day conditions.

Engine Start and Shutdown

A total of 10 engine starts were performed including one warm engine restart. Cold engine starts were characterized by a 5-11 sec. runup to 19% Compressor speed (Ng) where the engine speed lever was positioned to ground idle. Light off occurred in 2-4 sec. followed by a rapid rise in Turbine Temperature (T5) to 690-730 deg C. Engine acceleration to idle was normal in 11-14 sec. from lightoff. Engine shutdowns were characterized by a rapid decay in Ng, and T5 as the speed lever was moved to the shutoff position. The T5 would decrease to between 100 and 150 deg C in 25 to 30 sec. and then rise to 200-250 deg. The warm engine restart was accomplished 4 min after shutdown with the T5 at 210 deg. C and relative winds at 070 deg., at 6 kt. Approximately 10 sec. were required to cool the engine to 100 deg. C before fuel was introduced. Lightoff was characterized by a rapid rise in T5 towards 800 deg. C. The "ENG ST" button, forward of the cyclic grip, was employed to stabilize T5 between 760 and 780 deg. until the temperature decayed to the idle T5 of 480-500 deg. C. The ENG ST button is designed to reduce fuel flow to the engine during engine start to prevent T5 from exceeding 840 deg. C. The engine start procedure is a two handed operation requiring the pilot to keep one hand on the speed lever and the other hand on the ENG ST button. The requirement to closely monitor T5 during engine start distracts the pilot from monitoring for cut signals from the

ground crew in the event of an engine fire on start. The requirement for the pilot to closely monitor and modulate T5 was annoying. An engine control system should be incorporated which could automatically modulate T5 on start.

Torque Matching

The capability of the engines to maintain matched torque during collective changes was qualitatively evaluated during flight. During small power additions from 60% to 70%, an 8% split between engine torques was noted. A moderate power addition from 45% to 70% torque produced a torque split of 25%. Reducing power from 70 to 35% produced an 18% torque split. Maintaining matched engine torque inflight was annoying, requiring the co-pilot to adjust the speed levers after power changes greater than 5%. During dip to dip operations which require large power changes, the co-pilot will be frequently distracted from mission tasks in order to maintain matched engine torques. The insufficient torque matching capability of the engines during power changes is annoying and engine controls should be procured which would automatically match engine torque during normal operations.

Static Droop

Static droop data are presented in appendix G, figure 20. The total static droop between the minimum collective pitch power point and the military power was 4% rotor speed. The decreasing character of the curve with increasing power

reflects an under cancelled gradient. Standard procedure was to set 104% Rotor RPM (Nr) prior to takeoff to compensate for the static droop characteristics. During flight, large power changes to transition to a climb or descent resulted in the necessity to adjust rotor speed to maintain between 100 and 102% Nr. During transitions to a hover, the co-pilot will be required to modulate the engine levers distracting him from mission tasks. An engine control-power management characteristics of the helicopter are an annoying characteristic. Rotor governing could be improved by the addition of an upgraded engine power management system.

HOVER

General

Hover performance characteristics were evaluated using the free hover technique under the conditions of appendix F, table I. Test day hover performance data were collected for 40 ft. IGE hover height and 100 ft. OGE hover heights, at 100% rotor RPM, standard day (15 deg. C at sea level) and hot day (40 deg. C at sea level) conditions, for the maximum aircraft gross weight (21,000 lb.) and the specification mission gross weight (20,800 lb.). Two typical mission gross weight configurations are outlined in table V. Recommendations for correction of hover power deficiencies are presented at the end of this section.

Table V
MISSION CONFIGURATIONS

Anti-submarine Warfare			Search and Rescue	
Parameter	Weight (lb.)		Parameter	Weight (lb.)
Basic Aircraft	13,886		Basic Aircraft	13,886
Crew (4)	800		Crew (5)	1,000
MK46 Torpedo (1)	530		Mission Equipment	214
Sonobuoy (6)	174		Fuel	2,300
Fuel	3,000	2,000		
Total	18,390	17,390	Total	17,400

Engine power available was determined from specification data presented in reference 5 at the military (30 min.) power rating.

Out of Ground Effect (OGE)

a. Hot Day. OGE hot day hover power required and hover ceiling data are presented in appendix G, figures 21 and 23. The power required to hover at sea level hot day (40 deg. C) and the power deficiency based upon 1948 Engine Shaft Horsepower (ESHP) available are shown in table VI.

Table VI

SEA LEVEL HOT DAY HOVER
OGE POWER REQUIREMENTS

Weight (lb)	Power Required (ESHP)	Power Surplus	
		ESHP	Percent
17,400	2,267	-319	-14
18,400	2,437	-489	-20
20,800	2,863	-915	-32
21,000	2,899	-951	-33

At test day conditions, the aircraft was also unable to hover OGE at sea level, 20 deg. C OAT, and a gross weight of 20,062 lb. The inadequate hot day hover performance at mission gross weights will prevent the aircraft from deploying the active sonar, which will excessively expose the carrier battle group to hostile submarine threats. The inadequate 40 deg. C. hot day OGE hover performance at gross weights between 17,400 lb. and 21,000 lb. is a serious deficiency which should be corrected as soon as possible.

b. Standard Day. OGE standard day hover power required and hover ceiling data are presented in appendix G, figures 22 and 23. The power required to hover at sea level standard day data based upon available power of 2,470 ESHP are presented in table VII.

Table VII

SEA LEVEL STANDARD DAY
OGE HOVER POWER REQUIREMENTS

Weight (lb)	Power Required (ESHP)	Power Surplus	
		ESHP	Percent
17,400	2,218	252	11
18,400	2,381	89	4
20,800	2,789	-319	-11
21,000	2,825	-355	12

The maximum OGE hover gross weight of the helicopter at sea level standard day was 18,750 ESHP which would allow, based upon the typical mission profile presented in Chapter I, 1,184 lb. of fuel available for sonar search hover operations. The 1,184 lb. of fuel reduced the number of sonar dip evaluations from six to three, a reduction of 50% mission capability. The limited fuel load of the aircraft will require the carrier group commander to shuttle extra aircraft to the area of operations to prosecute and track a submarine contact, diverting the helicopter from other fleet duties. The limited sea level standard day OGE hover capability of the helicopter is a serious deficiency which should be corrected as soon as possible.

In Ground Effect

a. Hot Day. 40 ft IGE hover power required and hover ceiling data are presented in appendix G, figures 24 and 26. The power requirements for a sea level hot day (40 deg. C) IGE hover capability based on 1948 dual ESHP available are presented in table VIII.

Table VIII
SEA LEVEL HOT DAY
40 FT IGE POWER REQUIREMENTS

Weight (lb.)	Power Required ESHP	Power Surplus	
		ESHP	%
17,400	2,212	-264	-12
18,400	2,380	-432	-18
20,800	2,804	-856	-31
21,000	2,840	-892	-31

The minimum SAR mission weight, 17,400 lb., allows for 2,300 lb. of fuel minus 600 lb. for startup and fuel reserve and minus 300 lb. for a 15 min. hover, which permits SAR coverage out to 56 miles from the carrier based on transit at 114 KTAS (.0780 nm/lb. specific range). During hot day conditions the aircraft had inadequate IGE hover capability at the minimum SAR gross weight. The inadequate SAR capability during hot day conditions will require the carrier air wing commander to severely restrict fixed wing operations around the carrier. The negligible sea level hot day (40 deg. C.) IGE hover

capability of the helicopter at mission gross weight is a serious deficiency which should be corrected as soon as possible.

b. Standard Day. 40 ft. IGE standard day hover power required and hover ceiling data are presented in appendix G, figures 25 and 26. The power requirements for a sea level standard day IGE hover based on 2470 ESHP available are presented in table IX.

Table IX
SEA LEVEL STANDARD DAY
40 FT IGE POWER REQUIREMENTS

Weight (lb.)	Power Required ESHP	Power Surplus	
		ESHP	%
17,400	2,159	311	14
18,400	2,321	149	6
20,800	2,727	-257	9
21,000	2,762	-292	11

The maximum sea level standard day IGE hover gross weight, 19,000 lb., allows for a fuel load of 3,900 lbs. The 3,900 lb. minus start, taxi, and hover fuel, as described in paragraph a, leaves 3,000 lb. available for SAR support which will provide for 2 hr. of endurance at 68 KTAS while performing plane guard duties or provide for a mission radius out to 120 nm from the carrier at 114 KTAS (.0780 nm/lb.

specific range) for extended SAR operations. During sea level standard day conditions, the aircraft will be capable of providing adequate SAR capability for the carrier air wing. The sea level standard day 40 ft. IGE hover performance of the helicopter is satisfactory for the CV-SAR mission.

LEVEL FLIGHT PERFORMANCE

General

Level flight performance was evaluated during one flight at 5 airspeed increments from 43 to 118 KCAS under the conditions of appendix F, table I. Airspeed position error corrections were accomplished using data from reference 2. Test day data were generalized to a reference (W/σ) weight of 20,800 lb., then reduced to sea level standard day conditions. Data for the sea level standard day condition are presented in appendix G, figures 27 and 28. Fuel and endurance requirements for a typical mission profile were 4678 lb. and 3.43 hr. Typical Mission Profile data are presented in appendix F, table III. Typical mission configuration and weights are presented in table X. Level flight performance data for 20,800 lb. gross weight is presented in table XI.

Table X

SH-3H DESIGN MISSION CONFIGURATION

Typical aircraft weight	13,450 lb.
Aircrew (4)	800 lb.
MK46 Torpedoes (2)	1,060 lb.
Sonobuoys (10)	290 lb.
Fuel	5,200 lb.
Total	20,800 lb.

Table XI

LEVEL FLIGHT PERFORMANCE DATA

Maximum Level Flight Airspeed	122 KTAS
Maximum Range Airspeed	104 KTAS
Maximum Endurance Airspeed	68 KTAS
Cruise Airspeed	114 KTAS
Maximum Range	370 nm
Maximum Radius	168 nm (Vcruise)
Maximum Endurance	3.8 hr.
Maximum Specific Range	0.079 nm./lb.

Maximum Level Flight Airspeed

Level flight data show that power required equals maximum power available at 122 KTAS. The maximum level flight airspeed achieved during the test, 118 KIAS, was characterized by a choppy medium frequency vertical vibration, accompanied by high noise levels in the cockpit (VAR 6). While transiting to the operating area to prosecute a known submarine contact at 118 KIAS or greater, the crew will be distracted by the high noise levels and vibrations in the aircraft which will interfere with their pre-dip preparations. The distracting vibration levels and noise levels in the aircraft at 118 KIAS are annoying. Recommendations to improve this characteristic are outlined in the Vibrations section.

Maximum Range

Maximum range performance was evaluated based on the typical mission fuel load (5,200 lb.) minus 150 lb. for engine start, rotor engagement, takeoff and climb, and 450 lb. minimum fuel at final landing, leaving 4,600 lb. useable mission fuel. Using the 104 KTAS which corresponds to the maximum specific range (.079 nm/lb SR) the aircraft range was calculated to be 370 nm at an endurance of 3.5 hr. The aircraft will be capable of transiting 370 nm to a picket destroyer which is equipped for refueling the aircraft, to conduct ASW operations. The maximum range performance of the helicopter is satisfactory for the carrier ASW mission.

Mission Radius

The mission radius of the aircraft was evaluated at cruise airspeed (114 KTAS) and maximum range airspeed (104 KTAS) allowing for a useable mission fuel load of 4,300 lb. which is the 5,200 lb. engine start fuel minus 150 lb for start and takeoff, 450 lb. reserve and 300 lb. for one 15 min. hover. Mission radius and endurance data are presented in table XII for the two airspeeds tested.

Table XII
MISSION RADIUS

Airspeed (KTAS)	Radius (nm)	Endurance (hr)
104	173	3.3
114	171	2.9

At 114 KTAS, the aircraft will reach its mission radius approximately 10 min. faster than when flying at 104 KTAS and will return to the carrier approximately 20 min. faster at the expense of only 2 miles in total radius capability. During a personnel transfer to a picket destroyer at the mission radius of the aircraft, the availability of the helicopter to the air wing for on-call missions, i.e., ASW, SAR, or fleet support, will be increased when the aircraft cruises at 114 KTAS vs. 104 KTAS. The mission radius corresponding to 114 KTAS is satisfactory for the CV-ASW mission.

Maximum Endurance Airspeed

Maximum endurance airspeed evaluated at 68 KTAS with 4,300 lb. useable fuel was 3.8 hr. Vibration levels in level flight from 60 to 90 KIAS consisted of a noticeable medium frequency vertical vibration (VAR-5) which is distracting to the crew. Vibration levels at 100 to 110 KIAS consisted of the same approximate frequency but at a lower amplitude (VAR-4). During plane guard detail around the carrier, the pilot will increase speed to 110 KIAS to avoid the vibration levels at 70 KIAS which will cut the aircraft endurance by 28 min. (12%).

FORWARD FLIGHT CLIMBS AND DESCENTS

Forward flight climb and descent performance was qualitatively evaluated at 70 and 90 KIAS under the conditions of appendix C, table I. The maximum rates of climb measured at 70 KIAS and 90 KIAS using maximum rated power were 1050 fpm and 750 fpm, respectively, up to 5000 ft. pressure altitude. The maximum rate of descent at 90 KIAS 4% torque and 103% rotor speed was 2200 fpm. Vibration levels in the climb were medium frequency vertical vibrations (VAR-4). Vibration levels in the descent were characterized by a choppy low frequency vibration (VAR-5). The ASW pilot will be capable of climbing to 5000 ft. pressure altitude for sonobuoy data link transmissions to the ship in 4.7 min. and then descending quickly to 150 feet to commence his sonar

search cycle. The forward flight climb and descent performance of the helicopter is satisfactory for the CV-ASW mission.

VERTICAL CLIMB PERFORMANCE

Vertical climb performance was evaluated under the conditions of appendix F, table I. Rate of climb data at standard day conditions for 17,400 and 18,400 lb. gross weight are presented in appendix G, figure 29. At these gross weights, the aircraft was incapable of climbing vertically under hot day conditions due to the negligible hot day performance characteristics of the helicopter. The maximum vertical climb rates at sea level standard day including, a 5% power margin for fleetwide engine variations for 17,400 and 18,400 lb. was 680 fpm and 300 fpm, respectively. A malfunction requiring the helicopter to break dip immediately would necessitate a vertical climb to pull the sonar clear of the water freestream recovery procedure) prior to transitioning to forward flight within 30 sec. An 18,400 lb. aircraft hovering with the sonar submerged to 500 ft. search depth will require approximately 1.7 min. to pull the sonar clear of the water. Due to the slow vertical climb rate of the helicopter, the ASW pilot will jettison the sonar and cable in order to transition to forward flight as soon as possible, which will deny the carrier battle group an airborne ASW asset until a new sonar can be fitted. The slow sea level standard day

vertical climb rate at a moderate dip weight of 18,400 lb. is a significant deficiency which should be corrected as soon as practicable.

PERFORMANCE SUMMARY

The overall performance characteristics of the SH-3H were the composite of those listed on pp. 51-66. The negligible sea level hot day hover performance and marginal standard day hover performance at minimum mission gross weights will prevent the carrier battle group from effectively utilizing the aircraft active sonar capability for anti-submarine protection. The current engine, the T58-GE-10 engine, manufactured by General Electric Aircraft Engines Group of Lynn, Massachusetts, has been installed in the SH-3 since 1973. A growth derivative of the engine, the T58-GE-16 has been available and has been in use in the CH-46E Sea Knight helicopter since 1980. Summaries of T58-GE-10 and -16 engine performance data from reference 7 are presented in appendix F, tables IV and V. Under static sea level standard day conditions, the -16 engine delivers an uninstalled power rating of 1870 SHP compared to 1400 SHP for the -10 engine. Assuming identical engine installation losses which reduce the output power of each engine by approximately 12 percent, the sea level Standard Day power available to the H-3 from two -16 engines would be as high as 3300 SHP, a 34 percent increase over the current power from the -10 engines. The -16 engines could provide approximately

500 SHP greater than the power required to hover OGE on a hot day at the aircraft maximum gross weight of 21000 lb., cited table VI above. Hover power available under standard and hot day conditions is graphically presented in appendix G, figures 22 and 25. Currently the SH-3 drive train is capable of sustaining only 2500 SHP. An upgraded transmission drive train and tail rotor could be designed to accommodate the additional torque loads of the -16 engines. Performance penalties incurred by upgrading engine and drive train would be an increase in specific fuel consumption of 8% for the engines upgrade, an increase in the weight of the engines by 93 lbs. per engine and an undetermined weight increase for the drive train upgrade. The increase in specific fuel consumption would reduce the range and endurance of the helicopter. Because of power limitation, the aircraft is currently operated at a reduced fuel load the added available power would permit the aircraft to carry more fuel. Further testing would be necessary to determine the actual performance of the helicopter with the -16 engines installed.

VIBRATIONS

Helicopter vibration levels were qualitatively evaluated during all tests. During hovering flight in light winds (less than 5 kt.) from the 225 to the 130 deg. relative azimuths produced medium frequency small amplitude vibrations (VAR-3). Vibration levels were still present but appeared to drop off slightly while translating forward at 10 to 20 KCAS

(VAR-2). Along other azimuths, vibration levels increased in intensity (VAR-4) as the relative wind increased through 10 kt. except in the right and rear azimuth (090-180 deg.) where the vertical vibrations seemed to increase to a higher amplitude and frequency seemed to decay slightly (VAR-5). The most noticeable vibrations were characterized by a low frequency vertical choppiness (VAR-8) which occurred in forward translations between 25-30 kt. and during approaches and departures. The onset of the much higher vibration levels, during these maneuvers, from VAR-2 to VAR-6 were slightly distracting in the cockpit. During a sonar search hover with 25 to 30 kt. head winds, the aircrew will be distracted from mission tasks and procedures by the choppy low frequency vertical vibrations. In forward flight the normal levels at 40 to 90 KIAS level flight were characterized by a moderate low to medium frequency vertical vibration felt in the airframe (VAR-5). The minimum vibrations levels encountered were during a 90 KIAS descent at 30% torque and were characterized by a slightly higher frequency at about the same relative amplitude (VAR-3). Cruise flight from 100 to 110 KIAS yielded slightly higher vibration levels (VAR-4). Vibration levels during left and right turns at 90 KIAS appeared to increase proportionately with angle of bank from the normal levels (VAR-5) at 15 deg. AOB to a choppy distracting low frequency vibration at 30 deg. AOB and then to choppy higher amplitude vertical

vibrations (VAR-7) at 40-45 deg. AOB, of which were particularly distracting to the pilot. The pilot will avoid bank angles above 35 deg. due to the distracting vibration levels which interfere with mission tasks. Although not of high enough intensity and/or long enough duration to be fatiguing to the crew or detrimental to the aircraft structure, a cost effective solution is available to reduce or minimize the helicopter vibration. An analysis of the vibration levels would likely show the noticeable frequencies to be at a multiple of the main rotor frequency. Improved rotor system balancing which optimize main and tail rotor blade tracking through the use of blade trim tab adjustments, control rod adjustments and tip or hub weight balance should be incorporated into the helicopter maintenance procedures. This would be a relatively inexpensive solution which can be implemented with a minimum of new equipment and a slight increase in training time for maintenance personnel.

CONCLUSIONS

The primary mission requirement of the Sea King Helicopter is to have an over-water hover capability at mission gross weights under a wide range of ambient conditions from a hover, the submersible sonar may be deployed down to search depths of several hundred feet to facilitate detection, identification and tracking of a hostile submarine. The helicopter demonstrated very limited potential to accomplish the phase of the mission except under low density altitude conditions and/or at a greatly reduced mission gross weight. Replacing the engines with the T58-GE-16 engines will significantly improve the hover performance of the helicopter under warm ambient temperature conditions and should be incorporated. The new engine could be procured at a unit cost of four to six hundred thousand dollars. Significantly below the 22 million dollar unit cost for the SH-60F.

While the aircraft displayed no adverse handling qualities which would preclude mission accomplishment, correction of the most significant deficiencies, the high directional control forces and lack of automatic turn

coordination would enhance mission effectiveness, allowing the pilots greater concentration on mission tasks.

The automatic departure characteristics which consumed an inordinate amount of pilot attention should also be corrected as soon as practicable.

The annoying deficiencies cited throughout this treatise are easily compensated for or occur infrequently enough that the benefits of their correction would not justify the costs involved. Funding for correction of annoying deficiencies could be better invested into other programs. Correction of the more serious deficiencies cited above however would allow the helicopter to remain as a formidable ASW weapon thereby assuring the U.S. Navy of the capability to continue its global presence in the interest of world peace.

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DEFINITION OF DEFICIENCIES

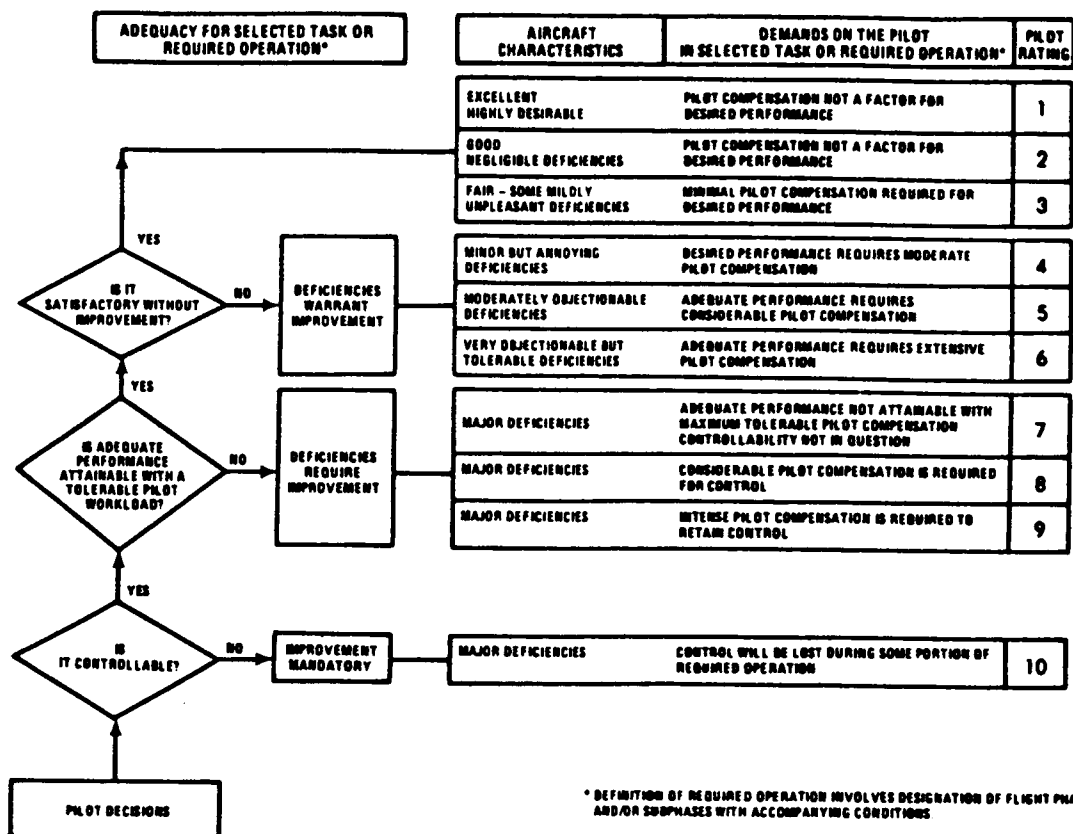
"Serious" indicates a deficiency, the correction of which is necessary because it adversely affects:

- 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.

"Significant" indicates a deficiency of lesser severity than "Serious" 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.

"Annoying" 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.

