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FLYING QUALITIES AND PERFORMANCE EVALUATION
OF THE SH-60B HELICOPTER

A Thesis

Presented for the

Master of Science

Degree

The University of Tennessee, Knoxville

Neil Berryman

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DISCLAIMER

The technical data included in this thesis are actual flight test data collected during a limited scope flying qualities and performance evaluation of a United States Navy SH-60B helicopter. However, all enhancing characteristics and deficiencies attributed to this aircraft, the aircraft systems, and/or the aircraft subsystems are the sole opinion of the author and do not necessarily represent the official position of the Rotary Wing Aircraft Test Directorate, the Naval Air Test Center, the Naval Air Systems Command, or the United States Navy. Similarly, the recommended corrections to the documented deficiencies are presented by the author for the sole purpose of fulfilling the requirements of this thesis, and are not attributable to any of the aforementioned authorities.

ABSTRACT

The United States Navy currently utilizes the SH-60B helicopter for conduct of the antisubmarine warfare/anti-ship surveillance and targeting (ASW/ASST) mission. A limited scope flying qualities and performance evaluation of the SH-60B helicopter was conducted to verify mission suitability prior to additional procurements of the aircraft. Testing was accomplished during two ground assessments totaling 7.0 hr and four flights totaling 6.0 hr. During all performance tests the aircraft was loaded with an inert MK-46 torpedo mounted on the left side of the fuselage. During forward flight and low airspeed flying qualities tests the aircraft was operated in a clean configuration. A limited mission systems evaluation was also conducted with the aircraft in the clean configuration.

The radar altimeter (RADALT) hold feature of the SH-60B automatic flight control system (AFCS) proved safe and accurate during low altitude maneuvering. Since a significant portion of the SH-60B mission profile could be performed at night or in adverse weather conditions, the feature was determined to be an enhancing characteristic which should be incorporated in future designs. Flight testing revealed severe transient droop characteristics of the main rotor system during rapid engine torque

increases. As a result, the aircraft will be susceptible to impact with the ship or the water in the event of a ship landing wave off. During automated hover operations the AFCS heading hold feature exhibited oscillatory divergent yaw characteristics which will result in loss of aircraft control during shipboard recovery, thereby endangering the aircraft and crew. These are Part I deficiencies which must be corrected as soon as possible. Restricted cockpit field of view will require significant pilot compensation when performing high workload tasks (i.e. takeoffs and landings) near the ship. This and six other Part II deficiencies should be corrected as soon as practicable. Testing also revealed four Part III deficiencies which are either too impractical or too costly to be corrected in this model, but which should be avoided in future designs.

Within the scope of this test, the SH-60B helicopter demonstrated excellent potential for the ASW/ASST mission and will be deemed satisfactory upon correction of the Part I deficiencies.

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1. INTRODUCTION

1.1 BACKGROUND

The SH-60B Light Airborne Multi-Purpose System (LAMPS) MK III helicopter is currently utilized for integration with surface combatants in order to fulfill the Sea Control requirements of the United States Navy. The aircraft is deployable from a guided missile frigate for conduct of the antisubmarine warfare (ASW) mission and is designed to detect, localize, entrap, and destroy hostile enemy submarine targets. The SH-60B is capable of carrying and deploying 24 sonobuoys and two MK-46 torpedoes. The secondary mission of the SH-60B is anti-ship surveillance and targeting (ASST), where the aircraft provides a mobile, elevated platform to extend detection and tracking ranges of the parent ship, thus enabling targeting of hostile platforms capable of launching missile attacks. Detailed profiles of the ASW and ASST missions are included in appendix E, table I. Tertiary missions include medical evacuation (MEDEVAC) and search and rescue (SAR). In order to fulfill its primary, secondary, and tertiary missions, the aircraft must exhibit good climb, descent, and forward flight performance and handling qualities; high speed dash and medium speed loiter capabilities; and low airspeed and hover flying qualities and performance suitable for

recovery assist, securing, and traversing (RAST) interface upon return to the parent ship. The aircraft should also be capable of operation at night and under instrument meteorological conditions (IMC).

1.2 PURPOSE

The purpose of this investigation was to conduct limited scope flying qualities and performance testing of the SH-60B helicopter in order to verify mission suitability, and to recommend changes whose incorporation might correct any outstanding deficiencies.

1.3 DESCRIPTION OF TEST AIRCRAFT

The SH-60B Seahawk is a single-piloted, single main rotor helicopter designed and manufactured by the Sikorsky Aircraft Division of the United Technologies Corporation in Stratford, Connecticut. The SH-60B, depicted in appendix F, figure 1, is powered by two General Electric T700-GE-401 turboshaft engines, each rated at 1690 SHP (30 min power) and 1437 SHP (maximum continuous power) under sea level, standard day conditions. The aircraft incorporates a four-bladed, fully articulated main rotor and a four-bladed tractor type tail rotor canted 20 deg to the vertical in order to extend the aircraft center of gravity (CG) envelope. A rotor brake is designed to hold the rotor during initial engine start-up and to expedite

aircraft shutdown. The main rotor blades are designed to fold and unfold automatically and the tail pylon to fold and unfold manually. Non-retractable landing gear consists of two single-wheel main gear and a lockable, dual-wheel, swivel-type tail gear. Each gear incorporates long stroke shock struts designed to dissipate high sink speed energy. The aircraft also incorporates a RAST system to provide mechanical assistance for shipboard landing and deck handling in conditions up to and including Sea State Five. The SH-60B cockpit provides side-by-side seating for the pilot and Airborne Tactical Operator (ATO), while the main cabin incorporates seating for a Sensor Operator (SENSO) immediately aft of the ATO. The main cabin also houses advanced electronics and avionics designed to integrate with a parent ship during the course of the ASW/ASST mission. The flight control system consists of dual cyclic, collective, and directional pedal controls through which inputs are transferred to the main and tail rotors via mechanical linkages and hydraulic servos. An automatic flight control system (AFCS) includes a stability augmentation system (SAS) designed to provide rate damping and short term four axis dynamic stability, an electronic flight control system (EFCS) designed to provide long term stability and various autopilot functions, and an automatic stabilator control system designed to eliminate

undesirable nose-up attitudes during low speed flight. A detailed description of the aircraft is presented in reference (1). A detailed description of the flight control system is included as appendix A. The test aircraft, Bureau Number 162092, was a Lot II production model.

1.4 SCOPE OF TESTS

The flying qualities and performance of the SH-60B aircraft were evaluated during two ground assessments totaling 7.0 hr and four flights totaling 6.0 hr. The ground assessment was performed at the Rotary Wing Aircraft Test Directorate (RWATD) of the Naval Air Test Center (NATC) under ambient lighting conditions. Test flights were conducted in the NATC local flying area under day visual meteorological conditions (VMC). A summary of tests and test conditions is presented in appendix E, table II. Individual flight loadings, configurations, and profiles are presented in appendix E, table III. All normal operating limits contained in reference (1) were observed. Flight restrictions as outlined in appendix B were observed. The aircraft was evaluated against the requirements of Military Specifications (references (2) through (4)), Military Standards, (references (5) through (8)), and a Detail Specification (reference (9)). Applicable specification paragraphs are presented in

appendix E, table IV. Due to the limited scope of this evaluation, further testing is recommended under a more extensive range of flight conditions.

1.5 METHOD OF TESTS

Methods and procedures for all performance testing were in accordance with references (10) and (11). Methods and procedures for all flying qualities testing were in accordance with reference (12). Handling qualities ratings (HQRs) and aircraft vibrations were assigned in accordance with the Cooper-Harper Ratings Scale and the Subjective Vibrations Assessment Scale, references (13) and (14), respectively. Copies of both rating scales are presented in appendix C. During conduct of the SH-60B flight testing, all noted problems and discrepancies were assigned deficiency classifications (Part I, Part II, or Part III) in accordance with the definitions outlined in appendix D. During the ground evaluation, protractors, string, masking tape, and paper clips were used as necessary to facilitate data collection. Cloth tape measures (calibrated to 1/10 in.) were used during ground evaluations and again inflight for control displacement measurements. All flight tests were conducted using normal cockpit instrumentation with the addition of a vertical accelerometer (calibrated to 0.25 g) mounted on the glare shield for dynamic flying qualities testing.

Hand held instrumentation included a stopwatch (calibrated to 0.01 sec), a control fixture (calibrated to 0.5 in.), and a force gauge (calibrated to 0.25 lb). Data were recorded on hand held data cards and a portable cockpit voice recorder. Performance data were reduced on the Hewlett-Packard HP-9845A computer. Due to the lack of specialized flight test instrumentation, it is recommended that any further testing be conducted with a full complement of flying qualities and performance instrumentation.

2. RESULTS AND DISCUSSION

2.1 COCKPIT EVALUATION

2.1.1 General

A cockpit evaluation was conducted during one 2.5 hr ground assessment and concurrent with all test flights under the conditions presented in appendix E, table II. The ground assessment was performed outdoors under day VMC conditions while performing simulated normal and emergency procedures as outlined in reference (1). Anthropometric measurements of the evaluating engineer are presented in appendix E, table V. Flight gear worn by the evaluating engineer is listed in appendix E, table VI. Linear measurements were made with retractable cloth tape measures and angular measurements with a protractor and string arrangement. The pilot seat was positioned for optimum comfort except during field of view measurements. These were made with the ATO seat occupied and the pilot seat adjusted such that the eye level of the evaluator was at design eye position (DEP), as specified in reference (7). The pilot seat harness was locked for all takeoff, landing, and emergency procedures.

2.1.2 Field of View

External field of view was quantitatively evaluated from the pilot DEP in the vertical, horizontal, and

lateral planes during a ground evaluation under day VMC conditions. Field of view was qualitatively evaluated concurrent with all applicable flight tests under the conditions presented in appendix E, table II. Data are presented in appendix F, figures 2 through 4. Forward field of view was obstructed by windshield center support posts from 17 to 21 deg left and from 35 to 47 deg right of aircraft center line; upper support posts from 35 to 45 deg above the horizon; and the glare shield and instrument panel at 12 deg below the horizon. The pilot chin bubble provided limited field of view below the instrument panel. Lateral field of view was obstructed by the upper console from 43 to 74 deg left of vertical and by the right upper door post from 60 to 78 deg right of vertical. The restricted forward field of view improved with the nose down pitch attitudes characteristic of high airspeed flight, but was considered uncomfortable during simulated shipboard landings. The restricted right lateral field of view required the pilot to significantly alter his position in order to initiate safe turns to the right, thereby making normal mission operations difficult. The restricted field of view will result in the pilot reducing his aggressiveness to ensure safe flight operations, thereby degrading his overall mission effectiveness. The restricted cockpit field of view of the SH-60B helicopter during right turns and low airspeed flight is a Part II

deficiency which should be corrected as soon as practicable.

2.1.3 Engine Control Quadrant

2.1.3.1 Accessibility and displays. The engine control quadrant was evaluated for pilot and ATO accessibility and display of settings. The control quadrant, as depicted in appendix F, figure 5, is labeled with OFF, IDLE, FLY, and LOCKOUT position indications for setting the engine power condition levers (PCLs). Each PCL is scribed with a hash mark for line-up with the position indications. Neither the position indications or line-up lines could be seen from the pilot's normal seating position, thereby requiring the pilot to lean approximately 1.5 ft forward and left for visual acquisition. This will be an annoying inconvenience which will result in the pilot relying on PCL detents for control positioning during engine run up. The obscured labeling of the engine control quadrant power condition levers is a Part III deficiency which should be avoided in future designs.

2.1.3.2 Functional operation. The engine control quadrant was also evaluated for functional operation of the PCLs. The engine control quadrant, as depicted in appendix F, figure 5, is equipped with mechanical detents at the IDLE and FLY positions for PCL setting. The size

and shape of the detents is such that rapid PCL movement allowed the levers to pass the indentations without being stopped. The engine characteristics of the SH-60B are such that rapid engine run up can be initiated if necessary. During engine run up for a Ready Alert launch, the pilot will rely on the PCL detents to correctly position the levers due to the obscured labeling of the engine control quadrant (paragraph 2.1.3.1). The inadequate stopping capabilities of the detents will allow the pilot to inadvertently advance the PCLs into LOCKOUT, resulting in engine over speed, over temperature, and subsequent damage. The inadequate stopping capability of the engine control quadrant PCL detents is a Part II deficiency which should be corrected as soon as practicable.

2.1.4 Environmental Control System Ventilation

Environmental control system (ECS) ventilation was evaluated for location, functional operation, and cooling effectiveness. An ECS vent is located on either side of the overhead panel directly above the center avionics/electronics console, as depicted in appendix F, figure 6. During a simulated mission flight in overcast weather, cool air provided by the ECS was condensed into water vapor at the mouth of both vents. This water then dripped onto the center console for approximately 15 min. During

operations in ambient conditions conducive to water vapor condensation, the location of the ECS vents will allow water to drip from the vents onto various aircraft electronics and avionics, thereby increasing the possibility of an electrical short circuit that will necessitate termination of the mission. Exposure of the center avionics/electronics console to dripping condensation from the ECS vents during ECS operation is a Part II deficiency which should be corrected as soon as practicable.

2.2 FLYING QUALITIES

2.2.1 General

Mechanical characteristics, forward flight flying qualities, and low airspeed flying qualities of the SH-60B helicopter were evaluated in accordance with techniques described in reference (15), under the conditions presented in appendix E, tables II and III. Data are presented in appendix F, figures 7 through 26. Mechanical characteristics were assessed quantitatively during a ground evaluation and qualitatively during flight. Where appropriate, forward flight flying qualities tests were conducted at 80 and 140 knots indicated airspeed (KIAS), representative of climb and loiter airspeeds and dash airspeed, respectively. Sideslip angles with trim ball displacement were determined by flying constant headings

and noting compass variations with differential ball settings.

2.2.2 Mechanical Characteristics

2.2.2.1 General. Test configuration during the mechanical characteristics ground assessment was with the rotor system statically spread utilizing external alternating current (AC) electrical power and backup pump hydraulics for flight control operation. Boost, SAS 2, and trim systems were engaged; all remaining AFCS components were off. Test method and procedures were in accordance with reference (12), with results presented graphically in accordance with reference (15). Displacement and force measurements were taken from the pilot and ATO controls, respectively. To determine total system freeplay, oscillatory inputs of incrementally decreasing magnitude were applied to the respective cockpit controls until blade movement at the rotor head was no longer apparent. Cyclic and collective centering and dynamics were evaluated by displacing the respective controls 3.0 in. from trim and then observing their behavior after release. Directional pedals were evaluated similarly from 2.0 in. displacement. Control jump was measured by displacing the respective controls 1.0 in. from trim and observing their behavior upon trim release via the appropriate button. Beeper trim rate was

determined by timing beep-induced control travel through 3.0 in. of the trim envelope. Trim lag characteristics were evaluated by operating the beeper trim in a single direction and then reversing direction to determine hysteresis. Data are presented in appendix F, figures 7 through 13, and summarized in appendix E, table VII. Results were evaluated against the requirements of reference (4), with applicable specifications and compliance outlined in table VII.

2.2.2.2 Longitudinal cyclic. Mechanical characteristics of the longitudinal cyclic were evaluated with the collective and directional controls centered. Additional control envelope measurements were taken with the collective lever full down and full up, and with the pedals full left and full right. Data are presented in appendix E, table VII, and appendix F, figures 7, 8, and 11. With the collective and pedals centered, longitudinal control travel measured 9.4 in. Total travel varied with collective and pedal position, but was apparently unaffected by lateral cyclic. Total and trim system freeplay measured 0.1 and 0.2 in., respectively. Breakout force (plus friction) was 1.25 lb and 1.0 lb in the forward and aft directions, respectively. Average force gradient in the forward direction was 1.6 lb/in. over the first inch of travel and 1.1 lb/in. overall. Average

force gradient in the aft direction was 1.8 lb/in. over the first inch and 1.2 lb/in. overall. Limit forces were 6.25 and 6.5 lb in the forward and aft directions, respectively. Control centering was absolute and control dynamics were lightly damped, exhibiting three overshoots when released from either side of trim. Stick jump measured 0.1 in. for both forward and aft displacement. The mechanical characteristics of the longitudinal cyclic met the requirements of the specification against which they were tested, with the exception of paragraph 3.2.3 in that stick jump was noticeable.

2.2.2.3 Lateral cyclic. Mechanical characteristics of the lateral cyclic were evaluated with the collective and directional controls centered. Additional control envelope measurements were taken with the collective lever full down and full up, and with the pedals full left and full right. Data are presented in appendix E, table VII, and appendix F, figures 7, 8, and 12. With the collective and pedals centered, lateral control travel measured 9.0 in. Total travel varied with collective and pedal position and with aft longitudinal cyclic. Total and trim system freeplay each measured 0.1 in. Breakout force (plus friction) was 1.25 lb left and right. Average force gradient was 1.5 lb/in. over the first inch of left and right travel, 0.8 lb/in. overall to the left,

and 0.9 lb/in. overall to the right. Limit forces were 5.0 and 4.25 lb to the left and right, respectively. Control centering was absolute and control dynamics were moderately damped, exhibiting two overshoots when released from either side of trim. Stick jump measured 0.1 in. for both left and right displacement. The mechanical characteristics of the lateral cyclic met the requirements of the specification against which they were tested, with the exception of paragraph 3.2.3 in that stick jump was noticeable.

2.2.2.4 Directional pedals. Mechanical characteristics of the directional pedals were evaluated with the cyclic and collective controls centered. Additional control envelope measurements were taken with the collective lever full down and full up, and with cyclic full forward, aft, left, and right. Data are presented in appendix E, table VII, and appendix F, figures 9 and 13. With the collective and cyclic centered, directional pedal control travel measured 5.3 in. This travel decreased with different collective positions, and the decreased control bands were off-centered such that full down collective allowed more right pedal deflection and full up collective more left pedal deflection. Cyclic position had no apparent effect on directional control travel. Total and trim system freeplay measured 0.2 in. and 0.3 in.,

respectively. Breakout force (plus friction) was 3.0 lb left and right. Average force gradients over the first inch of travel were 15.75 and 15.5 lb/in. to the left and right, respectively. Overall average force gradients were 11.7 and 13.0 lb/in. to the left and right, respectively. Limit forces were 30.0 and 37.0 lb to the left and right. Control centering was positive and control dynamics were deadbeat when released from either side of trim. Control jump measured 0.3 in. for both left and right pedal displacement. The mechanical characteristics of the directional pedals met the requirements of the specification against which they were tested, with the exception of paragraph 3.2.3 in that control jump was noticeable, and paragraph 3.3.12 in that pedal limit forces were 100% and 147% greater than required for left and right pedal, respectively.

2.2.2.5 Collective. Mechanical characteristics of the collective were evaluated with the cyclic and directional pedals centered. Additional control envelope measurements were taken with the pedals full left and full right, and with cyclic full forward, aft, left, and right. Data are presented in appendix E, table VII, and appendix F, figure 10. Collective control travel measured 8.8 in. with the cyclic and pedals centered and remained constant regardless of cyclic or pedal position. Total and trim

system freeplay each measured 0.1 in. Breakout force (plus friction) was 2.25 lb upward and 2.0 lb down. Control centering was absolute and control dynamics were deadbeat when released from either side of trim. Control jump measured 0.6 in. for both up and down collective displacement. The mechanical characteristics of the collective control met the requirements of the specification against which they were tested, with the exception of paragraph 3.2.3 in that control jump was noticeable.

2.2.2.6 Cyclic beeper trim. The cyclic beeper trim was evaluated for authority, rate, and lag with the collective and directional pedals centered. Data are presented in appendix E, table VII. The trim envelope was essentially square, with a total authority of 9.1 in. longitudinally and 9.0 in. laterally. Trim rate was 0.57 and 0.51 in./sec in the longitudinal and lateral axes, respectively. When applying cyclic trim changes in a single direction, trim system response was considered predictable with no noticeable lag exhibited. However, a hysteresis of 0.10 sec longitudinally and 0.15 sec laterally was apparent upon reversal of trim direction. The beeper trim system met the requirements of the specification against which they were tested.

2.2.3 Forward Flight

2.2.3.1 Trimmed flight control positions. Trimmed flight control positions were evaluated concurrent with performance tests during level flight, climbs and descents, and steady level turns. Data are presented in appendix F, figures 14 through 16. During level flight, longitudinal cyclic variation was essentially linear, translating forward with increasing airspeed through 1.5 in. (16%) of control travel. Control coupling was noticeable in that the lateral cyclic progressed left with increasing airspeed, but this movement had no discontinuities and was not considered uncomfortable by the pilot. Directional pedals were displaced to the left below 80 KIAS but remained centered at higher airspeeds. Margins for all controls were adequate throughout the airspeed band. When operating in excess of 70 KIAS, maintaining airspeed ± 2 kt was easy, requiring $\pm 1/4$ in. longitudinal cyclic inputs once every 4 to 5 sec (HQR-2). At airspeeds below 70 KIAS, the automatic stabilator system, as described in appendix A, paragraph 4.2, is designed to maintain a level cabin attitude. Below 50 kt the autopilot, described in appendix A, paragraph 4.4.5, converts to a pitch attitude hold mode. The combination of these effects resulted in greater difficulty when setting and maintaining a constant airspeed in this

regime, requiring $\pm 3/8$ in. longitudinal inputs every 2 sec (HQR-5). During the 80 kt climbs and descents, both longitudinal and lateral cyclic remained essentially constant while the directional pedals progressed left with increasing torque. Control margins remained adequate during climbing left turns and descending right turns at 30 deg angle of bank. Maintaining airspeed ± 3 kt and heading ± 5 deg during the 80 kt climbs and descents was relatively easy, requiring $\pm 1/4$ in. longitudinal cyclic inputs every 3 sec, negligible lateral cyclic, and $\pm 1/8$ in. pedal inputs every 4 sec (HQR-3). During steady level turns at 80 KIAS, lateral cyclic movement was small but predictable, while longitudinal cyclic remained essentially constant. During right turns at 140 KIAS, the lateral control exhibited a slight reversal by 45 deg angle of bank, but this was not considered uncomfortable by the pilot. Margins for all controls were adequate throughout the tested range. The ASW pilot will have adequate control margins for climb on course and cruise flight during transition between a parent ship and a detection zone. Within the scope of this test, the trimmed flight control position characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The trimmed flight control position characteristics met the requirements of the specification against which they were tested.

2.2.3.2 Static longitudinal stability. Static longitudinal stability characteristics were evaluated about trim airspeeds of 80 and 140 KIAS. Data are presented in appendix F, figure 17, with longitudinal force versus displacement values cross-plotted from appendix F, figure 11. The aircraft exhibited positive static stability about both airspeeds as indicated by the requirement for increasing forward cyclic (push force) with increasing airspeed and increasing aft cyclic (pull force) with decreasing airspeed. Speed trim bands of 7.0 and 3.0 kt were observed at 80 and 140 KIAS, respectively. The longitudinal control position versus airspeed gradients were shallow about both trim airspeeds, but breakout (plus friction) forces of 1.25 lb forward and 1.0 lb aft (paragraph 2.2.2.2) provided the pilot with adequate cues to off trim conditions. During transitions to detection zones and during loiter or search operations, the strong longitudinal cyclic forces will provide adequate airspeed cueing and thus enable the pilot to divert his attention outside the cockpit. Within the scope of this test, the strong force cues provided by the static longitudinal stability characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The static longitudinal stability characteristics met the requirements of the specification against which they were tested.

2.2.3.3 Dynamic longitudinal stability.

2.2.3.3.a Short term response. Short term longitudinal dynamic response was evaluated at 80 KIAS. Aircraft natural frequency was approximately 0.5 Hz, as determined by a cyclic frequency sweep. The short term was excited by a 1.0 in. longitudinal cyclic doublet introduced at a rate approximately equal to that of the natural frequency. Short term response was heavily damped as indicated by one small overshoot. This allowed pitch attitude changes of 5 ± 2 deg to be made with only $\pm 1/4$ in. longitudinal cyclic corrections every 3 sec (HQR-3). The heavily damped short term response characteristics will allow small, precise pitch attitude changes during IMC approaches to the ship. This will reduce overall pilot workload and permit him to perform alternate mission tasks as necessary. Within the scope of this test, the heavily damped short term response characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The short term response characteristics met the requirements of the specification against which they were tested.

2.2.3.3.b Long term response. Long term longitudinal dynamic response characteristics were evaluated from a level flight airspeed of 80 KIAS. Responses were observed following cyclic control releases with all AFCS components on; autopilot off; and autopilot and SAS off. With all AFCS modes operational, the aircraft long term mode was

not naturally excitable and converged aperiodically when artificially excited by release from an off trim condition. With autopilot off, the aircraft was still not naturally excitable and was oscillatory convergent about an airspeed of 70 KIAS after being artificially disturbed. With autopilot and SAS off, the aircraft was naturally excitable but remained oscillatory about 86 ± 4 KIAS, exhibiting a period of approximately 20 sec with no apparent tendency to diverge. This oscillation was relatively easy to suppress within ± 3 deg pitch attitude, requiring $\pm 1/4$ in. longitudinal cyclic inputs every 2 to 3 sec (HQR-4). During operations at loiter airspeeds with the AFCS operational, the convergent long term response characteristics will enable the pilot to release the cyclic control for extended periods of time, thus enabling him to perform alternate tasks and improving his overall mission effectiveness. In the event of a dual AFCS failure, the convergent long term characteristics will be adequate for the pilot to return to the ship and avoid ditching the aircraft. Within the scope of this test, the convergent long term longitudinal dynamic response characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The long term dynamic response characteristics met the requirements of the specification against which they were tested.

2.2.3.4 Maneuvering stability. Maneuvering stability was evaluated during steady turns, pull ups, and push overs at 140 KIAS. Data are presented in appendix F, figures 18 and 19. During steady turns the aircraft exhibited positive maneuvering stability as indicated by the requirement for aft cyclic displacement with increasing load factor. Longitudinal control position gradients were shallow, measuring only 0.5 and 1.0 in/g during left and right turns, respectively. However, the increasing longitudinal cyclic forces were considered adequate cueing to increasing load factor. Maintaining bank angle ± 5 deg and airspeed ± 2 kt was relatively easy, requiring $\pm 3/8$ in. lateral cyclic every 3 to 4 sec and $\pm 1/4$ in. longitudinal cyclic every 4 to 5 sec (HQR-3). The pull ups and push overs indicated positive maneuvering stability by requiring increased aft cyclic (pull force) to increase load factor and increased forward cyclic (push force) to decrease load factor. Steady state "g" rates were obtained for aft control inputs up to 2.0 in. and forward control inputs up to 1.5 in. As with steady turns, the cyclic forces during pull ups and push overs were adequate cues to increasing load factor. During the ASST mission, the pilot will perform high airspeed cruise at low altitudes to avoid enemy radar detection, initiating rapid pull ups to establish a target acquisition platform and then rapid push overs to return to safe cruise height.

The tactile cues provided by the positive maneuvering stability will allow the pilot to remain aggressive while performing low altitude, high airspeed flight. Within the scope of this test, the positive maneuvering stability characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The maneuvering stability characteristics met the requirements of the specification against which they were tested.

2.2.3.5 Static lateral-directional stability.

2.2.3.5.a General. Static lateral-directional stability was evaluated at trim airspeeds of 80 and 140 KIAS using steady heading sideslips, cyclic only turns, and pedal only turns. Data are presented in appendix F, figure 20. Longitudinal coupling was evidenced by the cyclic variation required to maintain airspeed during steady heading sideslips. This was more apparent during left sideslip, but pitch attitude excursions were not considered significant or uncomfortable to the pilot.

2.2.3.5.b Lateral static stability (effective dihedral). The aircraft exhibited positive lateral static stability at both test airspeeds as indicated by the requirement for increasing left lateral cyclic with increasing left sideslip and increasing right lateral cyclic with increasing right sideslip. Lateral stick position versus sideslip gradients were consistent for left and right

sideslips, requiring displacement and forces adequate to cue an off trim condition. However, effective dihedral was weak, as indicated during pedal only turns in either direction where trim displacements of 1 ball width generated little roll rate and resulted in a flat turn. The weak lateral static stability characteristics will force the pilot to make small lateral cyclic inputs during level flight in order to prevent off trim flight conditions which could result in Doppler navigation system inaccuracies. This will be an annoying inconvenience which will distract his attention from alternate tasks. The weak lateral static stability characteristics of the SH-60B helicopter during trimmed forward flight are a Part III deficiency which should be avoided in future designs. The effective dihedral characteristics met the requirements of the specification against which they were tested.

2.2.3.5.c Directional static stability. The aircraft exhibited positive directional stability as indicated by the requirement for increasing right pedal displacement with increasing left sideslip and increasing left pedal displacement with increasing right sideslip. Pedal position versus sideslip gradients were essentially equal for left and right sideslips, and at both airspeeds the pedal displacement and resultant force requirements provided adequate cues to an off trim condition. Strong

directional stability in both directions was apparent when transient sideslip returned to the inherent sideslip value almost immediately after initiating cyclic only turns to 20 deg angle of bank. Slight adverse yaw was apparent during rapid cyclic inputs in either direction regardless of airspeed, but was not considered uncomfortable by the pilot. During a ground controlled approach (GCA), maintaining a coordinated ($\pm 1/8$ ball width) descending turn at 30 ± 2 deg angle of bank was easy, requiring $\pm 1/8$ in. lateral cyclic inputs every 4 to 5 sec and negligible pedal inputs (HQR-2). The strong directional stability characteristics apparent at dash and loiter airspeeds will allow coordinated turns without the necessity for large pedal inputs while updating tactical plots atop a sonobuoy. This will reduce pilot workload and thus improve his overall mission effectiveness. Within the scope of this test, the positive directional stability characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The directional stability and adverse yaw characteristics met the requirements of the specification against which they were tested.

2.2.3.5.d Sideforce. The aircraft exhibited positive sideforce as indicated by the requirement for increasing right bank angle with increasing right sideslip and increasing left bank angle with increasing left sideslip.

During flight at both test airspeeds, sideslip in either direction generated approximately 4 deg of bank angle change with 5 deg of sideslip, resulting in consistent proprioceptive cues that were immediately apparent to the pilot. During target search operations at loiter airspeeds, the strong sideforce cues around trim will allow the ASW pilot to maintain coordinated flight without constantly monitoring the trim ball, thus facilitating his task and improving his overall mission effectiveness. Within the scope of this test, the strong sideforce characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.2.3.6 Dynamic lateral-directional stability.

2.2.3.6.a Lateral-directional oscillations.

Lateral-directional oscillation (LDO) characteristics were evaluated at 80 and 140 KIAS. The aircraft was flown controls free in light turbulence and exhibited no tendency to excite in the lateral-directional mode. When artificially excited, aircraft response was deadbeat with no visible overshoots in the roll or yaw axes. The heavily damped LDO mode, in conjunction with convergent long term longitudinal dynamic response characteristics (paragraph 2.2.3.3.b), will enable the pilot to release the cyclic in order to perform alternate tasks, thereby increasing his overall mission effectiveness. Within the

scope of this test, the heavily damped LDO characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The LDO characteristics met the requirements of the specification against which they were tested.

2.2.3.6.b Spiral stability. Spiral stability characteristics were evaluated at 80 and 140 KIAS following release from cyclic initiated bank angles of 20 deg left and right. The aircraft exhibited a convergent spiral mode in that roll attitude returned to zero upon return of the lateral cyclic to trim regardless of airspeed or bank angle direction. At 80 KIAS, the time to half amplitude was approximately 1.4 sec. This decreased to less than 1.0 sec at 140 KIAS. The convergent spiral mode will enable the pilot to easily return to a wings level attitude should he become disoriented at night or under IMC operations. Within the scope of this test, the convergent spiral stability characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.2.3.7 Longitudinal control response. Longitudinal control response characteristics were evaluated at 140 KIAS. Single axis control inputs of 0.5 in. increment were entered from 0.5 to 1.5 in. forward and aft. No time delays were noticed and a time to steady state of approximately 1.0 sec remained essentially constant

throughout the range of step inputs. High control sensitivity was apparent in that small cyclic inputs resulted in high pitch rates forward and aft. Control response was highly predictable in that increased input size resulted in increased pitch rates. During the ASW mission, the sensitive and predictable longitudinal control response characteristics will enable the pilot to remain aggressive during torpedo delivery against hostile targets. Within the scope of this test, the sensitive and predictable longitudinal control response characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The longitudinal control response characteristics met the requirements of the specification against which they were tested.

2.2.3.8 Lateral control response. Lateral control response characteristics were evaluated at 140 KIAS. Single axis control inputs of 0.5 in. increment were entered from 0.5 to 1.5 in. left and right. No time delays were noticed and a time to steady state of approximately 1.0 sec remained essentially constant throughout the range of step inputs. High control sensitivity was apparent in that small cyclic inputs resulted in large roll rates left and right. Control response was highly predictable in that increased input size resulted in increased roll rates. During the ASST

mission, the sensitive and predictable lateral control response characteristics will enable the pilot to aggressively turn the aircraft while operating at low altitudes in order to avoid detection by enemy radar. Within the scope of this test, the sensitive and predictable lateral control response characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The lateral control response characteristics met the requirements of the specification against which they were tested.

2.2.3.9 Gust response. Gust response characteristics were evaluated at trim airspeeds of 80 and 140 KIAS. The aircraft was flown "open loop" and remained undisturbed through light turbulence. When artificially excited by collective and pedal pulses, aircraft lateral and vertical accelerations were heavily damped and showed no tendency to excite the dynamic modes. During "closed loop" flight, maintaining airspeed ± 2 kt was easy, requiring $\pm 1/8$ in. longitudinal cyclic inputs every 3 sec (HQR-3). Maintaining wings level attitude ± 2 deg and trim ball position $\pm 1/8$ ball width was also easy, requiring $\pm 1/4$ in. lateral cyclic and directional pedal inputs every 3 to 4 sec (HQR-3). Ride quality remained adequate throughout the closed loop portion of the test. The heavily damped gust response characteristics will permit flight through

mild turbulence with adequate ride quality and minimal pilot workload. Within the scope of this test, the heavily damped gust response characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.2.4 Instrument Flight Capabilities

Instrument flight capabilities were evaluated under simulated IMC while conducting a GCA. Pilot inputs to the cyclic control and directional pedals were minimal throughout the approach. The static longitudinal stability characteristics (paragraph 2.2.3.2) provided cyclic force cues adequate for maintaining constant airspeed, while the longitudinal beeper trim characteristics (paragraph 2.2.2.6) enabled rapid speed changes when necessary. The heavily damped LDO characteristics (paragraph 2.2.3.6.a) minimized any roll or yaw excursions caused by natural excitations and the convergent spiral mode (paragraph 2.2.3.6.b) enabled the pilot to easily return to a wings level attitude following directional changes. The combination of flying qualities outlined above will provide the ASW pilot with instrument flight capabilities adequate for night or IMC approaches to the ship. Within the scope of this test, the instrument flight capabilities of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.2.5 Low Airspeed

2.2.5.1 General. Low airspeed flying qualities were evaluated under the conditions presented in appendix E, table II. Data are presented in appendix F, figures 21 through 26. In ground effect (IGE) tests were flown at 15 ft above ground level and velocities up to 30 knots true airspeed (KTAS) in the applicable direction. The ship system Doppler radar set and horizontal situation indicator (HSI) were utilized for ground speed and directional determination, respectively. Out of ground effect (OGE) tests were flown at 500 ft above ground level using outside references to control drift. All altitudes were measured via the ship system radar altimeter (RADALT). Pitch and roll attitudes were measured from the ATO attitude gyro, set to 0 deg pitch and roll while parked on a known level surface. Ambient winds during low airspeed tests were calm.

2.2.5.2 Trimmed flight control positions. Low airspeed trimmed flight control positions were evaluated in 10 kt increments from 0 to 30 KTAS forward, rearward, left, and right. Data are presented in appendix F, figures 21 and 22. Migrations of longitudinal control position with forward and aft translation and lateral control position with left and right translation were essentially linear. A minimum directional pedal control margin

of 1.1 in. (21%) left pedal remaining was noted at 20 and 30 KTAS right translation. All remaining control margins were adequate throughout the airspeed band. Control coupling was apparent in that forward flight required increasing left lateral cyclic and right lateral flight required increasing forward cyclic. Throughout the test the horizontal stabilator system was observed to be moving the stabilator, and pitch attitude throughout the forward and aft translations remained constant at 5 deg nose up. Maintaining forward airspeed ± 2 kt and height above ground ± 2 ft was easy, requiring $\pm 1/4$ in. longitudinal inputs every 3 sec and $\pm 1/4$ in. collective inputs every 4 sec (HQR-3). Maintaining heading ± 2 deg was also easy, requiring $\pm 1/8$ in. lateral cyclic inputs every 3 to 4 sec and negligible pedal inputs (HQR-2). The trimmed flight control position characteristics will provide the pilot with adequate control margins during RAST connection in a head wind, tail wind, or crosswind of up to 30 kt. Within the scope of this test, the trimmed flight control position characteristics of the SH-60B helicopter during low airspeed forward and aft translations and low airspeed left and right translations are satisfactory for the ASW/ASST mission. The control position characteristics met the requirements of the specification against which they were tested.

2.2.5.3 Critical azimuth. A critical azimuth determination was performed at 30 KTAS using 30 deg relative azimuth increments. Data are presented in appendix F, figures 23 and 24. Longitudinal and lateral control margins remained adequate throughout the variation of wind azimuth. Roll attitude varied from 8 deg left to 4 deg right, while pitch attitude remained essentially constant throughout. A minimum left directional pedal control margin of 0.7 in. (14%) was experienced at a relative wind azimuth of 60 deg. Heading control between 240 and 300 deg azimuth was moderately difficult, requiring $\pm 1/2$ in. pedal inputs every 3 sec (HQR-4). The control position characteristics will provide the pilot with adequate control margins during RAST connection in a wind of up to 30 kt from the critical azimuth of 60 deg. Within the scope of this test, the control position characteristics of the SH-60B helicopter during low airspeed flight on any azimuth are satisfactory for the ASW/ASST mission. The control position characteristics met the requirements of the specification against which they were tested.

2.2.5.4 Static stability. Static stability characteristics were evaluated about a trim airspeed of 0 KTAS. Data are presented in appendix F, figures 25 and 26. The aircraft exhibited positive static

lateral-directional stability as indicated by the requirement for increasing left cyclic and right pedal displacements for increasing left sideward airspeed, and increasing right cyclic and left pedal displacements for increasing right sideward airspeed. During forward and aft translation, neutral static longitudinal stability was noted between 4 and 20 KTAS forward and negative stability was noted between 20 and 30 KTAS aft. However, control excursions were minimal and not considered uncomfortable by the pilot. Cyclic displacements about trim were small throughout the test band, but breakout (plus friction) forces were adequate cues to off trim conditions. The strong force cues will prevent inadvertent displacement of the cyclic while in a hover, thereby enabling the pilot to concentrate on alternate tasks both inside and out of the cockpit. Within the scope of this test, the strong force cues provided by the low airspeed static longitudinal stability characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission. The low airspeed static longitudinal stability characteristics met the requirements of the specification against which they were tested.

2.2.5.5 Dynamic stability. Dynamic stability characteristics were evaluated in a 500 ft OGE hover. The aircraft showed no tendency to excite naturally and

exhibited only small longitudinal and lateral oscillations when artificially excited by pitch and roll displacements, respectively. The oscillations were easily suppressed, requiring $\pm 1/8$ in. longitudinal cyclic every 3 sec and $\pm 1/4$ in. lateral cyclic inputs every 4 to 5 sec (HQR-3). The easily suppressed dynamic response characteristics will enable the pilot to maintain an OGE hover during initial RAST connection without the necessity for constant corrective control inputs. Within the scope of this test, the easily suppressed oscillatory dynamic response characteristics of the SH-60B helicopter during low airspeed flight are satisfactory for the ASW/ASST mission. The low airspeed dynamic stability characteristics met the requirements of the specification against which they were tested.

2.2.5.6 Control response.

2.2.5.6.a Cyclic. Cyclic control response was evaluated in a 500 ft OGE hover. Single axis control inputs of 0.5 in. increment were entered from 0.5 to 1.5 in. forward, aft, left, and right of trim. Both longitudinal and lateral control response was adequate and predictable with no tendency for the pilot to overcontrol. Steady state rates increased with expanding step inputs while time to steady state remained essentially constant at approximately 1.0 sec. High control sensitivity was

apparent in that small inputs resulted in large pitch and roll rates. A slight (0.1 sec) time lag was noticeable between control input and aircraft response in both axes, but was not considered uncomfortable by the pilot. The predictable cyclic control response characteristics will enable the pilot to make precise longitudinal and lateral movements in response to Landing Signal Officer (LSO) instructions during RAST alignment. Within the scope of this test, the predictable cyclic control response characteristics of the SH-60B helicopter during low airspeed flight are satisfactory for the ASW/ASST mission. The cyclic control response characteristics met the requirements of the specification against which they were tested.

2.2.5.6.b Directional. Directional control response was qualitatively evaluated in IGE hover pedal turns. Pedal response was highly damped, resulting in a sluggish aircraft response and a tendency for the pilot to overcontrol. Longitudinal coupling was apparent in that left yaw required forward cyclic inputs and right yaw required aft cyclic inputs. Arresting a moderate (20 deg/sec) yaw rate on heading ± 5 deg was moderately difficult, requiring four 1/2 in. pedal inputs which resulted in 3 yaw overshoots (HQR-5). The highly damped directional response characteristics will adversely affect the pilot's ability to perform precise pedal turns during

RAST hook up. The highly damped directional control response characteristics of the SH-60B helicopter during low airspeed flight are a Part III deficiency which should be avoided in future designs. The directional response characteristics met the requirements of the specification against which they were tested.

2.2.6 Trimmability

Trimmability characteristics of the collective, cyclic, and directional pedals were evaluated during applicable portions of the forward flight and low airspeed testing. All control trim release switches (see appendix A, paragraph 3.2) functioned adequately by reducing control forces upon switch activation and re-establishing force trim upon switch release. Control jump during this procedure was negligible, enabling forces to be easily trimmed to zero. The cyclic beeper trim switch was used to "fine tune" longitudinal and lateral stick position, but was unpredictable, requiring three to four inputs when altering trim direction due to system hysteresis (paragraph 2.2.2.6). The necessity to make numerous beeper trim switch inputs to set a desired longitudinal or lateral trim attitude will be an annoying inconvenience that will distract pilot attention from alternate tasks. The unpredictable beeper trim response resulting from hysteresis in the cyclic beeper trim system is a Part III

deficiency which should be avoided in future designs. The longitudinal and lateral trim systems met the requirements of the specification against which they were tested.

2.3 ENGINE ASSESSMENT

2.3.1 General

A qualitative engine assessment was conducted during ground operations and throughout all applicable flight tests under the conditions presented in appendix E, table II. Static and transient droop were evaluated quantitatively during each flight, and engine performance was assessed concurrent with applicable testing. Engine power available data were calculated using inlet recovery and fuel flow characteristics from reference (16). These data were then used to calculate aircraft performance.

2.3.2 Engine Start

Engine start characteristics were evaluated during each flight. Start procedures as outlined in reference (1) were followed. All engine starts were simple, logically sequenced, and accomplished easily by using either the auxiliary power unit (APU) or alternate engine cross bleed. Time from starter engagement to drop out was rapid, averaging 21 sec with no critical engine parameters approaching a limitation. The uncomplicated engine start procedures will facilitate aircraft shipboard operations

by decreasing overall pilot workload. Within the scope of this test, the engine start characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.3.3 Engine Accelerations

Single engine acceleration characteristics were evaluated during each flight. Repositioning the PCLs from IDLE to FLY in as little as 8.0 sec resulted in a smooth acceleration with no apparent movement of the aircraft due to transient torque loads. However, the inadequate stopping capabilities of the PCL detents (paragraph 2.1.3.2) required excessive care be taken so as not to miss the FLY position during PCL run up. During shipboard operations, the pilot will be able to accelerate the engines smoothly and quickly without causing the aircraft to slide on the deck. Within the scope of this test, the engine acceleration characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.3.4 Engine Trim Response

The engine trim system was evaluated during ground operations and during all applicable flight tests. Beep trim authority ranged from 96 to 100%. Actuator response to trim switch engagement was immediate and resulted in predictable main rotor speed (Nr) changes of approximately 1% per second of activation. The engine trim response characteristics will enable the pilot to quickly and

efficiently adjust Nr during execution of the Engine Start checklist, thus facilitating shipboard operations. Within the scope of this test, the predictable engine trim system characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.3.5 Droop Characteristics

2.3.5.1 Static droop. Static droop was evaluated during all applicable flight tests. Static droop was negligible as indicated by the lack of permanent rotor decay in response to changing power requirements. At no time during testing was adjustment of the engine trim switch required. The static droop characteristics will enable the pilot to maintain Nr without the necessity for constant adjustments of the engine trim switch. Within the scope of this test, the static droop characteristics of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.3.5.2 Transient droop. Transient droop was evaluated in forward flight and in conjunction with all applicable flight tests. Torque was increased from a minimum value (collective bottomed) to approximately 105% in steadily decreasing time increments. Collective application times between 9.0 and 3.7 sec resulted in only 1% transient Nr decay. However, as the application time decreased to

approximately 2.8 sec, a transient Nr decay of more than 10% resulted. This severe response was sudden and unpredictable in that no warning was provided during power increases of identical magnitude applied over a slightly longer time interval. During shipboard landing operations, the pilot will rapidly reduce his collective while in a low IGE hover in order to ensure maximum weight on wheels at touchdown. An LSO wave off due to a missed trap will require the pilot to rapidly increase collective to expediently clear the deck. The severe transient droop characteristics will result in the pilot reducing power to regain Nr, thereby making the aircraft susceptible to a hard landing or water impact and endangering the aircraft and crew. The severe transient droop characteristics of the SH-60B helicopter during rapid engine torque increases are a Part I deficiency which must be corrected as soon as possible.

2.4 PERFORMANCE

2.4.1 General

Aircraft performance was evaluated during hover, vertical climbs, level flight, and forward flight climbs and descents under the conditions presented in appendix E, table II. Data are presented in appendix F, figures 27 through 37. Test day data were unreferred to International Civil Aviation Organization (ICAO) standard

day (15 deg C) and hot day (35 deg C) conditions for a referred gross weight of 20,800 lb. This weight is representative of the helicopter basic weight plus crew, full fuel, two torpedoes, and 24 sonobuoys. At the time of this investigation, the test aircraft had accumulated approximately 300 operational hours. Engine performance closely approximated design specifications; however, a 5% power available margin was incorporated into the performance calculations to account for engine discrepancies throughout the fleet.

2.4.2 Hover

2.4.2.1 IGE/OGE hover determination. The OGE transition height was determined by analyzing torque required to hover at 10 ft incremental wheel heights from 10 to 72 ft. Data are presented in appendix F, figure 27. Engine torque required to hover increased with vertical distance above the ground to a height of 45 ft, above which torque remained essentially constant. The IGE/OGE hover transition height was thereby determined to be 45 ft.

2.4.2.2 IGE hover. The IGE hover performance was evaluated during a 10 ft free hover. Power required with pressure altitude was computed for referred gross weights of 17,500; 19,460; 20,800; and 21,700 lb (maximum allowable gross weight) under standard and hot day

conditions. Data are presented in appendix F, figures 28 and 29. The power required data were then used in conjunction with engine power available data (paragraph 2.3.1) to determine hover ceilings, as presented in appendix F, figure 30. At a referred gross weight of 20,800 lb with 5% power remaining, the IGE hover ceilings were 9090 and 1860 ft for standard and hot days, respectively. The IGE hover performance will be adequate for hovering at a low wheel height over the ship deck during RAST operations under hot day sea level conditions at maximum gross weight. Within the scope of this test, the IGE hover performance of the SH-60B helicopter is satisfactory for the ASW/ASST mission. The IGE hover performance met the requirements of the specification against which it was tested.

2.4.2.3 OGE hover. The OGE hover performance was evaluated during a 70 ft free hover. Power required with pressure altitude was computed for referred gross weights of 17,500; 19,460; 20,800; and 21,700 lb under standard and hot day conditions. Data are presented in appendix F, figures 31 and 32. The power required data were then used in conjunction with engine power available data (paragraph 2.3.1) to determine hover ceilings, as presented in appendix F, figure 33. At a referred gross weight of 20,800 lb with 5% power remaining, the OGE hover ceilings

were 8640 and 1000 ft for standard and hot days, respectively. However, OGE hover at maximum gross weight under hot day ambient conditions would not be possible. An aircraft encountering an emergency after takeoff at maximum gross weight under hot day conditions will be unable to sustain OGE hover necessary for initial RAST hookup. This will force the pilot to jettison stores in order to decrease gross weight for hover operations. The inability of the SH-60B helicopter to hover OGE under hot day conditions at maximum gross weight is a Part II deficiency which should be corrected as soon as practicable. The OGE hover performance failed the requirements of reference (9) in that maximum gross weight for OGE hover was 4% less than required.

2.4.3 Vertical Climb

Vertical climb performance was evaluated at heavy and light gross weights. Standard and hot day data unreferred for a gross weight of 20,800 lb are presented in appendix F, figure 34. Standard day vertical rate of climb at sea level was 590 ft/min with a 5% power margin remaining. Hot day vertical rate of climb at sea level was 220 ft/min with a 5% power margin remaining. During takeoffs from a pitching or rolling ship, the ASW pilot will attempt a rapid vertical ascent in order to safely clear the deck. The vertical rate of climb performance

at 20,800 lb under hot day conditions will be inadequate to perform such an ascent, forcing the pilot to ascend more slowly and increasing the possibility of contact with the oscillatory deck. The inadequate vertical climb performance of the SH-60B helicopter under hot day conditions at 20,800 lb is a Part II deficiency which should be corrected as soon as practicable.

2.4.4 Level Flight

2.4.4.1 General. Level flight performance was evaluated concurrent with level flight flying qualities testing. Calculated data are presented in appendix F, figures 35 and 36. Level flight performance was flown at a constant referred gross weight of 20,800 lb. Data were reduced using engine assessment fuel flow and power available data (paragraph 2.3.1) as well as indicated to calibrated airspeed corrections from reference (1).

2.4.4.2 Maximum level flight airspeed. Maximum level flight airspeed (V_h) was calculated for the standard ASW configuration (with one MK-46 torpedo) in accordance with reference (9). Data are presented in appendix F, figure 35. Maximum level flight airspeed was 135 KIAS, as determined by the intersection of the power required curve with engine maximum continuous power available data (paragraph 2.3.1). It follows that at a lesser gross

weight and without the parasite drag of the torpedo, power required will decrease and Vh will be in excess of 135 KIAS for the ASST mission. During the ASW mission, where mission radius will be within 100 miles, the 135 kt Vh will provide an adequate ingress time from the ship to a detection zone and thus expedite the initiation of search operations. During conduct of the ASST mission, the aircraft will be able to launch from a parent ship and rapidly proceed to within range of a hostile platform before this platform is capable of detecting the parent ship. Within the scope of this test, the 135 kt maximum continuous level flight airspeed of the SH-60B helicopter is satisfactory for the ASW/ASST mission.

2.4.4.3 Maximum endurance. Maximum endurance of the SH-60B was calculated for the ASW and ASST missions in accordance with reference (9). Data are presented in appendix F, figure 36. Maximum endurance airspeed was 77 KTAS as determined by the minimum fuel flow requirement at that airspeed. The character of the true airspeed versus fuel flow curve was such that airspeed deviations up to 5 kt resulted in negligible fuel flow increases. Mission radius is defined as 100 nmi and 40 nmi for the ASW and ASST missions, respectively. Following transit to a target zone, the pilot will fly at the maximum endurance airspeed in order to conserve fuel and maximize time on

station. Loitering at 77 KTAS resulted in a maximum computed endurance of 3 hr and 50 min for the ASW mission and 4 hr and 35 min for the ASST mission. This was sufficient to complete the respective mission profiles as outlined in appendix E, table I. When operating at the maximum endurance airspeed of 77 kt, the pilot will have adequate endurance to complete his assigned mission. Within scope of this test, the maximum endurance of the SH-60B helicopter is satisfactory for the ASW/ASST mission. Maximum endurance met the requirement of the specification against which it was tested.

2.4.5 Forward Flight Climb and Descent

2.4.5.1 Maximum rate of climb. Test day maximum rate of climb was evaluated during climbs utilizing the 30 min power setting. Data are presented in appendix F, figure 37. Maximum rate of climb under test day conditions was approximately 2380 ft/min at an airspeed of 72 KIAS. The character of the curve was such that airspeed deviations up to 5 kt resulted in only a slight reduction in rate of climb. During transition from the ship to a detection zone, the forward flight climb performance characteristics will enable the pilot to rapidly ascend to cruise altitude without having to constantly monitor airspeed, thereby expediting the ASW mission. Within the scope of this test, the high forward

flight rate of climb obtainable in the SH-60B helicopter is satisfactory for the ASW/ASST mission.

2.4.5.2 Minimum rate of descent. Test day power off minimum rate of descent was evaluated during descents utilizing a minimum collective setting. Data are presented in appendix F, figure 37. Minimum rate of descent under test day conditions was approximately 1700 ft/min at an airspeed of 65 KIAS. The character of the curve was such that airspeed deviations up to 10 kt resulted in only a slight increase in rate of descent. During conduct of an autorotation, the 1700 ft/min descent will be a comfortable rate for the pilot, as will the relatively slow airspeed which generates the same. The combination of low rate of descent and low airspeed will provide the pilot and aircrew with a maximum amount of time to prepare for entry into the water and will help the pilot establish a 0 kt ground speed touchdown. Within the scope of this test, the minimum rate of descent obtainable in the SH-60B helicopter is satisfactory for the ASW/ASST mission.

2.5 DEGRADED MODES OF OPERATION

2.5.1 Stabilator Failure

Aircraft handling qualities during automatic mode and fixed position stabilator failures were evaluated

under the conditions presented in appendix E, table II. Failure of the stabilator automatic mode was insignificant in that the system could be easily overridden by the pilot via the MAN SLEW control (see appendix A, paragraph 4.2). Since the automatic mode is designed to position the stabilator in the up position at higher airspeeds, flying qualities with the stabilator fixed in this position were considered normal in the high speed regime. During deceleration and transition to hover, the aircraft pitch attitude increased significantly, but forward longitudinal control margin was adequate and the condition was considered tolerable to the pilot. Upon returning to the ship following a fixed position stabilator failure, the pilot will be able to safely land the aircraft with a tolerable pilot workload. Within the scope of this test, the handling qualities of the SH-60B helicopter during a fixed position stabilator failure are satisfactory for the ASW/ASST mission.

2.5.2 SAS Off Flight

Aircraft handling qualities during SAS and TRIM off flight were evaluated under the conditions presented in appendix E, table II. Continued flight was possible for extended periods of time with only a minimal pilot induced oscillation (PIO) tendency resulting from the degraded mode. Maintaining level flight was relatively easy,

requiring 3/8 in. longitudinal and 1/4 in. lateral cyclic inputs every 3 sec (HQR-4). Slight increases in task difficulty increased pilot workload proportionally. Upon returning to the ship following a SAS failure, the SAS off handling characteristics will enable the pilot to safely land the aircraft with a tolerable pilot workload. Within the scope of this test, the SAS off handling qualities of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.5.3 Boost Off Flight

Aircraft handling qualities during boost off flight were evaluated under the conditions presented in appendix E, table II. With boost off, hydraulic system pressure is diverted from the pilot-assist servos, as described in appendix A, paragraph 3.3. Due to degraded mechanical mixing (see appendix A, paragraph 3.4), collective inputs resulted in aircraft pitch, roll and yawing moments. However, while collective and pedal forces increased significantly with boost off, increases in cyclic forces were minimal. This imbalance in control force requirements was unpredictable and very uncomfortable to the pilot. Maintaining hover height ± 5 ft, drift ± 10 ft, and heading ± 5 deg was exceedingly difficult, requiring large (1.0 in. or greater) inputs to all controls every 1 to 2 sec (HQR-7). In the event of a

hydraulic system leak in the pilot-assist manifold, the pilot will deselect boost in order to preserve hydraulic fluid. Upon returning to the ship with boost off, the unpredictable handling qualities will result in a hard landing and possible aircraft damage. The dissimilar and unpredictable handling qualities of the SH-60B helicopter during boost off flight are a Part II deficiency which should be corrected as soon as practicable.

2.5.4 Single Engine Operation

Single engine performance capabilities were evaluated under the conditions presented in appendix E, table II. With a single engine inoperative, level flight at the test day gross weight of 18,000 lb was easily maintained with no noticeable differences in handling qualities. Although the aircraft was unable to hover OGE, the test day IGE hover capability facilitated aircraft landing. Following an engine failure, the single engine capabilities will enable the pilot to maintain level flight, return to the ship, and safely land the aircraft after dumping fuel as necessary to decrease gross weight. Within the scope of this test, the single engine capabilities of the SH-60B helicopter are satisfactory for the ASW/ASST mission.

2.6 AUTOMATIC FLIGHT CONTROL SYSTEM

2.6.1 Altitude Hold

The barometric and radar altitude hold features were evaluated for accuracy and reliability under the conditions presented in appendix E, table II. The RADALT hold, as described in appendix A, paragraph 4.4.6, was activated at 500 ft/130 KIAS. During aggressive maneuvering turns of up to 30 deg angle of bank, the system maintained constant altitude ± 25 ft. Applying a large forward cyclic input to increase airspeed 40 kt resulted in a transient altitude loss of approximately 70 ft, which was regained in approximately 15 sec. The RADALT hold feature will provide safe and accurate altitude hold during IMC operations. When operating for extended periods on station, this will significantly reduce pilot workload and greatly increase flight safety for the aircraft and crew. The safe and accurate operational characteristics of the SH-60B RADALT hold feature during low altitude maneuvering flight are an enhancing characteristic which should be incorporated in future designs.

2.6.2 Automatic Approach Coupler

The automatic approach coupler was evaluated for accuracy and reliability during an automatic approach to hover under the conditions presented in appendix E,

table II. The automatic approach coupler, as described in appendix A, paragraph 4.4.7, was activated at a radar altitude of 150 ft and an airspeed of 60 KIAS. The automatic sequence initiated deceleration and descent almost immediately, converting aircraft control to the automatic hover mode upon reaching a radar altitude of 40 ft at 0 KIAS. No pilot corrections were required during the automatic approach and time from system activation to steady hover was approximately 55 sec. The automatic approach coupler will perform a coordinated approach to a stable hover for survivor pickup during SAR operations, thereby decreasing overall pilot workload and enabling him to perform alternate tasks. Within the scope of this test, the automatic approach coupler of the SH-60B helicopter is satisfactory for the ASW/ASST mission.

2.6.3 Automatic Hover

The automatic hover mode was evaluated for attitude and altitude hold under the conditions presented in appendix E, table II. The automatic hover mode, as described in appendix A, paragraph 4.4.8, was activated manually once established in a steady IGE hover. When disturbed by a lateral wind gust of 5 to 10 kt, aircraft heading altered approximately 10 to 15 deg. Attempts by the AFCS to correct the heading deviation resulted in an oscillatory response in the yaw axis. If left unattended

this response diverged at an uncomfortable rate, reaching double amplitude in less than 8.0 sec. While hovering during RAST connection, the pilot will engage the heading hold in order to perform alternate tasks. Slight wind gusts will excite the oscillatory divergent response of the heading hold, and the sluggish response characteristics of the directional controls during low airspeed flight (paragraph 2.2.5.6.b) will adversely affect the pilot's ability to arrest the divergence. If yaw rates continue to build, directional pedal control limits will be reached and the aircraft will be uncontrollable, resulting in the loss of aircraft and crew. The oscillatory divergent yaw response characteristics of the SH-60B automatic hover heading hold are a Part I deficiency which must be corrected as soon as possible.

2.6.4 Blade Fold

Blade fold characteristics were evaluated for speed and reliability at the completion of the final test flight. The blade fold system, as described in appendix A, paragraph 5.1, incorporates the EFCS to move the blades to a "null" position prior to folding. During the fold test the EFCS searched for the null position for almost 3.0 min before the operation was aborted. Even when functioning properly the operation is slow, taking up

to 4.0 min for completion during which time the flight crew must remain in the aircraft to monitor the operation. Upon landing aboard ship in adverse weather conditions, the pilot will initiate the blade fold sequence so that the aircraft can be hangared. The long period of time required for blade fold will require the aircraft to remain on the pitching and rolling deck, thereby subjecting the aircraft, aircrew, and ground crew to potentially hazardous situations. The slow operating time of the SH-60B automatic position and blade fold system during blade fold operations is a Part II deficiency which should be corrected as soon as practicable.

3. CONCLUSIONS AND RECOMMENDATIONS

3.1 GENERAL

Within the scope of this test, the SH-60B helicopter demonstrates excellent potential for the ASW/ASST mission and will be satisfactory upon correction of the Part I deficiencies. Due to the limited scope of this evaluation and the lack of specialized flight test instrumentation, further testing is recommended under a more extensive range of flight conditions and with a full complement of flying qualities and performance instrumentation.

3.2 ENHANCING CHARACTERISTICS

The safe and accurate operation of the RADALT hold feature during low altitude maneuvering flight (paragraph 2.6.1) was an enhancing characteristic which should be incorporated in future designs.

3.3 DEFICIENCIES

3.3.1 Part I Deficiencies

3.3.1.1 General. This investigation revealed two Part I deficiencies which should be corrected as soon as possible as they adversely affect the safety of the aircraft and crew.

3.3.1.2 Severe transient droop during rapid engine torque increase. The magnitude of transient droop (paragraph 2.3.5.2) depends on the following factors: amount of change in rotor loading, rate of load change, Nr governing, basic engine response, and rotor inertia. In order to reduce transient droop to an acceptable level, one or more of these factors must be changed. Since the amount and rate of rotor load change depend upon the perceived requirements in a particular operational situation, they should be free from engineering control. For example, a collective rate limiting device such as a hydraulic damper could be incorporated to slow collective application. This would almost certainly eliminate the chance of transient droop, but would cause a far more dangerous situation by depriving the pilot of aircraft power when it is most needed. The basic response of the engine, specifically the rate of change of free power turbine speed with changing fuel flow, is inherent to the design of the engine. Altering this response would require the selection of an alternate engine, which is not considered feasible or cost effective since the majority of all other performance characteristics are considered acceptable. Similarly, changing the inertia of the main rotor system in order to decrease transient droop would require extensive aircraft design changes. This is unacceptable due to the enormous costs as well as the

potentially negative changes in flying qualities which could result. The most simple and cost effective method of countering the severe transient droop problem is to modify the Nr governor. Satisfactory static droop characteristics (paragraph 2.3.5.1) indicate that the Nr governor gain (fuel flow rate per unit speed change) is adequately defined. Therefore, the current Nr governor would either have to be modified with a mechanical load anticipator or replaced with a more advanced electronic digital fuel control system. A mechanical anticipator is designed to momentarily increase fuel flow as the collective is raised, effectively steepening the slope of the governor gain curve. As the anticipator induced effects are subsiding, the governor will have "caught up" with the required fuel flow. But although this system is relatively simple to design and easy to incorporate, it is very difficult to set correctly for all situations. The preferred method of improving Nr governing is an electronic digital fuel control system. While such a system is more expensive than a mechanical unit, the electronic unit has far greater flexibility in providing load anticipation for a wide variety of situations. In addition, it is easy to remove, install, and maintain. For this reason, it is recommended that the manufacturer design or procure an electronic digital fuel control system for retrofit into the existing SH-60B helicopter as

soon as possible and for incorporation into any future SH-60B aircraft prior to Navy acceptance.