COOPER-HARPER HANDLING QUALITIES RATING SCALE



APPENDIX B

VIBRATION ASSESSMENT RATING SCALE

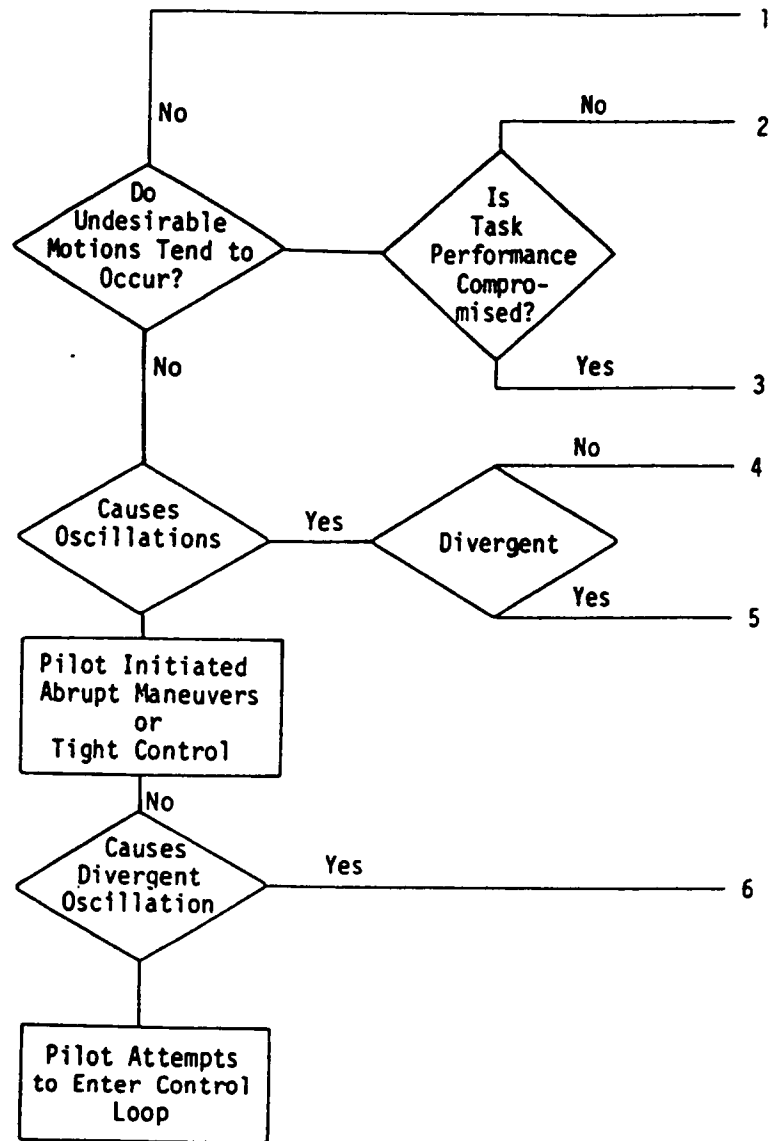
Rating	Term used for Deg	Description
0	No vibration	
1 2 3	Slight	Not apparent to experienced aircrew fully occupied by their tasks but noticeable if their attention is directed to it or if not otherwise occupied.
4 5 6	Moderate	Experienced aircrew are aware of the vibration but it does not affect their 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.

APPENDIX B

TURBULENCE CRITERIA

Intensity	Aircraft reaction	Reaction inside aircraft	Reporting term	Definition
LIGHT	Turbulence that momentarily causes slight changes in aircraft attitude, altitude, or heading. Report as <u>Light Turbulence</u> . OR Turbulence (light bumpiness) that causes slight aircraft fluctuations at rapid intervals without appreciable change in attitude, roll, or yaw. Report as <u>Light Chop</u> .	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects remain at rest, food service may be conducted, and little or no difficulty is encountered in walking.	Occasional Intermittent Continuous	Less than 1/3 of the time 1/3 to 2/3 More than 2/3
MODERATE	Turbulence that is similar to Light Turbulence but of greater intensity. Changes in aircraft attitude, altitude, or heading occur, but the aircraft remains in positive control at all times. Report as <u>Moderate Turbulence</u> . OR Turbulence that is similar to Light Chop but of greater intensity. It causes rapid bumps or jolts without appreciable change in aircraft attitude, roll, or yaw. Report as <u>Moderate Chop</u> .	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged. Food service and walking are difficult.	NOTE 1. Pilots should report location(s), time, intensity, altitude, type of aircraft and, when applicable, duration of turbulence. 2. Duration may be based on time between two locations or over a single location. All locations should be readily identifiable. EXAMPLES a. Over Omaha, 1232Z, moderate turbulence, Flight Level 300, 8707 b. From 50 miles south of Albuquerque to 30 miles north of Phoenix, 1235Z to 1250Z, occasional Moderate Chop, Flight Level 320, DC-8	
SEVERE	Turbulence that causes large changes in aircraft attitude, altitude, or heading. It may cause large variations in indicated airspeed. Aircraft may be momentarily out of control. Report as <u>Severe Turbulence</u> .	Occupants are forced violently and repeatedly against seat belts or shoulder straps. Unsecured objects are tossed about. Food service and walking are impossible.		
EXTREME	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage. Report as <u>Extreme Turbulence</u> .			

PIO TENDENCY CLASSIFICATION



APPENDIX B

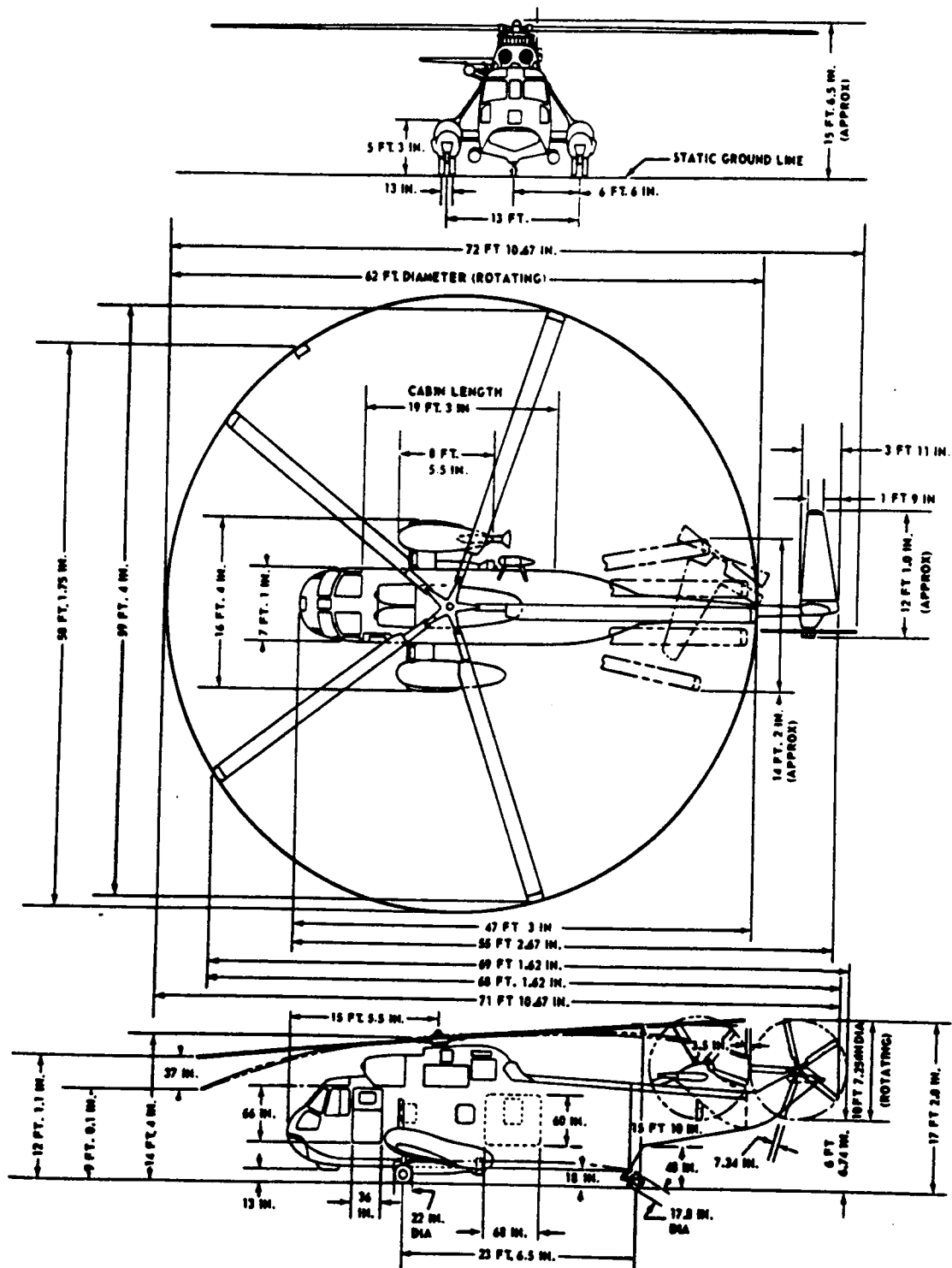


Figure 1
THREE VIEW DRAWING OF THE SH-3H HELICOPTER

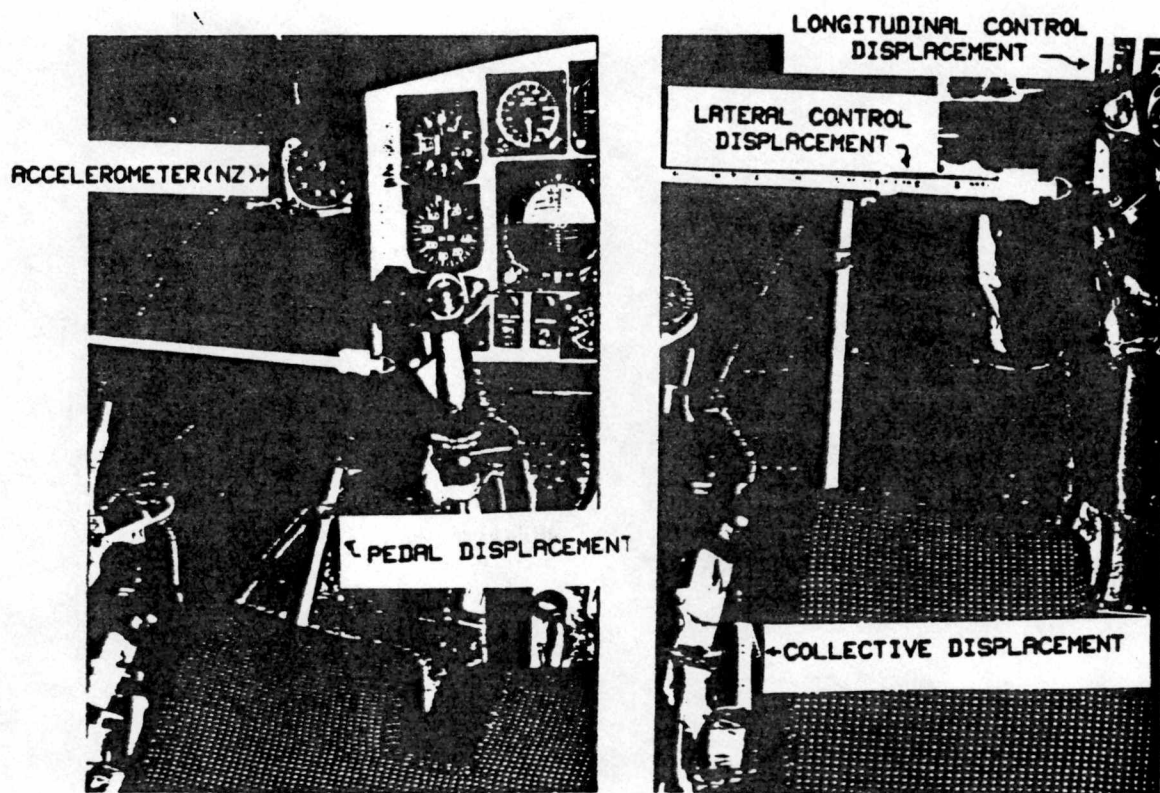
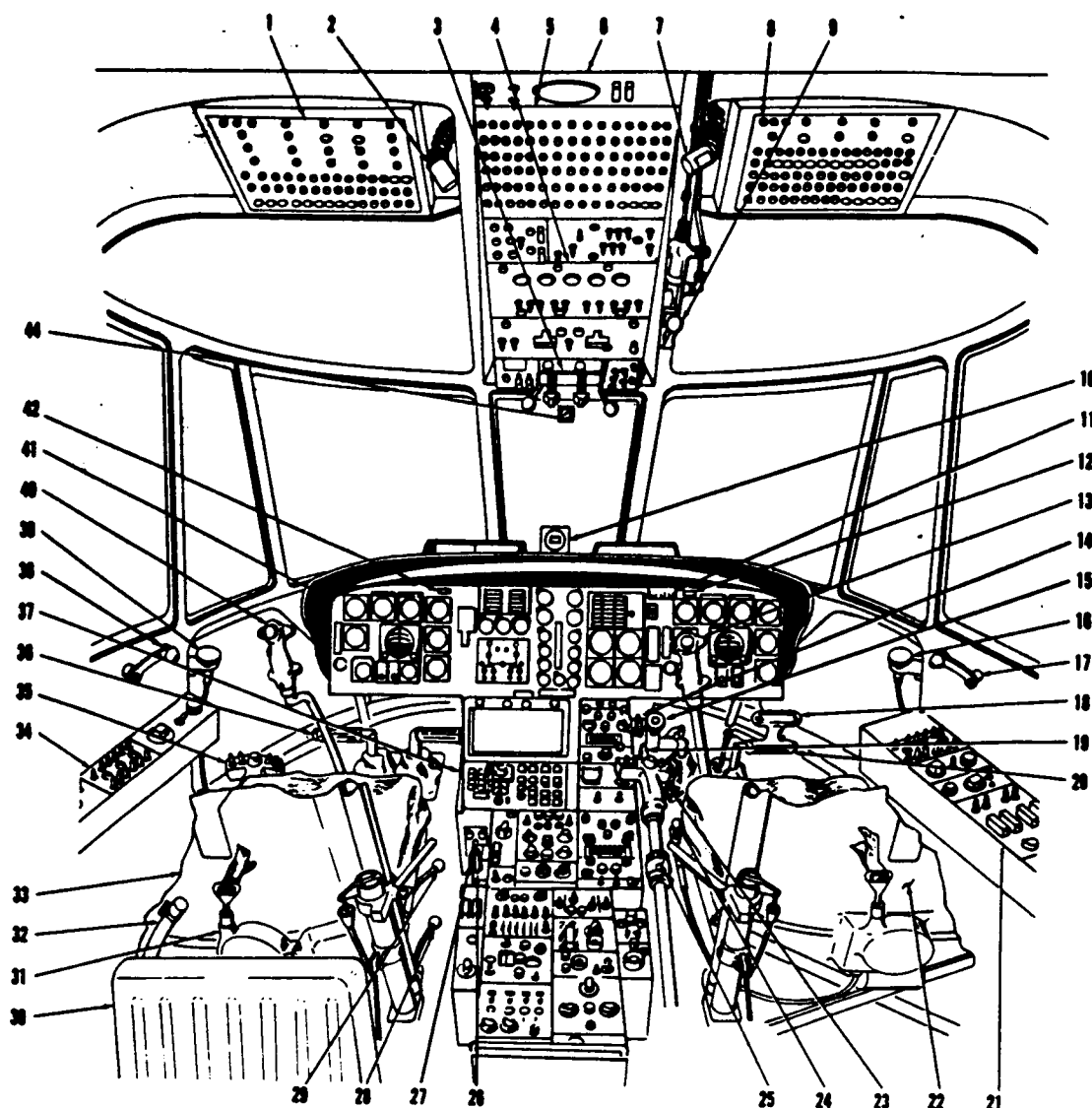


Figure 2
COCKPIT INSTRUMENTATION

APPENDIX C



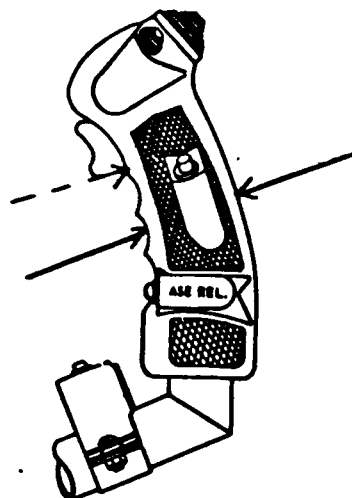
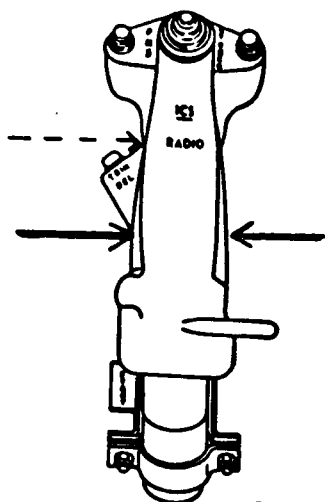
1. COPILOT'S CIRCUIT BREAKER PANEL
2. COPILOT'S SPOTLIGHT
3. ENGINE CONTROL QUADRANT
4. OVERHEAD SWITCH PANEL
5. CENTER CIRCUIT BREAKER PANEL
6. PILOT'S COMPARTMENT DOME LIGHT PANEL
7. PILOT'S SPOTLIGHT
8. PILOT'S CIRCUIT BREAKER PANEL
9. ROTOR BRAKE LEVER
10. STANDBY COMPASS
11. FIRE WARNING LIGHT
12. MASTER CAUTION LIGHT
13. PILOT'S CYCLIC STICK
14. PARKING BRAKE HANDLE
15. WINDSHIELD WIPER KNOB
16. PILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
17. PILOT'S WINDOW EMERGENCY RELEASE HANDLE
18. PILOT'S TOE BRAKES
19. TAILWHEEL LOCK HANDLE
20. PILOT'S TAIL ROTOR PEDALS
21. PILOT'S CONSOLE
22. PILOT'S SEAT

23. PILOT'S COLLECTIVE PITCH LEVER
24. PILOT'S SHOULDER HARNESS LOCK LEVER
25. COLLECTIVE STICK FRICTION CONTROL
26. EMERGENCY LANDING GEAR EXTENSION HANDLE
27. LANDING GEAR CONTROL PANEL
28. COPILOT'S SEAT HEIGHT ADJUSTMENT LEVER
29. COPILOT'S SEAT FORWARD AND AFT ADJUSTMENT LEVER
30. SEA ANCHOR
31. COPILOT'S SHOULDER HARNESS INERTIA REEL (TYPICAL)
32. ANTI-EXPOSURE CONNECTION
33. COPILOT'S SEAT
34. COPILOT'S CONSOLE
35. COPILOT'S COLLECTIVE PITCH LEVER
36. COPILOT'S TAIL ROTOR PEDALS
37. COCKPIT CONSOLE
38. COPILOT'S WINDOW EMERGENCY RELEASE HANDLE
39. COPILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
40. COPILOT'S CYCLIC STICK
41. INSTRUMENT PANEL
42. MASTER CAUTION LIGHT
43. DELETED
44. FREE-AIR TEMPERATURE GAGE

SH-3H

Figure 3
COCKPIT DIAGRAM

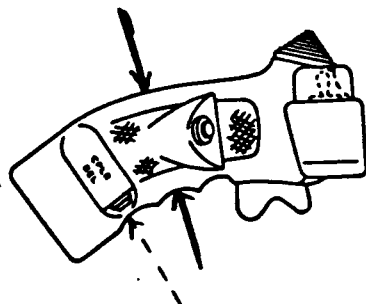
APPENDIX C



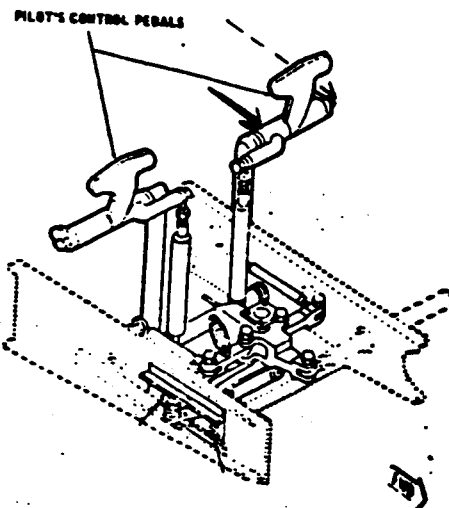
Cyclic Control Grip

FORCE MEASUREMENT ←

DISPLACEMENT MEASUREMENT ←---



Collective Control Grip



Pedals

Figure 4

FORCE/DISPLACEMENT MEASUREMENT POINTS

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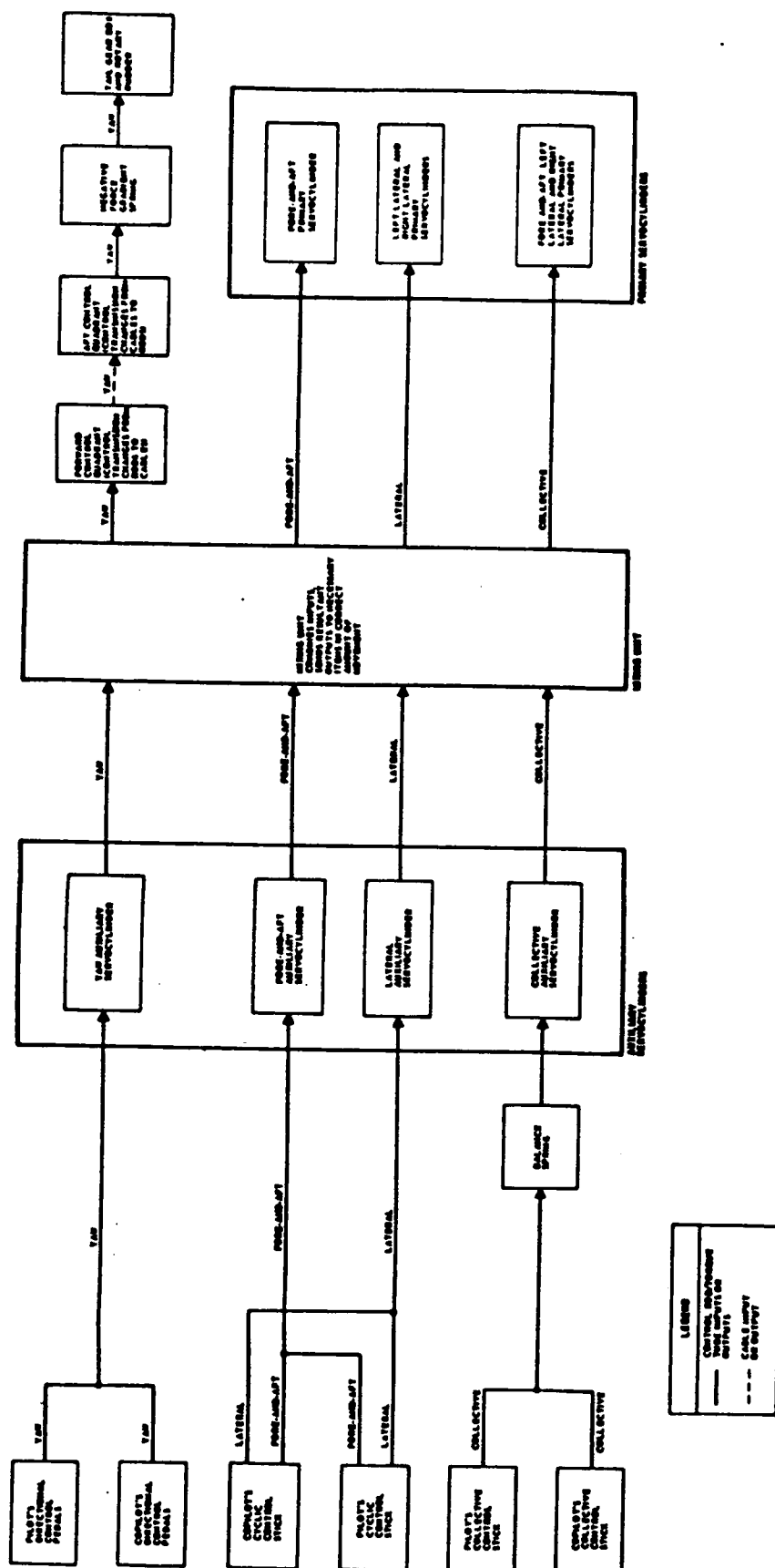


Figure 5
SIMPLIFIED FLIGHT CONTROL SCHEMATIC

FLIGHT CONTROL DESCRIPTION

INTRODUCTION

The flight control system of the SH-3H helicopter is a mechanically actuated hydraulically powered irreversible system designed to facilitate six degrees of freedom control of the helicopter in and about the longitudinal, lateral, and vertical axes. The flight controls incorporate three major subsystems, the main rotor control system, the tail rotor control system, and the flight control hydraulic power supply system. The aircraft is equipped with a foldable five-bladed fully articulated main rotor and a five-bladed tail rotor. Control augmentation is provided by the Automatic Stabilization Equipment (ASE) which is designed to enhance helicopter handling characteristics inflight and operate in conjunction with the coupler system to provide "hands off" approach and hover capability during ASW mission tasks. A simplified block diagram of the flight control system is presented in appendix C, figure 5.