3.3.1.3 Oscillatory divergent yaw response of the automatic hover heading hold. Solution of this deficiency (paragraph 2.6.3) could be accomplished by either a hardware or software modification. A hardware alteration, such as a decrease in size of the vertical tail area, would make the aircraft less directionally stable and therefore less susceptible to lateral wind gusts. However, such a major modification is not economically feasible since it would require significant aeronautical and structural redesign and re-qualification. In addition, such a change would almost certainly degrade the forward flight flying qualities of the aircraft and would not be worth the trade off. For this reason, the practical solution to this problem lies in software modification. The most desirable solution, and the one that is recommended, is for the manufacturer to increase the response time of the Digital Stability Augmentation System by providing faster yaw rate sensing and/or quicker computer analysis of rate information. This will enable the system to detect, arrest, and correct yaw excursions before they have time to saturate the AFCS and get out of control. However, if such a software improvement is not possible or is cost prohibitive, the alternate

recommendation would be for the manufacturer to program the AFCS to disable the heading hold portion of the automatic hover mode prior to driving the aircraft oscillatory divergent. This could be accomplished by incorporating a software code designed to disable heading hold once a pre-determined yaw rate has been reached. In this case the pilot should also be provided with a visual indication (i.e. warning light) alerting him to the degraded condition of the automatic hover function.

3.3.2 Part II Deficiencies

3.3.2.1 General. This investigation identified a total of 8 Part II deficiencies which should be corrected as soon as practicable in order to significantly improve the effectiveness and maintainability of the aircraft.

3.3.2.2 Restricted cockpit field of view during right turns and low airspeed flight. This particular deficiency (paragraph 2.1.2) is common to many military rotary wing aircraft, due in part to large instrument panels, but also to the necessity of strong structural supports and ballistic protection around the pilot and copilot seating areas. The structural supports and seat armor are considered essential for mission safety and are therefore not candidates for redesign. Decreasing the size of the instrument panel or glare shield is one obvious solution

to the field of view problem, but it too is unrealistic. Reducing the dimensions of the instrument panel would require removal or down sizing of some flight or mission instruments, neither of which is acceptable. The SH-60B instrument panel already incorporates a multi-purpose display which significantly reduces the number of cockpit instruments. All remaining instruments are considered essential for safe operation of the aircraft and/or successful conduct of the mission. Therefore, the only feasible method of improving cockpit field of view is to reduce the high nose up pitch attitude in the hover and low airspeed flight regime. This could be accomplished by altering the loading of the aircraft (i.e. shifting the CG forward), but would result in an uncomfortable nose down attitude when cruising at high forward flight airspeeds. However, a great deal of the nose up attitude during hover and low airspeed flight is already negated by the moveable stabilator system, described in appendix A, paragraph 4.2. In the low airspeed flight regime the stabilator is positioned at 40 deg trailing edge down to minimize the effect of rotor downwash on the stabilator surface. This in turn decreases the nose up pitching moment provided by the stabilator, resulting in a more comfortable cabin attitude. For this reason, it is recommended that the manufacturer increase the total travel of the stabilator to 45 or 50 deg trailing edge down in order to maximize

the effect of its design. This could be accomplished by increasing stabilator actuator travel and repositioning the stabilator down stops.

3.3.2.3 Inadequate stopping capability of the engine control quadrant PCL detents during PCL run up. This deficiency (paragraph 2.1.3.2) can be eradicated by incorporating mechanical improvements into the PCL hardware. This would best be accomplished by rounding the forward edge of the PCL detents such that the spring stop can more reliably "find" the detent and lock the levers during rapid movement. In addition, the spring force of the spring stops could be increased so as to insure the movement of the stop into the detent. Care should be taken not to increase the spring force to the extent that the operator has difficulty releasing the PCL from the detent. Recommend manufacturer incorporate changes as outlined above.

3.3.2.4. Exposure of the center avionics/electronics console to dripping condensation from the ECS vents during ECS operation. This deficiency (paragraph 2.1.4) could be solved by relocating the affected avionics/electronics systems, moving or extending the ECS vents, or by installing a filter system to remove the condensed water from the incoming air. Relocation of avionics/electronics systems components is not feasible since space in the

cockpit is at a premium, and relocating any other equipment to this location would simply subject the new equipment to the original problem. Since the location of the ECS vents is considered optimal for maximum air delivery to the pilots, the vents should not be moved to an alternate location where they would be less effective. For these reasons, it is recommended that the manufacturer modify the vent ducting to include a filter or collection portal to remove any condensed water before this water can drip onto the exposed components below.

3.3.2.5 Inability to hover OGE under hot day conditions at maximum gross weight. Helicopter hover performance is determined by comparing engine power available with total power required to hover under a given set of conditions. Since all data documenting this particular deficiency (paragraph 2.4.2.3) are considered accurate, the basic solution would be to find a method of reducing power required or increasing power available.

3.3.2.5.a Power required. Total power required to hover at a specific gross weight is a function of main rotor, tail rotor, and accessory power requirements; engine installation losses; and rotor downwash effects on the fuselage. These can be further broken down to functions of disk areas, operating RPMs, number of rotor blades, and fuselage shape. With the exception of N_r , all factors are

inherent to the basic design of the aircraft and are not feasible to change. Decreasing N_r by a nominal amount (usually about 4%) will maximize rotor system efficiency and minimize rotor system power requirements, but is unacceptable due to the detrimental effect of the lower N_r on autorotational capability. In the event of an aircraft emergency, operation at a decreased N_r may result in excessive N_r decay at the initiation of an autorotation and insufficient rotor energy to safely recover the aircraft upon completion of the maneuver.

3.3.2.5.b Power available. Power available is a function of engine performance and engine inlet performance. For a given airspeed and specific atmospheric conditions, engine performance is a function of gas generator and power turbine speeds. These are inherent to the basic design of the engine, therefore changing either requires major engine modification or the incorporation of an entirely different power plant. Since the magnitude of the deficiency is not considered hazardous to the aircraft or aircrew, engine redesign or replacement is not considered economically practical. Therefore, the only realistic method of increasing aircraft power available would be to maximize the engine inlet performance characteristics. Inlet performance can be summarized as the effect of the engine inlet on free stream air flow. The inlet is comprised of the ducting, shaping, guiding, etc. between

the free stream air and the engine compressor face. Since airflow around a helicopter is very disturbed compared to the free stream air, inlets are very difficult to design for optimum performance over a wide range of operating conditions (i.e. hover, level flight, climbs and descents, etc.). In order to maximize performance, the inlet should be designed such that engine inlet pressure ratio and engine inlet temperature rise are minimized. Therefore, it is recommended that the manufacturer investigate the original engine inlet design to ensure inlet performance has been maximized for the T700-GE-401 engines. If the original design was not optimized the manufacturer should redesign as necessary. If in fact the design is deemed optimal, it is recommended that a change be incorporated into reference (1) limiting aircraft operating weight during training exercises and non-essential missions (i.e. down load fuel as necessary for the aircraft to have an OGE hover capability at the applicable atmospheric conditions).

3.3.2.6 Inadequate vertical climb performance under hot day conditions at 20,800 lb. This particular deficiency (paragraph 2.4.3) is directly related to the aircraft OGE hover performance deficiency (paragraph 2.4.2.3), in that the power which can be utilized for a vertical climb is the net difference between the OGE hover power available

and the OGE hover power required. For this reason, the analysis presented in paragraph 3.3.2.5 is valid, and the same recommendations are applicable.

3.3.2.7 Dissimilar and unpredictable handling qualities during boost off flight. This particular deficiency (paragraph 2.5.3) is predicated not on the increased collective and pedal forces during boost off flight, but rather on the imbalance of control force requirements between the collective, the pedals, and the cyclic stick. The discontinuity is due to the fact that the boost system provides no augmentation in roll and a disproportionately small amount in pitch. Since the amount of boost provided is determined by normal operational requirements, it is unfeasible to alter the amount of control boost such that the aircraft flies better following a system failure. Instead, the most logical method of dealing with this discrepancy is to guarantee system operation by providing system redundancy. Therefore, it is recommended that the manufacturer replace the current pilot-assist servos (see appendix A, paragraph 3.3) with units similar in concept to the flight control primary servos (appendix A, paragraph 3.5). With one stage pressurized by the No.1 hydraulic system and the second stage by the No.2 system, the pilot-assist servos could continue providing hydraulic control boost in the event of a primary system failure.

3.3.2.8 Slow operating time of the automatic position and blade fold system during blade fold operations. This deficiency (paragraph 2.6.4) would be best solved by increasing the rate of all portions of the blade fold sequence. As currently designed, the speed index drive unit can take as long as 42 sec to position the main rotor blades in the "null" position. Even once the blades begin to fold, the fold motors are very slow, further prolonging the process. Equally important is the incorporation of a more accurate detection capability for when the blades reach the null position. This will prevent the blades from missing the correct fold position and having to complete another full revolution. For these reasons, it is recommended that the manufacturer incorporate hardware changes as necessary to double the current rate of the speed index drive unit and blade fold motors. In addition, the manufacturer should incorporate hardware and software alterations as necessary to more accurately define the blade null position. Finally, the manufacturer should incorporate software changes to better define the null position search envelope (i.e. decrease speed of the index drive unit when within 10 vice 30 deg of null).

3.3.3 Part III Deficiencies

3.3.3.1 General. Four Part III deficiencies were identified as a result of this investigation. These

should be avoided in future designs since they are annoying inconveniences which will distract pilot attention from conduct of primary mission tasks.

3.3.3.2 Obscured labeling of the engine control quadrant PCLs. This deficiency (paragraph 2.1.3.1) can be solved by the addition of PCL position labeling (OFF, IDLE, FLY, and LOCKOUT) outboard each PCL. In this location both the pilot and ATO will be able to read the nomenclature without physically repositioning himself. However, adequate correction of the PCL detent stopping capability (paragraph 2.1.3.2) will effectively negate the requirement for improved labeling since the pilot will have adequate cueing to each PCL position. Therefore, it is recommended that the manufacturer incorporate the changes outlined in paragraph 3.3.2.3, at which time this deficiency will be cancelled.

3.3.3.3 Weak lateral static stability characteristics during trimmed forward flight. This particular deficiency (paragraph 2.2.3.5.b) documents weak effective dihedral (roll due to sideslip). Effective dihedral is affected by main rotor lateral blowback, fuselage rolling moments, tail rotor forces, and rolling moments due to the vertical stabilizer. Although the amount of effective dihedral could be strengthened by correctly altering any of these, none of these alterations is considered practical.

Changes to lateral blowback or fuselage rolling moments would require major aircraft and rotor system redesign and are therefore not viable. Tail rotor thrust alteration is also not a viable solution since any gain in strength of effective dihedral in one turn direction would be offset by a comparable reduction in strength of effective dihedral in the opposite direction. Increasing aircraft rolling moment induced by the vertical stabilizer could be accomplished by increasing the surface area of the same; however, maximizing this rolling moment would require the additional surface area to be placed as far above the aircraft CG as possible. This is unfeasible since the additional tail surface area would be directly behind the tail rotor where surface area is intentionally minimized to optimize tail rotor effectiveness. Since no cost effective, feasible solution to this problem is available, and since the deficiency is only an annoying characteristic, the best alternative is to leave the deficiency outstanding.

3.3.3.4 Highly damped directional control response characteristics during low airspeed flight. Since the yaw boost servo provides all pedal rate damping, this deficiency (paragraph 2.2.5.6.b) can most easily be solved by mechanically decreasing the amount of this damping. This will in turn decrease sluggishness of the aircraft

yaw control response. However, care must be taken so as not to decrease damping too greatly, as this will result in the aircraft being too easily disturbed by pilot inputs or atmospheric turbulence. Recommend the manufacturer decrease pedal rate damping by making appropriate mechanical adjustments to the yaw boost servo (appendix F, figure 45).

3.3.3.5 Unpredictable beeper trim response resulting from hysteresis in the cyclic beeper trim system. Since beeper trim system characteristics are predictable for consecutive applications in one direction, this deficiency (paragraph 2.2.6) can be eradicated by reducing freeplay in the mechanical segments of the system. By reducing freeplay between the trim motor, gearing, and clutch assemblies, the system will not exhibit undesirable delays in movement upon reversal of trim direction. Recommend the manufacturer conduct an engineering evaluation of the cyclic beeper trim system to determine where freeplay exists, and initiate action to mechanically reduce this freeplay.

3.4 SPECIFICATION COMPLIANCE

3.4.1 General

The test aircraft met all requirements against which it was evaluated with the exception of two

mechanical characteristics specifications and one hover performance specification.

3.4.4.1 MIL-H-8501A. The aircraft failed the requirements of paragraph 3.2.3 in that stick jump was noticeable in all cockpit flight controls; and paragraph 3.3.12 in that pedal limit forces were 100% and 147% greater than required for left and right pedal, respectively.

3.4.1.2 DCP-85. The aircraft failed the requirements of paragraph C-1-1 in that maximum gross weight for hot day OGE hover was 4% less than required.

LIST OF REFERENCES

LIST OF REFERENCES

- (1) NATOPS Flight Manual A1-H60BB-NFM-000, "Navy Model SH-60B Aircraft", of 15 March 1985 with Interim Change 22 of 16 December 1986.
- (2) Military Specification MIL-C-5011A, "Charts, Standard Aircraft Characteristics and Performance, Piloted Aircraft", of 5 November 1951.
- (3) Military Specification MIL-D-23222, "Demonstration Requirements for Rotary Wing Aircraft (Helicopters)", of 27 May 1963 with Change 12 of May 1971.
- (4) Military Specification MIL-H-8501A, "Helicopter Flying and Ground Handling Qualities; General Requirements for", of 3 April 1962.
- (5) Military Standard MIL-STD-250D, "Aircrew Station Controls and Displays for Rotary Wing Aircraft", of 28 August 1974.
- (6) Military Standard MIL-STD-850B, "Aircrew Station Vision Requirements for Military Aircraft", of November 1970.
- (7) Military Standard MIL-STD-1333A, "Aircrew Station Geometry for Military Aircraft", of 30 June 1976.
- (8) Military Standard MIL-STD-1472C, "Human Engineering Design Criteria for Military Systems, Equipment, and Facilities", of 2 May 1981.
- (9) Decision Coordinating Paper DCP-85, "Destroyer - Helicopter System (LAMPS MK III)", of 15 March 1979.
- (10) Flight Test Manual USNTPS-FTM-No. 106, "Rotary Wing Performance", of 31 December 1987.
- (11) Technical Memorandum TM-83-1, "Summary of Helicopter Performance Flight Test Techniques", undated.
- (12) Flight Test Manual USNTPS-FTM-No.105, "Helicopter Stability and Control, Theory and Flight Test Techniques" (Preliminary Edition), of November 1983.
- (13) NASA Technical Note TN-D-5153, "The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities", of April 1969.

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- (14) Rating Scale, "Subjective Vibration Assessment", Aeroplane and Armament Experimental Establishment, Boscombe Down, England, undated.
- (15) Technical Note ASNF-TN-68-3, "Flying and Ground Handling Qualities of MIL-H-8501A", of November 1968.
- (16) Technical Report USNTPS-3R-85, "Engine Assessment of the UH-60A Helicopter", of 20 September 1985.

APPENDICES

APPENDIX A

FLIGHT CONTROL DESCRIPTION

FLIGHT CONTROL DESCRIPTION

1 INTRODUCTION

The SH-60B helicopter incorporates a mechanical, hydraulically boosted, irreversible flight control system. Aircraft control is accomplished through either the pilot or the ATO controls in the side-by-side cockpit, as depicted in appendix F, figure 38. Each control set consists of conventional collective, cyclic, and tail rotor (directional) control systems. These transmit pilot inputs to the respective control surfaces via a system of push rods, bellcranks, pulleys, cables, actuators, and servos. The AFCS includes the SAS, designed to provide rate damping and short term dynamic stability; the EFCS, designed to provide long term stability and various autopilot functions; and a stabilator system, designed to augment aircraft pitch attitude as a function of airspeed. Flight control system malfunctions are indicated to the pilot by the appropriate lights on the Caution/Advisory Panel.

2 HYDRAULIC SYSTEM

The hydraulic system, as depicted schematically in appendix F, figure 39, consists of three interrelated systems designed to provide flight control hydraulic pressure (3,000 to 3,100 psi) and redundancy to the pilot-assist, primary, and tail rotor servos. The No.1 hydraulic system supplies pressure to the first stage of the primary and tail rotor servos through pump and transfer modules. Similarly, the No.2 system supplies pressure to the second stage of the primary servos and to all of the pilot-assist servos. Both pump modules are mounted on and driven by the main transmission. Pressure is monitored by a system designed to detect hydraulic leaks and isolate them by re-routing fluid flow. This is accomplished through a series of pressure switches, fluid level switches, and shut-off valves whose control is governed by one of two electronic logic modules. In the event of a pressure loss, an electrically driven backup pump is designed to supply emergency pressure to either system as necessary.

3 FLIGHT CONTROLS

3.1 General

Aircraft control is accomplished by varying the pitch of the main and tail rotor blades. Vertical control

is provided through the collective, which increases or decreases pitch of the four main rotor blades simultaneously, resulting in aircraft climb or descent. Longitudinal and lateral control is provided through the cyclic, which causes the main rotor blades to pitch in such a manner that the tip path plane is tilted in the direction of desired movement. Yaw control is provided through the directional pedals, whose movement collectively alters the pitch of the tail rotor blades and results in a yawing moment about the vertical axis of the aircraft. Component locations and a schematic block diagram of the flight control system are presented in appendix F, figures 40 and 41, respectively.

3.2 Cockpit Controls

The cockpit controls, as depicted in appendix F, figures 42 and 43, consist of dual collective levers, cyclic sticks, and directional pedals. Each collective grip, as depicted in appendix F, figure 44, incorporates switches for the search light, primary servos, contingency power, heading trim, and altitude hold/collective trim release. Each cyclic grip, also depicted in figure 44, incorporates switches for cyclic beeper trim, trim release, intercommunication system/radio transmit, hover departure, cargo release, and emergency stores release. Microswitches on the directional pedals provide disengagement for the pedal trim. The collective, cyclic, and directional pedals transmit pilot inputs to the pilot-assist servos through a series of bellcranks, control rods, and torque shafts. The controls are routed vertically up either side of the aircraft, aft and outboard of the respective pilot seats. All of the dual controls are interconnected via control rods atop the forward upper cabin such that movement of any control results in the identical movement of the corresponding control.

3.3 Pilot-Assist Servos

Outputs from the cockpit controls are transmitted to the pilot-assist servos, as depicted in appendix F, figure 45. These are designed to utilize pressure from the No.2 hydraulic system (paragraph 2) to reduce cockpit control forces and SAS feedback in the collective, pitch, and yaw axes. The yaw boost servo also provides pedal rate damping when the pedal trim system is disengaged. Boost servo valves are electrically powered and can be turned on and off via the SAS/BOOST button on the AFCS control panel. Pilot control forces will increase without

hydraulic boost but are designed to remain manageable. The pilot-assist servos also serve as a point to introduce trim system inputs and electrical signals from the AFCS. Outputs from the pilot-assist servos are transferred to a control mixing assembly.

3.4 Mixing Unit

Collective, cyclic, and directional outputs from the pilot-assist servos are transferred via control rods to the mixing unit, as depicted in appendix F, figure 45. This unit is designed to produce uncoupled airframe response characteristics by mechanically combining controls through bellcranks and levers within the unit. An electrical collective to yaw coupling is also introduced when the trim system is engaged. Respective control mixing channels are presented in appendix E, table VIII. An electromechanical pitch bias actuator (PBA) located at the mixing unit is designed to provide positive stick gradient with changes in airspeed (see paragraph 4.4.4). Control outputs from the mixing unit are transmitted to the primary servos and the tail rotor control system.

3.5 Primary Servos

Outputs from the mixing unit are transferred to the forward, aft, and lateral primary servos via the primary servo input control rods, as depicted in appendix F, figure 45. These servos are designed to provide hydraulic boost to the main rotor system flight controls. Each has two independent, redundant stages powered by the No.1 and No.2 hydraulic systems (paragraph 2) with both stages sharing a common input linkage. In the event that one stage becomes inoperative due to a pressure loss, an internal bypass valve is designed to prevent a hydraulic lock, and the malfunctioning stage can be shut down using the primary servo switch located on either of the collective grips (paragraph 3.2). Electrical interlocks prevent both servo stages from being shut down simultaneously. Primary servo outputs are transmitted to the main rotor head assembly via control rods and bellcranks.

3.6 Main Rotor System

The main rotor system, as depicted in appendix F, figure 46, transmits rotor control motions to the main rotor blades. Inputs from the primary servos tilt the swashplate assembly, which in turn moves the pitch control

rods and pitch horns to alter main rotor blade pitch. The main rotor system is fully articulated, incorporating two elastomeric bearings per blade to allow flap, lead and lag, and feather. Main rotor hydraulic dampers are designed to restrain lead and lag motion and to absorb heavy rotor head loads during blade engagement at start up. Bifilar absorbers incorporate floating weights to absorb rotor vibrations and stress. Anti-flap stops and droop stops are installed to limit blade flapping at low Nr. These restraints are designed to disengage under centrifugal force when Nr reaches 35% and 55%, respectively. An electromechanical blade fold system is located at the spindle fold hinge to permit automatic fold through four blade fold actuators and mechanical gearings.

3.7 Main Rotor Blades

The main rotor blades, as depicted in appendix F, figure 47, are constructed of a titanium spar, honeycomb core, fiberglass skin, and metal abrasion strips. Each rotor blade has a span of 23.5 ft and a chord of 20.7 in. The outer portion of the blade tips are swept 20 deg to increase rotor blade efficiency and provide sound attenuation. Trim tabs are incorporated on the outer portion of the trailing edge. The blade main spar is pressurized with nitrogen to help detect cracks via the blade inspection method (BIM) indicators, also depicted in figure 47. These color coded indicators are designed to signal a nitrogen pressure loss and thereby an unsafe operating condition. Heating mats designed to provide blade deicing are incorporated into each blade.

3.8 Tail Rotor Control System

3.8.1 Tail rotor quadrant. Directional control outputs from the mixing unit (paragraph 3.4) are transmitted to the tail rotor quadrant via redundant cables and pulleys, as depicted in appendix F, figure 48. The tail rotor quadrant, as depicted in appendix F, figure 49, is located at the tail rotor gear box. Spring cylinders on the quadrant are designed to permit full yaw control in the event of a single control cable failure and neutral tail rotor blade positioning for a "fly home" capability in the event of a dual cable failure. A tail rotor servo powered by the No.1 hydraulic system (paragraph 2) is designed to provide boost to the tail rotor pitch change beam, which in turn changes the collective pitch of the tail rotor blades. Unlike the primary control servos, only one stage of the tail rotor servo is pressurized at one time.

3.8.2 Tail rotor. The four-bladed tail rotor, also depicted in figure 49, is a bearingless, cross-beam type assembly with collectively controllable blade pitch. The tail rotor is tilted 20 deg to the vertical to extend the aft CG envelope of the aircraft and to provide additional lift in a hover. Blade construction is of graphite and fiberglass with bonded polyurethane and metal abrasion strips. Heating mats designed to provide blade deicing are incorporated into each blade.

4 AUTOMATIC FLIGHT CONTROL SYSTEM

4.1 General

The AFCS consists of three independent yet complementary subsystems; the stabilator control system, the analog SAS, and the digital EFCS. A simplified block diagram of the AFCS is presented in appendix F, figure 50. Pilot control is provided through the AFCS and stabilator control panels, depicted in appendix F, figures 51 and 52, respectively. As a complete system, the AFCS provides pitch, roll, and yaw stability augmentation; cyclic, pedal, and collective trim; attitude, airspeed, heading, and altitude hold; and turn coordination at airspeeds greater than 50 kt. Other features include longitudinal stick gradient augmentation; automatic approach, hover, and departure; hover augmentation; and a crew hover capability. All three AFCS subsystems incorporate signals from common pitch rate, airspeed, and lateral acceleration sensors. All airspeed data utilized by the AFCS are provided through the aircraft Pitot-static system, which incorporates two nose mounted Pitot tubes and two static ports mounted flush with the fuselage aft of either cockpit door.

4.2 Stabilator System

The stabilator system, as depicted schematically in appendix F, figure 53, is designed to provide aircraft pitch stability about the longitudinal axis. By altering pitch of the stabilator control surface, the system limits pitch attitude excursions with changes in airspeed and undesirable nose high attitudes caused by rotor downwash effects during hover and low airspeed flight. Major system components include stabilator amplifiers, pitch rate gyros, accelerometers, airspeed transducers, stick position sensors, and electromechanical actuators which move the stabilator control surface. The system is designed to operate automatically, with each amplifier receiving independent airspeed, pitch rate, lateral

acceleration, and collective stick position inputs from the respective sensors. These signals are compared, analyzed, and computed to determine the best stabilator position for the existing flight conditions. The appropriate extend or retract signal is transmitted to the stabilator actuators, which position the control surface between 40 deg trailing edge down for hover and low airspeed flight and 8 deg trailing edge up for cruise and maneuvering flight. Collective stick position and lateral acceleration signals are gradually faded out of the positioning logic as airspeed decreases below 60 kt and are completely disregarded below 30 kt. A manual stabilator mode is incorporated to allow pilot override in the event of a system malfunction. Stabilator control and monitoring is provided through the stabilator control panel and position indicator (figure 52). Manual control is via the manual slew (MAN SLEW) switch.

4.3 Stability Augmentation System

4.3.1 General. The SAS, as depicted schematically in appendix F, figure 54, consists of two separate, independent channels designed to provide short term, limited authority corrections to the pitch, roll, and yaw axes of the aircraft. Functions of the analog SAS (SAS 1) include rate damping for dynamic stability and turn coordination at airspeeds greater than 50 kt. The digital SAS (SAS 2) is incorporated into the EFCS (see paragraph 4.4.2) and is designed to provide the same features as SAS 1. In addition, SAS 2 provides hover augmentation and gust alleviation as well as a hover coupler (see paragraphs 4.4.2 and 4.4.8, respectively). With dual SAS engagement, each SAS has a 5% flight control movement authority at a rate of 100% per sec. With single SAS engagement, control authority remains at 5% but SAS amplifier gain is doubled to compensate for the inoperative channel.

4.3.2 Analog stability augmentation system. Incorporated into SAS 1 are rate, acceleration, and airspeed signals from the No.1 stabilator system, and attitude signals from the ATO attitude heading reference system (AHRS). In addition to parameter sensors, major SAS 1 components include amplifiers, servovalves, and hydraulic actuators (figure 54). Changes in pitch rate are sensed by the No.1 pitch rate gyro, filtered by the No.1 stabilator amplifier, and applied to the SAS amplifier. Long term pitch rate signals are removed and short term corrective signals are transmitted to the pitch SAS servovalve. Changes in roll rate are sensed by the AHRS and applied to

the SAS amplifier. The inputs are combined with signals from an internal roll rate gyro and the resultant corrective signal is transmitted to the roll SAS servovalve. At airspeeds less than 50 kt, yaw rates are sensed by the SAS amplifier internal yaw rate gyro. Long term yaw rate signals are removed and short term corrective signals are transmitted to the yaw SAS servovalve. At airspeeds greater than 50 kt, yaw rates sensed by the internal gyro are combined with signals from the No.1 lateral accelerometer as pre-filtered by the No.1 stabilator amplifier. The resultant corrective signal, designed to provide turn coordination, is applied to the yaw SAS servovalve. Electrical control signals to the pitch, roll, and yaw SAS servovalves are converted to hydraulic commands and transferred to the SAS actuators. These actuators, located at the pilot-assist servos (paragraph 3.3), use pressure supplied by the No.2 hydraulic system (paragraph 2) to provide corrective movements to the appropriate flight control linkages. These in turn direct changes in main or tail rotor pitch without moving the corresponding cockpit controls. The system is engaged and disengaged by depressing the SAS 1 button on the AFCS control panel (figure 51), and can also be disengaged by depressing the AFCS REL button on either cyclic grip.

4.4 Electronic Flight Control System

4.4.1 General. The EFCS, as depicted schematically in appendix F, figure 55, is a digital system designed to provide stability augmentation, hover augmentation, trim, and pitch bias. Autopilot functions are incorporated for airspeed, heading, attitude, and altitude hold. Coupler functions are utilized for automatic hover approach and departure and crew hover control. During blade fold, the system repositions the flight controls via the pilot-assist servos. The EFCS control is either "inner-loop", during which no cockpit control movement occurs, or "outer-loop", during which cockpit controls are repositioned. Inner-loop control is designed to be fast in response but limited in authority, while outer-loop control has 100% authority rate limited to 10% per sec. In the event of a system malfunction, both types of control permit complete pilot override through normal flight control operation. The EFCS utilizes airspeed, attitude, and altitude sensors; rate gyros; accelerometers in the various control axes; and a digital computer to analyze inputs from control position indicators. The computer central processing unit (CPU) is designed to rapidly process the respective signals, determine flight

path errors, and apply corrective flight control inputs via inner-loop SAS and outer-loop trim system and pitch bias actuator inputs. The computer has the capability to self-monitor, isolate faults, and provide failure advisory. With the exception of the PBA, all EFCS functions can be engaged and disengaged through the appropriate buttons and switches on the AFCS control panel and can also be disengaged by depressing the AFCS REL button on either cyclic grip (figures 51 and 44, respectively).

4.4.2 Digital stability augmentation system. The SAS 2, as depicted schematically in figure 54, is designed to provide the identical short term, limited authority pitch, roll, and yaw corrections as SAS 1 (paragraph 4.3.2). In addition, hover augmentation and gust alleviation are provided through increased pitch and roll inner-loop stability resulting from SAS 2 compensation of pitch and roll excursions. The system is also utilized in the hover coupler mode (see paragraph 4.4.8). The SAS 2 incorporates pitch rate and lateral acceleration signals provided by the No.2 stabilator system and analyzed by the EFCS computer. The resultant corrective control inputs are transmitted to the SAS servovalves, which direct flight control system changes as described in paragraph 4.3.2.

4.4.3 Trim system. A trim system is incorporated into all cockpit controls. Hydraulic powered longitudinal and lateral trim actuators and electromechanical collective and yaw trim servos are designed to provide a control force gradient about the control trim position. These force gradients maintain each control in a set position until the respective trim is released. The entire system is engaged and disengaged via the TRIM button on the AFCS control panel (figure 51). Collective and cyclic trim can be individually disengaged by depressing the appropriate trim release switch (paragraph 3.2). Directional trim disengagement is accomplished via the pedal microswitches below 50 kt and by the pedal switches in conjunction with the cyclic trim release above 50 kt. The EFCS computer monitors the system and transmits appropriate signals to the trim actuators and servos in response to pilot induced trim changes or autopilot commands. The trim system commands full control authority but is rate limited to 10% per sec in all four channels. The system can be overridden through the trim control force gradient by normal control application.

4.4.4 Pitch bias actuator. The PBA is designed to provide static longitudinal stability by producing a positive stick position gradient throughout the entire airspeed range. The EFCS computer analyzes pitch attitude, pitch rate, and airspeed via the appropriate sensors and transmits corrective control inputs to the PBA. The electromechanical actuator, located at the mixing unit (paragraph 3.4), operates with 15% control authority rate limited to 5% per sec. The PBA functions automatically through the EFCS computer and is not selectable on the AFCS control panel.

4.4.5 Autopilot. An autopilot mode is designed to provide airspeed, pitch attitude, and roll attitude hold during cruise flight and heading hold regardless of airspeed or translational azimuth. The autopilot also assists the SAS in turn coordination at airspeeds greater than 50 kt. During operations in the autopilot mode, the EFCS computer analyzes inputs from the air data transducers, rate gyros, accelerometers, collective position sensors, and AHRS pitch attitude and heading sensors. To maintain attitude, airspeed, and heading the computer transmits open loop corrective control inputs to the appropriate trim actuators (paragraph 4.4.3). Pitch attitude hold is only operational at airspeeds less than 50 kt. Once above this airspeed, the computer is designed to automatically convert to the airspeed hold mode. Roll attitude hold operates regardless of airspeed, but is programmed to allow turn coordination above 50 kt. The cyclic beeper trim switch (figure 44) can be used to initiate small changes in pitch and roll attitude below 50 kt and airspeed and roll attitude above 50 kt. Heading hold is designed to be operational in a hover and at any airspeed in any translational direction. The autopilot mode is only operational when the trim system and one or both SAS modes are engaged.

4.4.6 Altitude hold. The altitude hold system is designed to maintain either the barometric or radar altitude at which the system is engaged. During operation in the barometric altitude mode, the EFCS computer analyzes inputs from air data transducers, vertical accelerometers, and the RADALT. Operation in the RADALT mode utilizes inputs from the RADALT and the vertical accelerometers. Computer generated altitude hold command signals are incorporated into the collective inner-loop actuator for rapid, limited authority corrections, and to the collective outer-loop trim actuator for rate limited, full authority control. Barometric hold engagement is limited to altitudes between -1,000 and +20,000 ft, while

RADALT hold is limited to operation below 5,000 ft. Engagement of either system will automatically disengage the other. Changes in altitude setting can be accomplished by depressing the ALT REL button on either collective grip (figure 44), climbing or descending with normal cockpit control inputs, and releasing the button once the new altitude has been attained. If the RADALT hold feature becomes unreliable as indicated by large fluctuations in the RADALT signal, the computer is designed to disengage that mode and engage barometric altitude hold.

4.4.7 Automatic approach coupler. The automatic approach mode is designed to provide a coordinated approach to a stabilized hover from any airspeed or altitude within the aircraft flight envelope. Input signals from the Doppler, RADALT, rate sensors, accelerometers, and hover altitude selector controls are utilized by the EFCS computer to establish an approach profile line. The approach coupler will provide a constant airspeed descent of 360 ft/min until 50 ft above this profile line, at which time rate of descent is decreased to 120 ft/min. Once the approach line is intersected, the system begins a 1 kt/sec deceleration while maintaining the computed rate of descent. When the aircraft is within 1 ft of selected altitude and 1 kt of selected airspeed, the computer switches control to the automatic hover mode and the aircraft accelerates to the selected longitudinal ground speed (see paragraph 4.4.8). If the aircraft is initially below the computed approach line entry height, the system is designed to provide a constant altitude, 1 kt/sec deceleration until the line is intersected, at which time the computed approach phase is initiated. Minimum hover altitude at the termination of an automatic approach is 40 ft. Altitude selector settings below this will be attained by an automatic vertical descent once a 40 ft, 0 kt hover has been established.

4.4.8 Automatic hover. The automatic hover mode is designed to provide longitudinal and lateral ground speed control; longitudinal, lateral, and directional stabilization about the selected ground speed; and automatic altitude retention. Input signals from the Doppler, RADALT, accelerometers, and ground speed selector controls are analyzed by the EFCS computer. The computer then transmits appropriate corrective signals to the SAS 2 and pitch and roll trim systems (paragraphs 4.4.2 and 4.4.3, respectively). The automatic hover mode can be engaged manually or will engage automatically at the termination of an automatic approach. Longitudinal and

lateral velocities are selected via rotary dials on the AFCS panel (figure 51). Once established, the selected ground speed can be beeped ± 10 kt in any direction with the cyclic trim switch (figure 44).

4.4.9 Automatic hover departure. The depart mode is designed to provide an automatic takeoff and departure to an altitude of 500 ft and a ground speed of 100 kt. Input signals from the Doppler, RADALT, rate sensors, accelerometers, and engine torque sensors are utilized by the EFCS computer to establish a 2 kt/sec acceleration and a 480 ft/min rate of climb once 20 kt ground speed has been attained. Upon reaching the correct airspeed and altitude, the acceleration and ascent functions are cancelled and airspeed/barometric altitude hold is activated. At any point during the departure profile, acceleration or ascent can be interrupted by depressing the cyclic or collective trim release, respectively.

4.4.10 Crew hover mode. The crew hover feature can be engaged while operating in the hover coupler mode and is designed to allow the crewman 5 kt of ground speed control during hoist or rescue operations. The control is provided through a trim switch on the Crew Hover-Trim Panel, as depicted in appendix F, figure 56. Switch inputs are combined with the Doppler, velocity, and accelerometer signals to maneuver the aircraft in the longitudinal and lateral axes.

5 FOLD SYSTEM

5.1 Blade Fold

The blade fold system is designed to automatically fold the main rotor blades to facilitate shipboard operations. Once initiated by the pilot, the blade fold system indexes the rotor head to a known position via an index drive unit. The drive unit rotates the blades at 2.5 RPM until within 30 deg of the index position, at which time rotation decreases to 0.25 RPM. Once the blades are properly positioned, the pilot initiates the remainder of the sequence, which includes EFCS repositioning of cockpit flight controls via the trim actuators. When this is completed, the blade pitch control horns are locked and then the rotor blades unlocked and folded by blade fold actuators and mechanical gearings (paragraph 3.6).

5.2 Pylon Fold

Pylon folding is performed manually with a blade folding pole. When the pylon is folded open to approximately 5 deg, a switch is activated and power is applied to a tail rotor blade position actuator. This extends, indexes, and gust locks the tail rotor prior to folding. The actuator also prevents initiation of the pylon fold procedure until the automatic main blade fold sequence has been completed.

APPENDIX B

FLIGHT TEST ENVELOPE

FLIGHT TEST ENVELOPE

During the flying qualities and performance evaluation of the SH-60B helicopter, all normal limitations as outlined in reference (1) were observed. In addition, the following restrictions were followed:

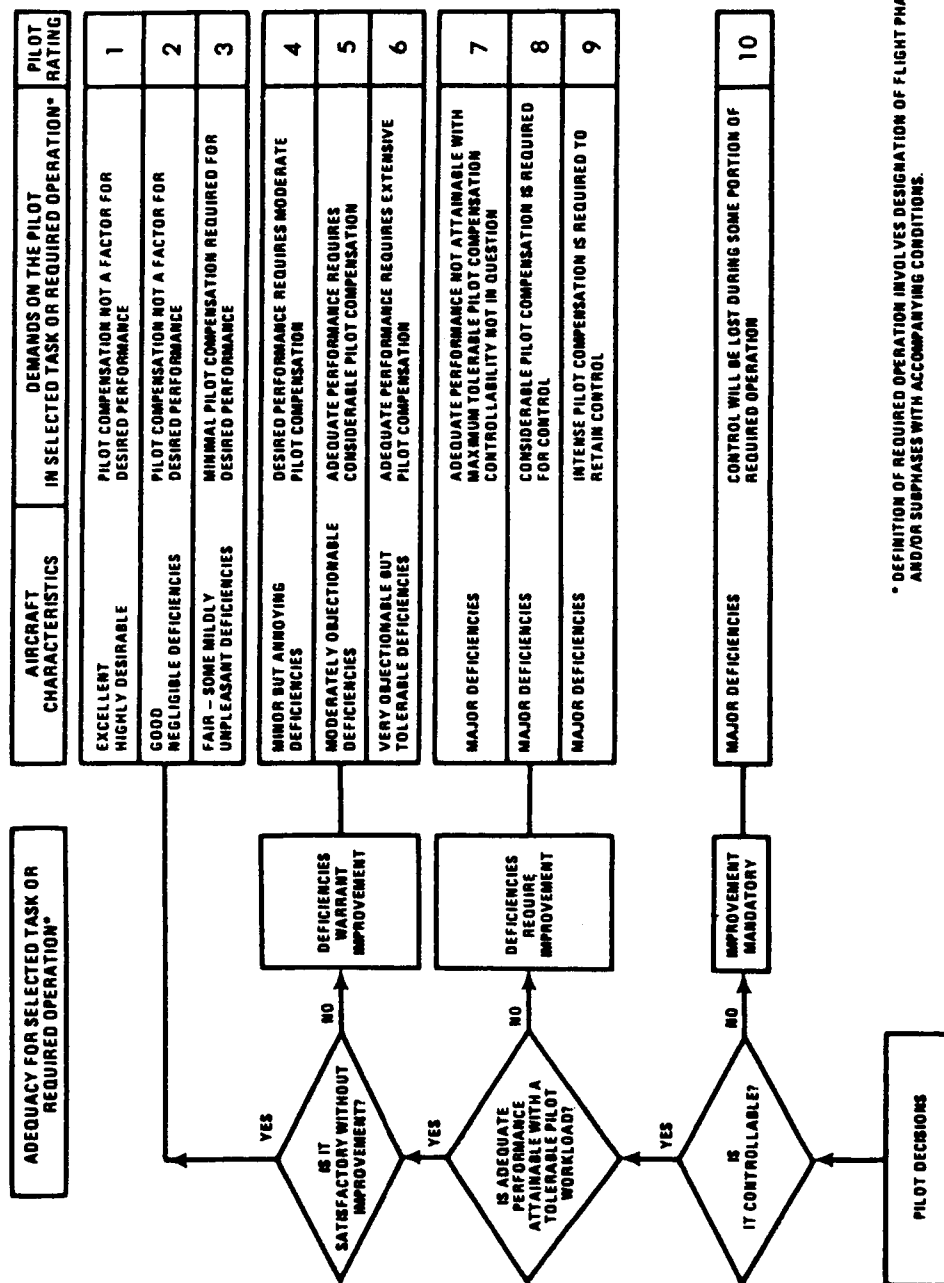
- a. Minimum normal acceleration: 0.0 g
- b. Maximum normal acceleration: 2.5 g
- c. Maximum pitch attitude: +45 deg
- d. Pitch recovery: Initiated at +30 deg of pitch
or before if necessary to avoid
exceeding the absolute limit
- e. Maximum roll attitude: +45 deg
- f. Roll recovery: Initiated at +30 deg of roll
or before if necessary to avoid
exceeding the absolute limit
- g. Maximum cyclic control step input: 2.0 in.
- h. Maximum control input increment: 0.5 in.
- i. Sideslip limit: 2 ball widths

In addition, the severity of any maneuver was reduced with the onset of control system force feedback.

APPENDIX C

RATING SCALES

COOPER-HARPER HANDLING QUALITIES RATING SCALE



* DEFINITION OF REQUIRED OPERATION INVOLVES DESIGNATION OF FLIGHT PHASE AND/OR SUBPHASES WITH ACCOMPANYING CONDITIONS.

VIBRATION 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 D

DEFINITION OF DEFICIENCIES

DEFINITION OF DEFICIENCIES

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

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

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

Part III indicates a deficiency that appears to 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.

APPENDIX E

TABLES

Table I
SH-60B MISSION PROFILES (1)

MISSION	MISSION SEGMENT	SEGMENT PROFILE	TIME (MIN)	TOTAL TIME (MIN/HR)
ASW (100 NM)	TRANSIT TO DATUM	ENGINE START TO TAKEOFF CLIMB ON COURSE CRUISE	5 25 20	220/3.67
	REDETECTION	SEARCH LOITER	30 50	
	LOCALIZATION ATTACK RE-ATTACK	DESCENT/COMBAT	40	
	RETURN TO SHIP	CLIMB ON COURSE CRUISE/DESCENT/LAND	25 25	
ASST (40 NM)	TRANSIT TO STATION	ENGINE START TO TAKEOFF CLIMB ON COURSE CRUISE	5 7.5 7.5	250/4.17
	SURVEILLANCE CLASSIFICATION	LOITER	210	
	RETURN TO SHIP	CRUISE/DESCEND/LAND	20	

NOTE: 1) ASW MISSION CONFIGURATION INCLUDES FULL LOAD OF SONOBUOYS AND TORPEDOES
ASST MISSION CONFIGURATION IS WITHOUT STORES

Table II

TESTS AND TEST CONDITIONS (1)(2)(3)

TEST	AIR SPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	ROTOR RPM (%)	REMARKS
COCKPIT EVALUATION	TEST RANGE						EVALUATED ON GROUND AND IN FLIGHT CONCURRENT WITH OTHER TESTS AS APPROPRIATE FIELD OF VIEW MEASUREMENTS FROM DESIGN EYE POSITION WITH AIRCRAFT LEVEL AND UNDER SIMULATED DAYLIGHT VMC
MECHANICAL CHARACTERISTICS	TEST RANGE						EVALUATED QUANTITATIVELY ON GROUND AND QUALITATIVELY IN FLIGHT CONCURRENT WITH OTHER TESTS AS APPROPRIATE FORCE/DISPLACEMENT MEASUREMENTS TAKEN WITH ROTORS STATIC/SPREAD, USING EXTERNAL AC POWER AND BACKUP PUMP HYDRAULICS (BOOST, TRIM, AND SAS 2 ON)
ENGINE ASSESSMENT	TEST RANGE						ENGINE CHARACTERISTICS EVALUATED QUALITATIVELY: START UP - APU AND ENGINE CROSS BLEED PCL RUN UP - IN 12, 10 AND 8 SECONDS SHUT DOWN - WITH ROTOR BRAKE
ENGINE PERFORMANCE	TEST RANGE						ENGINE CHARACTERISTICS EVALUATED QUANTITATIVELY: ENGINE TRIM - CONCURRENT WITH OTHER TESTS STATIC DROOP - CONCURRENT WITH VERTICAL CLIMB PERFORMANCE/MISSION TASKS
ICE/DGE HOVER DETERMINATION	TEST RANGE						HOVER LADDER METHOD RADALT USED FOR HOVER HEIGHT MEASURING DEVICE OUTSIDE REFERENCE USED TO CONTROL DRIFT

Table II (Con't)

TEST	AIR SPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	ROTOR RPM (%)	REMARKS
HOVER PERFORMANCE	0	40 (IGE) 100 (OGE)	19,170	359.8	22	96	FREE FLIGHT METHOD 10 FT AGL (IGE)/70 FT AGL (OGE) (AS DETERMINED BY RADALT) OUTSIDE REFERENCE USED TO CONTROL DRIFT WINDS 5 KT AT 0 DEGREES RELATIVE TO AIRCRAFT NOSE
		90 (IGE) 150 (OGE)	16,400	352.5	26	98 100	
VERTICAL CLIMB PERFORMANCE	0	100	18,980	359.3	22	100	AVERAGE CLIMB TORQUE SETTINGS: 90, 100, 106 % (HEAVY GROSS WEIGHT) 80, 90, 105 % (LIGHT GROSS WEIGHT) OGE TRIM POINT TAKEN BETWEEN EACH CLIMB CHANGE IN PRESSURE ALTITUDE MEASURED IN STABLE CLIMB OVER AT LEAST 30 SECONDS TO MINIMIZE ALTIMETER LAG ERRORS
		150	16,300	352.2	26		
LEVEL FLIGHT PERFORMANCE	40, 61, 79, 99, 120, 143 (V _H)	2700 TO 3200	18,680	358.5	16	100	CONSTANT REFERRED GROSS WEIGHT METHOD W/ ₀ = 20,800 LB
CLIMB AND DESCENT PERFORMANCE	40, 60, 80, 100, 120	2000 TO 3000	17,490	355.3	19	100	SAWTOOTH CLIMB/DESCENT METHOD AVERAGE TORQUES: MINIMUM - 9% MAXIMUM - 100%
TRIMMED FLIGHT CONTROL POSITIONS - LEVEL FLIGHT	40, 61, 79, 99, 120, 140, 160, 180 (V _{H0})	3000	18,680	358.5	16	100	TEST CONDUCTED THROUGH 1000 FT ALTITUDE BAND COLLECTIVE FIXED DIVE USED TO ACHIEVE V _{H0}
TFCP - CLIMBS AND DESCENTS	80	2000 TO 3000	17,490	355.3	19	100	AVERAGE TORQUES: 9, 20, 48, 77, 100 %

Table II (Con't)

TEST	AIR SPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	ROTOR RPM (%)	REMARKS
TFCP STEADY TURNS	80, 140	2000	18,200	357.2	20	100	ANGLE OF BANK SETTINGS: 15, 30, 45 DEGREES LEFT AND RIGHT
STATIC LONGITUDINAL STABILITY	80, 140 (TRIM)	2000	18,290	357.5	20	100	AIR SPEED VARIATIONS: -3, -5, -10, -19; +4, +7, +10, +17, +21 (80 KT TRIM) -4, -11, -18, -19; +3, +6, +10, +14, +20 (140 KT TRIM)
STATIC LATERAL - DIRECTIONAL STABILITY	80, 140	2000	17,890	357.0	20	100	STEADY HEADING SIDESLIP METHOD
							SIDESLIP ANGLES: 7, 13, 18 DEG LEFT 5, 12, 15 DEG RIGHT (80 KT TRIM) 3, 6, 13 DEG LEFT 5, 9, 13 DEG RIGHT (140 KT TRIM)
							CYCLIC ONLY TURNS 20 DEGREES ANGLE OF BANK LEFT AND RIGHT
DYNAMIC LONGITUDINAL STABILITY	80	2000	17,180	354.5	20	100	PEDAL ONLY TURNS
							SIDESLIP ANGLES: 13 DEG LEFT/12 DEG RIGHT (80 KT TRIM) 6 DEG LEFT/9 DEG RIGHT (140 KT TRIM)
							SHORT TERM: FREQUENCY SWEEP USED TO DETERMINE NATURAL FREQUENCY FOR LONGITUDINAL DOUBLET INPUT
SPIRAL STABILITY	80, 140	2000	17,100	354.3	20	100	LONG TERM: EVALUATED CONTROLS FREE ALL AFCS COMPONENTS ON AUTOPILOT OFF/AUTOPILOT AND SAS OFF PILOT INDUCED EXCITATION USED THROUGHOUT
							LATERAL CYCLIC USED TO GENERATE 20 DEGREE ANGLE OF BANK LEFT AND RIGHT

Table II (Con't)

TEST	AIRSPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	ROTOR RPM (%)	REMARKS
LATERAL - DIRECTIONAL OSCILLATION	80, 140	2000	17,020	355.2	21	100	EVALUATED CONTROLS FREE PILOT INDUCED EXCITATIONS USED: RELEASE FROM STEADY HEADING SIDESLIP SINUSOIDAL PEDAL INPUT COLLECTIVE PULSE
MANEUVER STABILITY	140	2000	18,530	353.9	22	100	ACCELERATED TURNS ANGLE OF BANK SETTINGS: 15, 30, 45 DEGREES LEFT AND RIGHT
			18,310	353.3			PULL UPS/PUSH OVERS STICK DISPLACEMENTS: 0.5, 1.0, 1.5, 2.0 INCHES (PULL UPS) 0.5, 1.0, 1.5 INCHES (PUSH OVERS)
CONTROL RESPONSE	140	2000	18,200	352.9	22	100	QUALITATIVE ASSESSMENT STICK DISPLACEMENTS: 0.5, 1.0, 1.5 INCHES (FORWARD/AFT AND LEFT/RIGHT)
GUST RESPONSE	80, 140	500	18,010	352.4	25	100	QUALITATIVE ASSESSMENT OPEN/CLOSED LOOP METHODS PILOT INDUCED EXCITATIONS USED: COLLECTIVE PULSE PEDAL PULSE
DEGRADED MODES	80 TO 140	1000	18,230	358.9	22	100	QUALITATIVE TESTS: AFCS FAILURE SAS FAILURE STABILATOR FAILURE SINGLE ENGINE CAPABILITIES/AUTOROTATION ASSESSMENT

Table II (Con't)

TEST	AIR SPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	ROTOR RPM (%)	REMARKS
LOW AIRSPEED TRIMMED FLIGHT CONTROL POSITIONS	0, 10, 20, 30	60	18,710	359.8	20	100	FORWARD/AFT AND LEFT/RIGHT 15 FT AGL (AS DETERMINED BY RADALT) GROUND SPEED DETERMINED VIA SHIP SYSTEM DOPPLER WIND SPEED APPROXIMATELY 0 KT
CRITICAL AZIMUTH	30	60	18,420	359.0	20	100	WIND AZIMUTH EVERY 30 DEG (AS DETERMINED BY RMI) 15 FT AGL (AS DETERMINED BY RADALT) GROUND SPEED DETERMINED BY SHIP SYSTEM DOPPLER WIND SPEED APPROXIMATELY 0 KT
LOW AIRSPEED STATIC STABILITY	0 (TRIM)	60	18,150	358.3	20	100	AIRSPEED VARIATIONS: 4, 10, 20, 30 KTAS (FORWARD FROM 0 KT TRIM) 6, 10, 20, 30 KTAS (AFT FROM 0 KT TRIM) 3, 6, 10, 20, 30 KTAS (LEFT/RIGHT FROM 0 KT TRIM) GROUND SPEED DETERMINED BY SHIP SYSTEM DOPPLER WIND SPEED APPROXIMATELY 0 KT
LOW AIRSPEED DYNAMIC STABILITY	0	550	17,920	357.7	20	100	EVALUATED DURING TRIMMED 0 KT OGE HOVER 500 FT AGL (AS DETERMINED BY RADALT) PILOT INDUCED EXCITATIONS: 5 DEGREE PITCH DISPLACEMENT (NOSE UP AND NOSE DOWN) 5 DEGREE ROLL DISPLACEMENT (RIGHT AND LEFT)
LOW AIRSPEED CONTROL RESPONSE	0	550	17,900	357.6	20	100	QUALITATIVE ASSESSMENT STICK DISPLACEMENTS: 0.5, 1.0, 1.5 INCHES (FORWARD/AFT AND LEFT/RIGHT)

Table II (Con't)

TEST	AIR SPEED (KIAS)	PRESSURE ALTITUDE (FT)	AVERAGE GROSS WEIGHT (LBS)	AVERAGE CG (IN)	OAT (DEG C)	NOTOR RPM (%)	REMARKS
SYSTEMS	0	40					LIMITED AFCS EVALUATION
	TO	70	18,230	358.9	22	100	AUTOMATIC APPROACH TO HOVER
	130	500					AUTOMATIC HOVER AUTOMATIC DEPARTURE FROM HOVER ALTITUDE HOLD BAROMETRIC ALTITUDE HOLD RADAR ALTITUDE HOLD

NOTE: 1) SAS: ON EFCS: ON STABILATOR: AUTO (UNLESS OTHERWISE NOTED)
 2) CONTROL DISPLACEMENT MEASURED WITH CLOTH MEASURING TAPE
 3) CONTROL FORCES MEASURED WITH HAND HELD FORCE GAUGE

Table III
FLIGHT PROFILE

FLT NO.	TAKEOFF GW (LB)	LANDING GW (LB)	CONFIGURATION	PROFILE
1	19,390	18,080	CREW: PILOT ENGINEER STORES: (1) MK-46 TORPEDO	ENGINE PERFORMANCE HOVER PERFORMANCE VERTICAL CLIMB PERFORMANCE LEVEL FLIGHT PERFORMANCE FLYING QUALITIES TFCP LEVEL FLIGHT STEADY TURNS STATIC LONGITUDINAL STABILITY
2	18,050	16,260	CREW: PILOT ENGINEER STORES: (1) MK-46 TORPEDO	CLIMB/DESCENT PERFORMANCE FLYING QUALITIES TFCP CLIMBS/DECENTS DYNAMIC LONGITUDINAL STABILITY
3	19,180	17,694	CREW: PILOT ENGINEER SENSO STORES: 2 SONOBUOYS	DEGRADED MODES MISSION TASKS HOVER ICE/OGE PEDAL TURNS SONOBUOY DEPLOYMENT MARK ON TOP (OTPI) SIMULATED IFR GCA AFCS EVALUATION ALTITUDE HOLD AUTO APPROACH AUTO DEPARTURE HEADING HOLD IN HOVER BLADE FOLD ASSESSMENT
4	18,920	15,980	CREW: PILOT ENGINEER NO STORES	FLYING QUALITIES STATIC LAT-DIR STABILITY SPIRAL STABILITY LAT-DIR OSCILLATIONS MANEUVER STABILITY CONTROL RESPONSE CUST RESPONSE LOW AIRSPEED FLYING QUALITIES

Table IV
MILITARY SPECIFICATION REQUIREMENTS

SPECIFICATION	TITLE	APPLICABLE PARAGRAPHS
MIL-C-5011A	CHARTS, STANDARD AIRCRAFT CHARACTERISTICS AND PERFORMANCE, PILOTED AIRCRAFT (5 NOVEMBER 1951)	3.2.2 - 3.2.4 3.2.8 - 3.2.11
MIL-D-23222A(AS)	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS (18 MAY 1971)	3.3.5
MIL-H-8501A	HELICOPTER FLYING AND GROUND HANDLING QUALITIES (7 SEPTEMBER 1961)	3.1 - 3.5.4.2 3.5.8 - 3.7.3
MIL-STD-2500	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT (28 AUGUST 1974)	4.2 - 4.10.1 5.1 - 5.7.10
MIL-STD-8508	AIRCREW STATION VISION REQUIREMENTS FOR MILITARY AIRCRAFT (3 NOVEMBER 1970)	4. 8.1
MIL-STD-1333A	AIRCREW STATION GEOMETRY FOR MILITARY AIRCRAFT (30 JUNE 1976)	3. 4.
MIL-STD-1472C	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT, AND FACILITIES (2 MAY 1981)	5.2 5.4

Table V
EVALUATOR ANTHROPOMETRIC MEASUREMENTS

PARAMETER	MEASUREMENT (Inches)	PERCENTILE
HEIGHT	72.0	80th
SHOULDER WIDTH	18.5	40th
TRUNK HEIGHT	24.8	80th
SITTING HEIGHT	36.8	66th
FUNCTIONAL REACH	34.5	97th
BUTTOCK-KNEE LENGTH	24.4	65th
SITTING EYE HEIGHT	33.2	90th

Table VI
COCKPIT EVALUATION FLIGHT GEAR

DESCRIPTION	PART NUMBER
HELMET, PROTECTIVE	81F5191-1
GLOVES, SUMMER FLYING	MIL-G-8118
VEST, SURVIVAL	MIL-V-81523
BOOTS, FLYING	MIL-B-21408
CLIPBOARD, PILOT'S	MIL-B-1170
COVERALLS, SUMMER FLYING	MIL-C-83141

Table VII

FLIGHT CONTROL MECHANICAL CHARACTERISTICS - SPECIFICATION COMPLIANCE (1)(2)

PARAMETER	CONTROL	MEASURED VALUE	SPECIFICATION MIL-H-8501A		
			PARAGRAPH	REQUIREMENT	CONFORMANCE
CONTROL TRAVEL (Inches)	LONGITUDINAL	9.4	-----	-----	-----
	LATERAL	9.0	-----	-----	-----
	DIRECTIONAL	5.3	-----	-----	-----
	COLLECTIVE	8.6	-----	-----	-----
CONTROL SYSTEM FREEPLAY (Inches)	LONGITUDINAL	0.1	3.5.10	0.2	MET
	LATERAL	0.1	3.5.10	0.2	MET
	DIRECTIONAL	0.2	3.5.10	0.2	MET
	COLLECTIVE	0.1	3.5.10	0.2	MET
BREAKOUT FORCE (PLUS FRICTION) (lb)	LONGITUDINAL	1.25 FWD/1.0 AFT	3.2.7	0.5 - 1.5	MET
	LATERAL	1.25 RT/1.25 LEFT	3.3.13	0.5 - 1.5	MET
	DIRECTIONAL	3.0 RT/3.0 LEFT	3.3.13	3.0 - 7.0	MET
	COLLECTIVE	2.25 UP/2.0 DOWN	3.4.2	1.0 - 3.0	MET
AVERAGE FORCE GRADIENT (OVER 1st INCH) (lb/in)	LONGITUDINAL	1.6 FWD/1.8 AFT	3.2.4	0.5 - 2.0	MET
	LATERAL	1.5 RT/1.5 LEFT	3.3.11	0.5 - 2.0	MET
	DIRECTIONAL	15.5 RT/15.75 LEFT	-----	-----	-----
	COLLECTIVE	-----	-----	-----	-----
LIMIT FORCE (lb)	LONGITUDINAL	6.25 FWD/6.5 AFT	3.2.6	6.0	MET
	LATERAL	4.25 RT/5.0 LEFT	3.3.12	7.0	MET
	DIRECTIONAL	37.0 RT/30.0 LEFT	3.3.12	15.0	FAILED
	COLLECTIVE	-----	-----	-----	-----
TRIM SYSTEM FREEPLAY (Inches)	LONGITUDINAL	0.2	-----	-----	-----
	LATERAL	0.1	-----	-----	-----
	DIRECTIONAL	0.3	-----	-----	-----
	COLLECTIVE	0.1	-----	-----	-----
BEEP TRIM AUTHORITY (Inches)	LONGITUDINAL	9.1	3.2.3	FORCES TRIM TO ZERO	MET
	LATERAL	9.0	3.3.10		MET

Table VII (Con't)

PARAMETER	CONTROL	MEASURED VALUE	SPECIFICATION MIL-H-8501A		
			PARAGRAPH	REQUIREMENT	CONFORMANCE
BEEPER TRIM LAG (3) (sec)	LONGITUDINAL LATERAL	0.10	-----	-----	-----
		0.15	-----	-----	-----
TRIM RATE (Inches/sec)	LONGITUDINAL LATERAL	0.57	-----	-----	-----
		0.51	-----	-----	-----
CENTERING	LONGITUDINAL LATERAL DIRECTIONAL COLLECTIVE	ABSOLUTE	3.2.3	POSITIVE	MET
		ABSOLUTE	3.3.10	POSITIVE	MET
		POSITIVE	3.3.10	POSITIVE	MET
		ABSOLUTE	3.4.2	NO CREEP	MET
CONTROL JUMP (4) (Inches)	LONGITUDINAL LATERAL DIRECTIONAL COLLECTIVE	0.1	3.2.3	NONE DESIRABLE	FAILED
		0.1	3.3.10	NONE DESIRABLE	FAILED
		0.3	3.3.10	NONE DESIRABLE	FAILED
		0.6	-----	NONE DESIRABLE	FAILED
CONTROL DYNAMICS (DAMPING) (5) (# of overshoots)	LONGITUDINAL LATERAL DIRECTIONAL COLLECTIVE	3 - LIGHT	-----	-----	-----
		2 - MODERATE	-----	-----	-----
		0 - DEADBEAT	-----	-----	-----
		0 - DEADBEAT	-----	-----	-----

NOTE: 1) ROTORS: STATIC/SPREAD ENGINES: OFF ELECTRIC POWER: EXTERNAL AC HYDRAULIC POWER: BACKUP PUMP

BOOST: ON TRIM: ON SAS 2: ON ALL REMAINING AFCS COMPONENTS: OFF

2) ALL DISPLACEMENTS MEASURED FROM PILOT CONTROLS ALL FORCES MEASURED FROM ATO CONTROLS

3) TRIM LAG MEASURED DURING TRIM DIRECTION REVERSAL

4) CONTROL JUMP FOR ALL CONTROLS FROM 1" DISPLACEMENT

5) CYCLIC/COLLECTIVE DYNAMICS FROM 3" DISPLACEMENT DIRECTIONAL PEDAL DYNAMICS FROM 2" DISPLACEMENT

Table VIII

FLIGHT CONTROL MIXING

FROM	TO	REASON FOR MIXING
COLLECTIVE	YAW	COUNTER YAWING MOMENT DUE TO ANTI-TORQUE
COLLECTIVE	LATERAL	COUNTER RIGHT ROLLING MOMENT DUE TO TAIL ROTOR THRUST
COLLECTIVE	LONGITUDINAL	COUNTER PITCHING MOMENT DUE TO ROTOR DOWNWASH ON STABILATOR
YAW	LONGITUDINAL	COUNTER PITCHING MOMENT DUE TO CANTED TAIL ROTOR LIFT VECTOR