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COCKPIT CONTROLS

The SH-3H is equipped with dual conventional helicopter flight controls. Each pilot station is equipped with a cyclic control stick, collective lever, and a pair of directional pedals. Each control is mechanically interconnected, below the cockpit deck, with the corresponding control in the opposite seat allowing them to move in unison. The cockpit controls are designed to allow the pilot to control the aircraft along six degrees of freedom.

The pilot controls the helicopter about the longitudinal and lateral axes with the cyclic control stick. The top of the cyclic is equipped with a handgrip which accommodates the various switches, as depicted in figure 1.

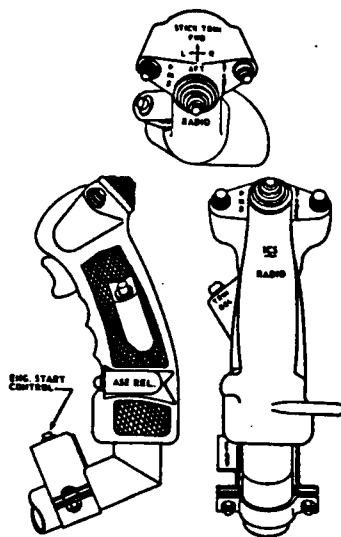


Figure 1

CYCLIC CONTROL GRIP

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As the cyclic pivots about its floor mount, the movement of the handgrip is designed to correspond to the desired direction of aircraft motion. The cyclic stick is held in a desired position by the cyclic trim system which is presented later in this section. The cyclic is designed to move automatically during coupled functions of the ASE.

Aircraft motion along the vertical axis and engine power output is controlled with the collective control lever. The collective is mounted to the cockpit deck left of each pilot seat and is inclined forward. The collective grips attached to the forward end of each lever contain a number of switches and buttons as depicted in figure 2.

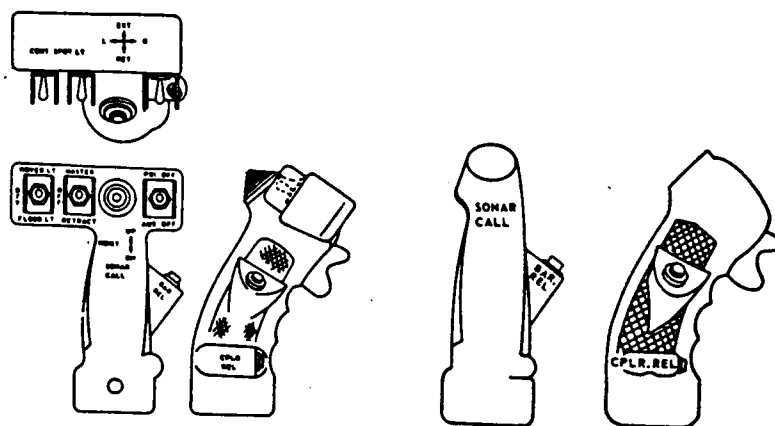


Figure 2

PILOT'S AND CO-PILOT'S COLLECTIVE GRIPS

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Movement of the collective handgrip is essentially in the vertical direction and is designed to correspond to desired aircraft motion during hover. During forward flight, the upward and downward motion of the collective lever, when coordinated with fore/aft movement of the cyclic, is designed to control altitude and airspeed. The collective levers are supported by a "balance spring" below the cockpit deck and an adjustable friction device is installed on the pilot's collective to provide positive force feel to both levers. The collective is designed to move automatically in response to altitude hold function of the ASE.

The directional pedals, mounted on the cockpit deck forward of each pilot seat, are designed to control aircraft rotation about the vertical axis. Forward movement of either pedal should rotate the nose of the aircraft in the corresponding direction. Each set of pedals are mechanically inter-connected so that they will move in opposition to each other. Each directional pedal incorporates a micro switch which is actuated by pressure from the pilot's foot and is designed to disengage the ASE heading hold feature. Each set of pedals incorporates a mechanical pedal position adjusting mechanism whilst only the pilot's pedals are equipped with toe brakes for the wheel brake system.

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LOWER CONTROLS

Flight control system inputs are routed from beneath the cockpit deck under the pilot's seat, aft, then upward behind the pilot station to the auxiliary hydraulic servocylinder module by a series of pushrods and bellcranks. The auxiliary servo cylinder module is an electrohydromechanical system designed to boost cockpit flight control inputs, provide control position feedback to the ASE, convert electrical correction signals from the ASE into mechanical inputs to the flight controls, and provide cyclic beeper trim capability. The yaw servo cylinder also incorporates a hydraulic pedal damper designed to prevent rapid pedal movement which could damage the airframe.

The auxiliary servo cylinders are hydraulically powered by the flight control auxiliary hydraulic system. 1500 psi hydraulic pressure is supplied to the four individual servo cylinders which boost the longitudinal lateral, directional and collective axes. Auxiliary hydraulic pressure is also reduced to 60 psi to power the longitudinal and lateral cyclic trim valves. An electrically closed solenoid valve is incorporated to allow the pilot to secure hydraulic power to the auxiliary servo module in the event of a servo malfunction or loss of system pressure. The solenoid is operated by the three-position flight control servo switch mounted on the pilot's collective handgrip as shown in

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figure 2. Control movements are still possible in the event of loss of auxiliary hydraulic system pressure, albeit at higher cockpit control forces. With the flight control hydraulic control switch in the AUX position, a pressure switch automatically opens the auxiliary hydraulic system solenoid valve, should pressure in the primary hydraulic system drop below 1000 psi. Loss of hydraulic power to the auxiliary servo module would result in loss of ASE correction inputs, cyclic beeper trim, and pedal damping. The directional control axis would become a purely mechanical unboosted system. The auxiliary servo module is depicted in figure 3.

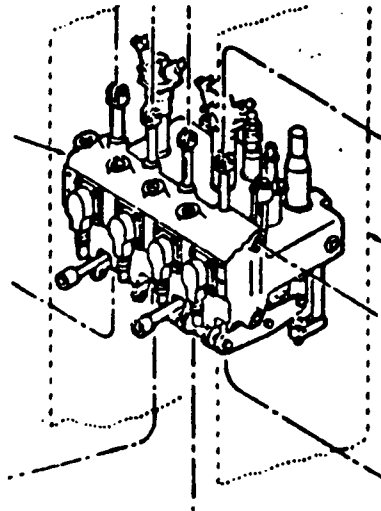


Figure 3

AUXILIARY SERVOS

INTERMEDIATE CONTROL/MIXING

Control output from the auxiliary servo module is

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transmitted via pushrods to the mechanical mixing unit. The purpose of mechanical control mixing is to reduce pilot workload by automatically compensating for off axis moments and forces resulting from individual control inputs.

The SH-3H flight controls incorporate two types of control mixing:

a. Collective to Pitch: Longitudinal tilt of the tip path plane is adjusted proportional to collective inputs, forward tilt for up collective and aft tilt for down collective. This coupling is employed to reduce pitch attitude variations due to collective movement especially during ASE transitions (coupled approaches and departures).

b. Collective to Yaw: Tail rotor blade pitch angles are adjusted proportionately to collective lever position to automatically compensate for changes in main rotor torque. Increasing the collective increases pitch on the main rotor blades requiring increased tail rotor thrust to offset increased main rotor torque. Decreasing collective reduces tail rotor blade pitch in the same manner. The process is irreversible while the auxiliary servos are pressurized, hence pedal movement will not affect collective lever position. The mechanism is designed to prevent tail rotor blade angle limits from being exceeded by the additive effects of both collective and pedal position. With full left pedal applied, raising the collective to full up will

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mechanically reduce left pedal displacement while tail rotor blade angle remains at maximum pitch. When full right pedal is applied, lowering the collective to full down will reduce right pedal displacement while tail rotor blade angle remains at minimum pitch. The mechanical mixing unit is depicted in figure 4.

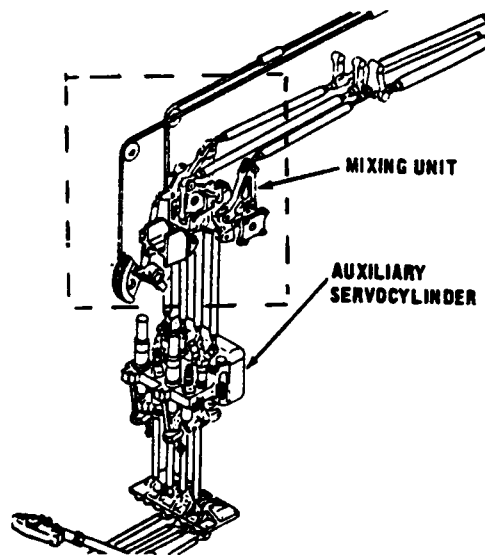


Figure 4

MECHANICAL MIXING UNIT

Mixing unit output is divided into two categories: Main rotor controls and tail rotor controls. Main rotor controls are routed aft via pushrods and bellcranks to the three primary servo cylinders. Tail rotor control output is routed via a single pushrod to the forward tail rotor control quadrant which is mounted below the mixing unit. The forward quadrant transmits tail rotor control inputs via dual

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directional flight control cables to the aft tail rotor control quadrant located in the aft fuselage forward of the pylon fold hinge line. Tail rotor inputs are transmitted from the aft quadrant through the pylon fold hinge line to the pitch change shaft on the tail rotor gearbox via a series of pushrods and bellcranks. The pitch change shaft is installed through the tail rotor gearbox and extends or retracts in response to directional control inputs. A five-pointed star assembly, mounted to the output end of the pitch change shaft, is connected to each of the five tail rotor blades via pitch change links. Movement of the pitch change shaft collectively changes pitch on each of the five tail blades, thereby changing the magnitude of the tail rotor thrust vector. A negative force gradient spring is installed in parallel between the aft most directional control bellcrank and the airframe. The spring is designed to absorb feedback loads exerted by the tail rotor system during flight and would be felt in the cockpit controls when the auxiliary hydraulic system is secured.

MAIN ROTOR UPPER CONTROLS

The purpose of the upper controls is to transmit cockpit flight control movements which have been boosted and mixed into the rotating components of the main rotor system. Inputs to the main rotor are applied through the three primary hydraulic servocylinders mounted around the main

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gearbox. The output shafts of the primary servos are attached to the stationary portion of the swashplate assembly. The swashplate assembly is designed to transmit control inputs from the stationary components mounted on the airframe to the rotating components on the rotor head. The swashplate assembly is mounted concentrically to the main gearbox output shaft and utilizes matched ball bearings to separate the stationary swashplate from the rotating swashplate. Mounted on the rotating swashplate are the five main rotor blade pitch change rods which connect the main rotor blade pitch change rods which connect the rotating swashplate with the pitch change horns on the main rotor sleeve and spindle assemblies. The swashplate is capable of sliding vertically along the main rotor shaft and tilting with respect to the horizontal plane. The heaving and tilting of the swashplate results in vertical motion of the main rotor blade pitch change links which rotate the sleeve and spindles and changes pitch on the main rotor blades. Heaving motion of the swashplate is generally the result of collective inputs which simultaneously increases or decreases pitch on all five main rotor blades, thereby changing the magnitude of the main rotor thrust vector. Tilting of the swashplate, generally the result of cyclic control inputs, changes the pitch on the main rotor blades as they pass through an azimuth angle causing the rotor disk to tilt,

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thereby changing the direction of the main rotor thrust vector. Movement of the swashplate assembly is accomplished by the fore-aft, right lateral and left lateral primary hydraulic servocylinders mounted around the main transmission as shown in figure 5.

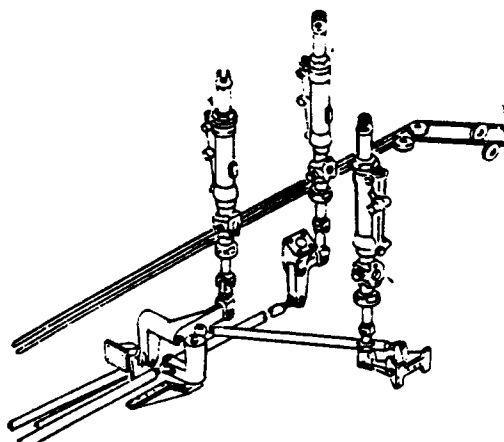


Figure 5

PRIMARY HYDRAULIC SERVOS

The primary servocylinders are powered by 1500 psi hydraulic pressure from the primary flight control hydraulic power supply system which is pressurized by a pump driven off the accessory section of the main transmission. The three-position electrically operated solenoid switch mounted on the pilot's collective handgrip is provided to secure pressurization to the primary servos in the event of a hydraulic leak downstream of the solenoid valve or primary servo malfunction. As with the auxiliary system, a pressure sensing switch is incorporated to ensure that the primary

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system receives hydraulic pressure should the auxiliary system pressure drop below 1000 psi, regardless of the position of the collective solenoid switch. The main rotor is not controllable if pressure to both hydraulic systems is lost.

ROTOR SYSTEMS

The aircraft is configured with a foldable, five-bladed fully articulated 62 ft diameter main rotor system and a five-bladed 10.6 ft diameter tail rotor. The main rotor is designed to convert cockpit control movement into changes in the direction and magnitude of the main rotor thrust vector for vertical, lateral and longitudinal control of the helicopter. The tail rotor is designed to convert control inputs into changes in the magnitude of the tail rotor thrust vector for countering main rotor torque and for directional control of the helicopter.

Major components of the main rotor system include the main rotor hub, sleeve and spindle assemblies (5), main rotor blades (5), and a bifilar vibration absorber. The hub assembly is mounted atop the main gearbox output shaft and consists of two star shaped plates which are fitted to splines on the main gearbox output shaft. Mounted on each arm of the star are the vertical and horizontal hinge assemblies which permit simultaneous flapping and hunting

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motion of the main rotor blades. Anti-flapping restrainers and droop restrainers are incorporated to limit the flapping angles of the blades during rotor engagement and disengagement. The droop stops are designed to support the blades at the horizontal hinge when the main rotor is stopped. Hydraulic damper-positioners, mounted to the hub, are designed to control rotor blade hunting (lead-lag) rates at operating RPM and during rotor engagement and disengagement. During blade fold the damper-positioners will automatically position the blades against their forward stops allowing pitch control locks to engage the sleeve and spindle assemblies, thereby supporting the blades during fold and spread operations. The rotor hub contains five reservoirs designed to provide lubrication for the horizontal and vertical hinges and for the sleeve and spindle assemblies. Mounted atop the main rotor hub is a fairing designed to serve as an airfoil to decrease drag and protect the hinge pins and reservoirs from the environment. Mounted to each horizontal hinge pin is a sleeve and spindle assembly. The sleeve and spindles allow spanwise rotation of the main rotor blades for blade pitch control. Each sleeve and spindle incorporates a pitch control horn which is connected to the rotating swashplate by an adjustable pitch control rod. Mounted to the outboard end of each sleeve and spindle is the blade fold hinge and associated lines and valves for the

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blade fold system. Attached to the outboard ends of the sleeve and spindles are the main rotor blades. Each main rotor blade is a symmetrical airfoil of all metal construction incorporating a hollow aluminum extruded spar and components, as shown in figure 6.

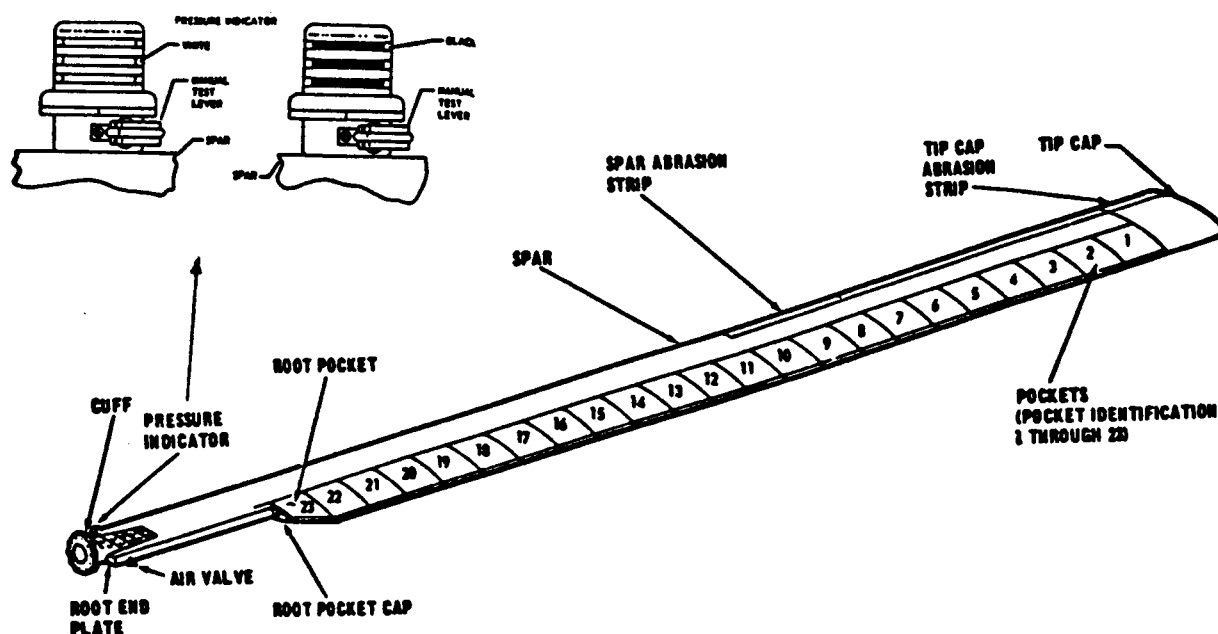


Figure 6

MAIN ROTOR BLADE

The aluminum spar is pressurized with dry nitrogen which pressurizes an indicator mounted to the root adjacent to the blade cuff. The pressure indicator shown in figure 6 will indicate a loss of spar pressure, possibly due to a crack in the spar, by showing three black rings which may be viewed during preflight and postflight only.

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The bifilar vibration absorber mounted atop the main rotor hub is designed to reduce fuselage vibration and rotor induced airframe structural loads. It is a star shaped aluminum forging with pivoting 17 lb. steel weights mounted at the end of each arm. The vibration absorber is shown in figure 7.

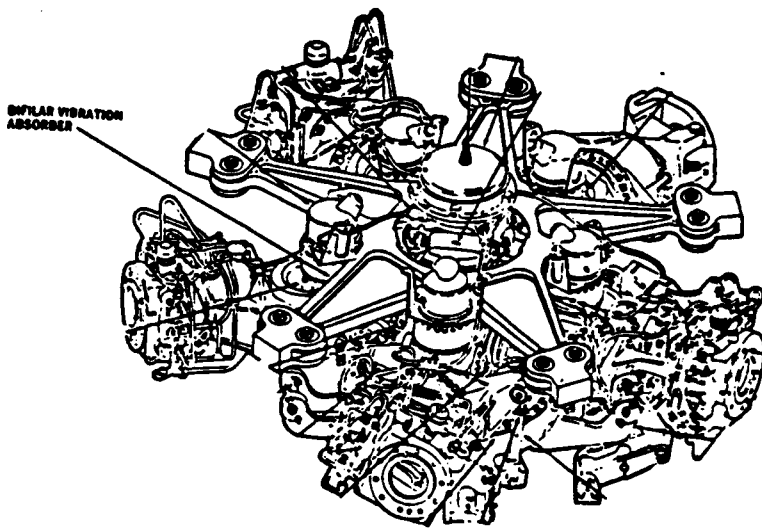


Figure 7

BIFILAR VIBRATION ABSORBER

The tail rotor assembly is mounted to the 90 deg tail rotor gearbox atop the vertical tail pylon. The tail rotor hub incorporates five flapping hinges for each of the tail rotor blades and five sleeve and spindle assemblies. The flapping hinges permit tail rotor flapping angles of up to 20 deg. The sleeve and spindles allow the tail rotor blades to

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rotate about their spanwise axis for changes in tail rotor blade pitch. The sleeve and spindles are connected to the star shaped tail rotor pitch beam by means of adjustable pitch change links. As the pitch beam, mounted on the tail rotor pitch change shaft, is extended or retracted, the pitch change links simultaneously decrease or increase the pitch on all five tail rotor blades, thereby decreasing or increasing tail rotor thrust. Five tail rotor blades are mounted to the sleeve and spindle assemblies of the tail rotor hub. Each blade is a symmetrical airfoil of all metal construction utilizing an extruded aluminum spar bonded to a single aluminum ribbed pocket. The associated components of the tail rotor blade are presented in figure 8.

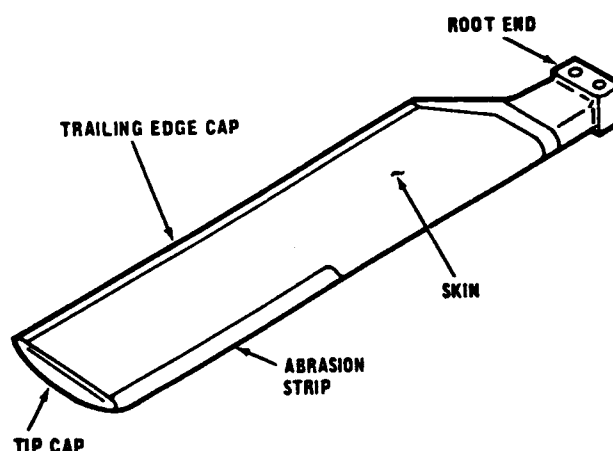


Figure 8
TAIL ROTOR BLADE

CYCLIC TRIM

The cyclic trim system is an electrohydraulic

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system designed to provide cyclic stick control force feel and permit a fine degree of control of the cyclic stick position. When used in conjunction with the ASE, the system is designed to permit a "hands-off" transition from forward flight to a hover. The trim system incorporates a master BEEPER TRIM electrical power switch located on the overhead console, cyclic mounted TRIM RELEASE buttons, and four-way STICK TRIM beeper switches, shown in figure 1, and longitudinal and lateral hydraulic trim actuators. Each hydraulic trim actuator is electrically controlled and incorporates a force gradient spring and a dual input trim valve. The actuators are mounted to the auxiliary hydraulic servo and are pressurized by 60 psi hydraulic pressure from the auxiliary hydraulic system. With the BEEPER TRIM switch ON, the trim valves are electrically closed and trapped hydraulic pressure will hold the cyclic in a constant position. The stick may be manually displaced without releasing the trim by applying 1 - 1-1/2 lb. of force in the desired direction. Resistance force is supplied by the force gradient spring which is designed to return the stick back to its trimmed position. The cyclic may be moved without resistance and repositioned by depressing the TRIM RELEASE button which opens both trim valves, placing the trim actuators in bypass. Releasing the TRIM RELEASE switch electrically closes the trim valves, recentering the trim

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system at its new position. Engaging the thumb-operated four-way STICK TRIM switch in the L, R, FWD or AFT positions electrically opens the appropriate trim valve, pressurizing one of the input ports causing the actuator piston to move, hence driving the stick in the corresponding direction. Releasing the stick trim switch closes the trim valve stopping the stick in its new position. Without auxiliary hydraulic pressure, the trim system utilizes trapped hydraulic fluid to hold stick position. The TRIM RELEASE function remains operable but the STICK TRIM function is cancelled. Without electrical power, the trim valves are opened and stick position must be manually maintained by the pilot. During ASE coupler operations the trim valves receive electrical signals from the ASE to drive the actuator pistons and hence reposition the stick to ASE modulated positions.

AUTOMATIC STABILIZATION EQUIPMENT

The Automatic Stabilization Equipment (ASE) is designed to improve helicopter handling qualities in all flight phases, to work in conjunction with the coupler system to provide a "hands-off" transition from forward flight to a hover, and provide "hands-off" hover capability in VMC and IMC conditions. The ASE provides two modes of operation: Basic ASE and coupler operation. During basic ASE operation the system is designed to stabilize the helicopter at the reference attitudes (pitch, roll, and yaw) selected by the

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pilot. Altitude hold capability is available by engaging the barometer altitude hold function (BAR ALT HOLD). Stability corrections from the ASE are introduced to the flight controls through the auxiliary servocylinders. Basic ASE inputs in pitch and roll are not fed back to cyclic stick, however, corrections in the yaw and collective (altitude hold engaged) channels will move the respective controls. The ASE gain and authority are limited to allow the pilot to override ASE corrective inputs at all times, both in the basic ASE and coupler modes. During coupler operations, the ASE references the longitudinal and lateral groundspeeds provided by the doppler system and radar altitude selected by the pilot through potentiometers on the ASE control panel. The system is also designed to allow the sonar operator or the hoist operator limited authority to maneuver the helicopter horizontally using controls on the cable angle control panel or the hover trim control panel, which are located in the cabin section of the helicopter.

Inflight, functions of the ASE are controlled primarily through the ASE control panel located on the cockpit center console and presented in figure 9.

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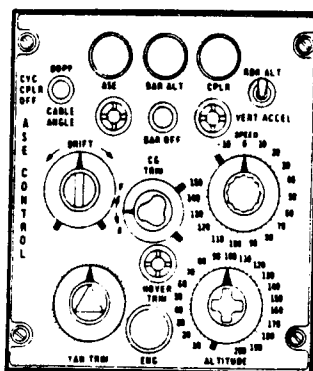


Figure 9

ASE CONTROL PANEL

A summary of individual switch and potentiometer functions is presented in appendix F, table IV. A cyclic mounted pushbutton switch is designed to completely disengage the ASE when depressed and released. Collective mounted switches include the coupler release switch, which disengages the coupler when pressed and released and the BAR ALT release switch which disengages the BAR ALT HOLD function only while it is depressed. The Channel Monitor Panel, located on the pilot's right console, allows the pilot to monitor auxiliary servo valve input or coupler output on the hover indicator (A mode) and to select between the port and starboard vertical gyros for input to the ASE.

To stabilize the aircraft during basic ASE and coupler operations, signals representing actual attitude, altitude and heading, rates and acceleration are continuously fed into the ASE amplifier from on-board vertical gyros, rate gyros,

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accelerometers, the radar altimeter, and doppler system. These signals are compared with corresponding signals which represent reference attitude, altitude, and heading. The amplifier processes all sensor, coupler, and control panel input signals to produce stability correction output signals to the appropriate channel of the auxiliary servos for input to the flight controls. ASE output signals are also provided to the hover indicators located on the pilot's and copilot's instrument panel. The hover indicators are designed to display information regarding four separate AFCS functions. The A mode displays ASE input to the auxiliary servos or coupler output depending on the position of the METER SELECT SWITCH. The C mode displays cable angle information during a coupled hover when the sonar is submerged and the D mode displays longitudinal, lateral and vertical doppler velocities. The hover indicator is presented in figure 10.

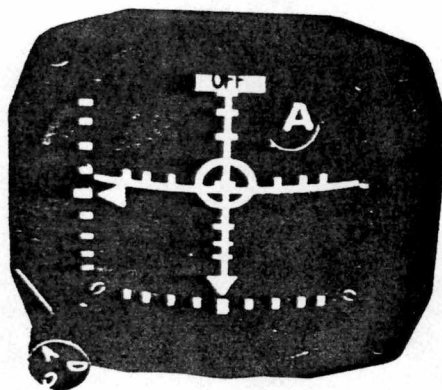


Figure 10

HOVER INDICATOR

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A summary of information displayed by the hover indicator is presented in appendix F, table V.

PITCH/ROLL CHANNELS

Basic ASE pitch and roll stabilization is accomplished by comparing vertical gyro position with the reference level position. A proportional error signal is produced, a portion of which is differentiated to produce a derived rate signal. The rate signal and proportional signal are summed and compared with output from the stick position sensors and ASE Control Panel C.G. TRIM CONTROL (pitch only). The resulting error signals are fed into the ASE amplifier which produces a correction signal which is fed to the pitch or roll servo valves of the auxiliary servo which moves the control surfaces. As the aircraft attitude returns to the reference attitude, the error signal diminishes and the vertical gyro output signal approaches reference "level" value.

When the coupler is engaged, pitch and roll error signals are compared using either doppler or cable angle reference signals. In the doppler mode, doppler output signals are compared with the SPEED and DRIFT signals selected on the ASE control panel. The resulting error signals along with an acceleration signal from the longitudinal or lateral acceleration signal from the longitudinal or lateral accelerometer are fed through a

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fade-in circuit which prevents large attitude changes in response to large velocity errors. An amplifier limiter limits the magnitude of the error signal being fed from the coupler to the ASE amplifier to prevent the ASE from saturating in that (pitch or roll) channel. The cyclic stick is designed to automatically reposition in response to coupler output to the cyclic trim valves to maintain ASE pitch or roll authority. The ASE amplifier processes coupler output signals similar to the basic ASE error signal until the error signal is diminished. The coupler mode is designed to give the helicopter a gradual decel from 60 ± 20 kt. ground speed to the hover speed set on the ASE control panel during an automatic coupled approach.

In the cable angle mode, the coupler is designed to maintain the submerged sonar transducer cable centered in the sonar well during a coupled hover. Off center cable position signals are compared with vertical gyro output. The resulting error signal represents the displacement of cable angle from true vertical. The signal is processed similar to a doppler error signal. During cable angle hover, an Inertial Velocity System (IVS) circuit combines outputs from the doppler, accelerometers and vertical gyro and compares the signal with the SPEED and DRIFT settings on the ASE control panel. The IVS circuit then produces an error signal which is combined with the proportional cable angle error

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signal to bias ASE corrections in order to maintain desired drift rates during a hover in the cable angle mode.

COLLECTIVE CHANNEL

The collective channel of the ASE is activated during basic ASE operations by engaging the BAR ALT HOLD button on the ASE control panel. The barometric altitude controller compares the current pressure altitude from the purifier chamber with the reference "engaged" altitude signal and issues a proportional error signal which is summed to a collective position sensor signal and fed to the ASE amplifier. The ASE amplifier produces a correction signal which is fed to the collective auxiliary servo for input to the flight controls allowing the aircraft to return to the reference altitude. The output portion of the collective auxiliary servo piston is connected to the collective open loop spring which helps support the weight of the collective at any position. As the auxiliary servo piston moves, the collective lever moves to maintain a position referenced to the servo piston displacement. As the altitude error diminishes, the position sensor proportional signal returns the collective to its original position. The pilot may override the ASE by displacing the collective against the force gradient of the open loop spring. By depressing the BAR REL button on the collective, the BAR ALT hold is disengaged and the proportional error signal is nulled.

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Releasing the BAR REL button re-references the altitude controller and the collective stick position sensor. When the coupler is engaged, BAR ALT HOLD is automatically engaged and the ASE is designed to return the aircraft to the engaged altitude when the coupler is disengaged. During coupler operation, the ASE control panel ALTITUDE signal is compared with the radar altitude signal. Any resultant error signal is processed through the coupler and compared with the Vz (vertical velocity) signal from the doppler system. The resultant signal is routed to the ASE amplifier which issues correction signals to the servo. The coupler output signal to the ASE is shaped to prevent the ASE from attempting to match radar altitude with the desired altitude too quickly. A portion of the radar altitude error signal is also summed with the doppler Vh (longitudinal velocity) signal. The resultant signal is called the pitch to collective cross-feed signal which allows the coupler to provide a programmed rate of descent vs. forward velocity. In the VERT ACCEL mode, the coupler bypasses the Vz signal and references signals from the vertical accelerometer which are compared to the BAR ALT proportional error signal and fed to the coupler. In the VERT ACCEL mode, vertical accelerations are dampened by the ASE to reduce collective pumping during coupler operations.

YAW CHANNEL

The ASE yaw channel consists of a heading hold function which is designed to maintain heading during all flight

phases when the ASE is engaged and the pilot's feet are off the pedal micro-switches. The system consists of a differential synchro, a yaw synchronizer, and a rate gyro. During level flight the differential synchro located in the ASE control panel receives inputs from the compass system and the position signal from the YAW TRIM control on the ASE control panel which is referenced to the original heading at the 12 o'clock position (see figure 9). The differential synchro develops an error signal proportional to the difference between compass heading and reference heading. The error signal is summed to a modulated signal from the yaw rate gyro and the resultant rate and proportional signal is fed into the ASE amplifier. The amplifier produces correction signals which are fed to the auxiliary yaw servo and input to the flight controls. Connected to the output piston of the yaw servo is the yaw open loop spring which will reposition the pedals and maintain reference to the position of the yaw auxiliary servo piston. The pilot has full overriding authority over ASE corrections in the yaw channel. During low airspeed and hover operations, the YAW TRIM control is designed to allow the pilot to adjust helicopter heading throughout 360 deg. left or right by biasing the differential synchro output signal to the ASE amplifier. During turns with the pilot's feet depressing the pedal switches, the differential synchro error signal is

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nulled to zero, however, the ASE amplifier continues to receive a yaw rate signal which produces a correction signal and drives the yaw auxiliary servo to oppose the yaw rate. The ASE correction is manifested to the pilot as an increased resistance to pedal movement which is proportional to the yaw rate as the pedals attempt to drive opposite the turn. The yaw channel operates independently of the coupler and is engaged at all times when the ASE is engaged.

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GLOSSARY

1. Breakout Force - The force necessary to cause initial motion of the flight control.
2. Centering - The tendency of a flight control to return to a trimmed position when displaced.
3. Controls Fixed - Flight controls are rigidly fixed in position during the aircraft response to a control input.
4. Controls Free - Flight controls are allowed to float during the aircraft response to a control input.
5. Control Margin - Relative distance between a control position and the control stop.
6. Critical Azimuth - Relative wind azimuth of the helicopter during hovering flight which represents the highest pilot workload, vibration levels, noise level, minimum control margins or uncommanded attitude changes.
7. Deadbeat Response - Dynamic aircraft response which quickly returns to or maintains a non-oscillatory steady state when subject to an sinusoidal excitation.
8. Design Eye Position - Used in cockpit measurement; the point in space located at the sitting eye height dimension of the 50th percentile aviator (31.5 inches) measured vertically from the intersection of the seatback tangent line and the seat surface when the seat is set to one half its vertical travel and measured horizontally, thirteen inches from the seat back tangent line.

APPENDIX E

9. Effective Offset - Relative distance between the main rotor shaft and the blade flapping hinge on an articulated rotor system.
10. Effective Translational Lift (ETL) - Velocity of the helicopter associated with a large reduction in power required characterized as lift produced by the forward motion of the rotor disk.
11. Free Hover Method - Hover performance determination method in which the helicopter maintains a constant altitude while varying rotor speed and power setting.
12. Freeplay - Lost motion between the flight control and the control surface.
13. Fully Articulated Rotor - Each rotor blade is hinged to allow movement vertically, horizontally as well as about the feathering axis.
14. Ground Effect - Reduction in power required when the rotor system is within a 1.5 rotor disk diameter of the surface.
15. Lateral Directional Oscillation - Helicopter dynamic mode which results in aerodynamic coupling between the roll and yaw axes of the aircraft.
16. Long Period Response - The response of the helicopter when displaced from trim airspeed in forward flight.
17. Magnetic Anomaly Detection (MAD) - Equipment which measures deviations in the earth's magnetic field caused by

APPENDIX E

- the presence of a large metallic body such as a submarine.
18. Percentile - Used in anthropometry as a relative measurement of an individual's body dimension as compared to the population.
 19. Short Period Response - Instantaneous response of the helicopter when displaced from a trim attitude.
 20. Spiral Stability - Characteristic of aircraft roll excursions from a trim bank angle.
 21. Static Droop - Relative deviation of rotor speed at various stable power settings.
 22. Stick Dynamics - Oscillatory response of a flight control when displaced from trim and released.
 23. Stick Jump - Uncommanded movement of the flight control when displaced against trim and then the trim is disengaged.
 24. Transient Droop - Relative deviation of rotor speed during power changes.
 25. Trim Speed Band - The airspeed range at which the aircraft will maintain at a trimmed control position.
 26. Vcruise - Cruise airspeed determined during level flight performance testing as the airspeed which corresponds to ninety-nine percent of the maximum specific range airspeed.
 27. Vne - Maximum forward flight airspeed (never exceeded).
 28. W/o Frontside/Backside Technique - Level performance test method whereby the helicopter maintains a constant ratio of gross weight over relative air density by varying density altitude as the gross weight is reduced by fuel consumption.

Table I

TESTS AND TEST CONDITIONS
BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
All	Cockpit Eval.	On deck and inflight	0-120	--	--	--	--	1. Field of View evaluated at DEP 2. Controls and displays 3. Normal/emergency procedures
1	Mechanical Character.	On deck rotor stopped	--	23	--	--	--	1. ASE-ON, BEEPER TRIM-ON External hydraulic, electrical power or #1 engine in accessory drive
1		On deck rotor turning 100% NR	--	23	-110	--	--	1. ASE-ON, BEEPER TRIM-ON 2. Both engines 100% NR 3. Relative winds 360 deg. 6 kt.
2		Inflight	100	20	1000	19,727	269.7	ASE-ON, BEEPER TRIM-ON

APPENDIX F

Table I (Continued)
TESTS AND TEST CONDITIONS
BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
All	Ground Handling	On deck rotors turning	--	--	--	--	--	Ground taxi Rotor engagement/ disengagement
HOVER/LOW AIRSPEED HANDLING QUALITIES								
1	Trimmed Control Positions	Forward rearward sideward flight	--	--	--	19,682	270.5	Pace truck method. 5 kt. increments; Relative wind 360,090,180 270 deg. Controls fixed/free
	Critical Azimuth		0-20	22	-120 (15 AGL)	19,257	268.6	10 kt increments Relative wind 045, 135,225,315 deg.
	Static Stability					18,649	268.9	Longitudinal, lateral
	Dynamic Stability	Hover	0 drift	21	400 (500 AGL)	18,377	269.0	Longitudinal, long term; Natural, artificial excitation

APPENDIX F

Table I (Continued)

TESTS AND TEST CONDITIONS
BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
HOVER/LOW AIRSPEED HANDLING QUALITIES (CONTINUED)								
1	Control Response	Hover	0 drift	21	400 (500 AGL)	17,867	269.9	Qualitative; Step inputs 0-2 in. at 1/4 in. increments (0-1.0 in. directional) longitudinal, lateral vertical, directional axes. Directional response at 15 ft. AGL
FORWARD FLIGHT FLYING QUALITIES								
2	Trimmed Flight Control Positions	Level flight turns Climbs and Descents	43-118 90	18	2000 (avg)	19,102	268.3	Turns: 0-45 deg. AOB left/right at 15 deg. increments Climb/Descend: Torque, 4-85% at 5 increments
	Static Stability: Longitud. Level flight		90±11	16	3000	18,777	267.8	Speed stability

APPENDIX F

Table I (Continued)
TESTS AND TEST CONDITIONS
BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
FORWARD FLIGHT FLYING QUALITIES (CONTINUED)								
2	Static Stability: Lateral-Direct.	Steady heading sideslips Turns on one control	90±11	16	3000	18,777	267.8	Directional stability Effective dihedral Sideforce characteristics; SHSS: ± 2 ball widths TOIC: Cyclic ± 20 deg. AOB; Pedal ± 2 ball widths
	Maneuver. Stability	Steady Turns Symmetrical Pullups/ Pushovers	90	15	3000	18,697	268.5	Turns: 30, 45 deg. AOB left/right Pullups/pushovers: 0-1.0 in. at 1/4 in. increments
	Dynamic Stability: Longitud.	Level flight	90	15	3000	18,597	268.9	1. Long Term: Controls free a. Natural & artificial excitation

Table I (Continued)
TESTS AND TEST CONDITIONS
BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
2	Dynamic Stability:							
	Longitud.	Level flight	90	15	3000	18,597	268.9	1. b. Light turbulence 2. Short Term: Controls fixed a. Pitch doublet input at 1 cycle/sec, 1 - 1-1/2 in. amplitude
	Lateral-Directional	Level Turns				18,477	269.0	1. Spiral stability at 20 deg. AOB, pedals fixed 2. Lateral-directional oscillation (LDO) a. Natural excitation b. Artificial excitation 1. Collective pulse, 1 - 1-1/2 in. 2. Pedal doublet 1/2 to 1 in. at 1/2 cycle/sec.

Table I (Continued)

TESTS AND TEST CONDITIONS

BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
2	Control Response	Level flight	90	15	3000	18,342	268.8	1. Longitudinal/lateral cyclic step inputs to 2.0 in. at 1/4 in. increments qualitative response evaluation
	Gust Response	Level flight	90	21	500 AGL	18,255	268.4	Natural excitation, light turbulence
PERFORMANCE Engine Assessment								
All	Engine Start/ Accel & Shutdown	On deck	0	17-23	-100 to 0	-	-	All starts IAW NATOPS 10 starts total including 1 battery start, one warm engine restart. Rel winds 330-180 deg, 3-11 kt.
4	Static Droop	Vertical climb	0	16	-100	19,377	268.8	7 torque increments 20 to 103%

Table I (Continued)

TESTS AND TEST CONDITIONS

BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
2	Transient Droop	Forward flight climbs	70, 90	16	3000	19,502	269.3	Qualitative
3	HOVER: In Ground Effect	Hover	0	25	25 AGL	18,672	270.6	Nr 96-106%
1	Out of Ground Effect			23	100 AGL	17,929	267.9	Nr 96-106%
2	Level Flight	Forward flight	43-118	18	2000	19,102	268.3	W/G 20,800
1	Vertical Climbs	Vertical climb	0	24	300 (avg)	17,582	269.9	4 torques 84-103%

Table I (Continued)

TESTS AND TEST CONDITIONS

BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
2	Forward flight climbs/descents	Forward flight	70,90	16	3000 (avg)	19,502	269.3	Qualitative maximum/minimum power
MISSION TASKS								
3,4	Hover	Hover	0	19	40 AGL	18,512	Mid	1. Simulated sonar dip manual, full coupler, altitude coupler 2. Simulated HIFR, mail transfer 3. Simulated SAR pickup 4. ASE/AUX-OFF
3,4	Approaches	Transition; forward flight to hover	100-0	19	200-0 AGL			1. Approaches to simulated sonar dip 2. Simulated ship approach 3. Single engine approach/landing

Table I (Continued)

TESTS AND TEST CONDITIONS

BUNO 149728

Flight #	Test	Condition	Airspeed (KIAS)	OAT (°C)	Pressure Altitude (ft)	Avg. Gross Weight (lb)	Avg. Long. C.G. (in)	Remarks
MISSION TASKS (CONTINUED)								
3,4	Approaches	Transition; forward flight to hover	100-0	19	200-0 AGL	18,512	Mid	4. Practice autorotation 5. Single engine waveoff 6. ASE/AUX-OFF approach/landing
3,4	Departures	Transition: hover to forward flight	0-100	19	0-500 AGL	18,512	Mid	1. Departure from 40 ft hover Automatic, manual 2. Shipboard takeoff (simulated) 3. Jump takeoff 4. Freestream recovery
2	Cruise	Forward flight	90,100	18	3000-500 AGL	18,678	268.4	1. Simulated IMC cruise Simulated GCA

NOTES: (1) ASE-ON, Nr 100%, Cyclic Beeper Trim-ON except where noted
(2) MK 46 Torpedo, Left-Front Station

Table II
MECHANICAL CHARACTERISTICS (1)

Parameter	Axis			
	Cyclic		Pedals Directional	Collective Vertical
	Longitudinal	Lateral		
Total Control Travel (in.)	13.6	14.6	6.4	7.5
Breakout Plus Friction Force (lb)	Forward/Aft 1.0 2.0	Left/Right 1.0 1.0	Left/Right 12.0 12.5	Up/Down 1.5 1.0
Average Force Gradient (lb/in.)	0.8 0.6	0.9 0.6	- -	- -
Total Freeplay (in.)	0.06	0.2	0.4	0.03
Trim Freeplay (in.)	0.06	0.06	0.1	0.06
Control Centering	Absolute	Absolute	Positive	- -
Stick Jump (in.)	0.1	0.1	- -	- -
Stick Dynamics (number of overshoots)	3 small	3	- -	- -

Table II (Continued)

MECHANICAL CHARACTERISTICS

Parameter	Axis			
	Cyclic		Pedals Directional	Collective Vertical
	Longitudinal	Lateral		
Beeper Trim Rate (in./sec)	Forward/Aft 0.2 0.2	Left/Right 0.3 0.3	- -	- -
Limit Force (lb)	6.0 6.5	6.0 5.5	- -	- -

NOTE: (1) ASE, Cyclic Trim-ON, Hydraulics: Both

Table III

TYPICAL MISSION PROFILE

Evolution	Power Rating	Duration (min)	Fuel Flow (lb/hr)	Specific Range (mi/lb)	Fuel Used (lb)
Start/takeoff	Idle	10	720	--	120
Climb to 150 ft	Max Continuous	1	1350	--	23
Transit to 60 nm at Vcruise	--	32	--	.0780	754
(15 min hover) x 6	Max Continuous	90	1350	--	2025
(10 nm transit at 100 KTAS) x 6	--	36	--	.0791	749
RTB 40 nm at Vcruise	--	21	--	.0780	503
Carrier controlled approach	90 KTAS	15	1160	--	29

Table III (Continued)

TYPICAL MISSION PROFILE

Evolution	Power Rating	Duration (min)	Fuel Flow (lb/hr)	Specific Range (mi/lb)	Fuel Used (lb)
Hover/land	Max Continuous	1	1350	--	25
Reserve	--	--	--	--	450

NOTE: Mission duration: 3.4 hr.
Total fuel required: 4,678 lb.

Table IV

T58-GE-10 ENGINE
STANDARD SEA LEVEL PERFORMANCE RATINGS
 (Ratings are at the Power-Turbine Output Shaft)

Rating	Min. SHP	Min. Torque (lb-in.)	Max. SFC (lb/shp/hr)	Max. Gas-Generator Speed (RPM)	Rated Power Turbine Speed (RPM)	Max. Power-Turbine Inlet Temp. (°F)	Max. Measured Power-Turbine Inlet Temp. (°C)
Military (30-min)	1400	4530	0.61	26,300	19,500	1285	(696)
Normal	1250	4043	0.62	-----	19,500	1220	(660)
90% Normal	1125	3638	0.64	-----	19,500	----	----
75% Normal	938	3032	0.68	-----	19,500	----	----
Ground Idle	----	37 lb-ft. (max.)	165 lb/hr.	-----	0	----	----

*Maximum allowable operating speed is 21,275 rpm at no-load conditions.

Table V

T58-GE-16 ENGINE
STANDARD SEA LEVEL PERFORMANCE RATINGS
(Ratings are at the Power-Turbine Output Shaft)

Rating	Min. SHP	Min. Torque* (lb-in.)	Max. SFC (lb/shp/hr)	Max. Gas-Generator Speed (RPM)	Rated Power Turbine Speed (RPM)	Max. Measured Power-Turbine Inlet Temp. (°F)	Max. Measured Power-Turbine Inlet Temp. (°C)
Military (30-min)	1870	5030	0.56	26,800	19,500	1481	(805)
Normal	1770	4760	0.56	-----	19,500	1445	(785)
90% Normal	1593	4280	0.57	-----	19,500	----	----
75% Normal	1328	3570	0.61	-----	19,500	----	----
Ground Idle	----	55 (Max)	180 lb/hr	15,400	0	----	----

*At rated power turbine speed.

Table VI
ASE CONTROL PANEL

Control	Function
ASE engage	Engages basic ASE pitch, roll, and yaw channels.
BAR ALT engage	Engages basic ASE collective channel.
BAR OFF	Disengages basic ASE collective channel.
CPLR engage	Engages basic ASE collective channel and coupler while basic ASE is engaged.
CYC CPLR	Selects coupler mode of operation.
DOPP	Selects doppler mode of pitch and roll coupler operation.
CABLE ANGLE	Selects cable angle mode of pitch and roll coupler operation.
ALT CPLR	Selects coupler mode of operation.
RAD ALT	Selects radar altitude mode of collective coupler operation.
CABLE ALT (On helicopters not modified by AFC 396)	Selects cable altitude mode of collective coupler operation.
VERT ACCEL	Selects vertical accelerometer mode of collective coupler operation.
HOVER TRIM ENG	Engages hover trim control panel during doppler mode.

APPENDIX F

Table VI (Continued)

ASE CONTROL PANEL

Control	Function
ALTITUDE	Preselects altitude to be maintained in collective coupler mode. Calibrated linearly up to 200 feet.
CG TRIM	Compensates for center of gravity changes in helicopter during ASE engagement. CG TRIM control is not calibrated.
DRIFT	Compensates for lateral drift with respect to ground during roll coupler mode. DRIFT control is not calibrated. Full scale sensitivity is low enough to allow small changes in drift to be made.
SPEED	Preselects desired ground speed to be maintained in pitch coupler mode. SPEED control is calibrated linearly from minus 20 to plus 150. Friction reduction allows for small speed changes to be made.
YAW TRIM	Makes heading changes without using rotary rudder pedals. YAW TRIM control is connected to compass system directional gyro ac output and is calibrated linearly. Each mark represents 1°, one complete YAW TRIM control revolution will cause 72° helicopter turn with ASE engaged. Friction reduction prevents abrupt heading changes.