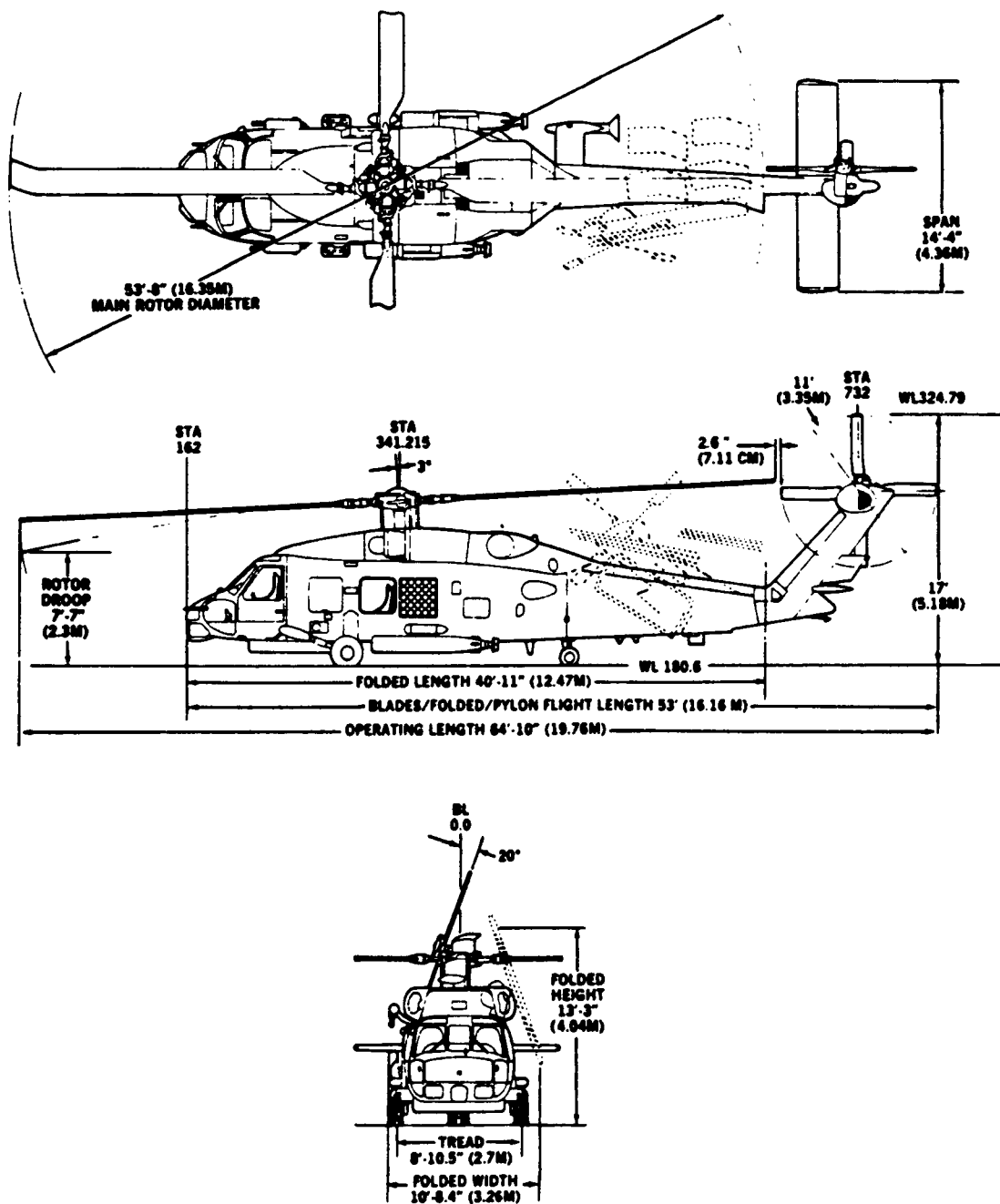


Figure 1
SH-60B PRINCIPLE DIMENSIONS

MODEL SH-60B HELICOPTER
BuNo 162092

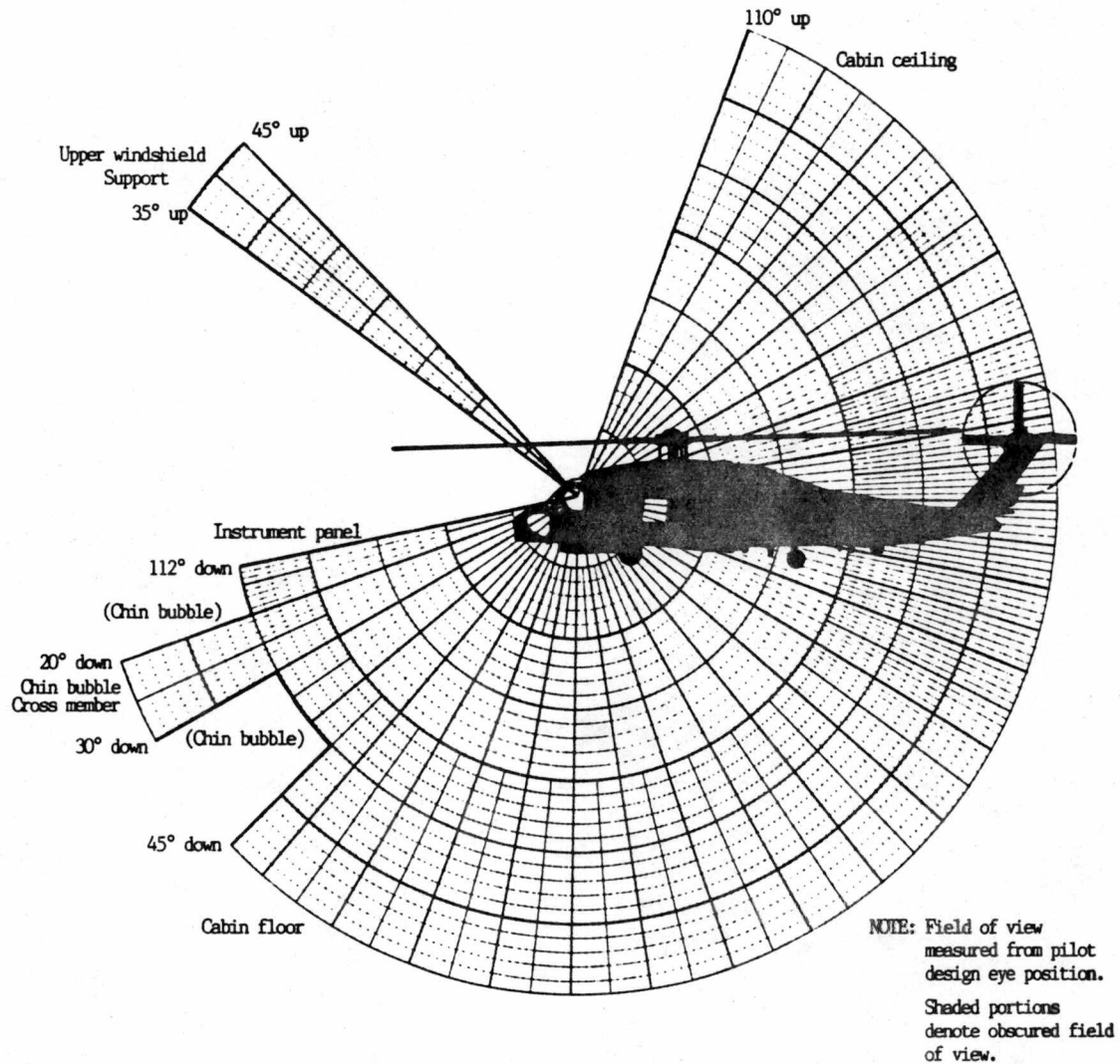


Figure 2
VERTICAL FIELD OF VIEW

MODEL SH-60B HELICOPTER
BuNo 162092

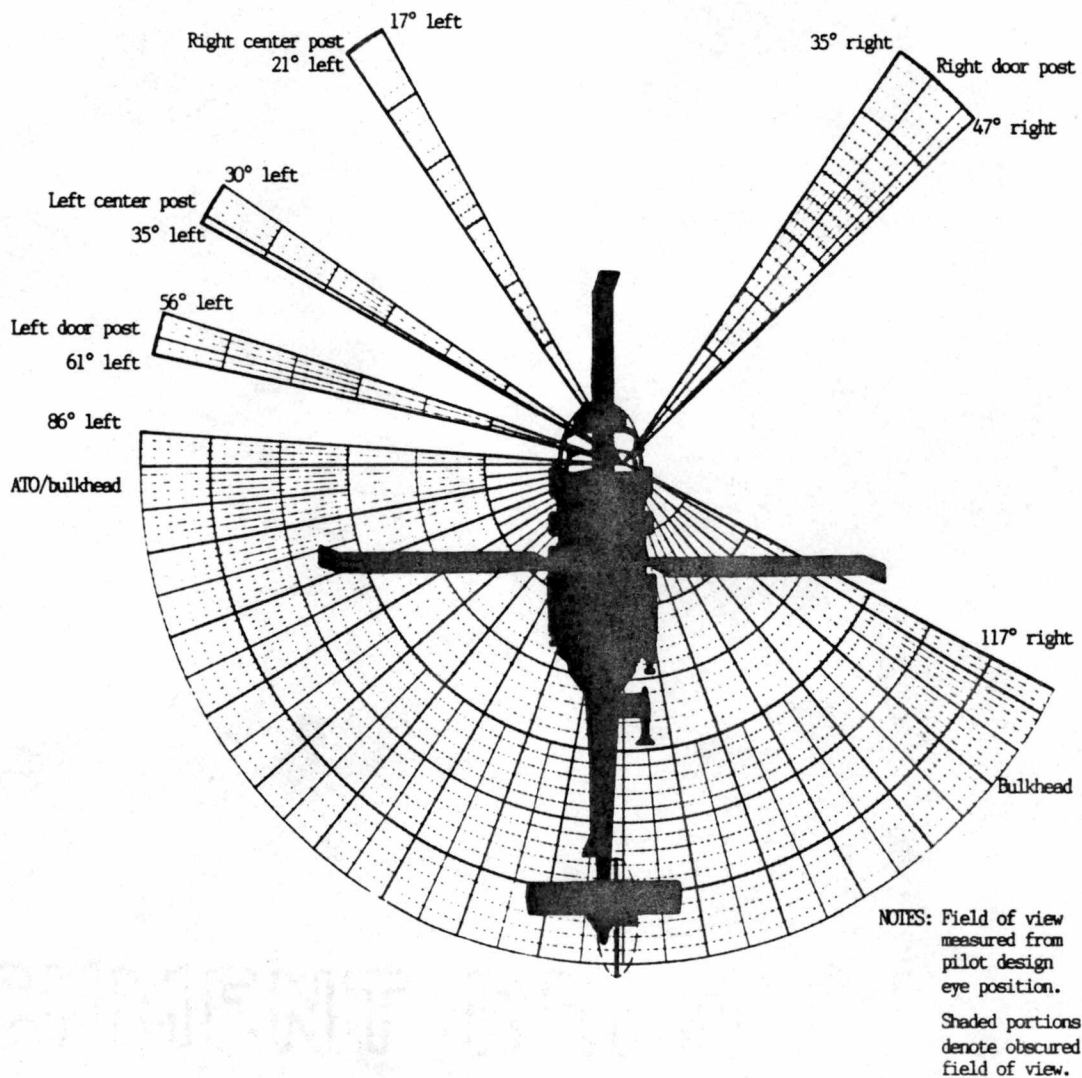


Figure 3
HORIZONTAL FIELD OF VIEW

MODEL SH-60B HELICOPTER
BuNo 162092

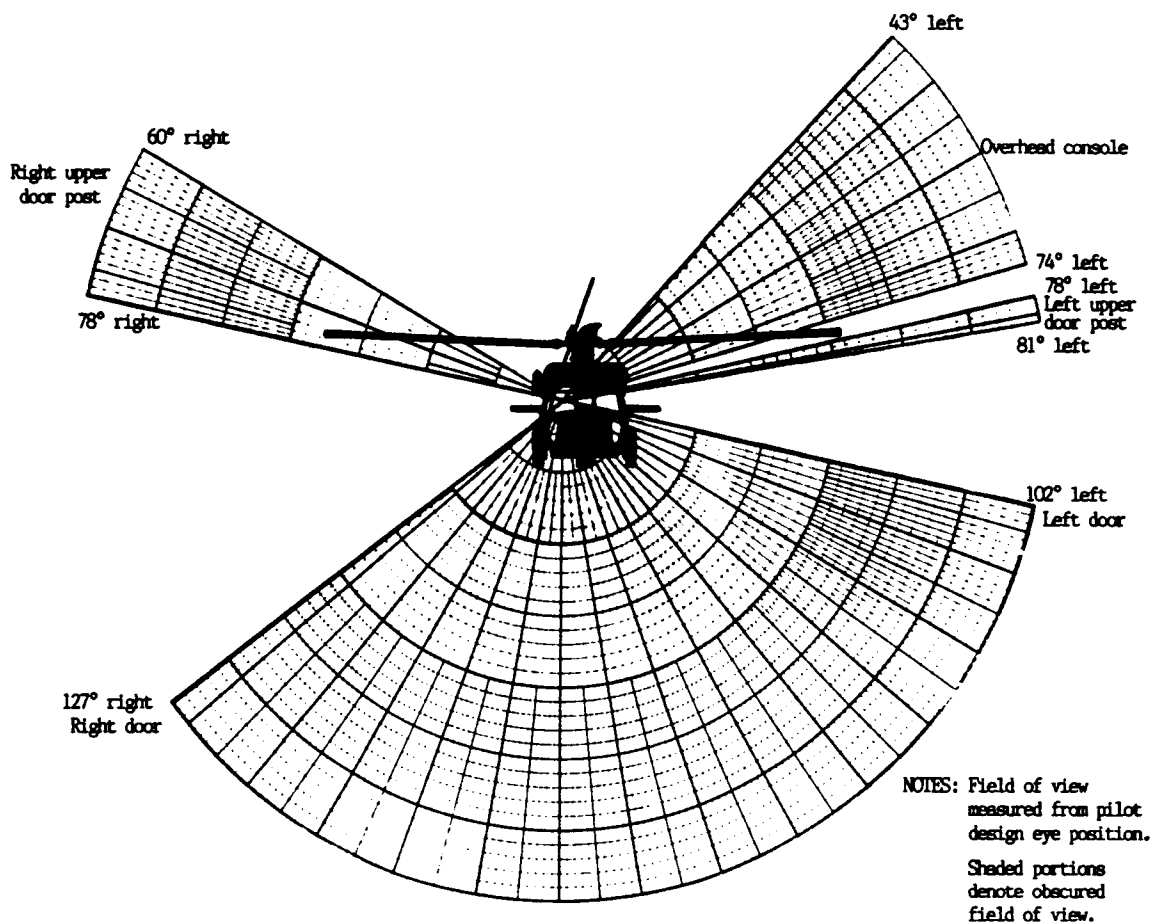


Figure 4
LATERAL FIELD OF VIEW

MODEL SH-60B HELICOPTER

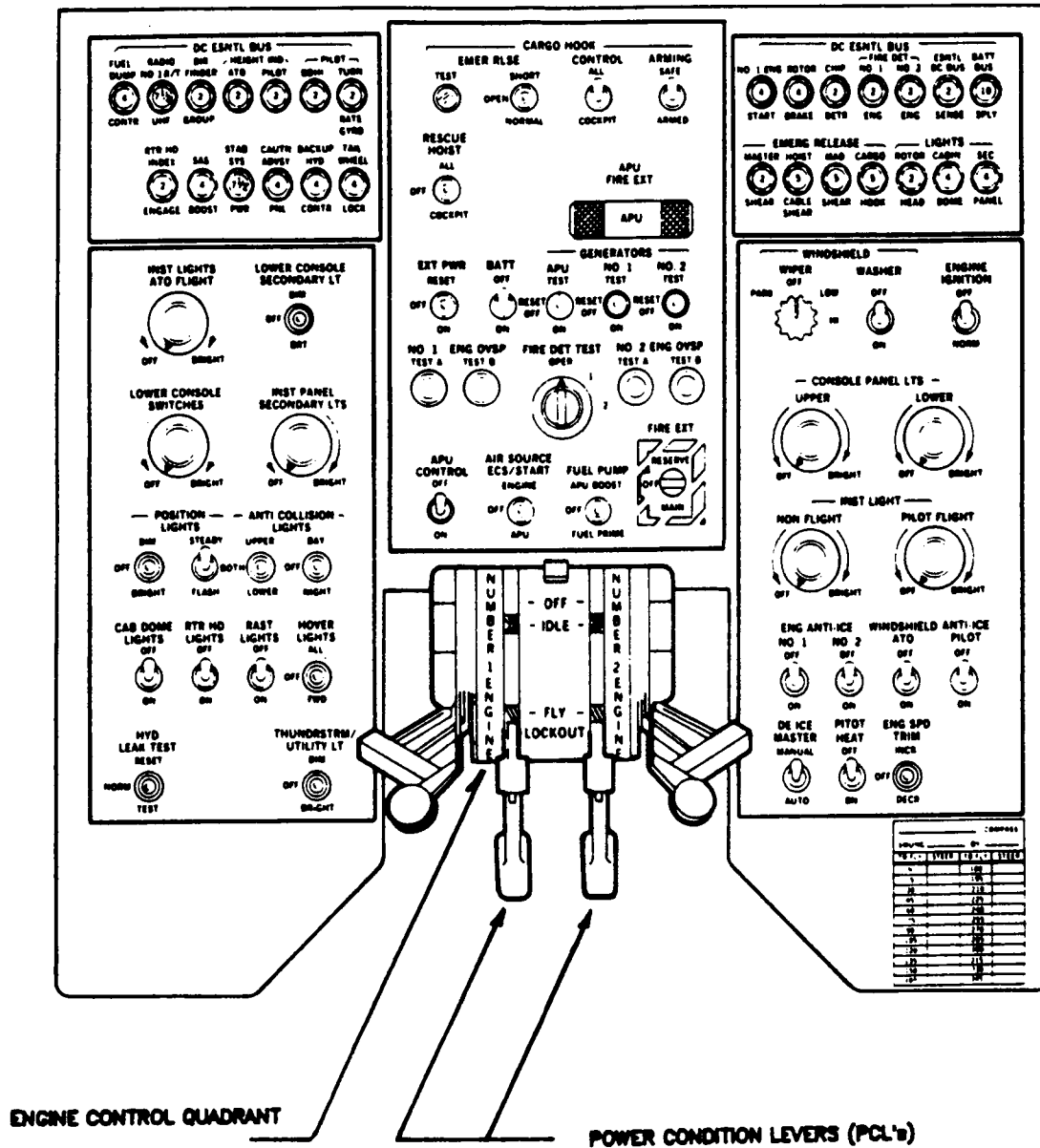


Figure 5
OVERHEAD CONSOLE

MODEL SH-60B HELICOPTER

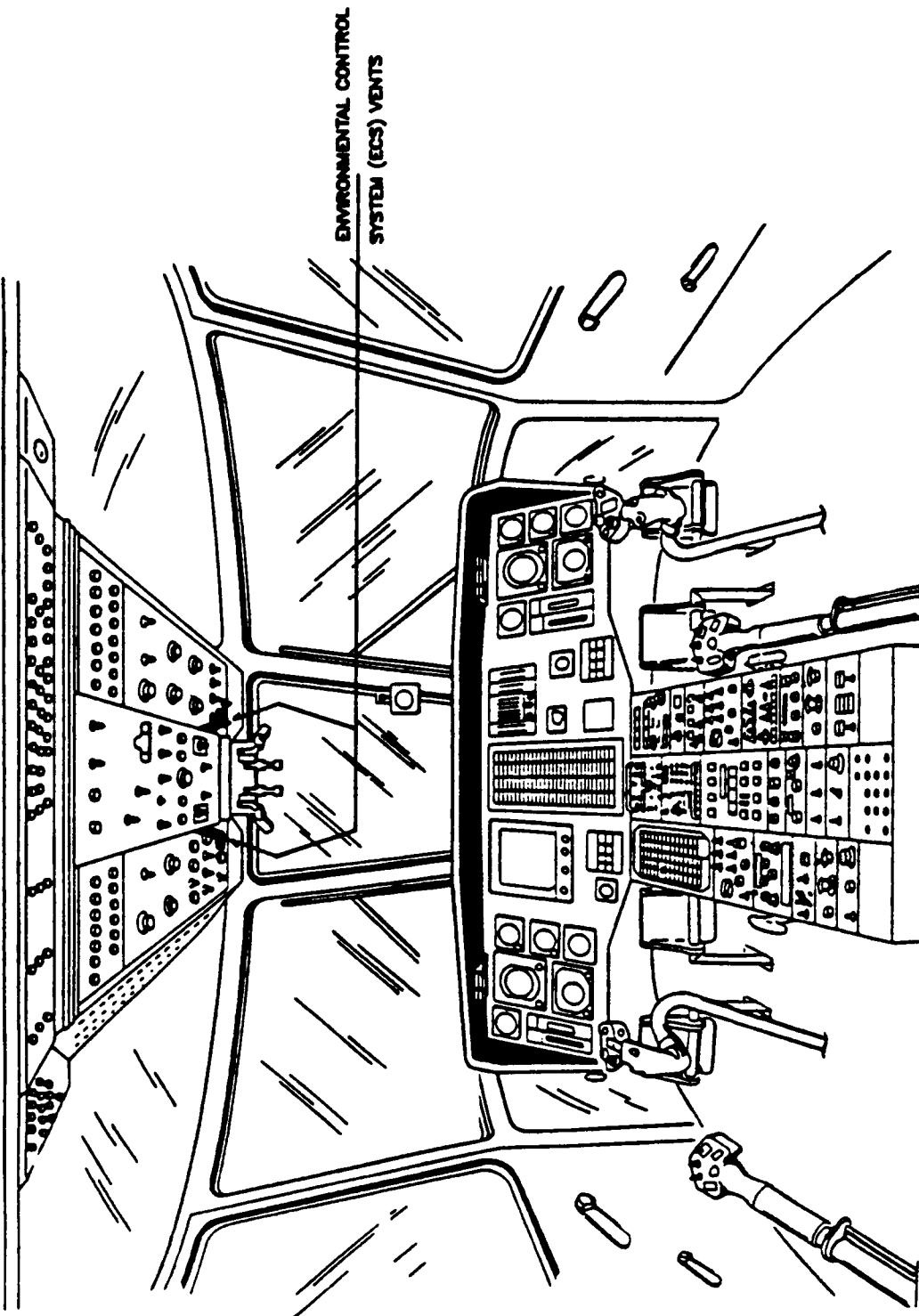


Figure 6
ENVIRONMENTAL CONTROL SYSTEM VENT LOCATIONS

MODEL SH-60B HELICOPTER BuNo 162092

ROTORS: SPREAD/STATIC
ENGINES: OFF
ELECTRIC POWER: EXTERNAL AC
HYDRAULIC POWER: BACKUP PUMP

BOOST: ON
TRIM: ON
SAS 2: ON
ALL REMAINING AFCS COMPONENTS: OFF
NOTE: DIRECTIONAL PEDALS CENTERED
..... COLLECTIVE FULL UP
———— COLLECTIVE MID-TRAVEL
----- COLLECTIVE FULL DOWN