APPENDIX F

Table VII
HOVER INDICATORS

	MODE A			MODE D	MODE C	
	Meter Selector - ASE ASE - Engaged	Meter Selector - CPLR CPLR - Engaged	Meter Selector - CPLR CPLR - Off		Meter Selector - CPLR CPLR - Engaged	Meter Selector - ASE CPLR - Engaged
Horizontal Bar	Pitch Servo Valve Differential Current (1 Division = 2 MA)	Pitch Coupler Output (1 Division = 2 MA Servo Valve Current)	Pitch Coupler Output Less Integral	Doppler V _h (1 Division = 10 Knots)	Pitch Cable Angle Error (1 Division = 2.5°)	Not Used
Vertical Bar	Roll Servo Valve Differential Current (1 Division = 2 MA)	Roll Coupler Output (1 Division = 2 MA Servo Valve Current)	Roll Coupler Output Less Integral	Doppler V _h (1 Division = 10 Knots)	Roll Cable Angle Error (1 Division = 2.5°)	Not Used
Horizontal Pointer	Yaw Servo Valve Differential Current (1 Division = 2 MA)	Not Used	Not Used	Not Used	Not Used	Not Used
Vertical Pointer	Coll Servo Valve Differential Current (1 Division = 2 MA)	Coll Coupler Output (1 Division = 2 MA Servo Valve Current)	Coll Coupler Output Less Integral (Radar Mode Only)	Doppler V _z (1 Division = 250 Ft/Min)	Coll Coupler Output (1 Division = 2 MA Servo Valve Current)	*Cable Error (1 Division = 10 ft)
Flag	ASE - Engaged	ASE - Engaged	ASE - Engaged	Doppler Transmitting	Sonar Power On	Sonar Power On

APPENDIX F

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED

PEDALS: FULL LEFT TO FULL RIGHT

EXTERNAL ELECTRICAL POWER: ON

COLLECTIVE: 0-100%

EXTERNAL HYDRAULIC POWER: ON

ASE: ON

CYCLIC TRIM: ON

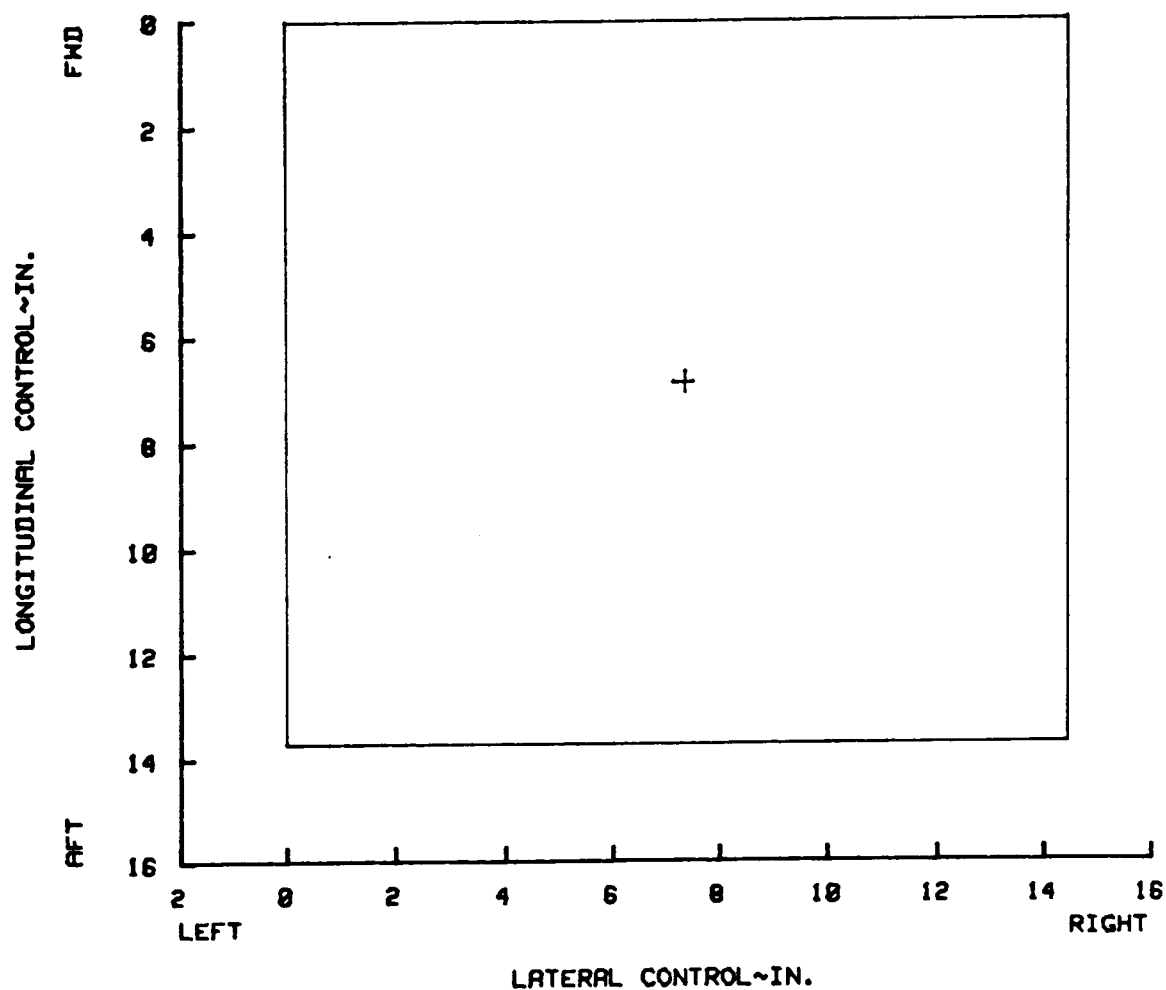


Figure 1

CYCLIC CONTROL ENVELOPE

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED

LONGITUDINAL/LATERAL CONTROL:

EXTERNAL ELECTRICAL POWER: ON

50%

EXTERNAL HYDRAULIC POWER: ON

ASE: ON

CYCLIC TRIM: ON

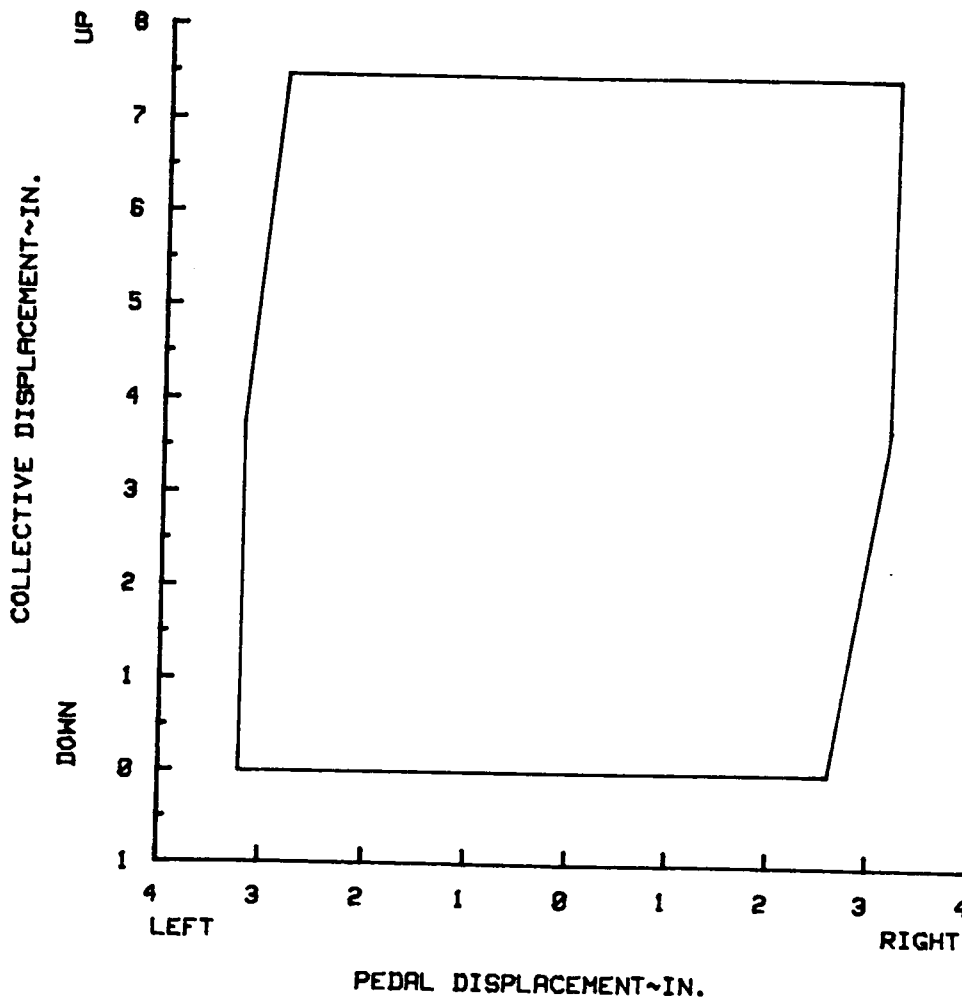


Figure 2

COLLECTIVE/PEDAL CONTROL ENVELOPE

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED
EXTERNAL ELECTRICAL POWER: ON
EXTERNAL HYDRAULIC POWER: ON
ASE: ON
CYCLIC TRIM: ON

COLLECTIVE: 50%
PEDALS: 50%
LATERAL CONTROL: 50%

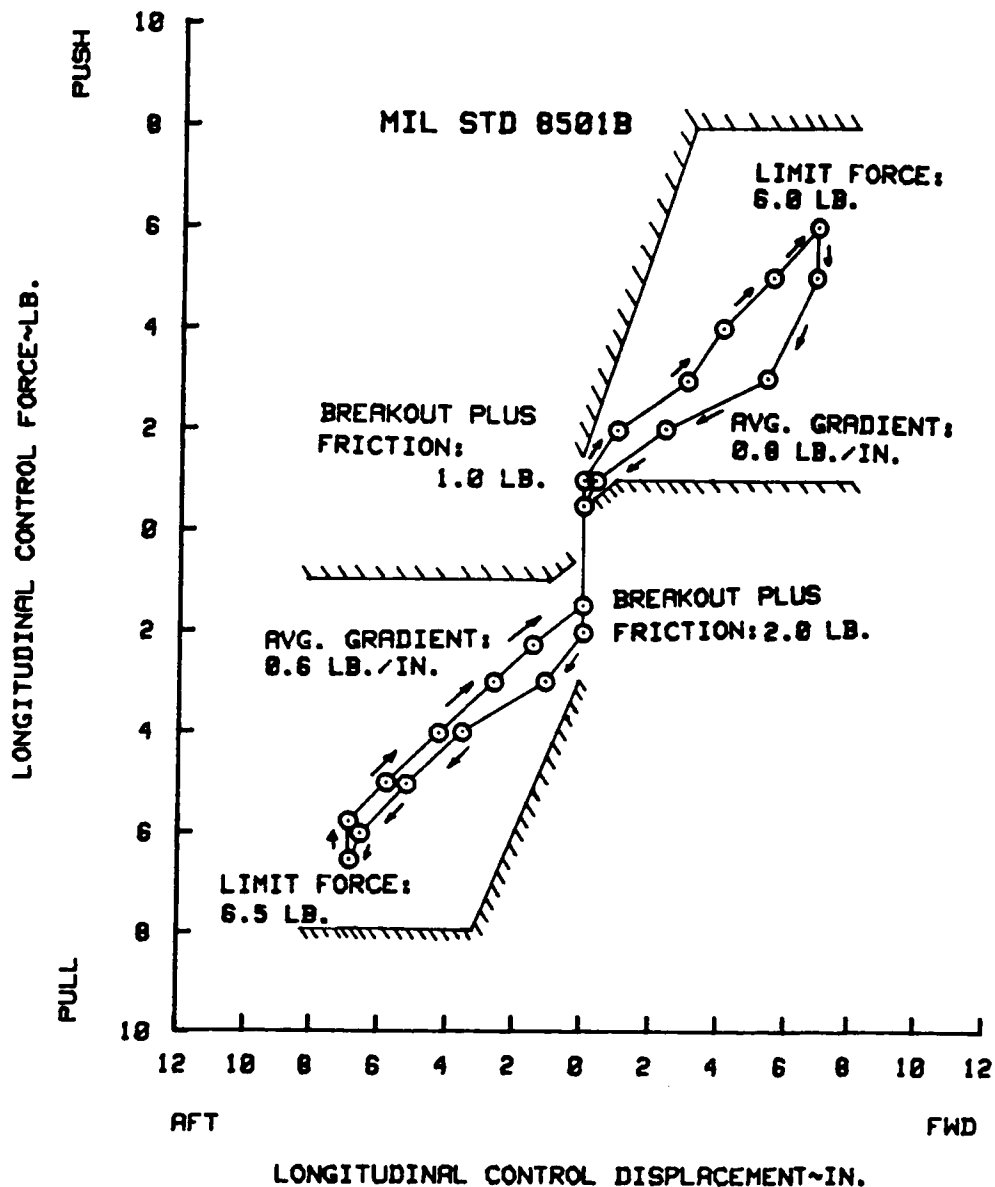


Figure 3
LONGITUDINAL CONTROL MECHANICAL CHARACTERISTICS

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED	COLLECTIVE: 50%
EXTERNAL ELECTRICAL POWER: ON	PEDALS: 50%
EXTERNAL HYDRAULIC POWER: ON	LONGITUDINAL CONTROL: 50%
ASE: ON	
CYCLIC TRIM: ON	

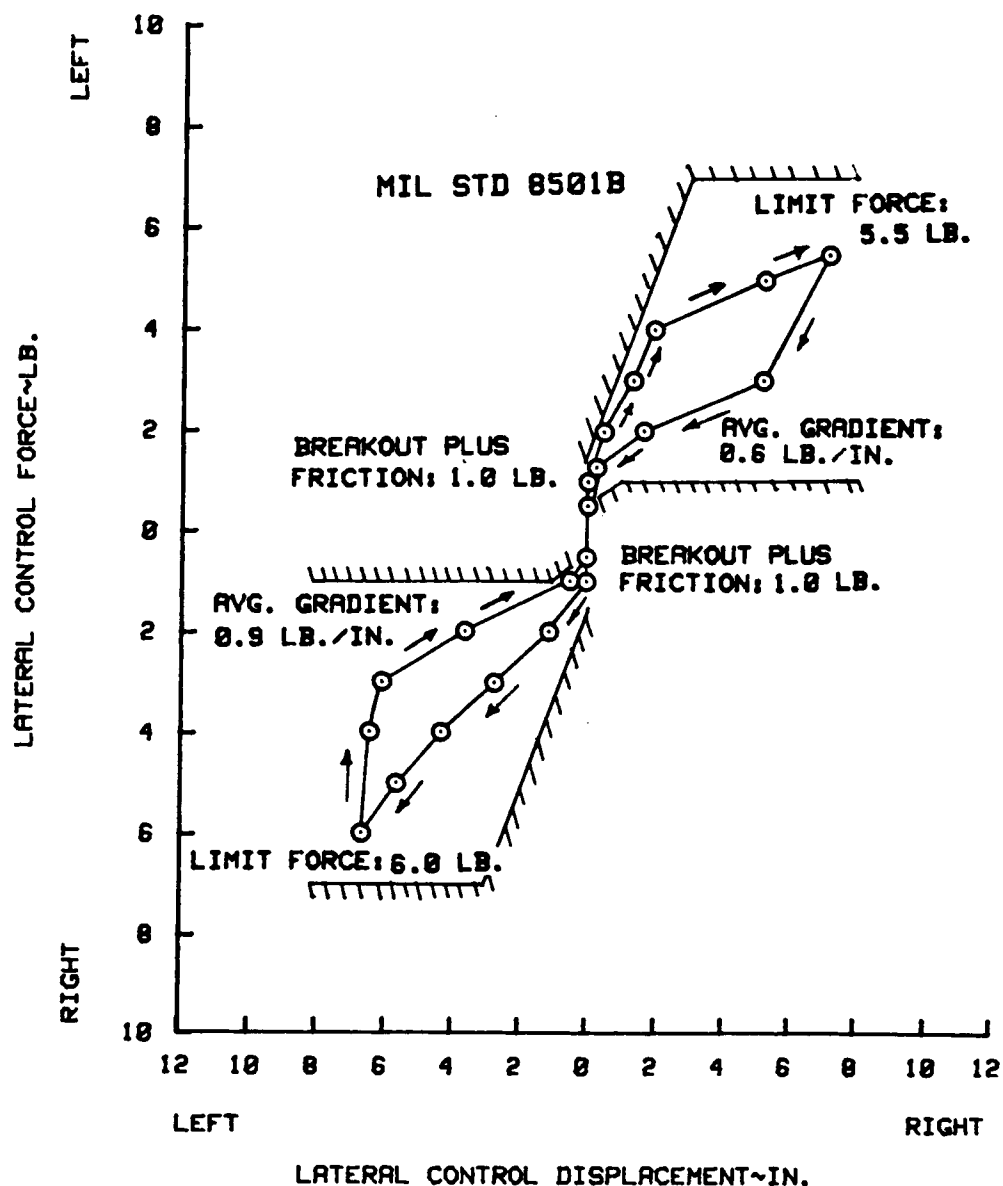


Figure 4
LATERAL CONTROL MECHANICAL CHARACTERISTICS

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED

COLLECTIVE: 50%

EXTERNAL ELECTRICAL POWER: ON

LONGITUDINAL/LATERAL CONTROL:

EXTERNAL HYDRAULIC POWER: ON

50%

ASE: ON

CYCLIC TRIM: ON

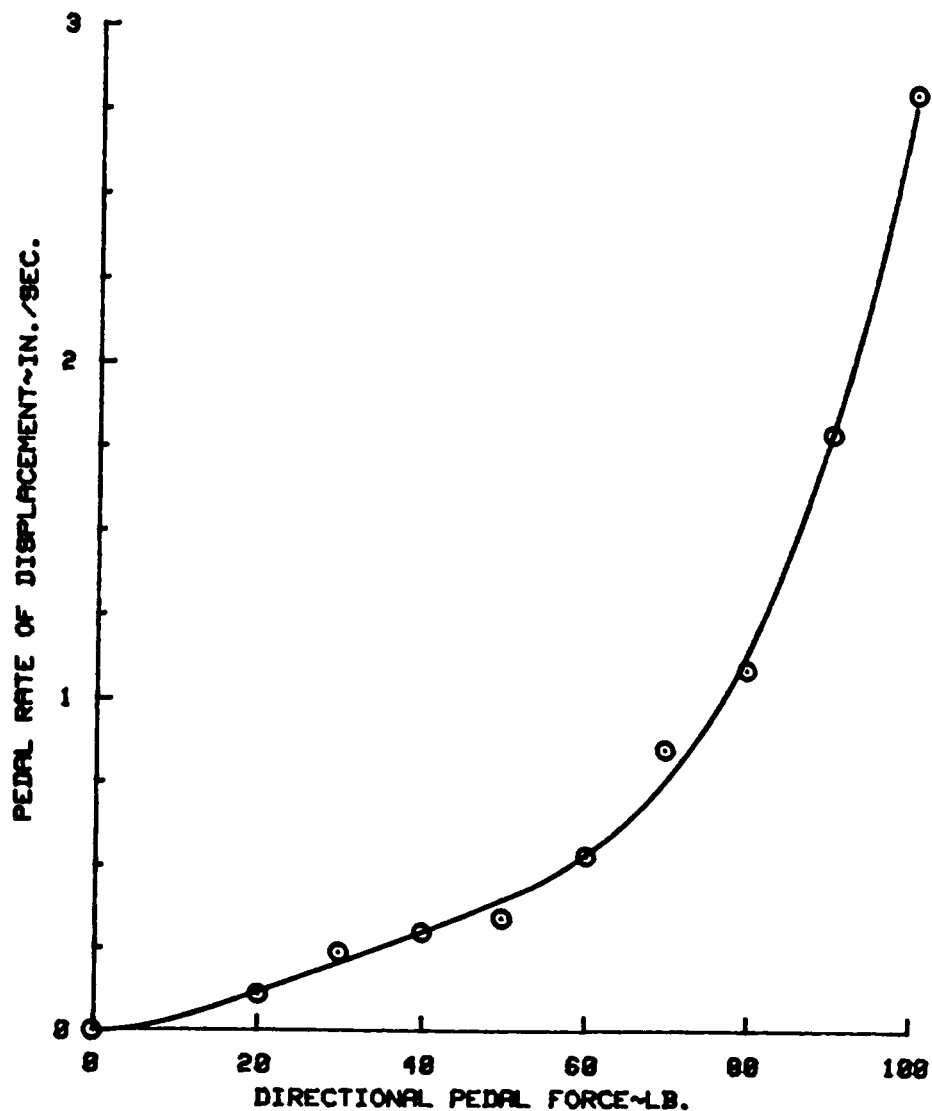


Figure 5
PEDAL VISCOUS DAMPING

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINES/ROTOR STOPPED
EXTERNAL ELECTRICAL POWER: ON
EXTERNAL HYDRAULIC POWER: ON
ASE: ON
CYCLIC TRIM: ON

LONGITUDINAL/LATERAL
CONTROL: 50%

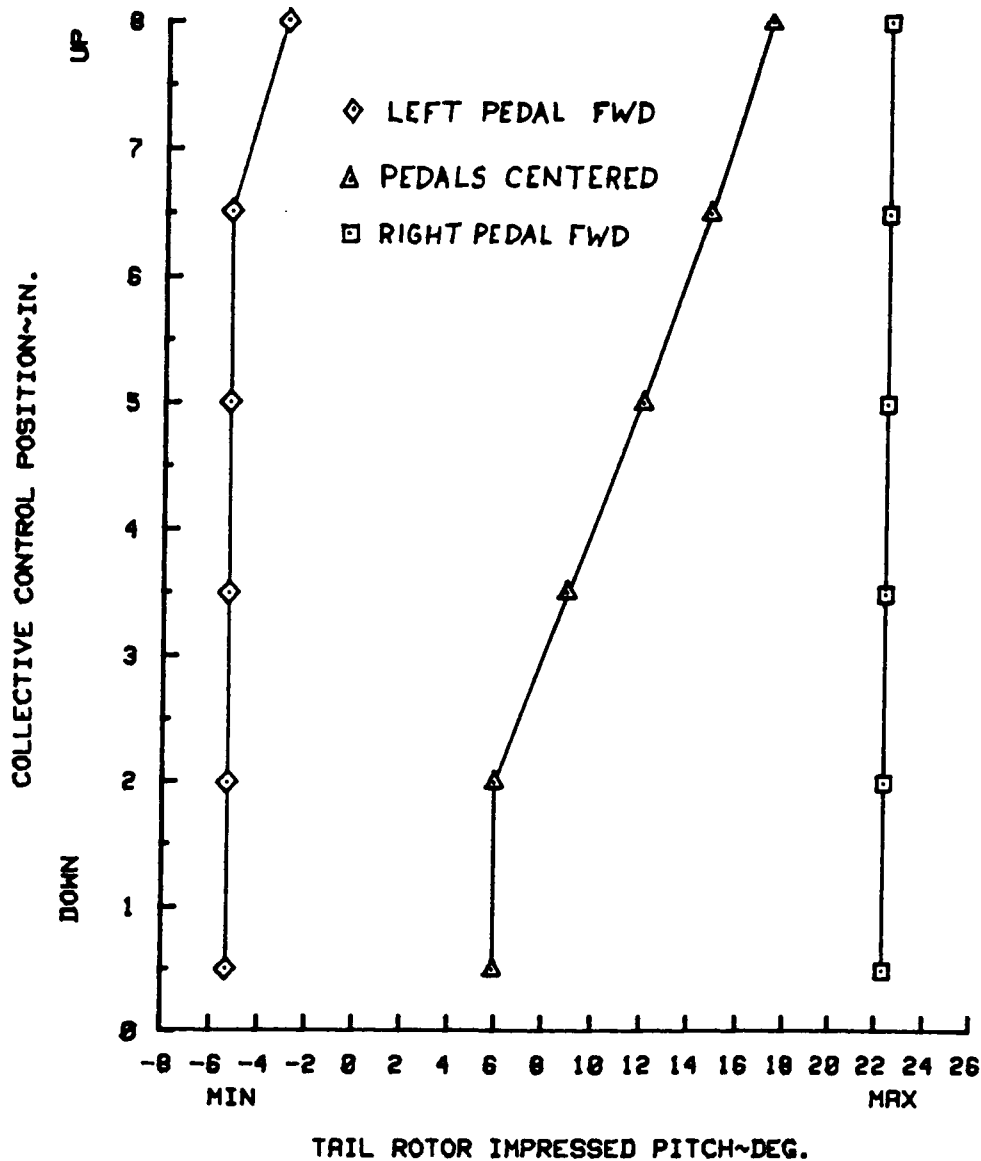


Figure 6
COLLECTIVE TO YAW MIXING

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 19,682 LB.
LONG. CENTER OF GRAVITY:
270.5 IN.
PRESSURE ALTITUDE: -120 FT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%

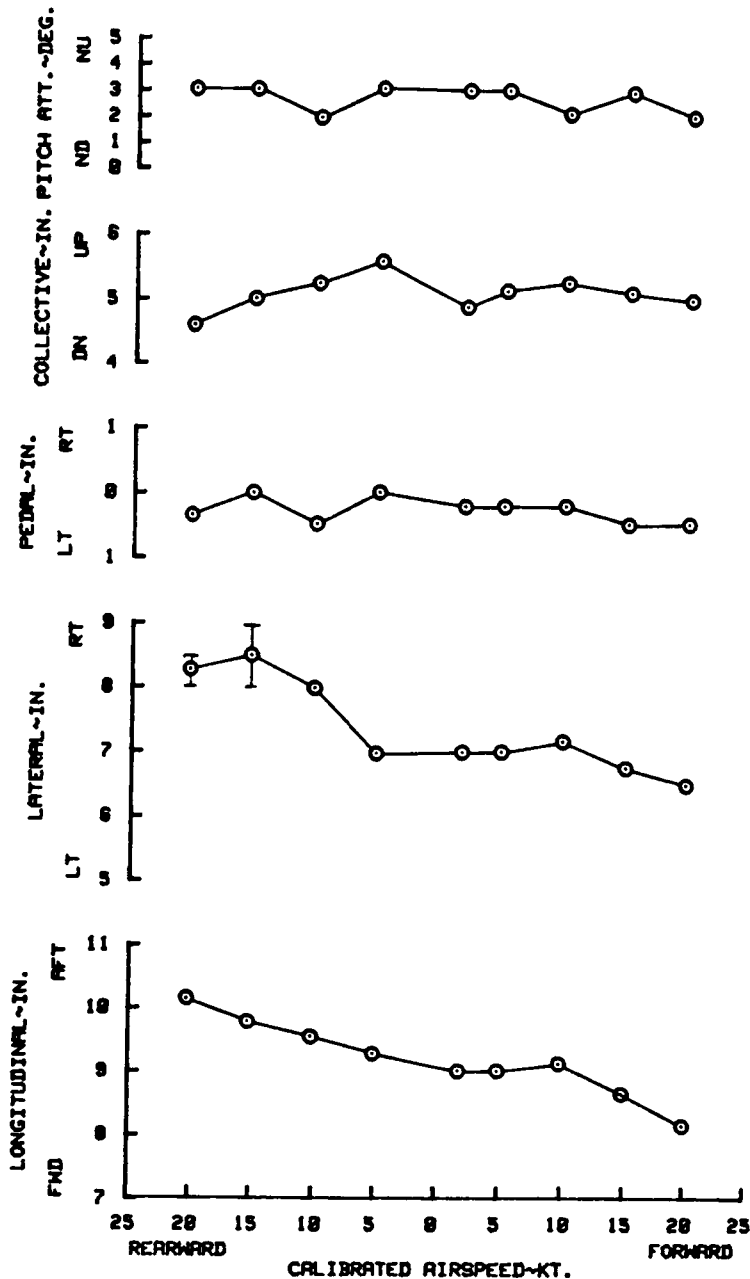


Figure 7
LOW AIRSPEED LONGITUDINAL TRIM FLIGHT CONTROL POSITIONS
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APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 19,682 LB.
LONG. CENTER OF GRAVITY:
270.5 IN.
PRESSURE ALTITUDE: -120 FT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%

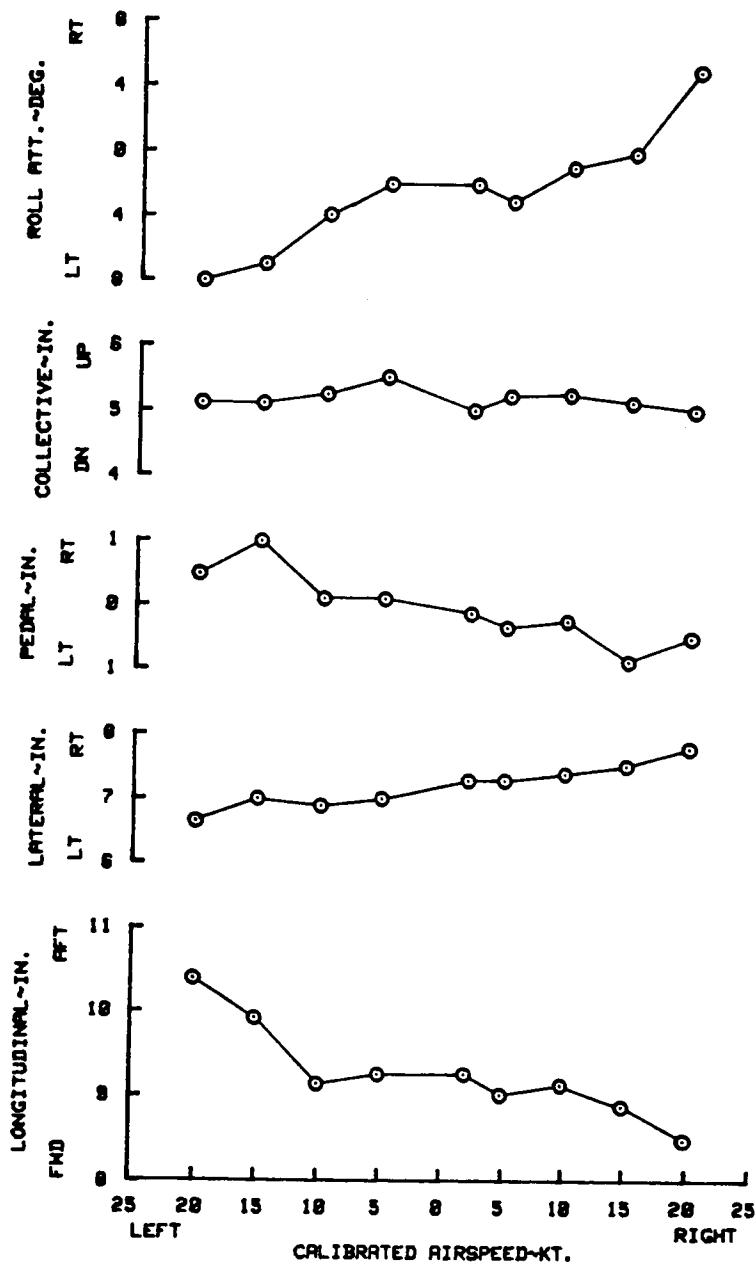


Figure 8

LOW AIRSPEED LATERAL TRIM FLIGHT CONTROL POSITIONS

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 19,257 LB.
LONG. CENTER OF GRAVITY:
268.6 IN.
PRESSURE ALTITUDE: -120 FT.
WIND SPEED: 20 KT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%

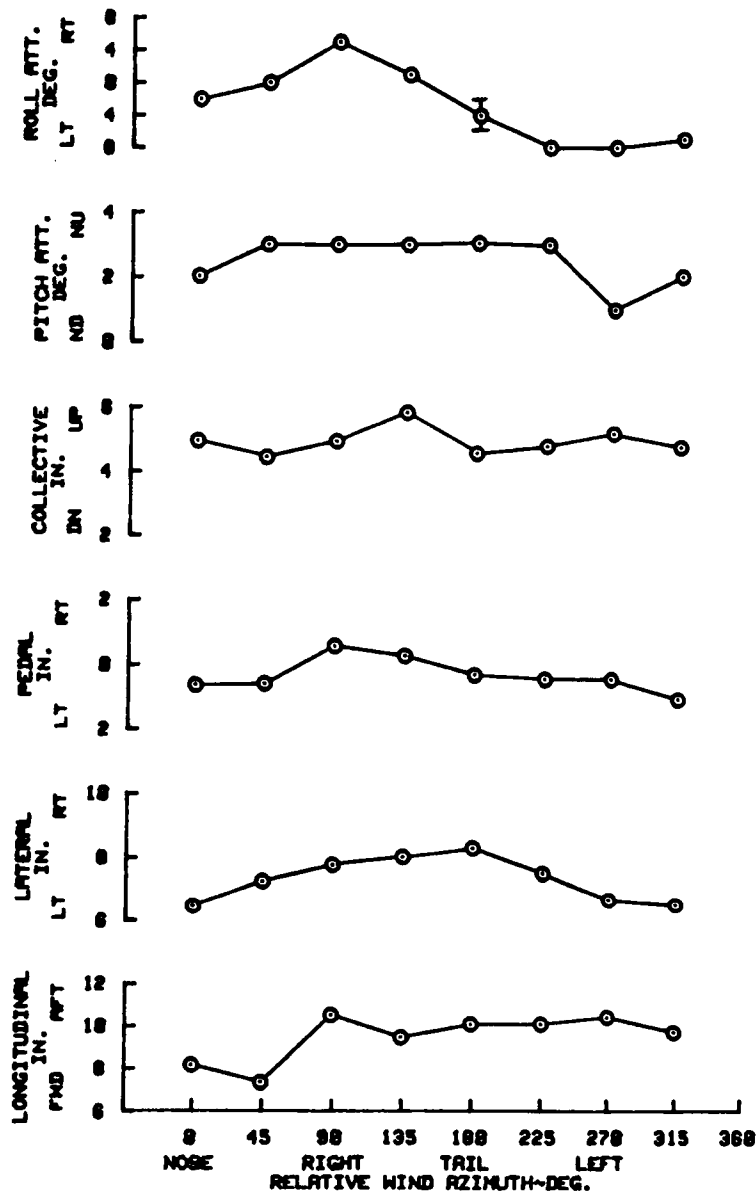


Figure 9
LOW AIRSPEED CRITICAL AZIMUTH
WIND AZIMUTH FROM NOSE OF AIRCRAFT

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 18,649 LB.
LONG. CENTER OF GRAVITY:
268.9 IN.
PRESSURE ALTITUDE: -120 FT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 1 KT. FORWARD

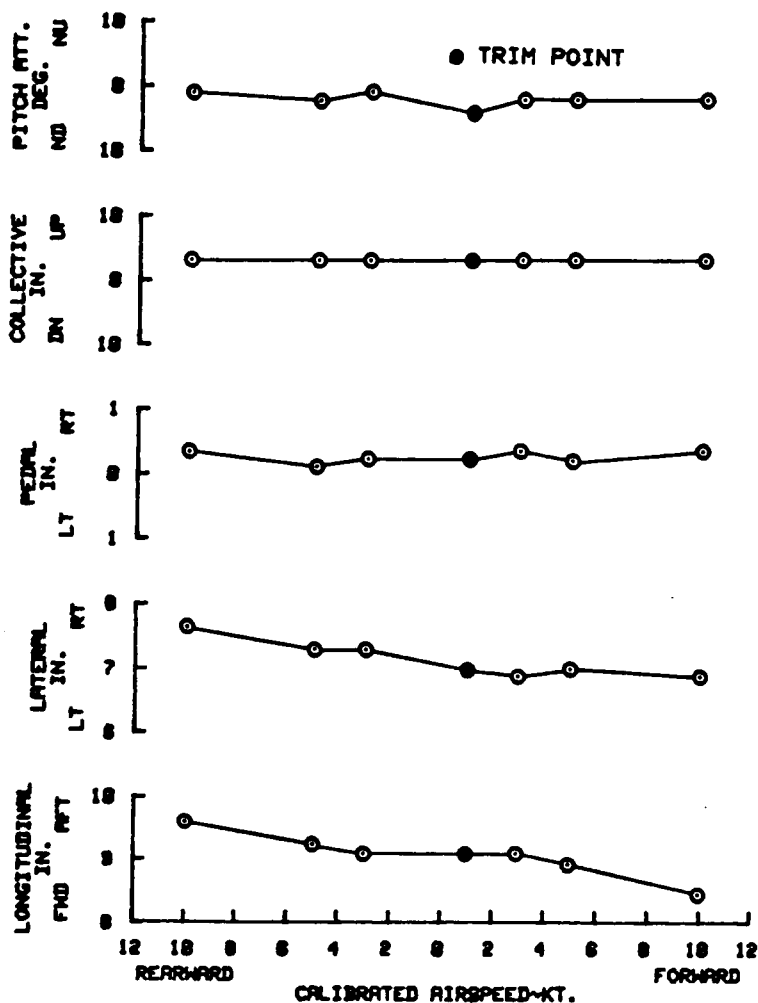


Figure 10
LOW AIRSPEED LONGITUDINAL STATIC STABILITY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 18,649 LB.
LONG. CENTER OF GRAVITY:
268.9 IN.
PRESSURE ALTITUDE: -120 FT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 2 KT. RIGHT

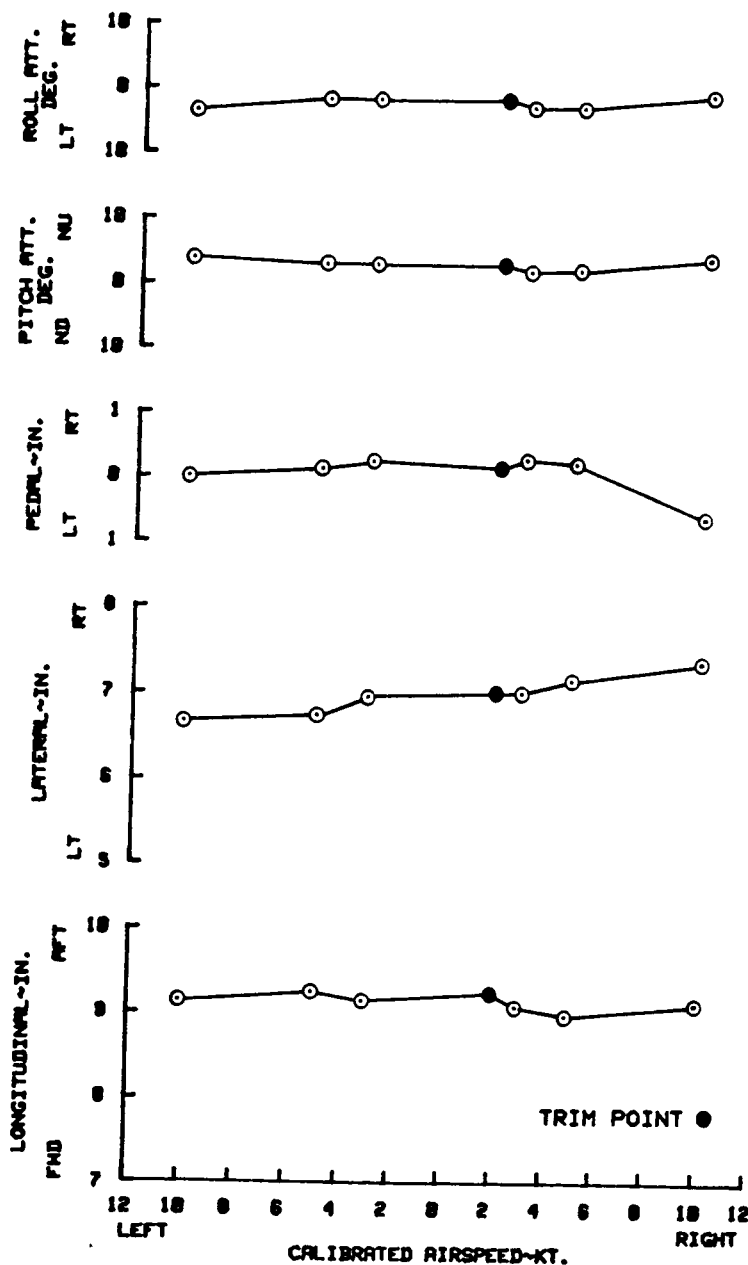


Figure 11
LOW AIRSPEED LATERAL STATIC STABILITY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: PACE VEHICLE

GROSS WT: 18,649 LB.
LONG. CENTER OF GRAVITY:
268.9 IN.
PRESSURE ALTITUDE: -120 FT.
WHEEL HEIGHT: 15 FT.
OAT: 22 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 2 KT. RIGHT
: 1 KT. FORWARD

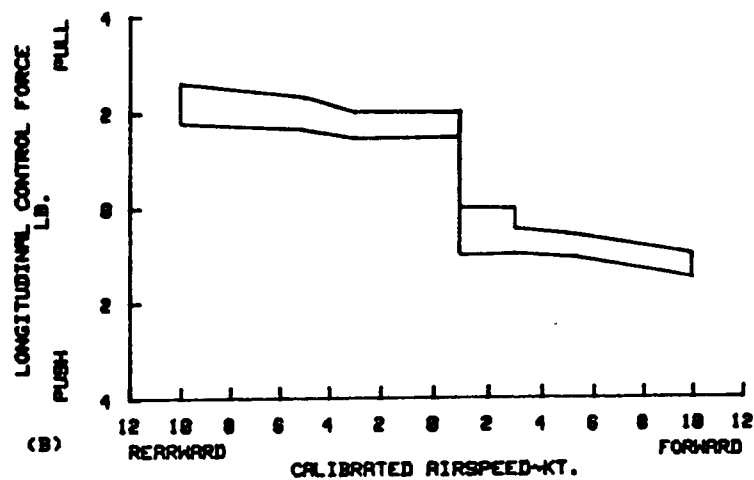
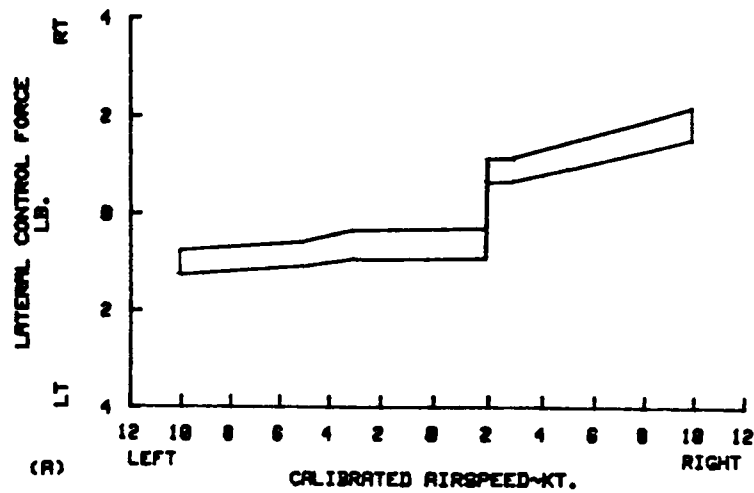


Figure 12
LOW AIRSPEED STATIC STABILITY

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: LEVEL TURNS

GROSS WT: 19,102 LB.
LONG. CENTER OF GRAVITY:
268.3 IN.
PRESSURE ALTITUDE: 2000 FT.
OAT: 18 DEG. C.
ROTOR SPEED: 100%

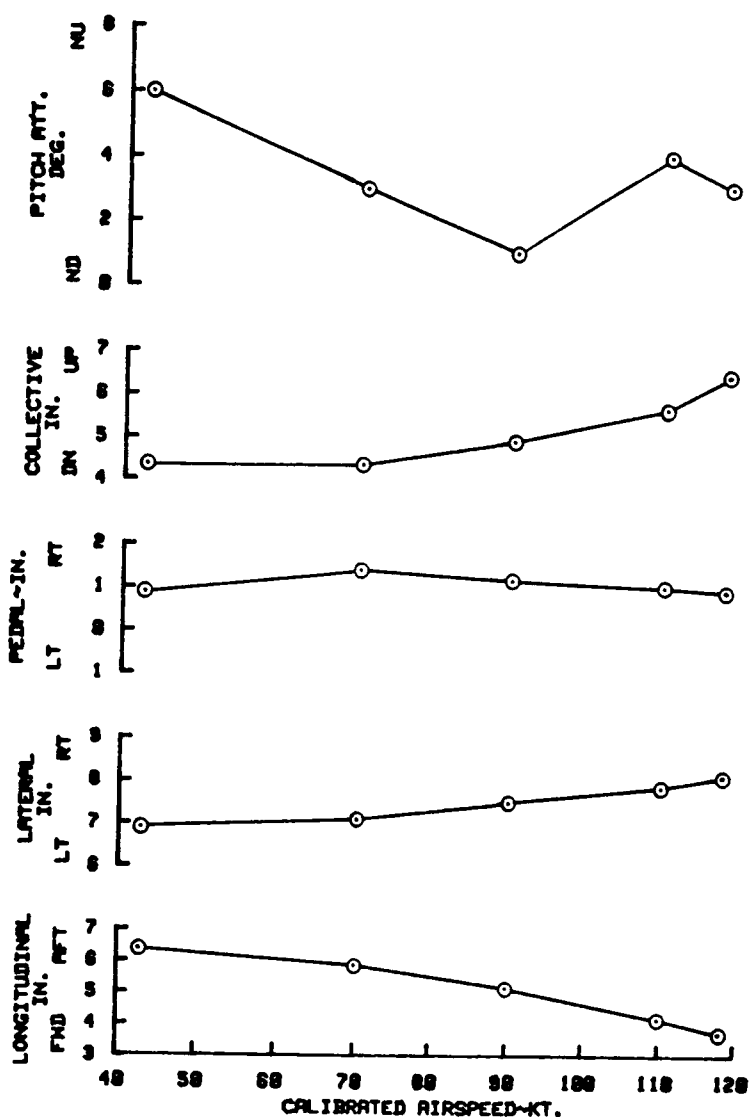


Figure 13
FORWARD FLIGHT TRIMMED FLIGHT CONTROL POSITIONS VS AIRSPEED

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: STEADY TURNS

GROSS WT: 19,102 LB.
LONG. CENTER OF GRAVITY:
268.3 IN.
PRESSURE ALTITUDE: 2000 FT.
OAT: 18 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 90 KIAS

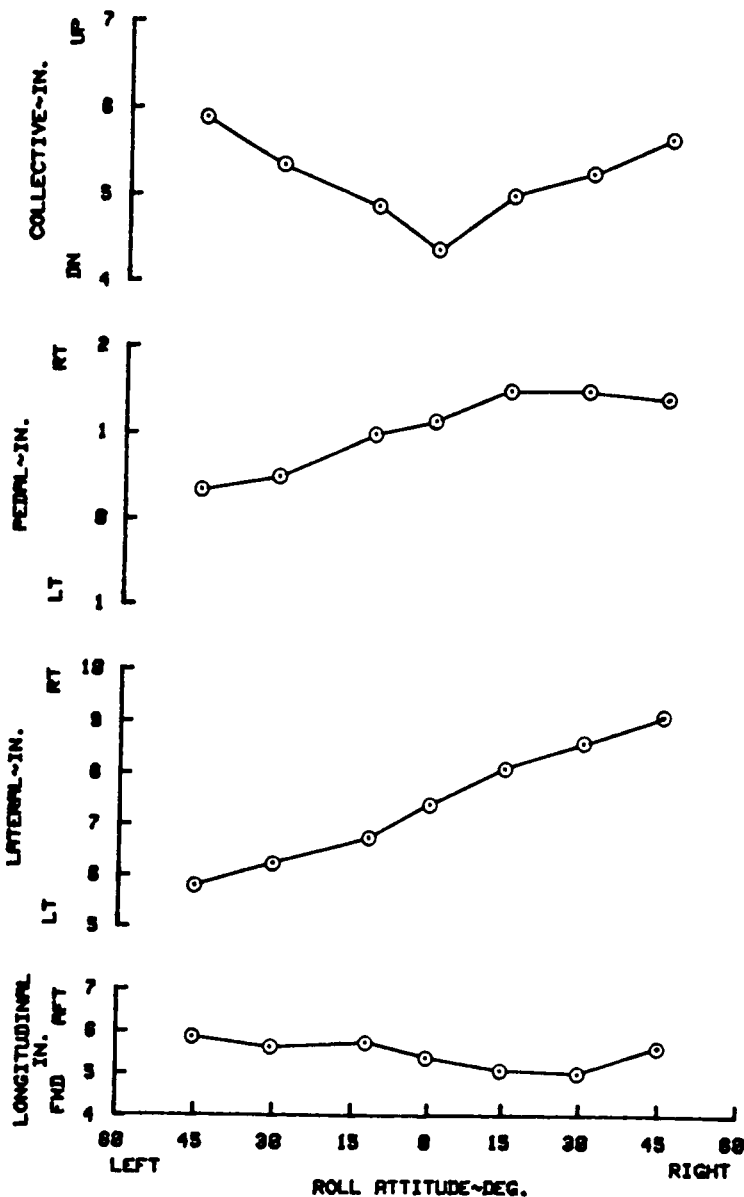


Figure 14

FORWARD FLIGHT TRIMMED FLIGHT CONTROL POSITIONS VS ROLL ATTITUDE

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: CLIMBS & DESCENTS

GROSS WT: 19,102 LB.
LONG. CENTER OF GRAVITY:
268.3 IN.
PRESSURE ALTITUDE: 2000 FT.
OAT: 18 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 90 KIAS