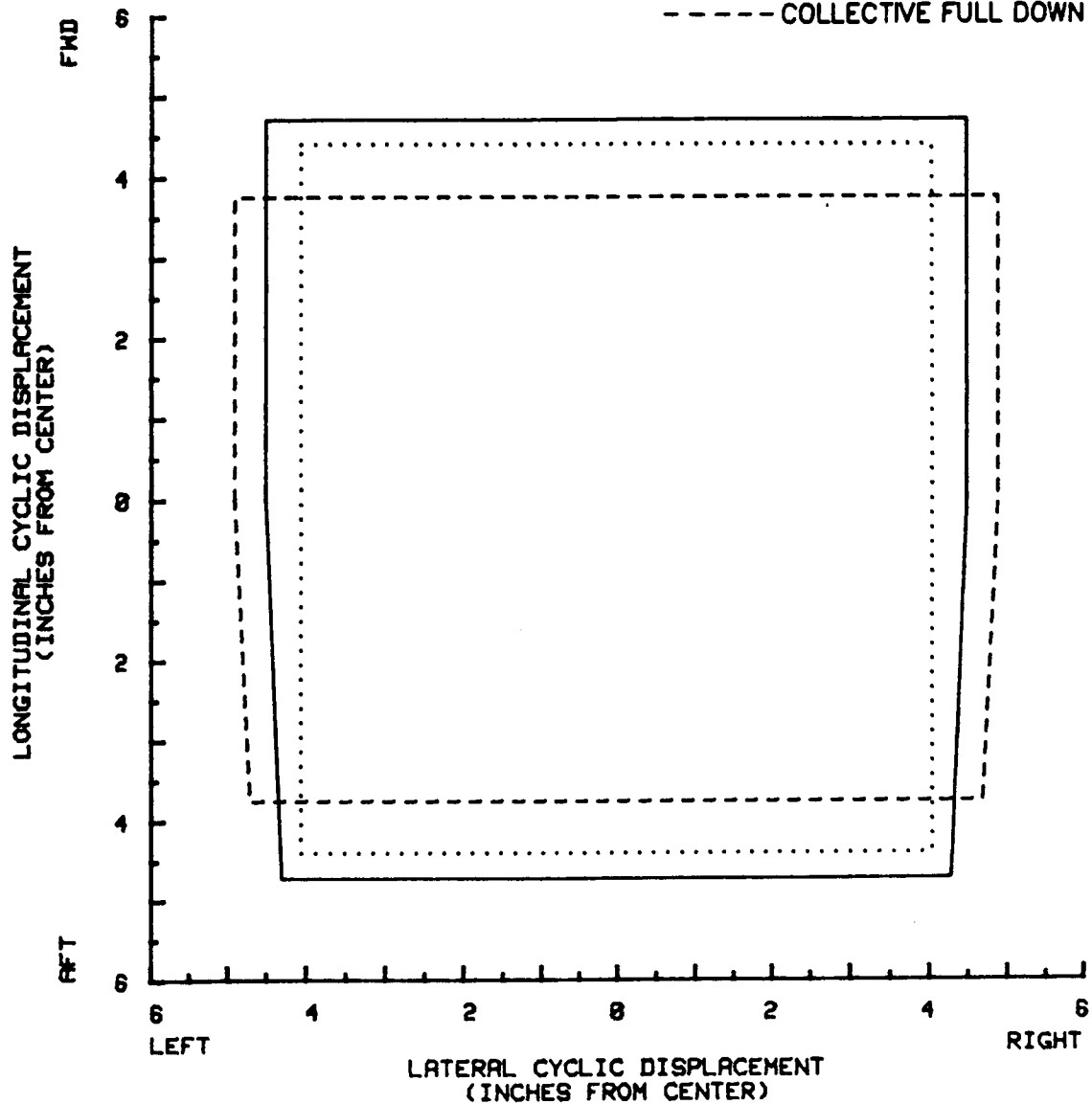


Figure 7
CYCLIC CONTROL ENVELOPE WITH COLLECTIVE DISPLACEMENT

APPENDIX F

MODEL SH-60B HELICOPTER BuNo 162092

ROTORS: SPREAD/STATIC

ENGINES: OFF

ELECTRIC POWER: EXTERNAL AC

HYDRAULIC POWER: BACKUP PUMP

BOOST: ON

TRIM: ON

SAS 2: ON

ALL REMAINING AFCS COMPONENTS: OFF

NOTE: COLLECTIVE CENTERED

..... FULL RIGHT PEDAL APPLIED

———— PEDALS CENTERED

----- FULL LEFT PEDAL APPLIED

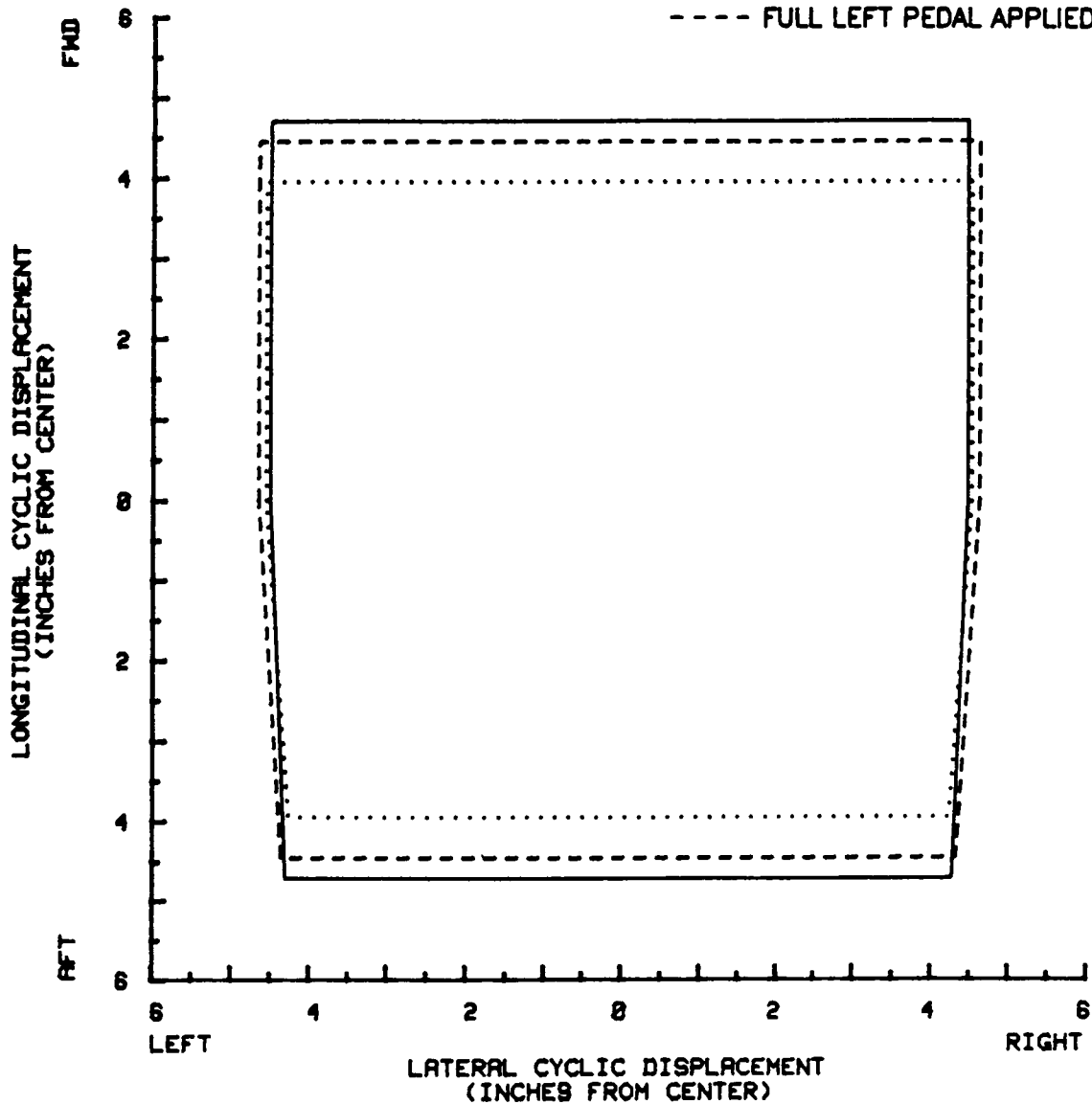


Figure 8
CYCLIC CONTROL ENVELOPE WITH PEDAL DISPLACEMENT

MODEL SH-60B HELICOPTER
BuNo 162092

ROTORS: SPREAD/STATIC

ENGINES: OFF

ELECTRIC POWER: EXTERNAL AC

HYDRAULIC POWER: BACKUP PUMP

BOOST: ON

TRIM: ON

SAS 2: ON

ALL REMAINING AFCS COMPONENTS: OFF

NOTE: CYLIC CENTERED

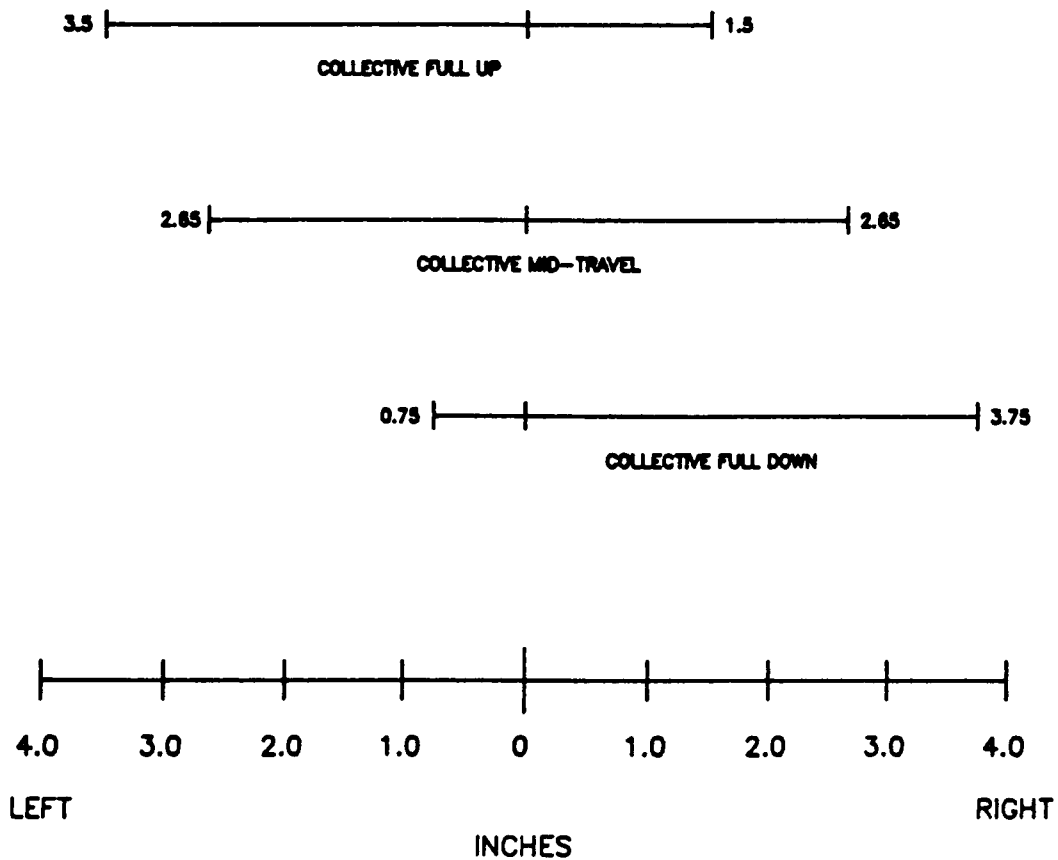


Figure 9
DIRECTIONAL PEDAL CONTROL ENVELOPE

MODEL SH-60B HELICOPTER

BuNo 162092

ROTORS: SPREAD/STATIC

ENGINES: OFF

ELECTRIC POWER: EXTERNAL AC

HYDRAULIC POWER: BACKUP PUMP

BOOST: ON

TRIM: ON

SAS 2: ON

ALL REMAINING AFCS COMPONENTS: OFF

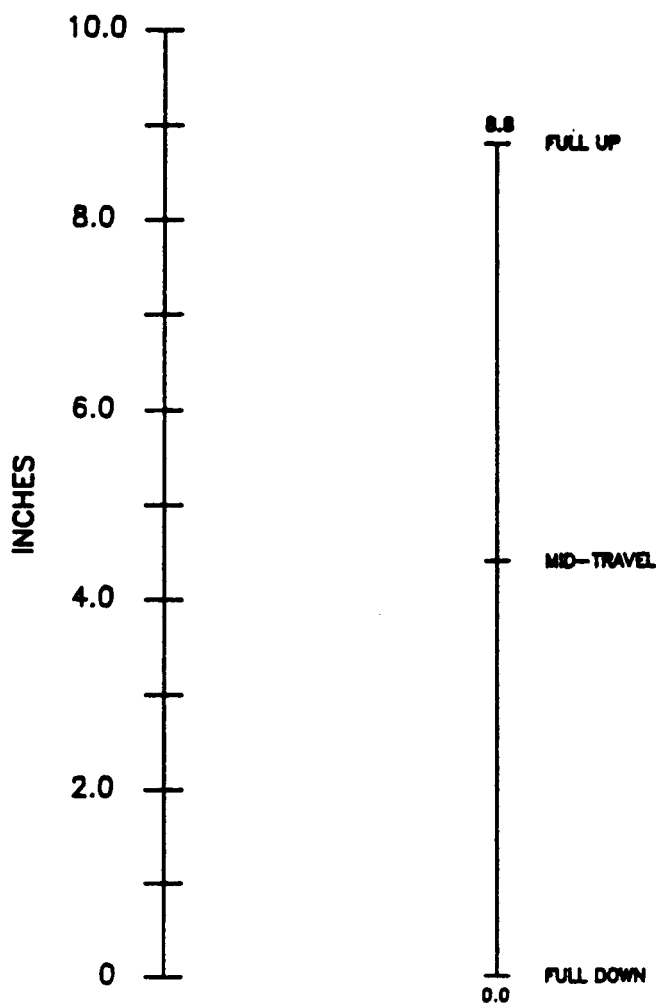


Figure 10
COLLECTIVE CONTROL ENVELOPE

APPENDIX F

MODEL SH-60B HELICOPTER BuNo 162092

ROTORS: SPREAD/STATIC
ENGINES: OFF
ELECTRIC POWER: EXTERNAL AC
HYDRAULIC POWER: BACKUP PUMP

BOOST: ON
TRIM: ON
SAS 2: ON
ALL REMAINING AFCS COMPONENTS: OFF

- NOTE: 1) DASHED LINES DENOTE
MIL-H-8501A SPECIFICATION
2) COLLECTIVE CENTERED
3) DIRECTIONAL PEDALS CENTERED

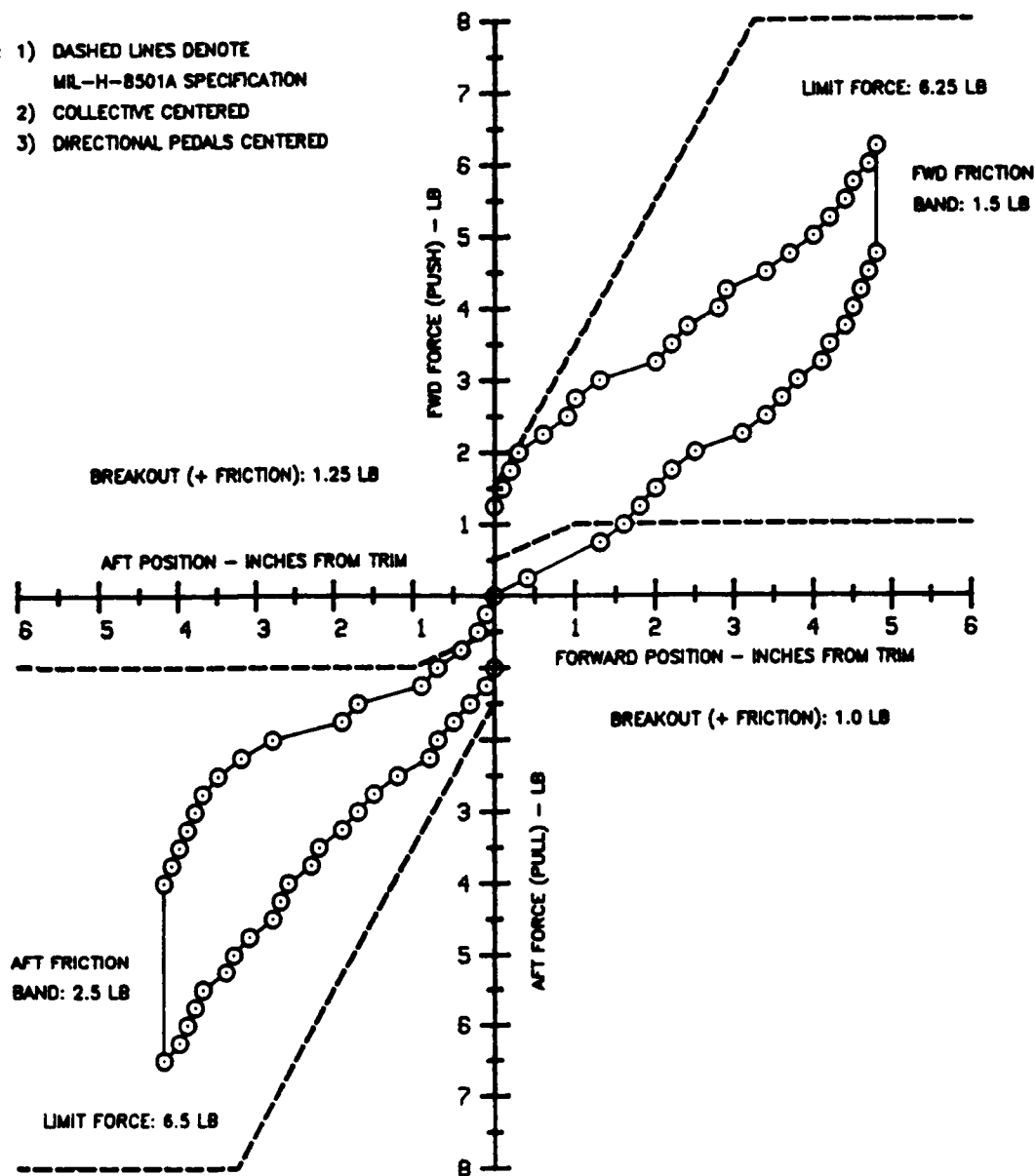


Figure 11
LONGITUDINAL CYCLIC FORCE GRADIENT

MODEL SH-60B HELICOPTER BuNo 162092

ROTORS: SPREAD/STATIC
ENGINES: OFF
ELECTRIC POWER: EXTERNAL AC
HYDRAULIC POWER: BACKUP PUMP

BOOST: ON
TRIM: ON
SAS 2: ON
ALL REMAINING AFCS COMPONENTS: OFF

- NOTE: 1) DASHED LINES DENOTE
MIL-H-8501A SPECIFICATION
2) COLLECTIVE CENTERED
3) DIRECTIONAL PEDALS CENTERED

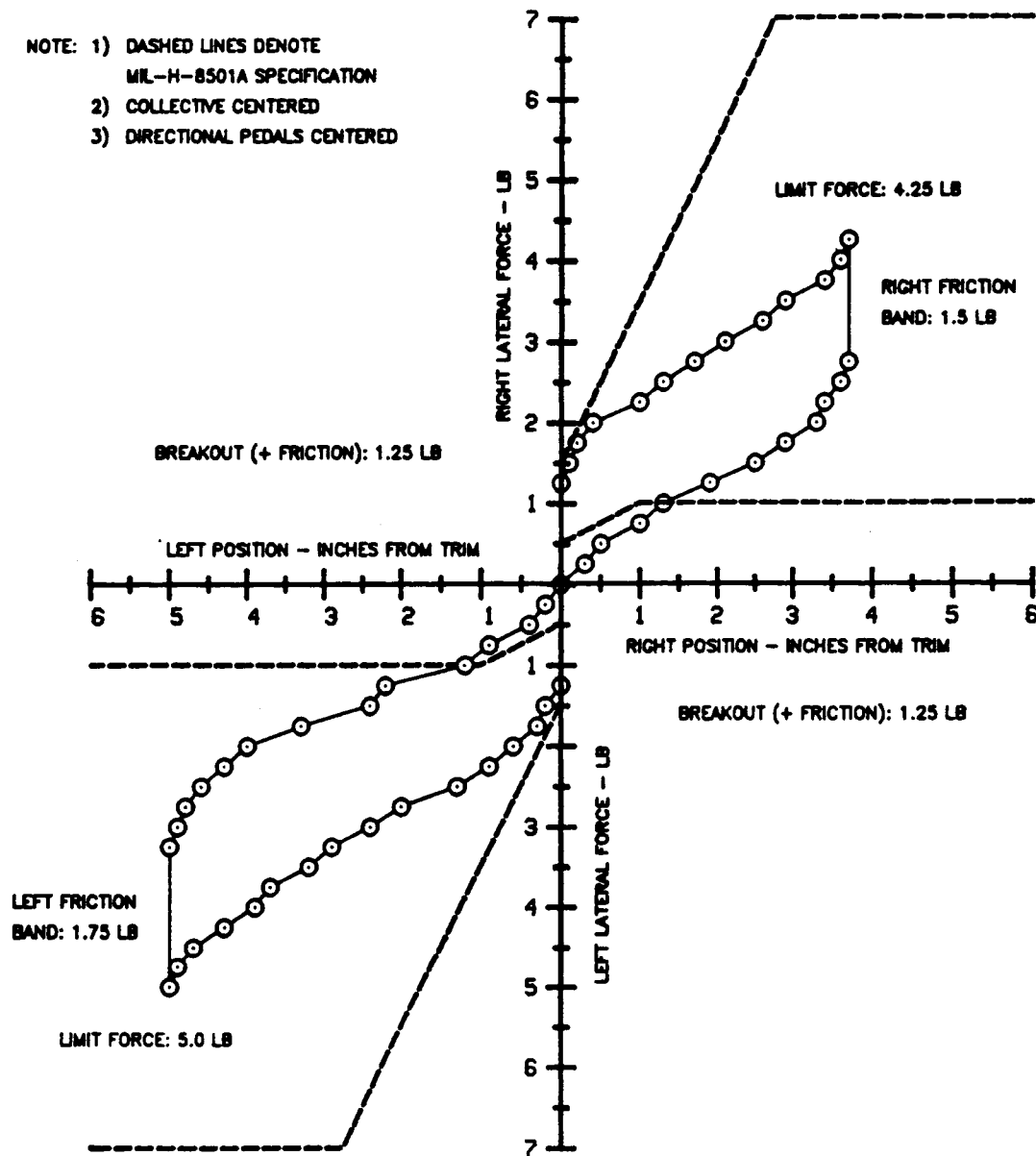


Figure 12
LATERAL CYCLIC FORCE GRADIENT

MODEL SH-60B HELICOPTER BuNo 162092

ROTORS: SPREAD/STATIC
ENGINES: OFF
ELECTRIC POWER: EXTERNAL AC
HYDRAULIC POWER: BACKUP PUMP

BOOST: ON
TRIM: ON
SAS 2: ON
ALL REMAINING AFCS COMPONENTS: OFF

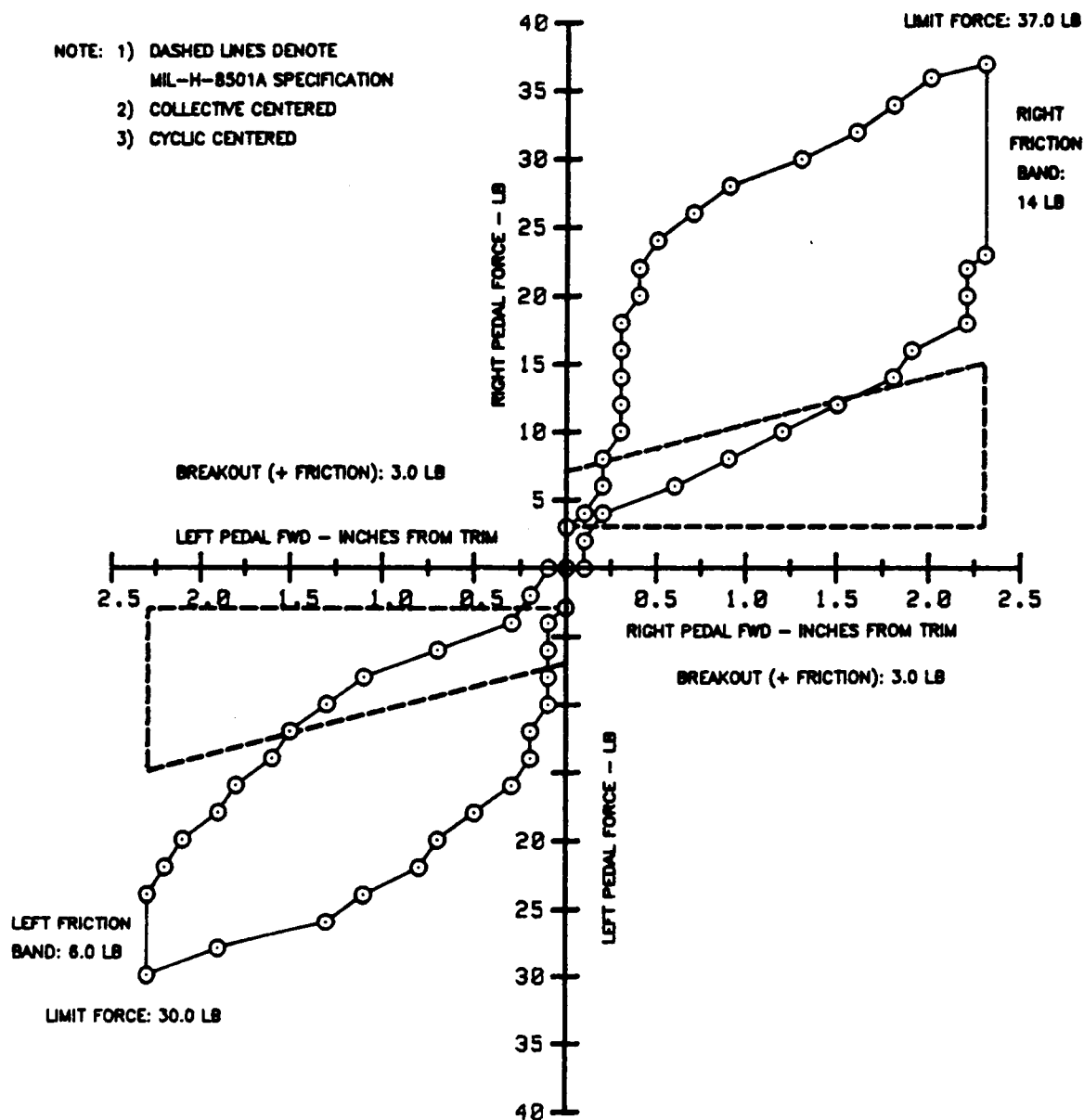


Figure 13
DIRECTIONAL PEDAL FORCE GRADIENT

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST:	ON	GWavg:	18,680 LB
SAS 1/SAS 2:	ON	CGavg:	358.5 IN
TRIM:	ON	Nr:	100%
AUTOPILOT:	ON	Hp:	3000 FT
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	OAT:	16 DEG C
		NOTE:	COLLECTIVE FIXED DIVE TO V _{HE}

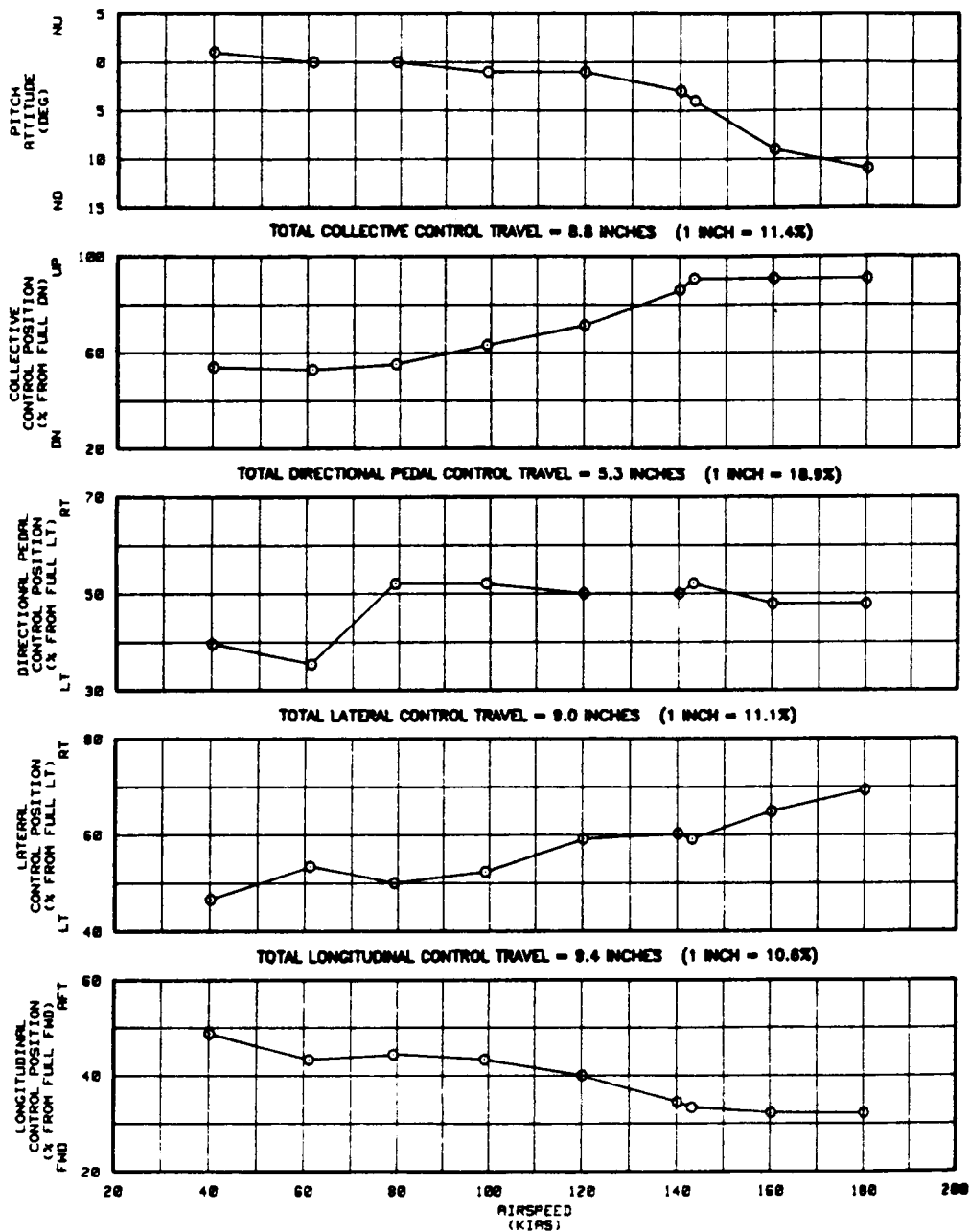


Figure 14
TRIMMED FLIGHT CONTROL POSITIONS - LEVEL FLIGHT

MODEL SH-60B HELICOPTER BuNo 162092

HYDRAULIC BOOST:	ON	GWavg:	17,490 LB
SAS 1/SAS 2:	ON	CGavg:	355.3 IN
TRIM:	ON	Nr:	100%
AUTOPILOT:	ON	Hp:	2000 TO 3000 FT
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	OAT:	19 DEG C
		A/S:	80 KIAS

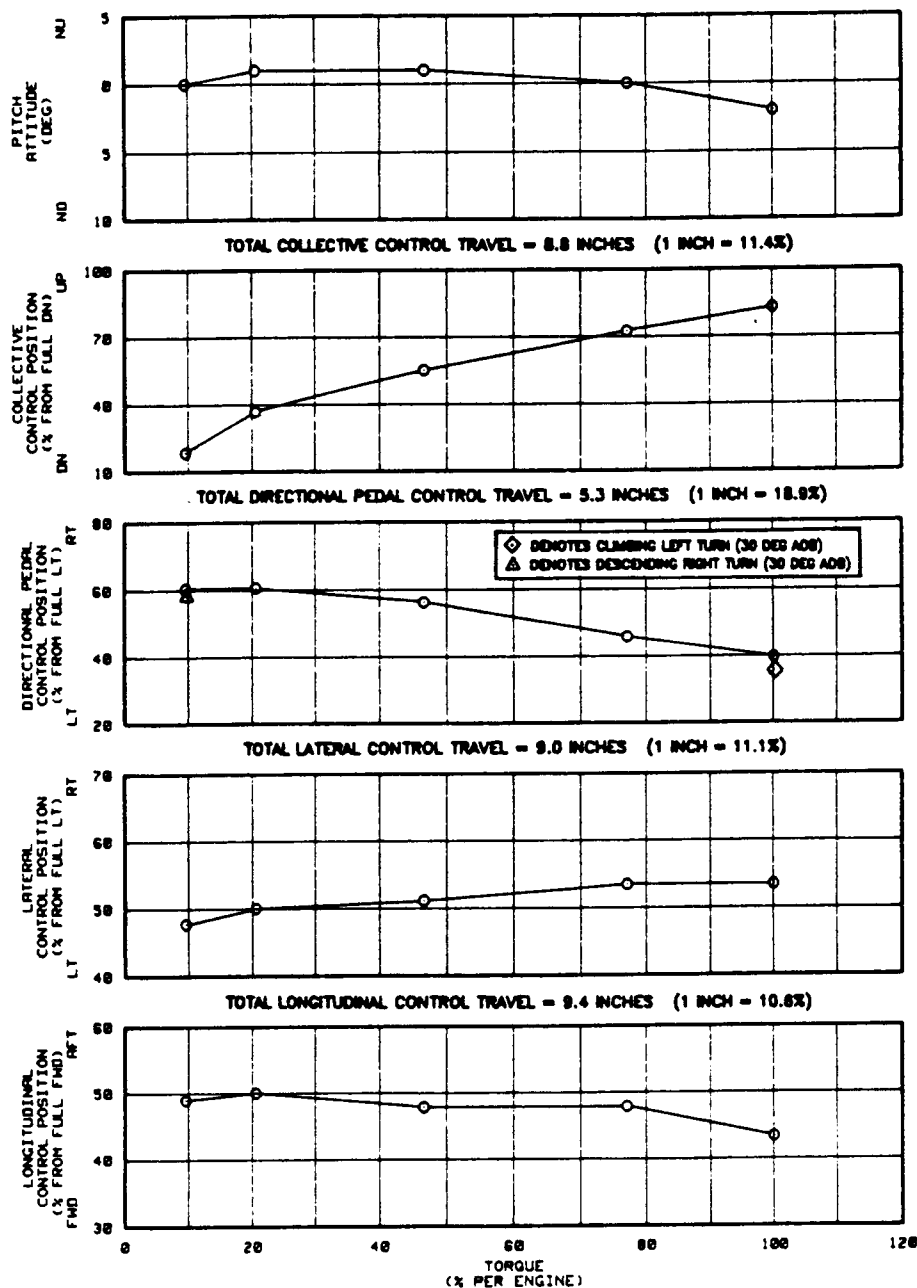


Figure 15
TRIMMED FLIGHT CONTROL POSITIONS - CLIMBS AND DESCENTS

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: 1 MK-46 TORPEDO (STATION 1)

GWavg: 18,200 LB
CGavg: 357.2 IN
Nr: 100%
Hp: 2000 FT
QAT: 20 DEG C
NOTE: $\circ$ = 80 KIAS $\square$ = 140 KIAS

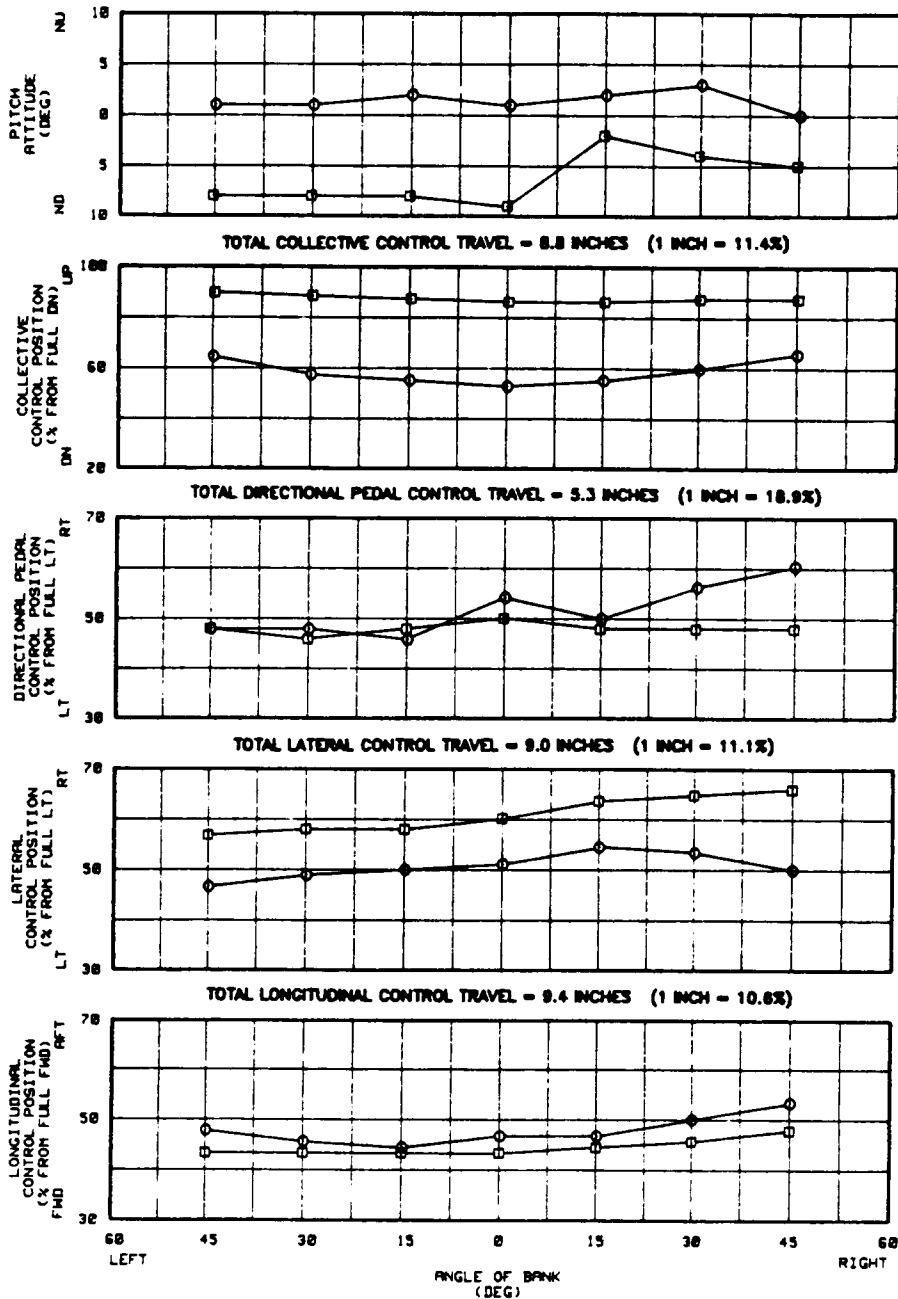


Figure 16
TRIMMED FLIGHT CONTROL POSITIONS - LEVEL TURNS

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: 1 MK-46 TORPEDO (STATION 1)

GWavg: 18,290 LB
CGavg: 357.5 IN
Nr: 100%
Hp: 2000 FT
OAT: 20 DEG C
NOTE: 80/140 KIAS TRIM
(SHADED POINTS DENOTE TRIM)