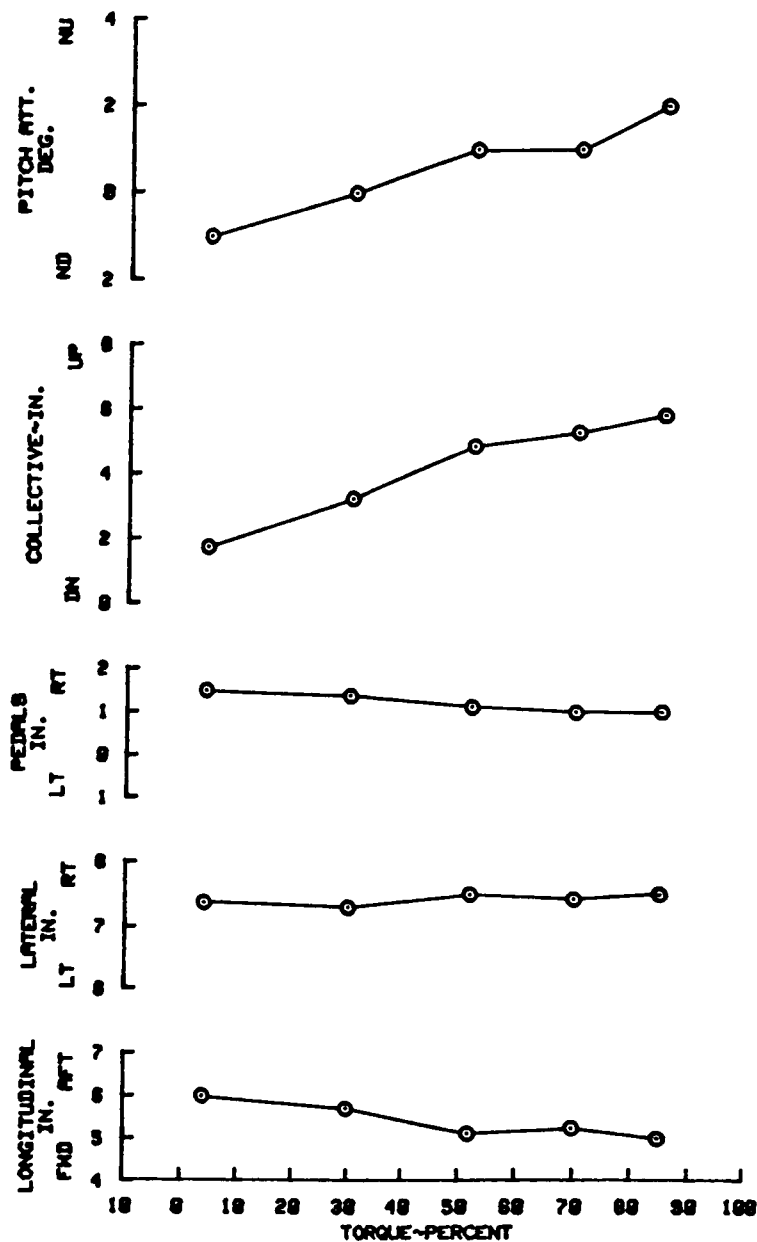


Figure 15

FORWARD FLIGHT TRIMMED FLIGHT CONTROL POSITIONS VS ENGINE TORQUE

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON

GROSS WT: 18,777 LB.
LONG. CENTER OF GRAVITY:
267.8 IN.
PRESSURE ALTITUDE: 3000 FT.
OAT: 16 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 90 KIAS

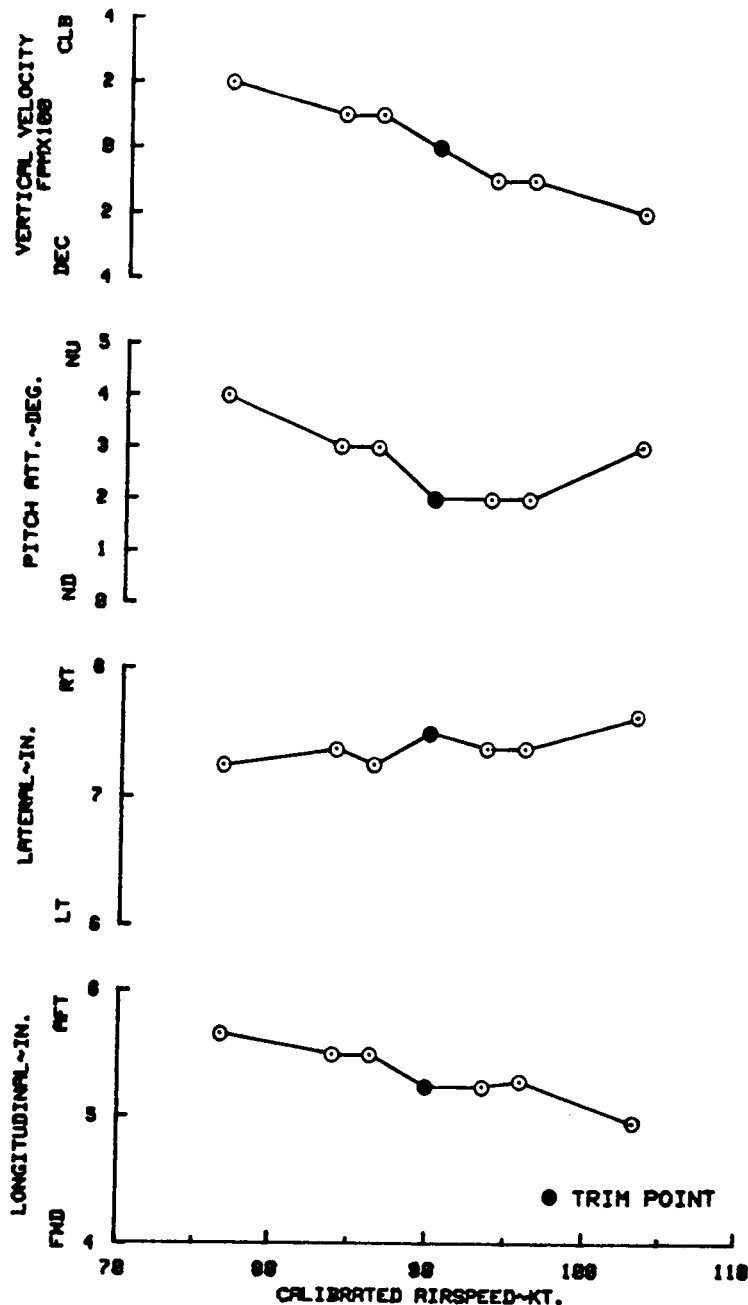


Figure 16

FORWARD FLIGHT LONGITUDINAL STATIC POSITION GRADIENT

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON

GROSS WT: 18,777 LB.
LONG. CENTER OF GRAVITY:
267.8 IN.
PRESSURE ALTITUDE: 3000 FT.
OAT: 16 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 90 KIAS



Figure 17

FORWARD FLIGHT LONGITUDINAL STATIC FORCE GRADIENT

MODEL SH-3H HELICOPTER
BUNO 149728

CONFIGURATION: MK46 TORPEDO,
LEFT FRONT
ASE: ON
CYCLIC TRIM: ON
METHOD: CONSTANT POWER TURNS

GROSS WT: 18,697 LB.
LONG. CENTER OF GRAVITY:
268.5 IN.
PRESSURE ALTITUDE: 3000 FT.
OAT: 15 DEG. C.
ROTOR SPEED: 100%
TRIM AIRSPEED: 90 KIAS

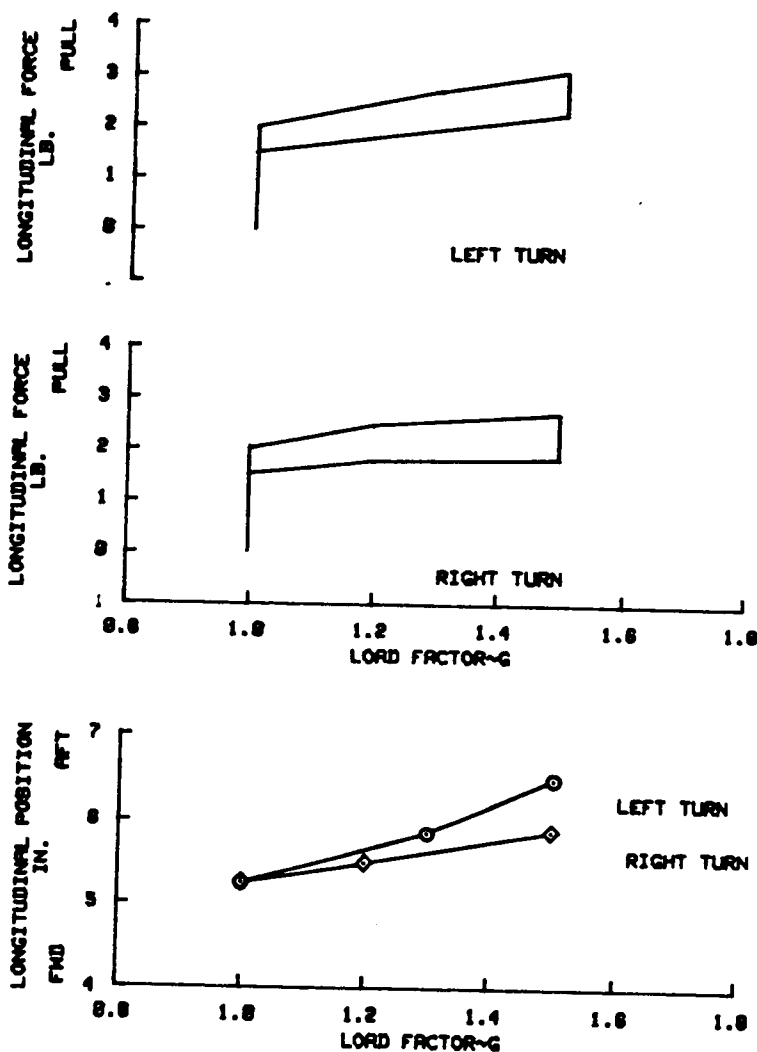


Figure 18
MANEUVERING STABILITY

APPENDIX G

MODEL SH-3H HELICOPTER

SOURCE: NA01-230HLH-1 dtd 15
Jan 83 (SH-3H Flight
Manual)

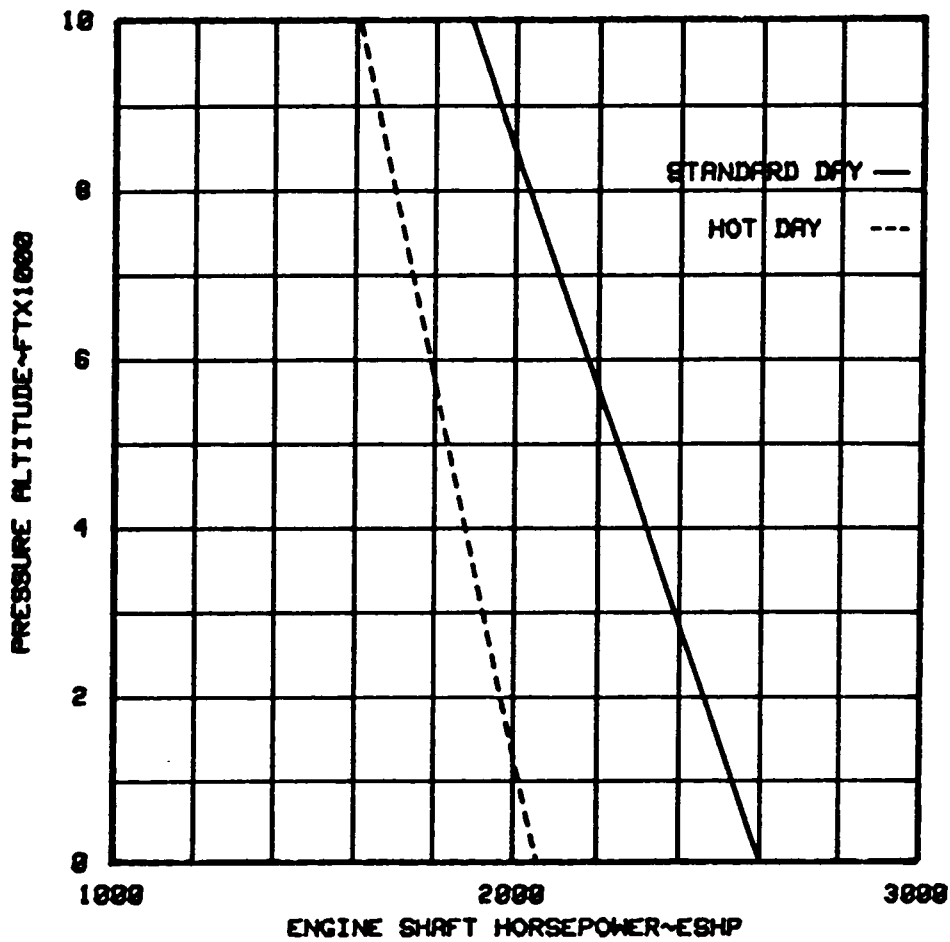


Figure 19
MILITARY POWER AVAILABLE

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 268.8 IN.
FUEL: JP-5 OAT: 16 DEG. C.
FUEL DENSITY: 6.8 LB./GAL. PRESSURE ALTITUDE: -100 FT.
ENGINE ANTI-ICE: OFF GROSS WEIGHT: 19,377 LB.
CONFIGURATION: MK46 TORPEDO;
L-F STA ICE SHIELD INSTALLED
METHOD: VERTICAL CLIMB

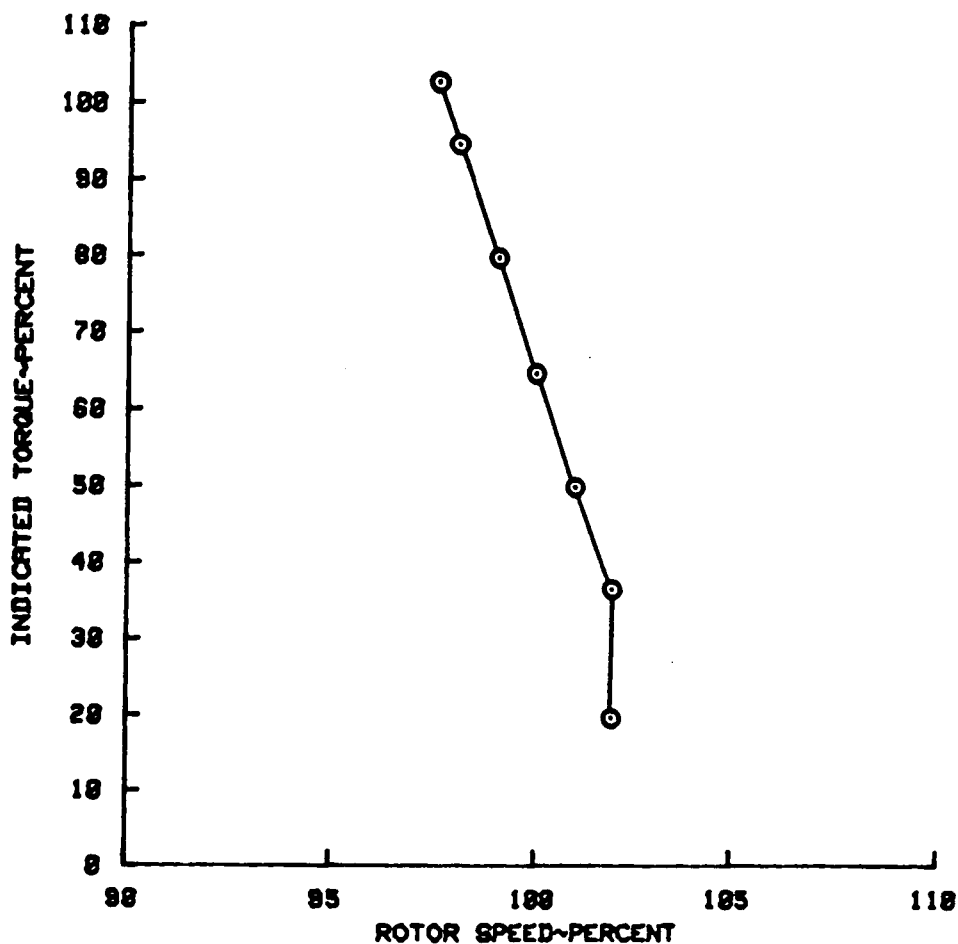


Figure 20
STATIC DROOP

MODEL SH-3H HELICOPTER

BUNO 149728

ENGINE: T58-GE-10; (2)

AVG. LONG. C.G.: 267.9 IN.

FUEL: JP-5

OAT: 40 DEG. C. @ SEA LEVEL

FUEL DENSITY: 6.8 LB./GAL.

ROTOR SPEED: 100%

ENGINE ANTI-ICE: OFF

POWER: MILITARY

CONFIGURATION: MK46 TORPEDO;

L-F STA ICE SHIELD INSTALLED

METHOD: FREE HOVER, OGE

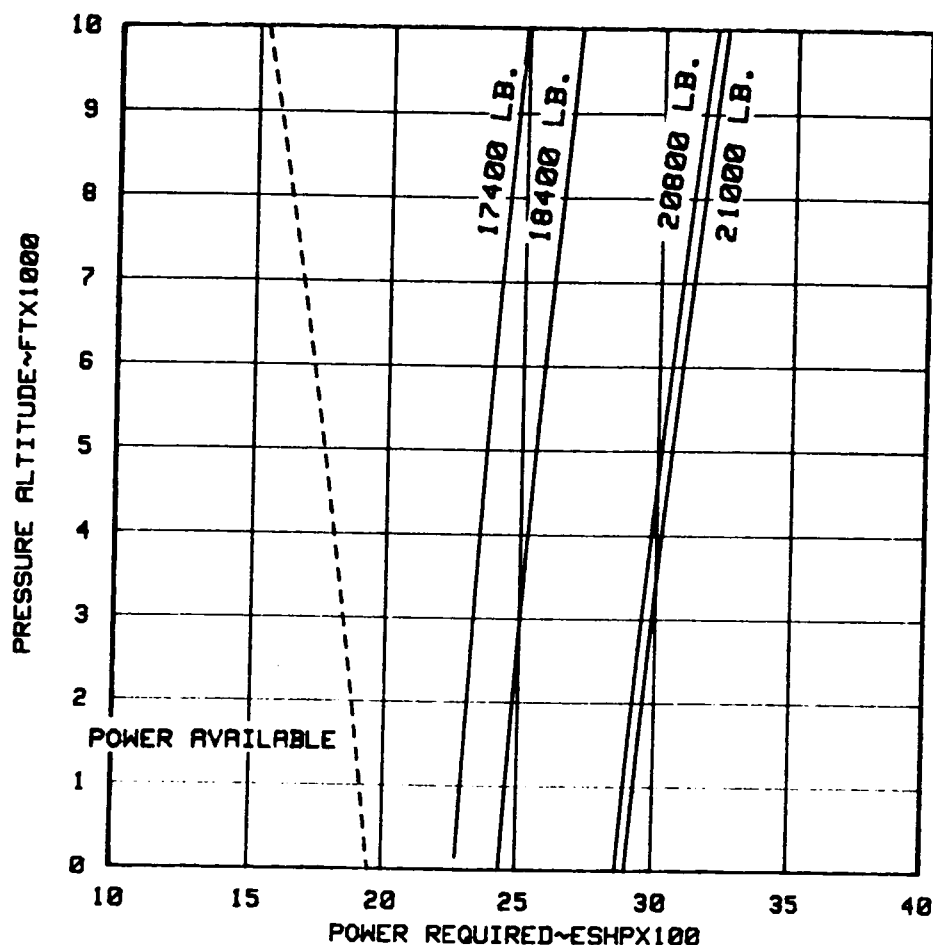


Figure 21

OGE HOVER POWER REQUIRED, HOT DAY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2)	AVG. LONG. C.G.: 267.9 IN.
FUEL: JP-5	OAT: 15 DEG. C. @ SEA LEVEL
FUEL DENSITY: 6.8 LB./GAL.	ROTOR SPEED: 100%
ENGINE ANTI-ICE: OFF	POWER: MILITARY
CONFIGURATION: MK46 TORPEDO;	
L-F STA ICE SHIELD INSTALLED	
METHOD: FREE HOVER, OGE	

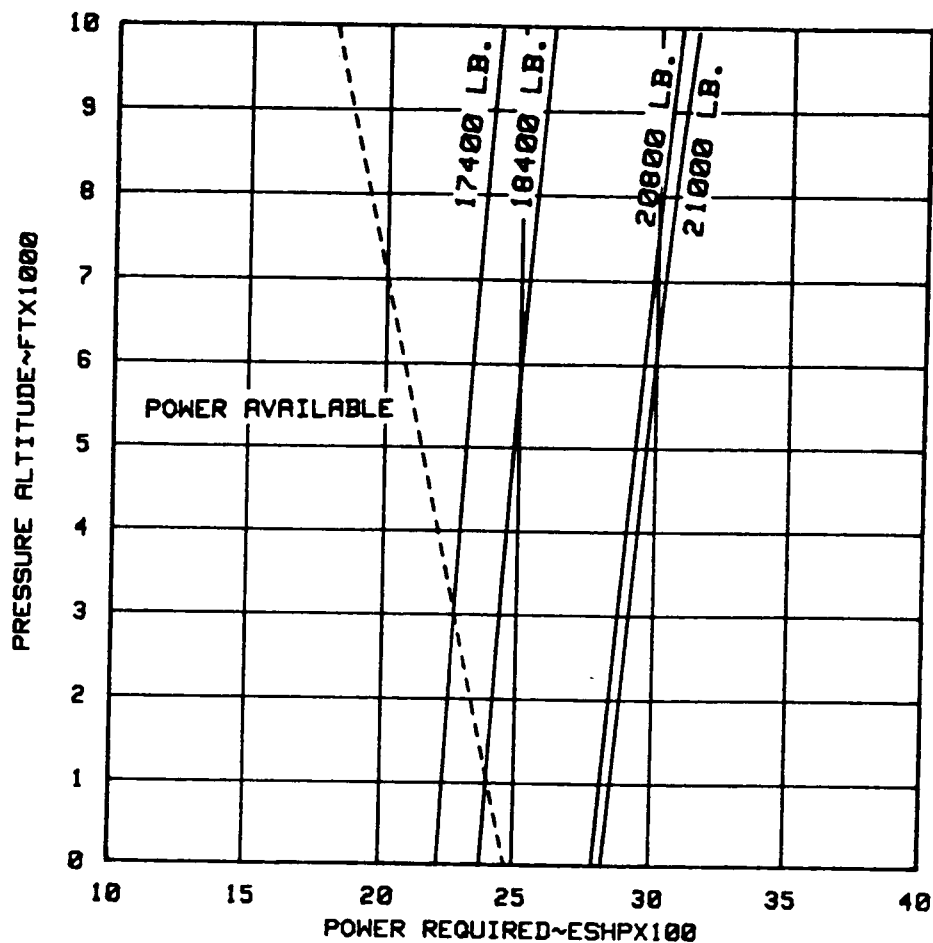


Figure 22
OGE HOVER POWER REQUIRED, STANDARD DAY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 267.9 IN.
FUEL: JP-5 ROTOR SPEED: 100%
FUEL DENSITY: 6.8 LB./GAL. POWER: MILITARY
ENGINE ANTI-ICE: OFF
CONFIGURATION: MK46 TORPEDO;
L-F STA ICE SHIELD INSTALLED
METHOD: FREE HOVER, OGE

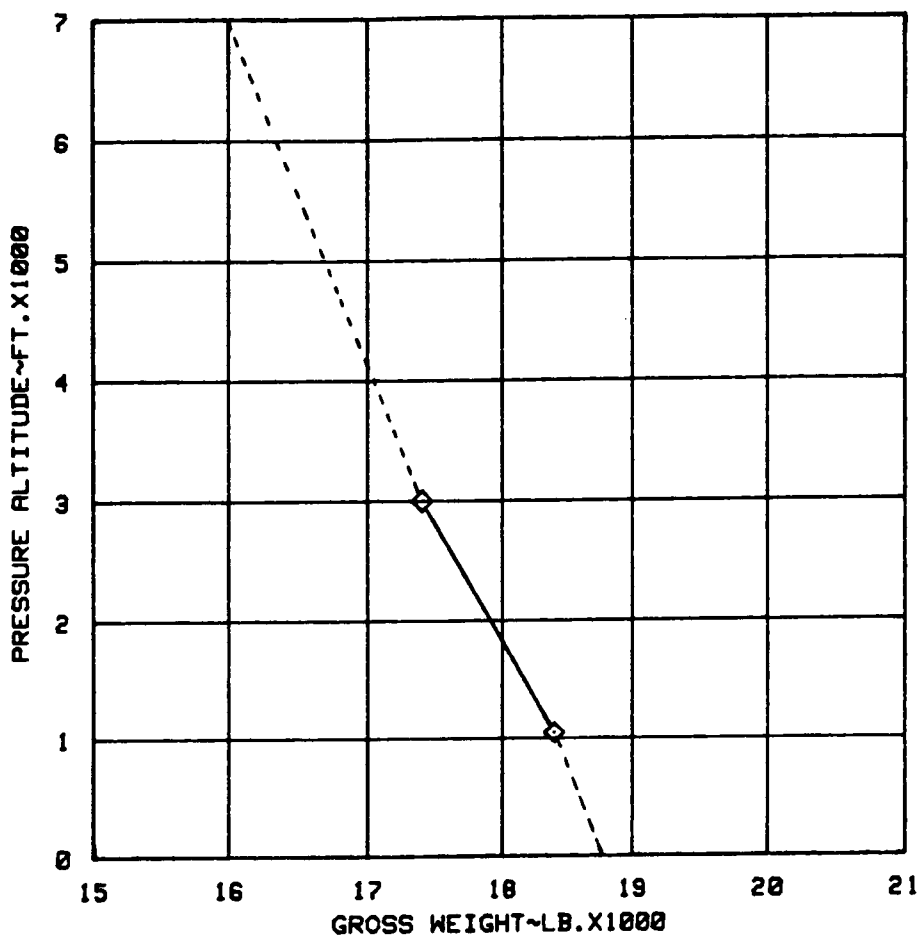


Figure 23
OGE HOVER CEILING

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 267.9 IN.
FUEL: JP-5 OAT: 40 DEG.C. @ SEA LEVEL
FUEL DENSITY: 6.8 LB./GAL. ROTOR SPEED: 100%
ENGINE ANTI-ICE: OFF POWER: MILITARY
CONFIGURATION: MK46 TORPEDO;
L-F STA ICE SHIELD INSTALLED
METHOD: FREE HOVER, IGE
WHEEL HEIGHT: 40 FT.