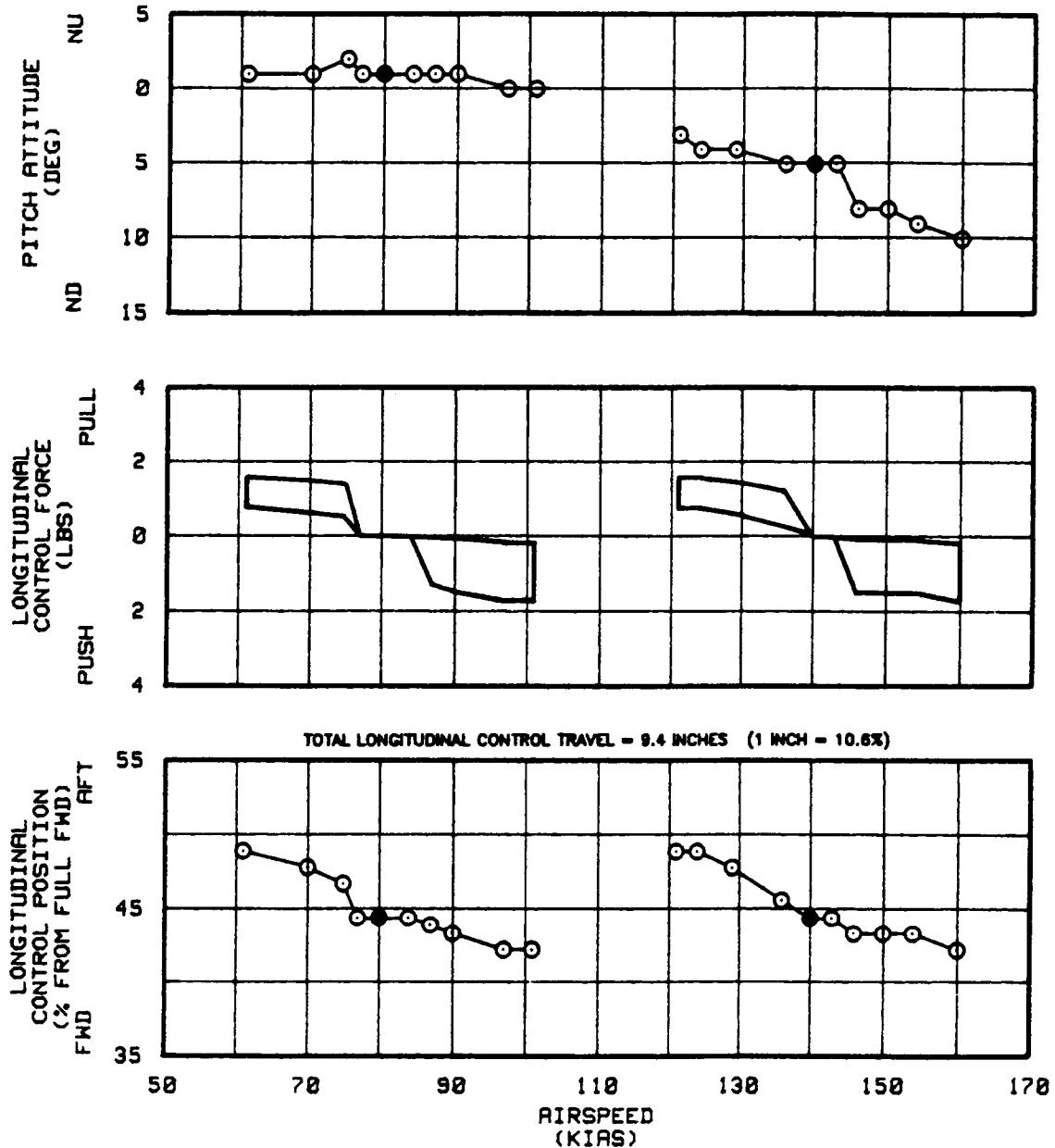


Figure 17
STATIC LONGITUDINAL STABILITY

MODEL SH-60B HELICOPTER

BuNo 162092

HYDRAULIC BOOST: ON

SAS 1/SAS 2: ON

TRIM: ON

AUTOPILOT: ON

CONFIGURATION: NO EXTERNAL STORES

GWavg: 16,530 LB

CGavg: 353.9 IN

Nr: 100%

Hp: 2000 FT

OAT: 22 DEG C

A/S: 140 KIAS

NOTE: □ = LEFT AOB ⊙ = RIGHT AOB
SHADED POINTS DENOTE TRIM

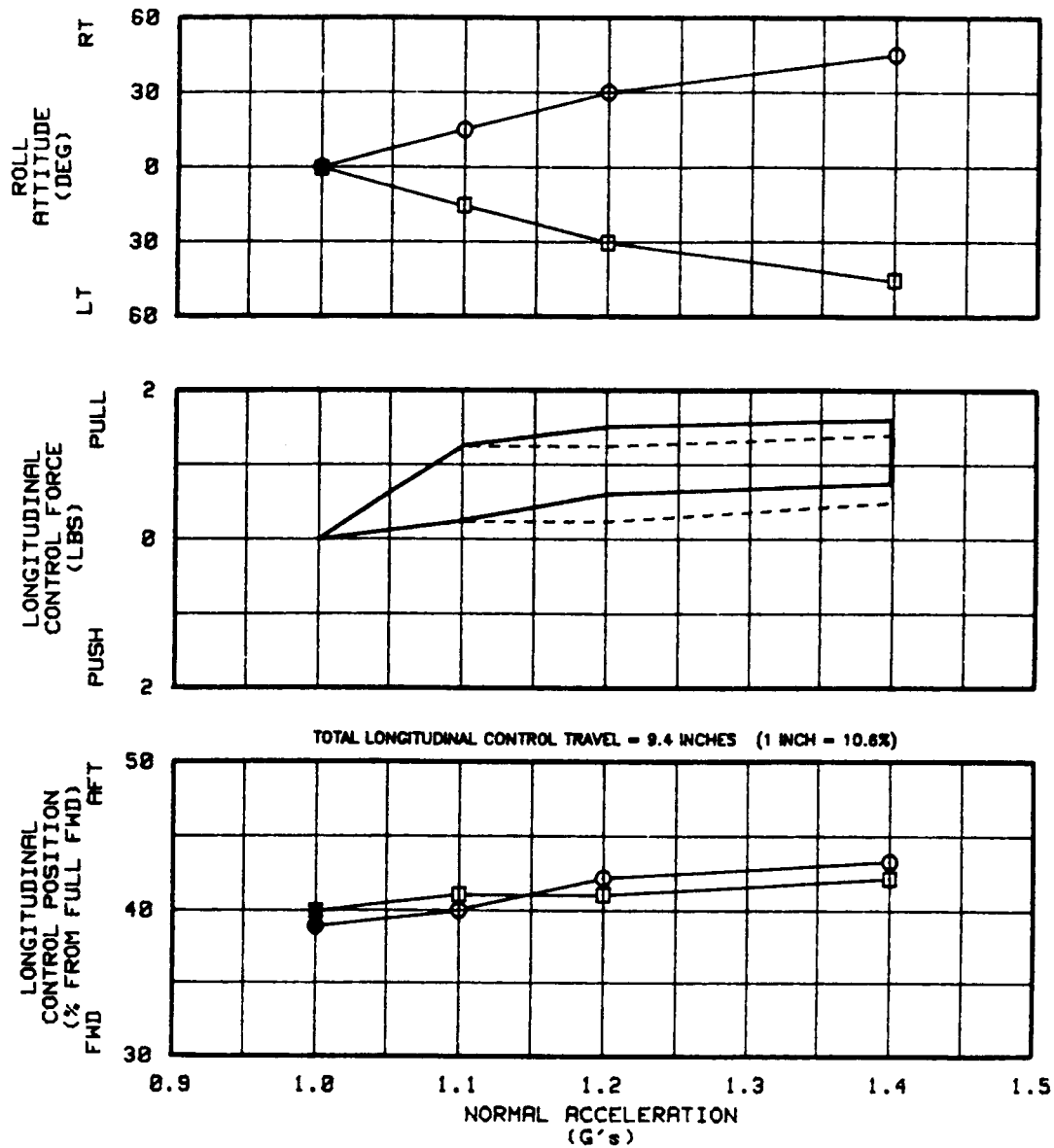


Figure 18
MANEUVERING STABILITY - STEADY TURNS

APPENDIX F

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 16,310 LB
CGavg: 353.3 IN
Nr: 100%
Hp: 2000 FT
OAT: 22 DEG C
A/S: 140 KIAS
NOTE: SHADED POINT DENOTES TRIM

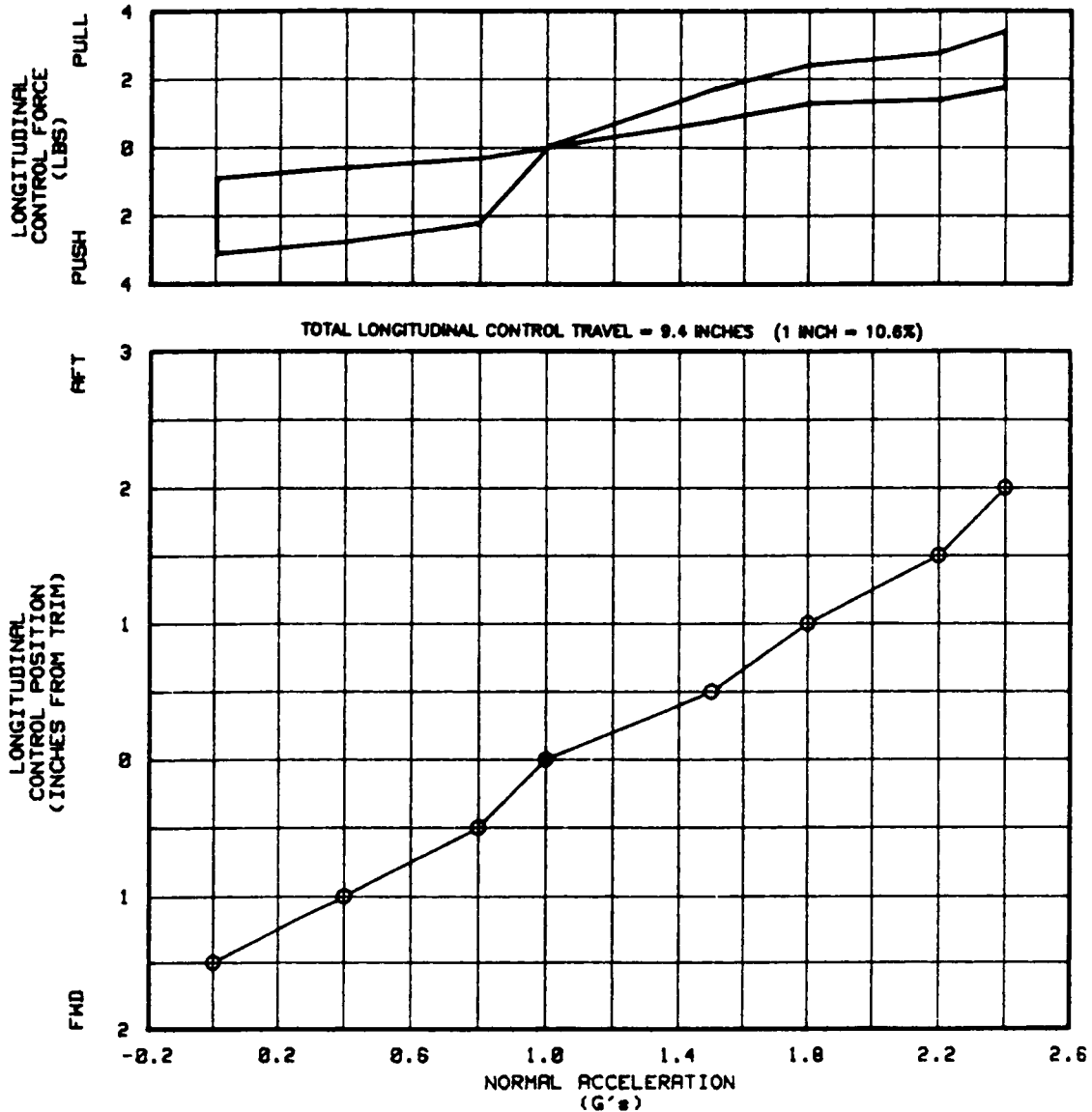


Figure 19
MANEUVERING STABILITY - PULL UPS AND PUSH OVERS

APPENDIX F

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 17,690 LB
CGavg: 357.0 IN
Nr: 100%
Hp: 2000 FT
QAT: 20 DEG C
NOTE: $\circ$ = 80 KIAS $\square$ = 140 KIAS
SHADED POINTS DENOTE TRIM

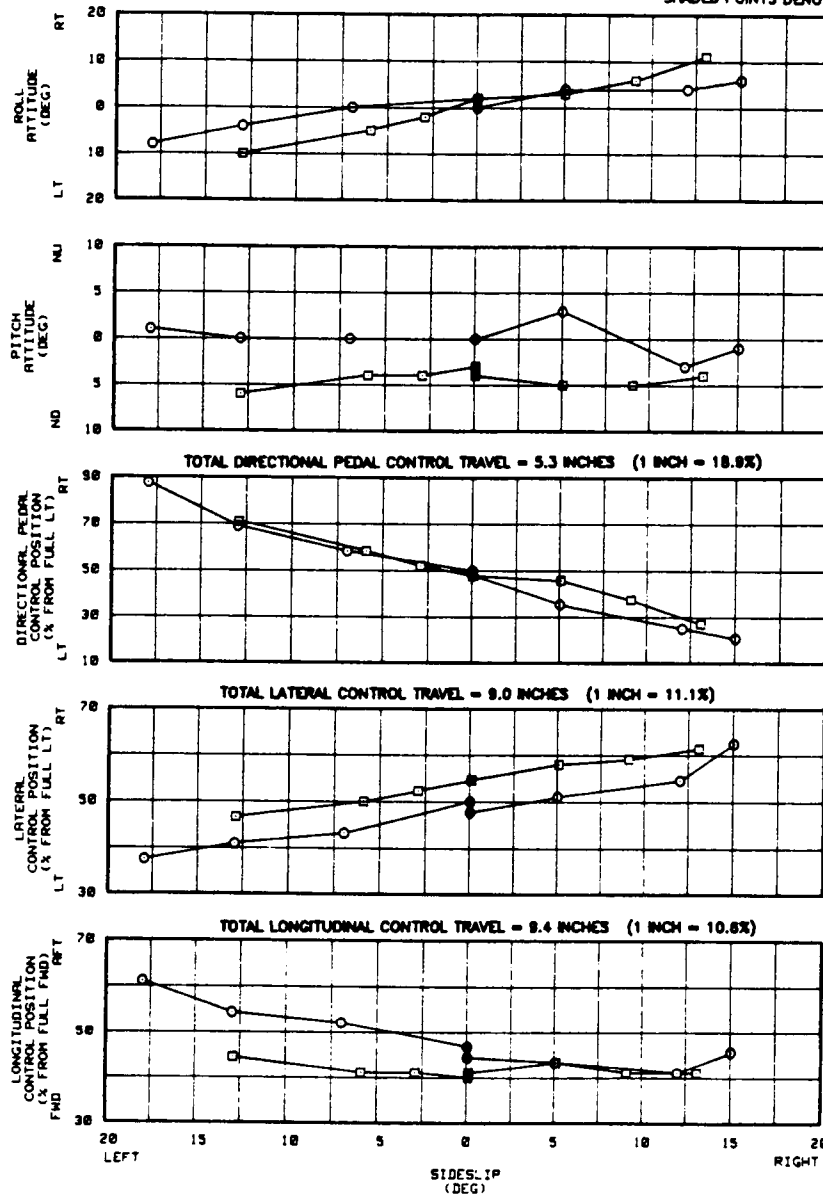


Figure 20
STATIC LATERAL-DIRECTIONAL STABILITY -
STEADY HEADING SIDESLIPS

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST:	ON	GWavg:	18,710 LB
SAS 1/SAS 2:	ON	CGavg:	359.8 IN
TRIM:	ON	Nr:	100%
AUTOPILOT:	ON	Hp:	80 FT
CONFIGURATION:	NO EXTERNAL STORES	OAT:	20 DEG C
		SKID HGT:	15 FTegl (RADALT)
		NOTE:	GROUND SPEED VIA A/C DOPPLER
			WIND SPEED APPROXIMATELY 0 KTS

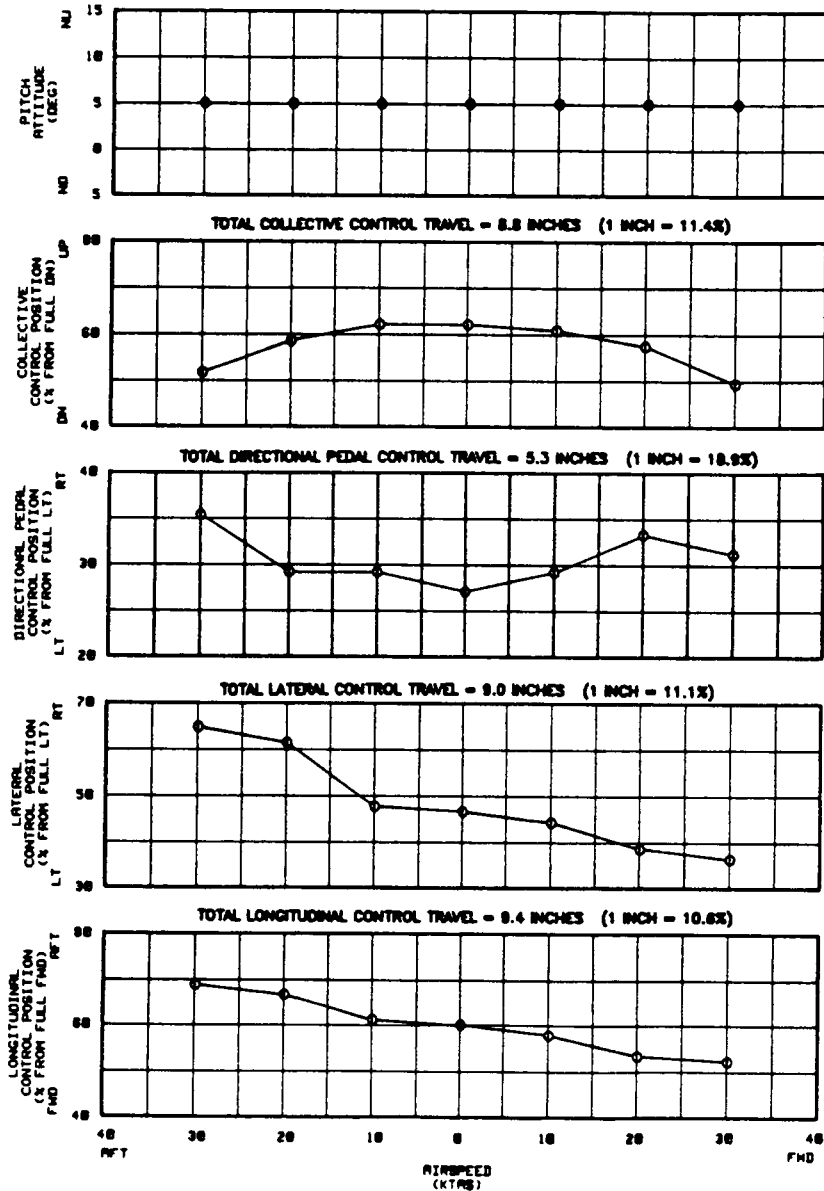


Figure 21
LOW AIRSPEED TRIMMED FLIGHT CONTROL POSITIONS -
FORWARD AND AFT

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 18,710 LB
CGavg: 350.8 IN
Nr: 100%
Hp: 80 FT
OAT: 20 DEG C
SKD HGT: 15 FTg (RADALT)
NOTE: GROUND SPEED VIA A/C DOPPLER
WIND SPEED APPROXIMATELY 0 KTS

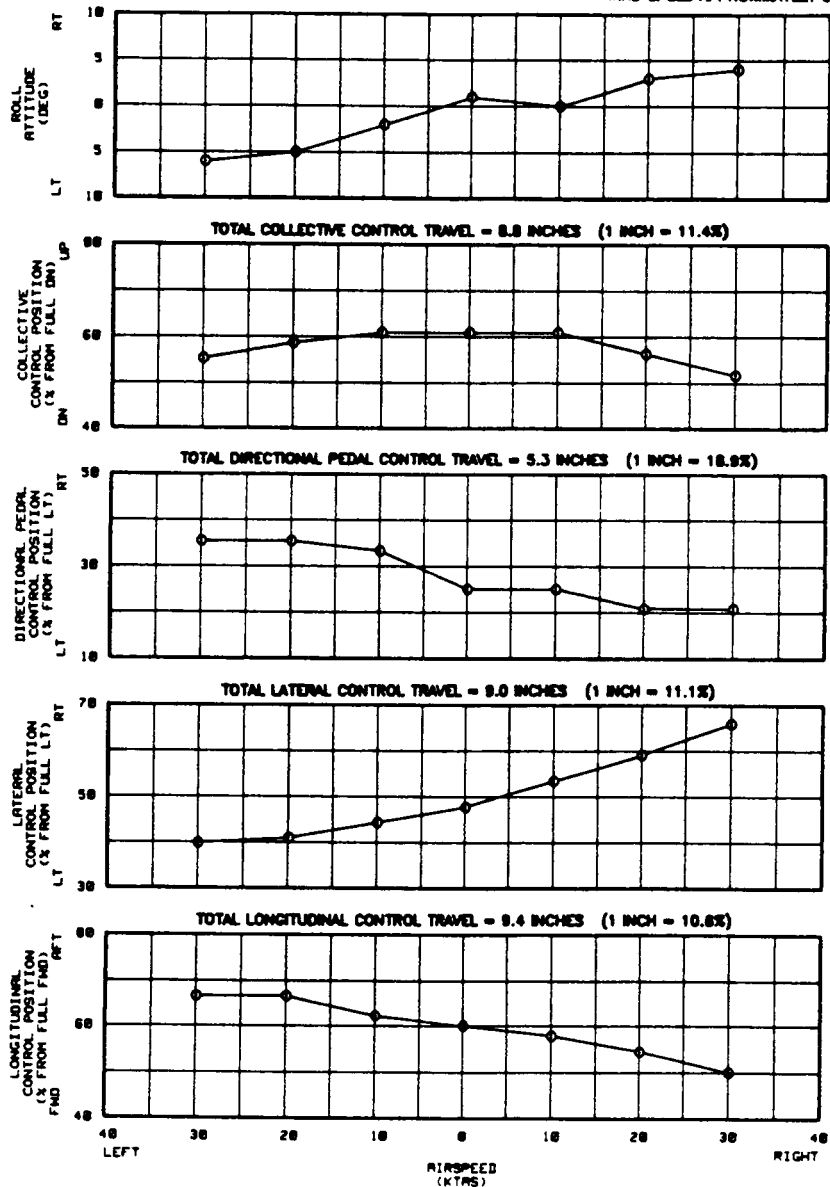


Figure 22
LOW AIRSPEED TRIMMED FLIGHT CONTROL POSITIONS -
LEFT AND RIGHT

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 18,420 LB
CGavg: 358.0 IN
Nr: 100%
Hp: 60 FT
OAT: 20 DEG C
SKD HGT: 15 FTegl (RADALT)
GRND SPD: 30 KTAS (DOPPLER)
NOTE: WIND SPEED APPROXIMATELY 0 KTS

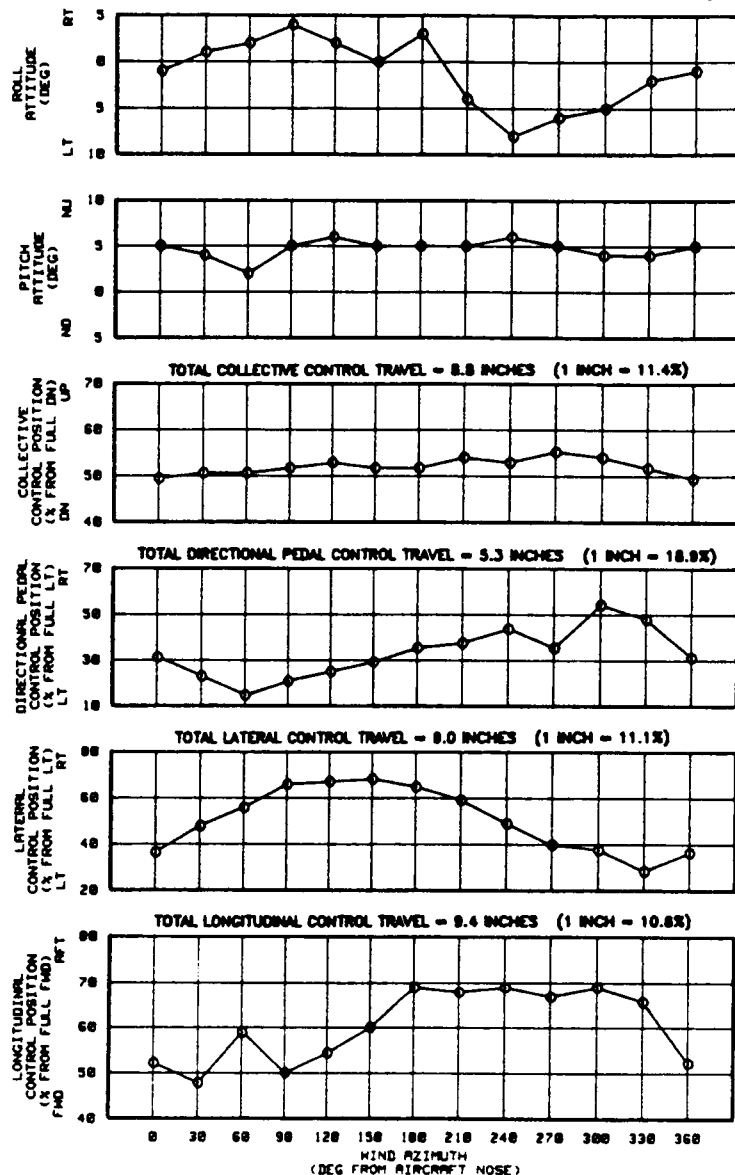


Figure 23
LOW AIRSPEED TRIMMED FLIGHT CONTROL POSITIONS
WITH WIND AZIMUTH

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 18,420 LB
CGavg: 358.0 IN
Nr: 100%
Hp: 60 FT
QAT: 20 DEG C
SKID HGT: 15 FTegl (RADALT)
GRND SPD: 30 KTAS (DOPPLER)
NOTE: WIND SPEED APPROXIMATELY 0 KTS

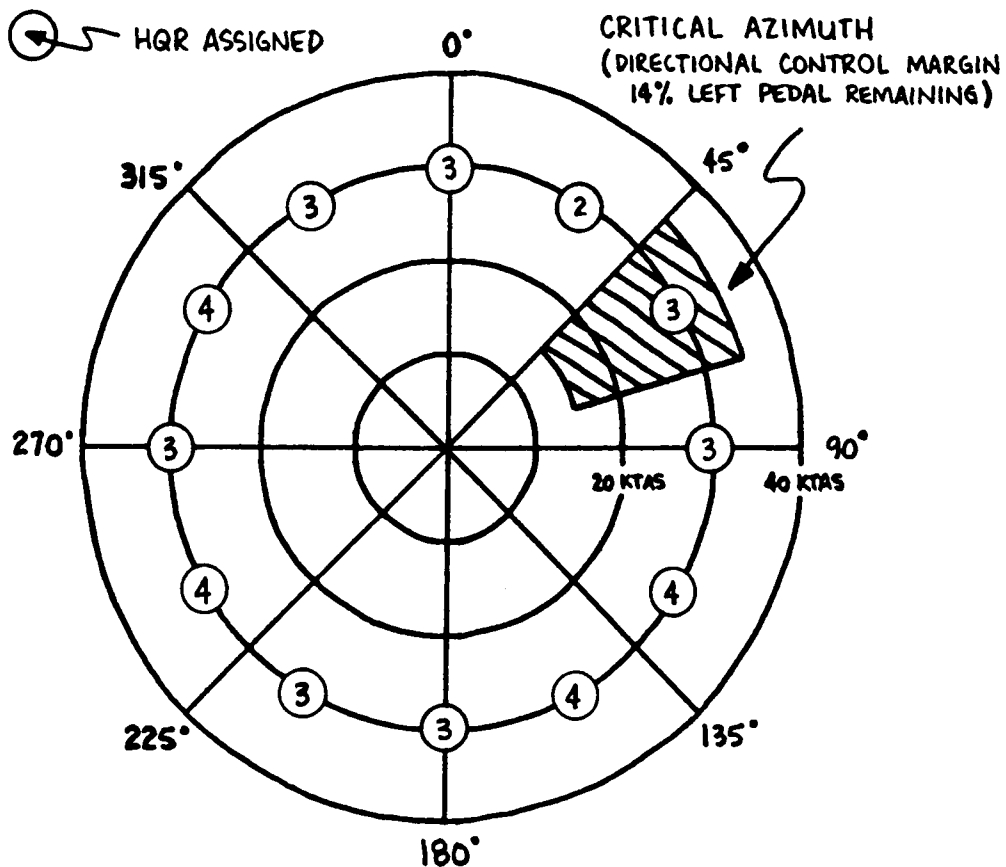


Figure 24
HANDLING QUALITIES RATINGS WITH WIND AZIMUTH

MODEL SH-60B HELICOPTER

BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 18,150 LB
CGavg: 358.3 IN
Nr: 100%
Hp: 60 FT
QAT: 20 DEG C
NOTE: GROUND SPEED VIA A/C DOPPLER
WIND SPEED APPROXIMATELY 0 KTS
SHADED POINTS DENOTE TRIM

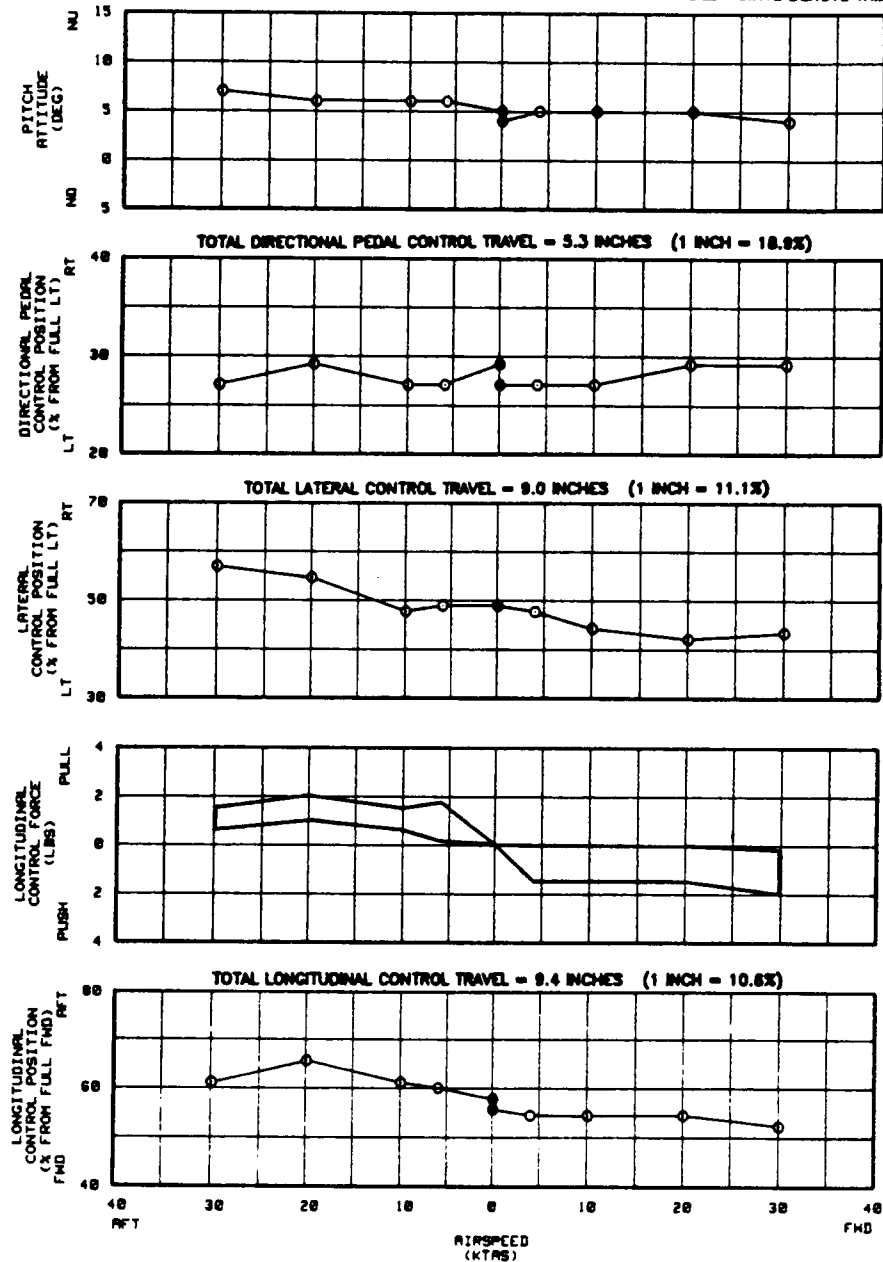
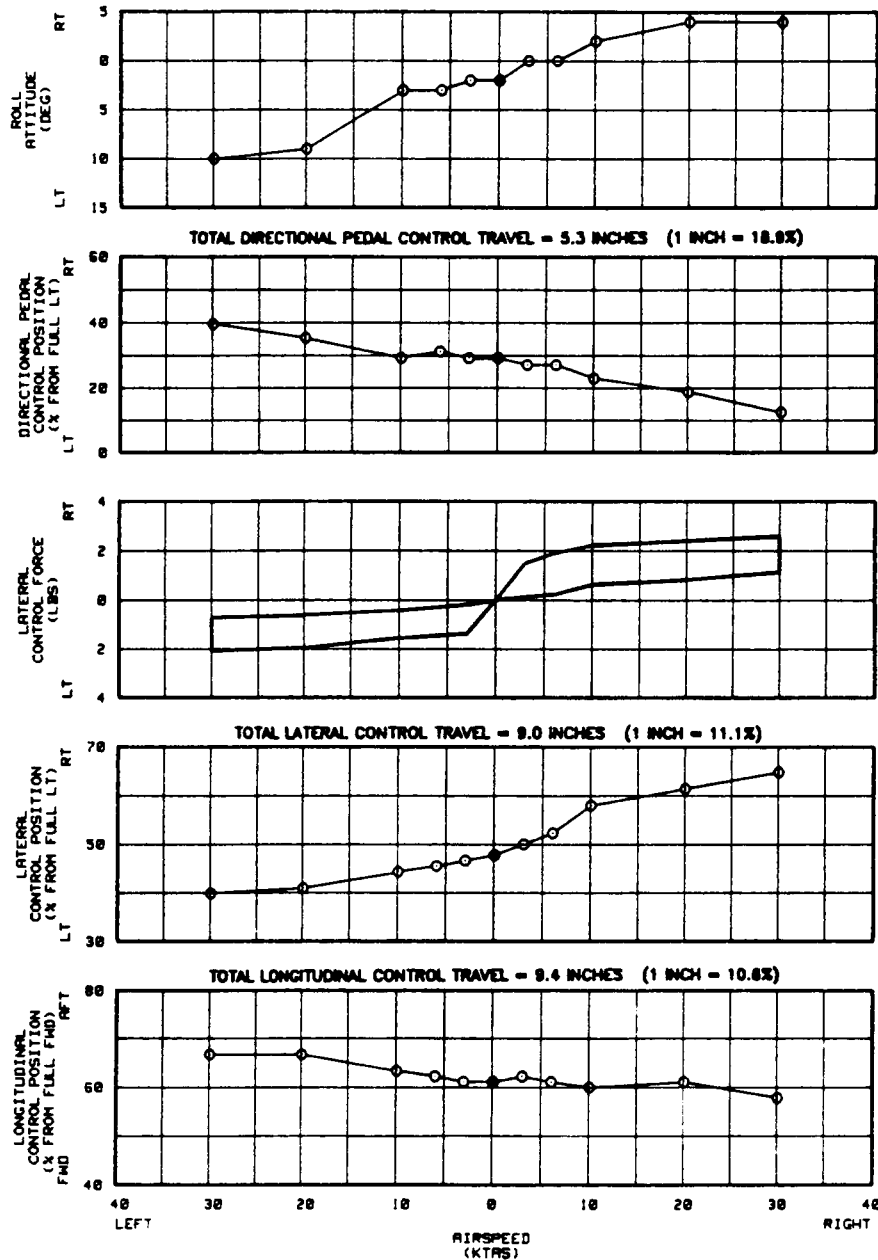