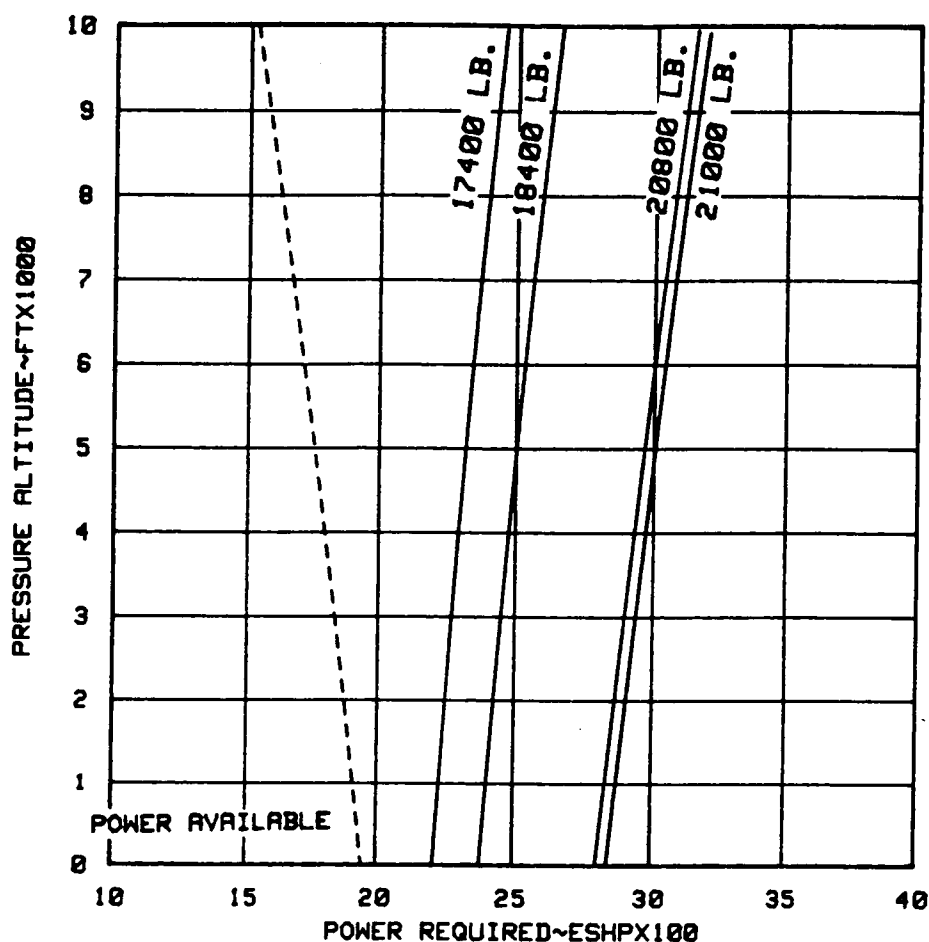


Figure 24
IGE HOVER POWER REQUIRED, HOT DAY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2)	AVG. LONG. C.G.: 267.9 IN.
FUEL: JP-5	OAT: 15 DEG.C. @ SEA LEVEL
FUEL DENSITY: 6.8 LB./GAL.	ROTOR SPEED: 100%
ENGINE ANTI-ICE: OFF	POWER: MILITARY
CONFIGURATION: MK46 TORPEDO; L-F STA ICE SHIELD INSTALLED	
METHOD: FREE HOVER, IGE	
WHEEL HEIGHT: 40 FT.	

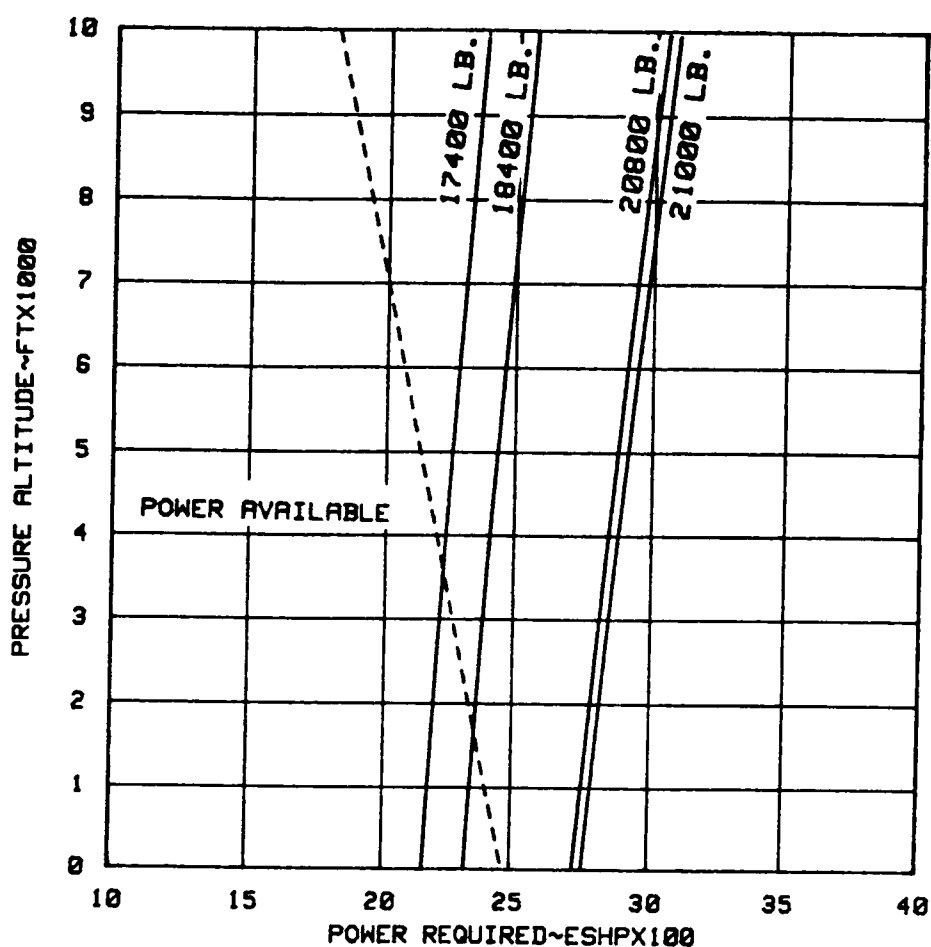


Figure 25
IGE HOVER POWER REQUIRED, STANDARD DAY

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 267.9 IN.
FUEL: JP-5 ROTOR SPEED: 100%
FUEL DENSITY: 6.8 LB./GAL. POWER: MILITARY
ENGINE ANTI-ICE: OFF
CONFIGURATION: MK46 TORPEDO;
L-F STA ICE SHIELD INSTALLED
METHOD: FREE HOVER, IGE
WHEEL HEIGHT: 40 FT.

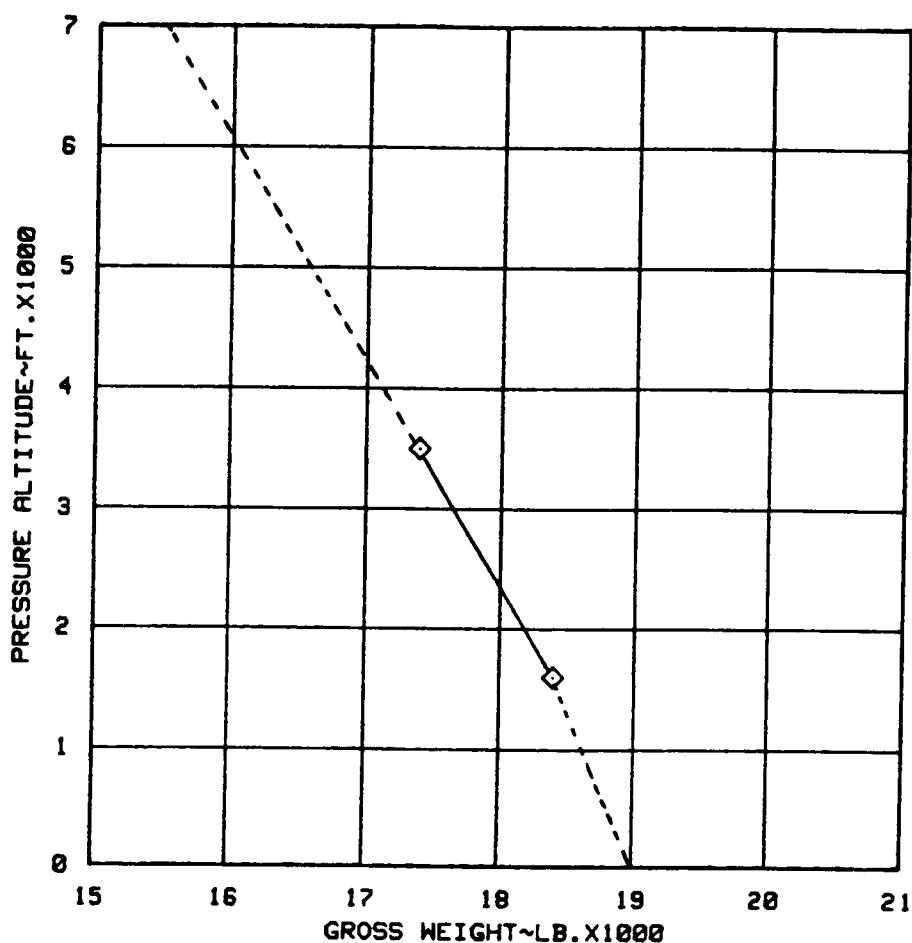


Figure 26
IGE HOVER CEILING

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 268.3 IN.
FUEL: JP-5 OAT: 15 DEG.C.
FUEL DENSITY: 6.8 LB./GAL. ROTOR SPEED: 100%
ENGINE ANTI-ICE: OFF GROSS WEIGHT: 20,800 LB.
CONFIGURATION: MK46 TORPEDO; PRESSURE ALTITUDE: SEA LEVEL
L-F STA, ICE SHIELD INSTALLED (0 FT.)
METHOD: W/o

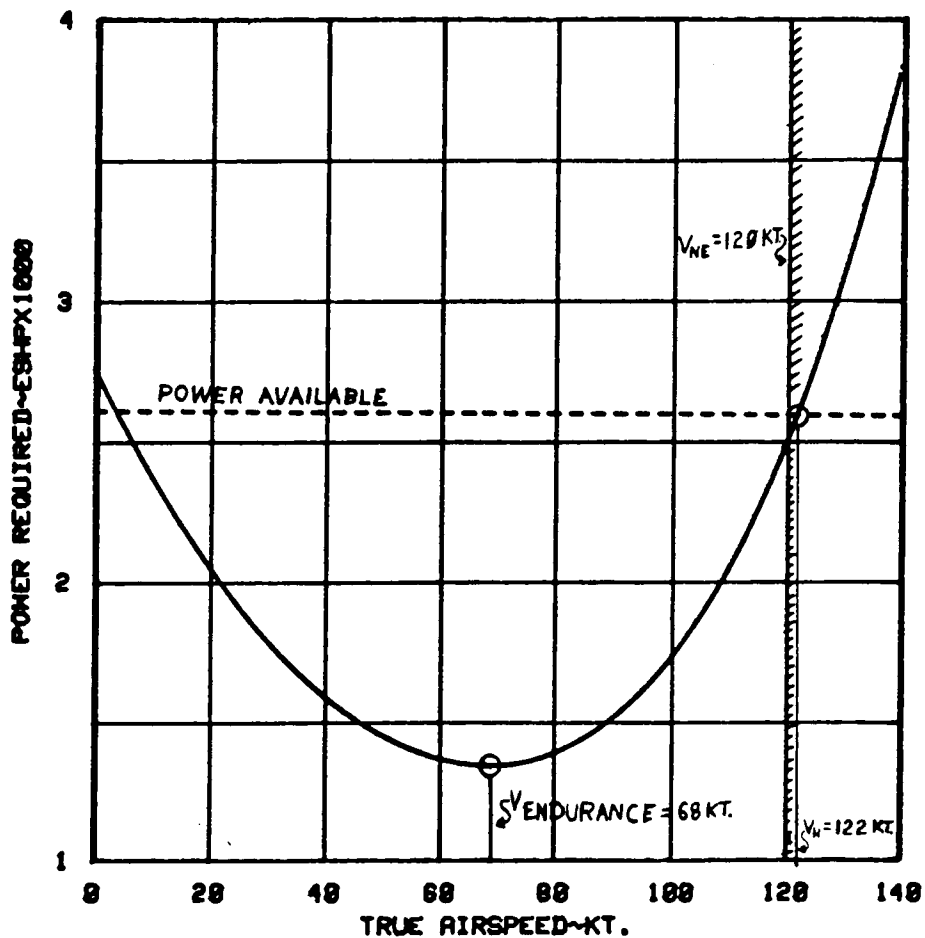


Figure 27
LEVEL FLIGHT PERFORMANCE

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2)

AVG. LONG. C.G.: 268.3 IN.

FUEL: JP-5

OAT: STANDARD DAY

FUEL DENSITY: 6.8 LB./GAL.

ROTOR SPEED: 100%

ENGINE ANTI-ICE: OFF

GROSS WEIGHT: 20,800 LB.

CONFIGURATION: MK46 TORPEDO;

PRESSURE ALTITUDE: SEA LEVEL

L-F STA ICE SHIELD INSTALLED

(0 FT.)

METHOD: W/o

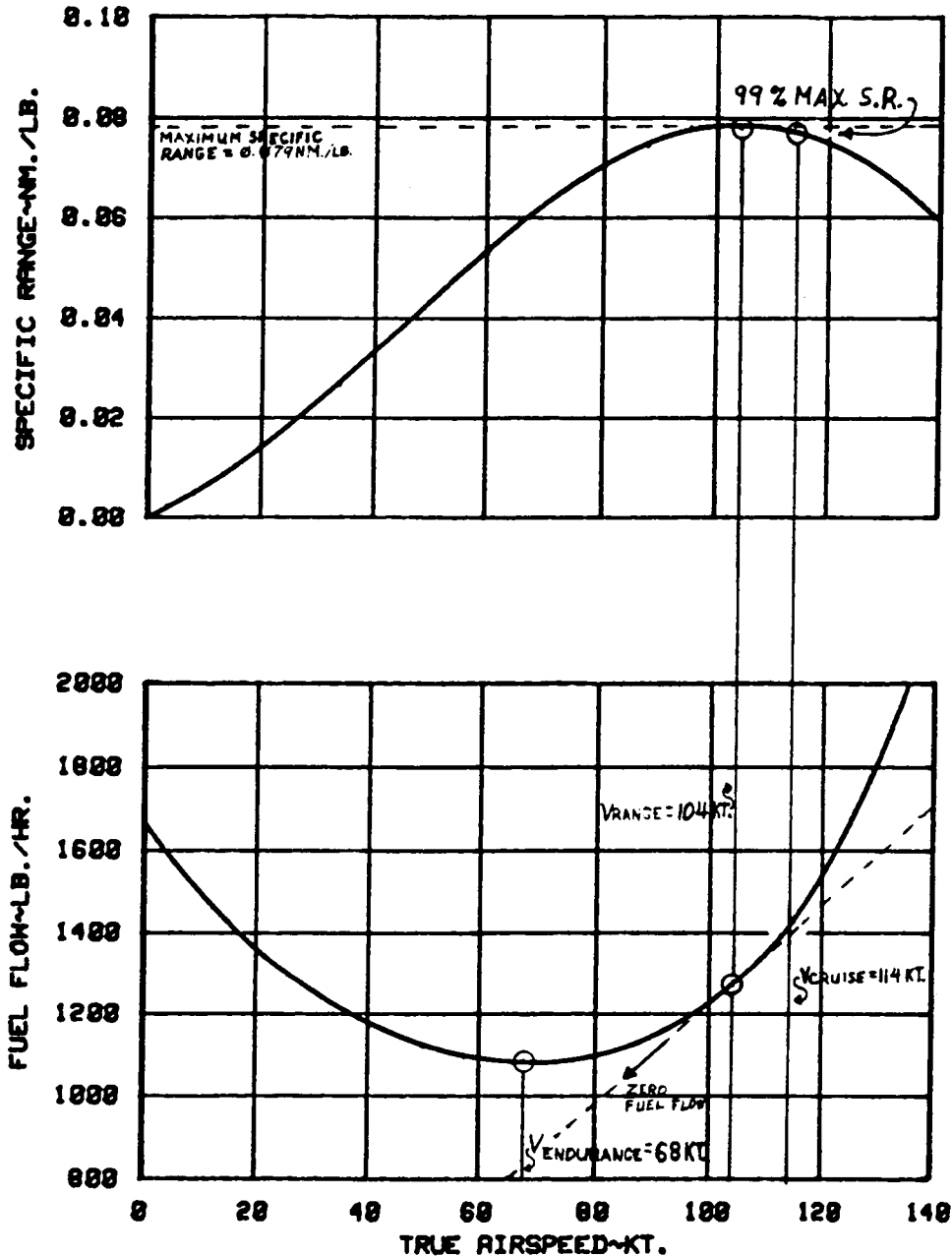


Figure 28

RANGE PERFORMANCE

APPENDIX G

MODEL SH-3H HELICOPTER
BUNO 149728

ENGINE: T58-GE-10; (2) AVG. LONG. C.G.: 269.9 IN.
FUEL: JP-5 OAT: 15 DEG. C. & SEA LEVEL
FUEL DENSITY: 6.8 LB./GAL. ROTOR SPEED: 100%
ENGINE ANTI-ICE: OFF POWER: MILITARY
CONFIGURATION: MK46 TORPEDO;
L-F STA ICE SHIELD INSTALLED
METHOD: VERTICAL CLIMB

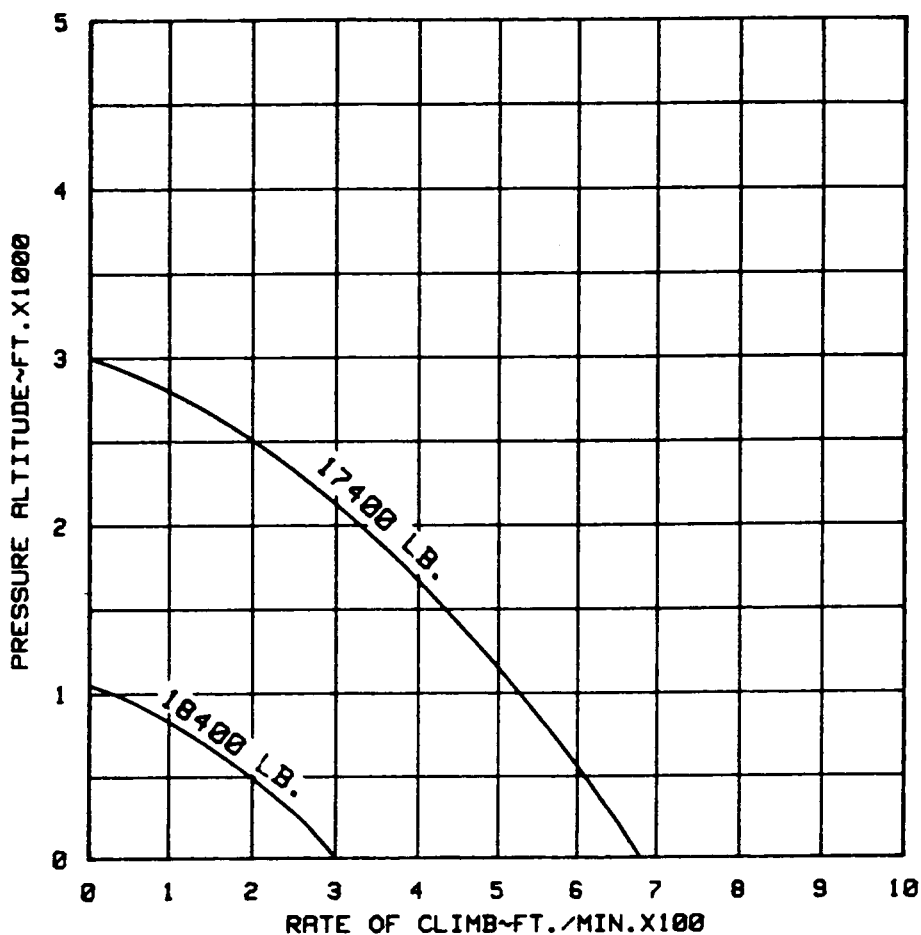


Figure 29
VERTICAL CLIMB PERFORMANCE, STANDARD DAY

APPENDIX G

VITA

John Damian Rudzis was born in Newport, Rhode Island on March 21, 1956. He received his primary and secondary education in the states of California, Maryland and Utah and graduated from East High School, Salt Lake City, Utah in June 1974. The following July he was admitted to the U.S. Naval Academy, Annapolis, Maryland and in June 1978 he was awarded a Bachelor of Science degree in Ocean Engineering and commissioned a Second Lieutenant in the U.S. Marine Corps. In June 1980 he was designated a Naval Aviator and subsequently served as a Marine Assault helicopter pilot in California and overseas. In October 1984 he transferred to Pensacola, Florida to serve as a flight instructor for the U.S. Navy. Following the tour of duty in Pensacola, he transferred to Patuxent River, Maryland in June 1986 and graduated from the U.S. Naval Test Pilot School in June 1987. Since that date Mr. Rudzis has served as a Project Test Pilot at the Naval Air Test Center at Patuxent River. In September 1987 he entered the University of Tennessee Space Institute extension program and received a Master of Science degree in Aviation Systems in August 1989. The author is a member of the Society of Experimental Test Pilots, the U.S. Naval Institute and the Naval Academy Alumni Association. He plans to continue to work in developmental test and evaluation of aircraft and aircraft systems.