Figure 25
LOW AIRSPEED STATIC LONGITUDINAL STABILITY

MODEL SH-60B HELICOPTER
BuNo 162092

HYDRAULIC BOOST: ON
SAS 1/SAS 2: ON
TRIM: ON
AUTOPILOT: ON
CONFIGURATION: NO EXTERNAL STORES

GWavg: 18,150 LB
CGavg: 358.3 IN
Nr: 100%
Hp: 80 FT
QAT: 20 DEG C
NOTE: GROUND SPEED VIA A/C DOPPLER
WIND SPEED APPROXIMATELY 0 KTS
SHADED POINTS DENOTE TRIM



MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	GWavg:	19,360 LB
SERIAL NUMBERS:	#1 212267	CGavg:	360.3 IN
	#2 212257	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	40 TO 100 FT
ENGINE BLEED AIR:	OFF	OAT:	22 DEG C
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG RELATIVE TO A/C
		METHOD:	FREE HOVER
		NOTE:	RADALT USED FOR HMMO

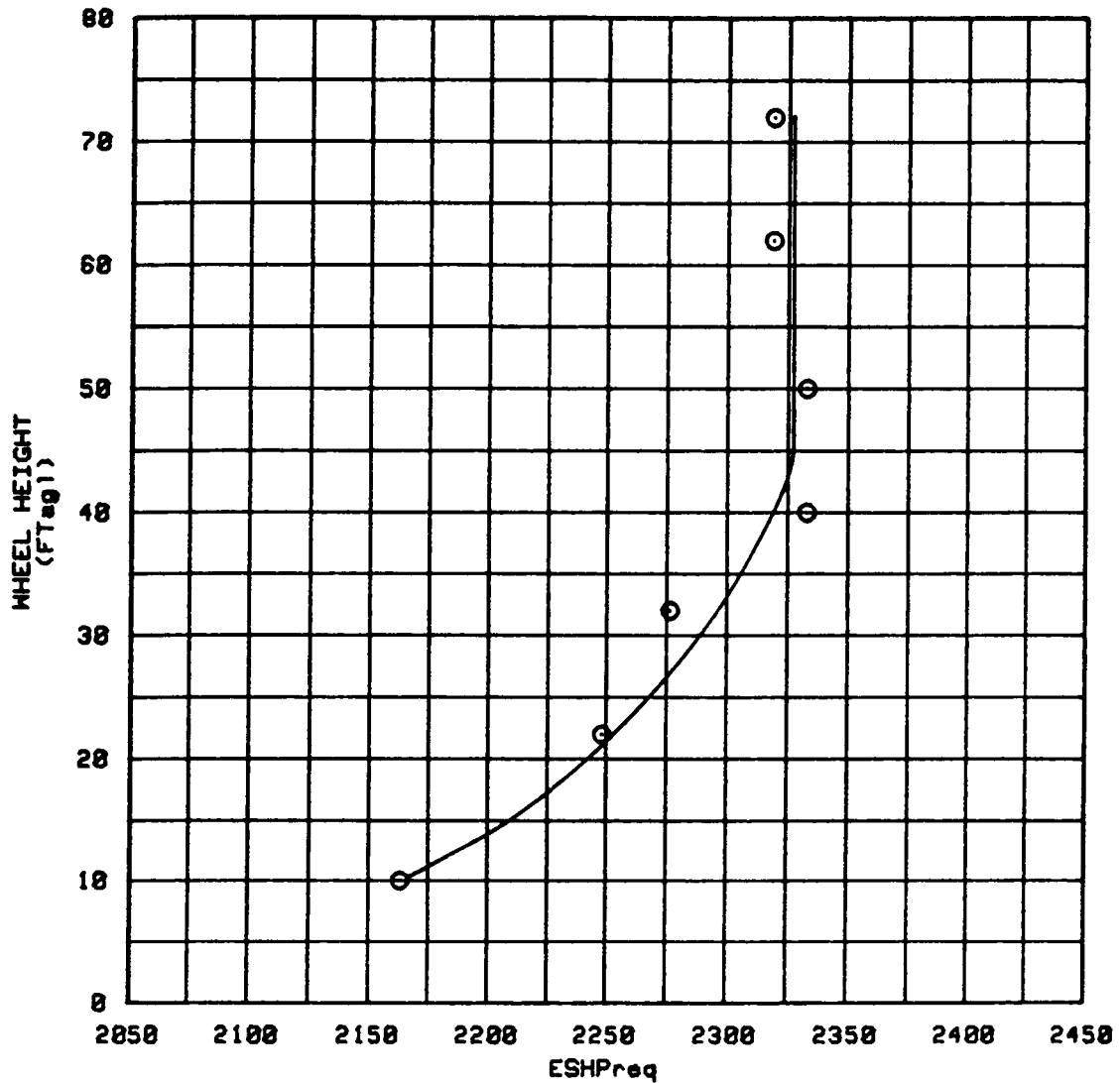


Figure 27
IGE/OGE TRANSITION DETERMINATION

MODEL SH-60B HELICOPTER BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (8.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	STD DAY (15 DEG C)
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
GWunreferred:	17,500 19,460	METHOD:	FREE HOVER (IGE - 15 FTagl)
	20,800 21,700 LB	NOTE:	RADALT USED FOR HHMD

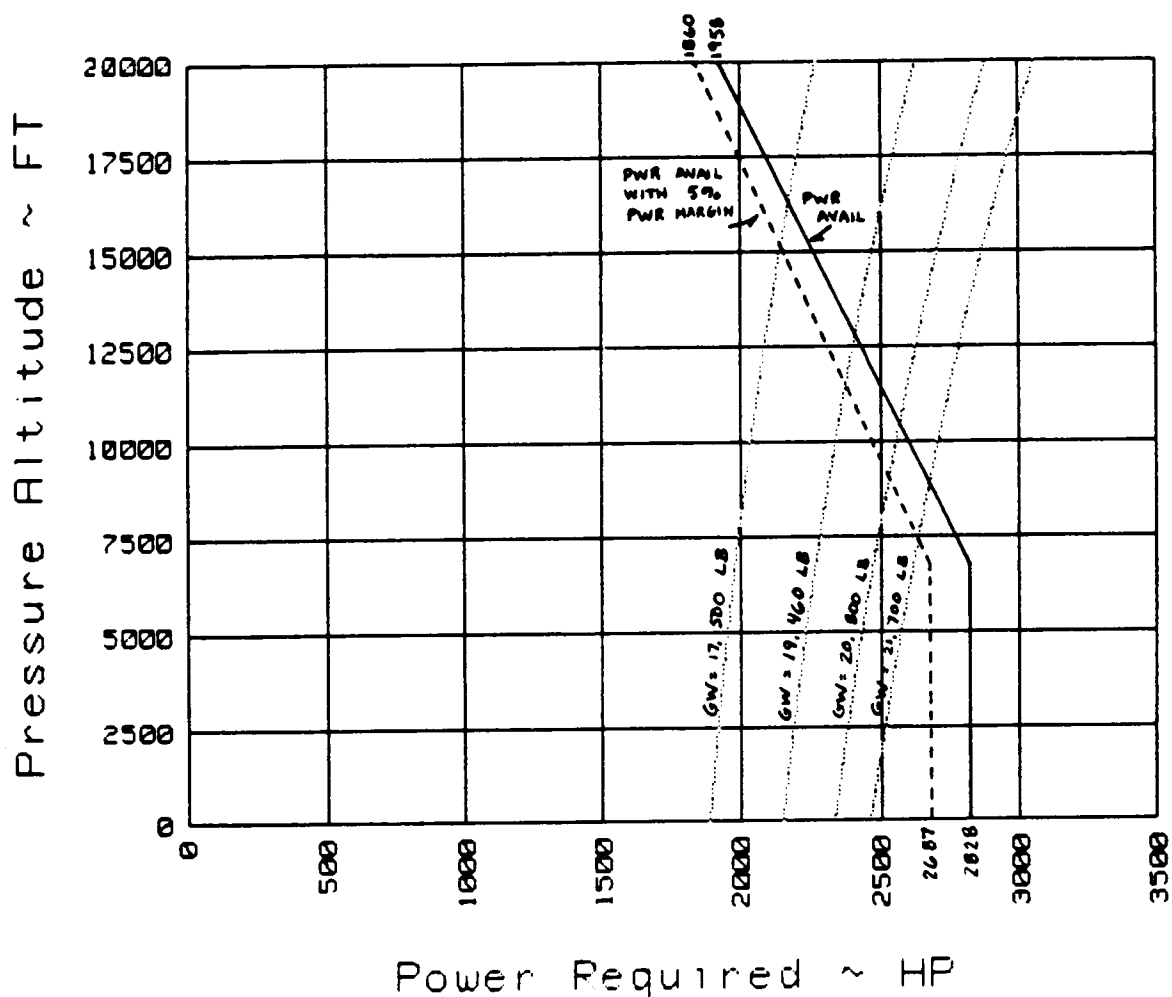


Figure 28
POWER REQUIRED WITH ALTITUDE - IGE STANDARD DAY

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	HOT DAY (35 DEG C)
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
GWunreferred:	17,500 19,480	METHOD:	FREE HOVER (IGE - 15 FTeg)
	20,800 21,700 LB	NOTE:	RADALT USED FOR HHMD

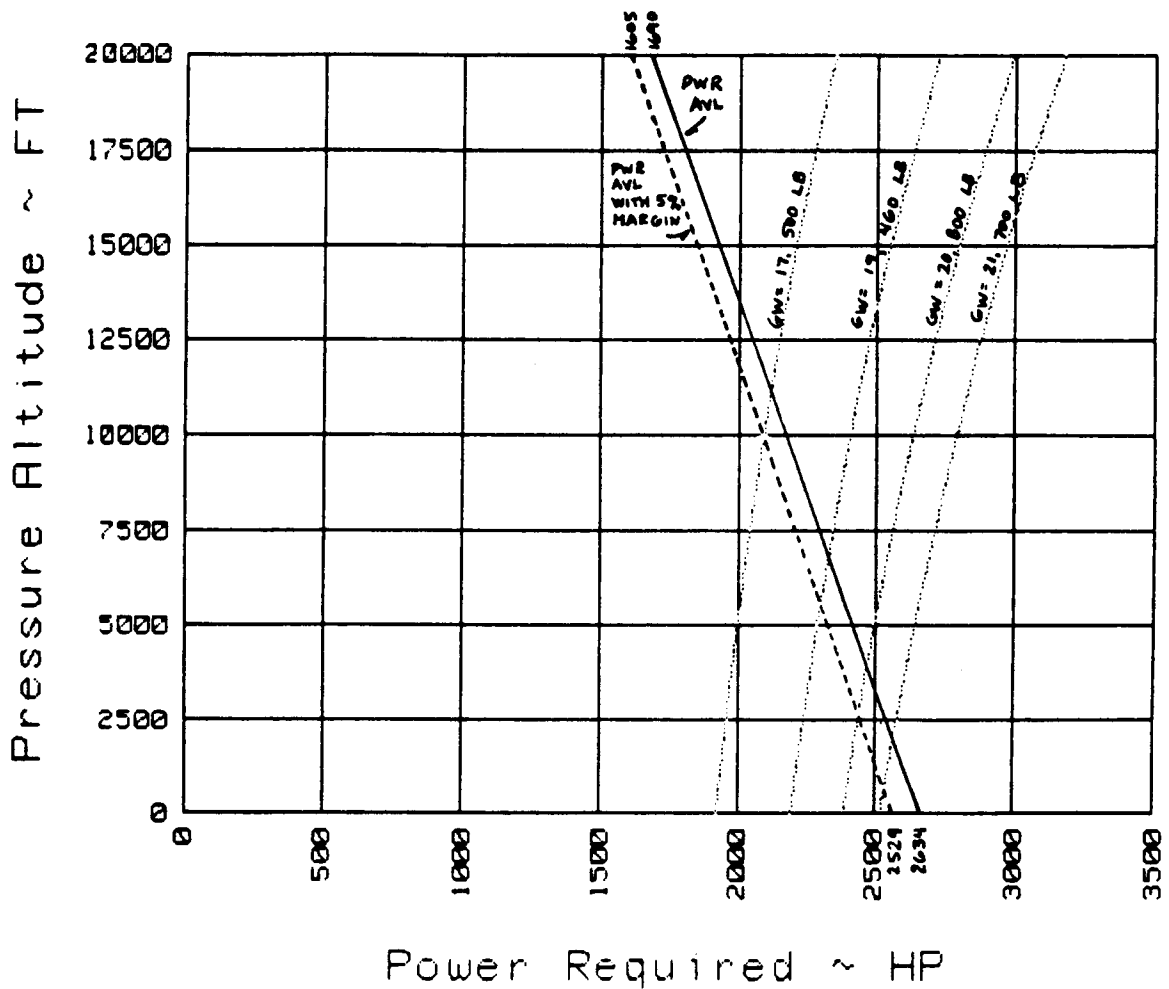


Figure 29
POWER REQUIRED WITH ALTITUDE - IGE HOT DAY

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	STD/HOT DAY
CONFIGURATION:	1 MK-48 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
		METHOD:	FREE HOVER (IGE - 15 FTagl)
		NOTE:	RADALT USED FOR HMMO

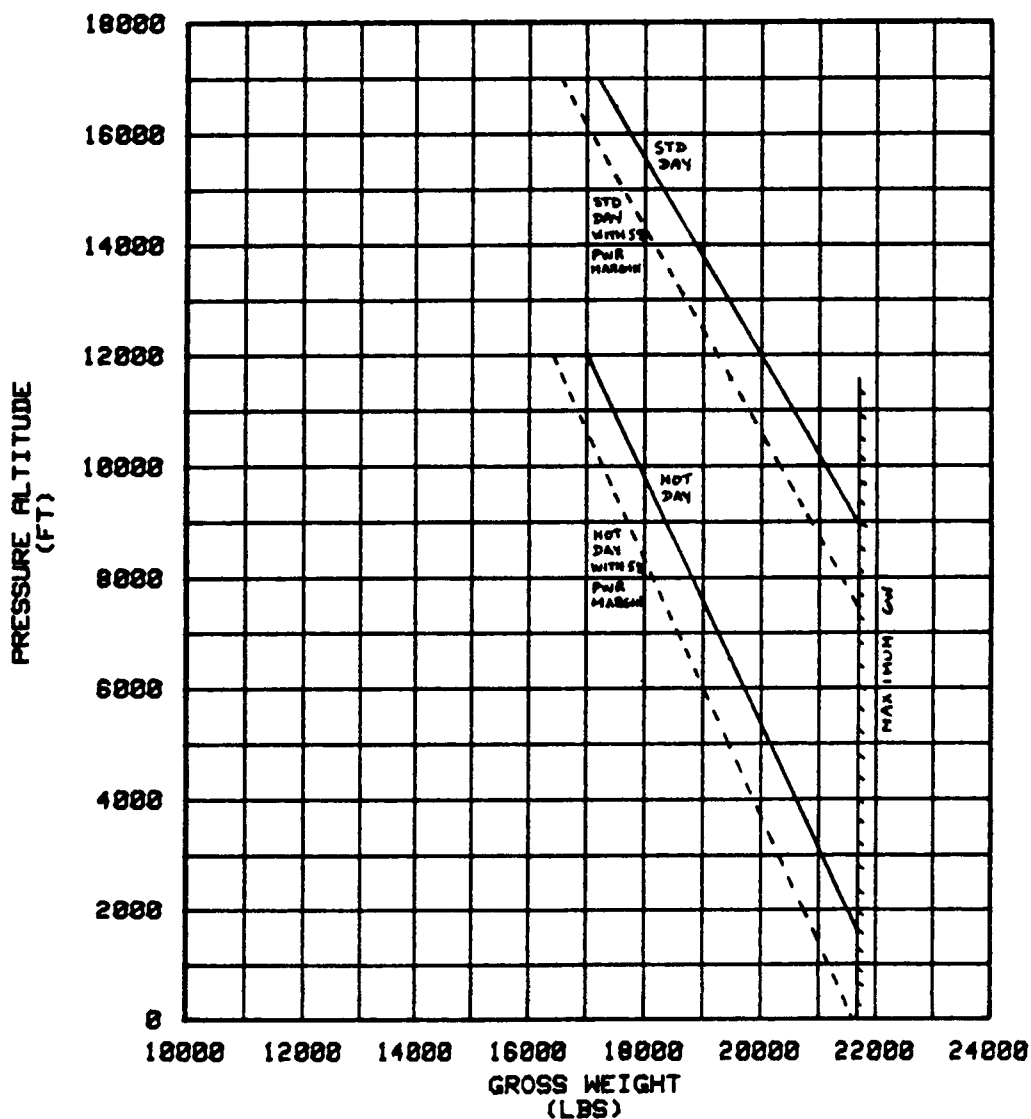


Figure 30
HOVER CEILING - IGE

APPENDIX F

MODEL SH-60B HELICOPTER
BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	STD DAY (15 DEG C)
CONFIGURATION:	1 MK-48 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
GWunreferred:	17,500 19,480	METHOD:	FREE HOVER (OGE)
	20,800 21,700 LB	NOTE:	RADALT USED FOR HHMD

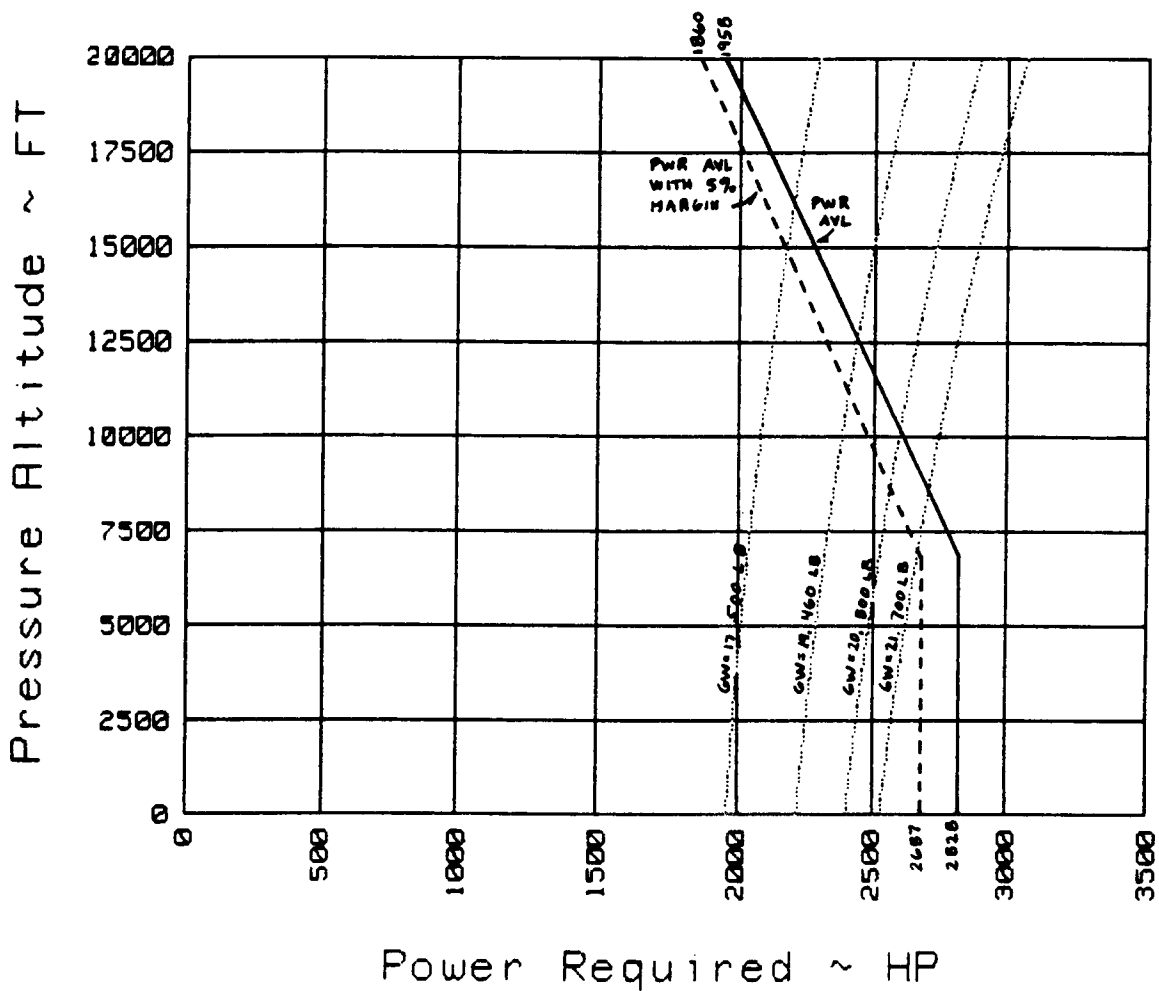


Figure 31
POWER REQUIRED WITH ALTITUDE - OGE STANDARD DAY

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	HOT DAY (35 DEG C)
CONFIGURATION:	1 MK-48 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
GWunreferred:	17,500 19,460	METHOD:	FREE HOVER (OGE)
	20,800 21,700 LB	NOTE:	RADALT USED FOR HHMD

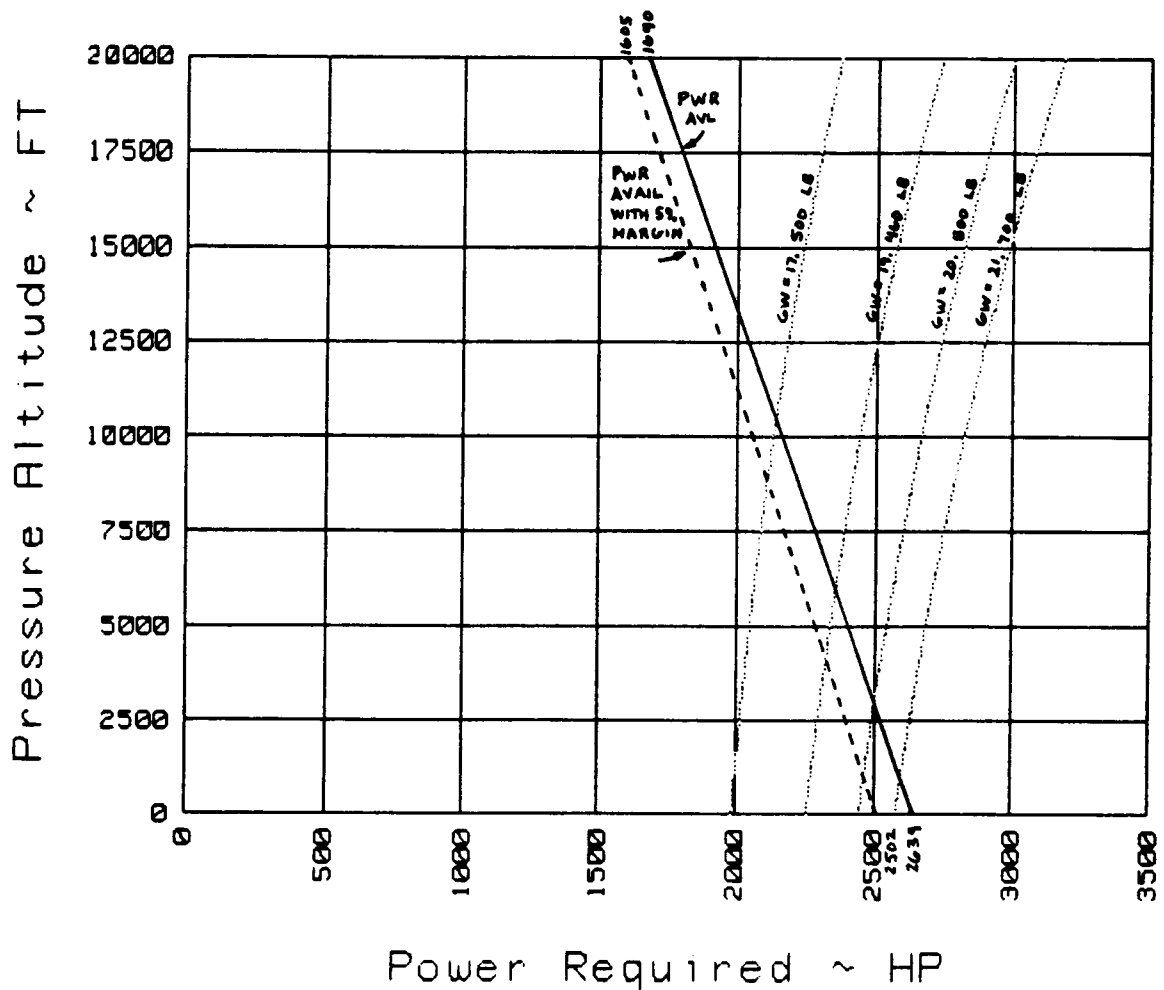


Figure 32
POWER REQUIRED WITH ALTITUDE - OGE HOT DAY

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	STD/HOT DAY
CONFIGURATION:	1 MK-48 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
		METHOD:	FREE HOVER (OGE)
		NOTE:	RADALT USED FOR HMMO

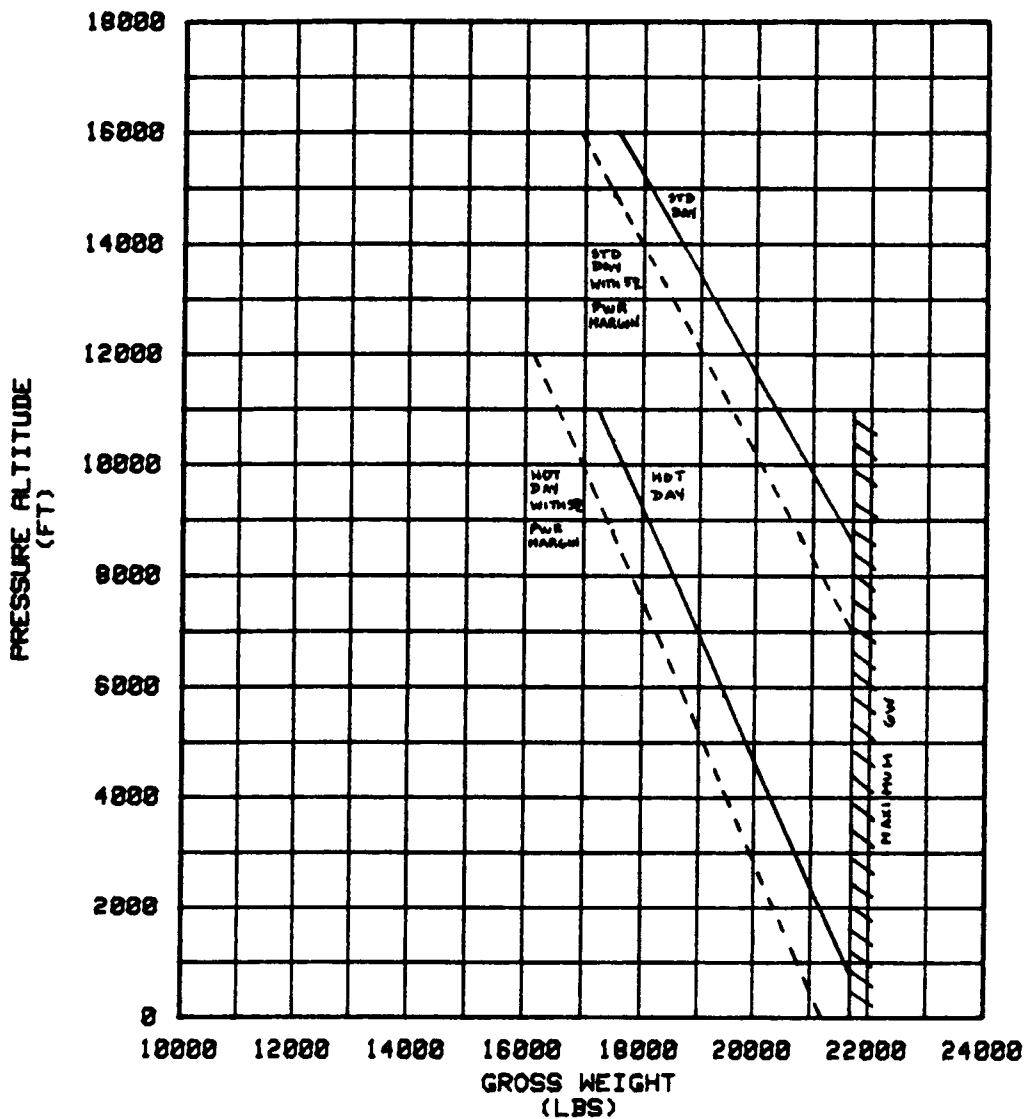


Figure 33
HOVER CEILING - OGE

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	Nr:	100%
FUEL:	JP-5 (6.8 lb/gal)	Hp:	STANDARD DAY
ENGINE BLEED AIR:	OFF	OAT:	STD/HOT DAY
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	WIND:	5 KT @ 0 DEG
GWunreferred:	20,800 LB	METHOD:	PARTIAL POWER VERTICAL CLIMBS

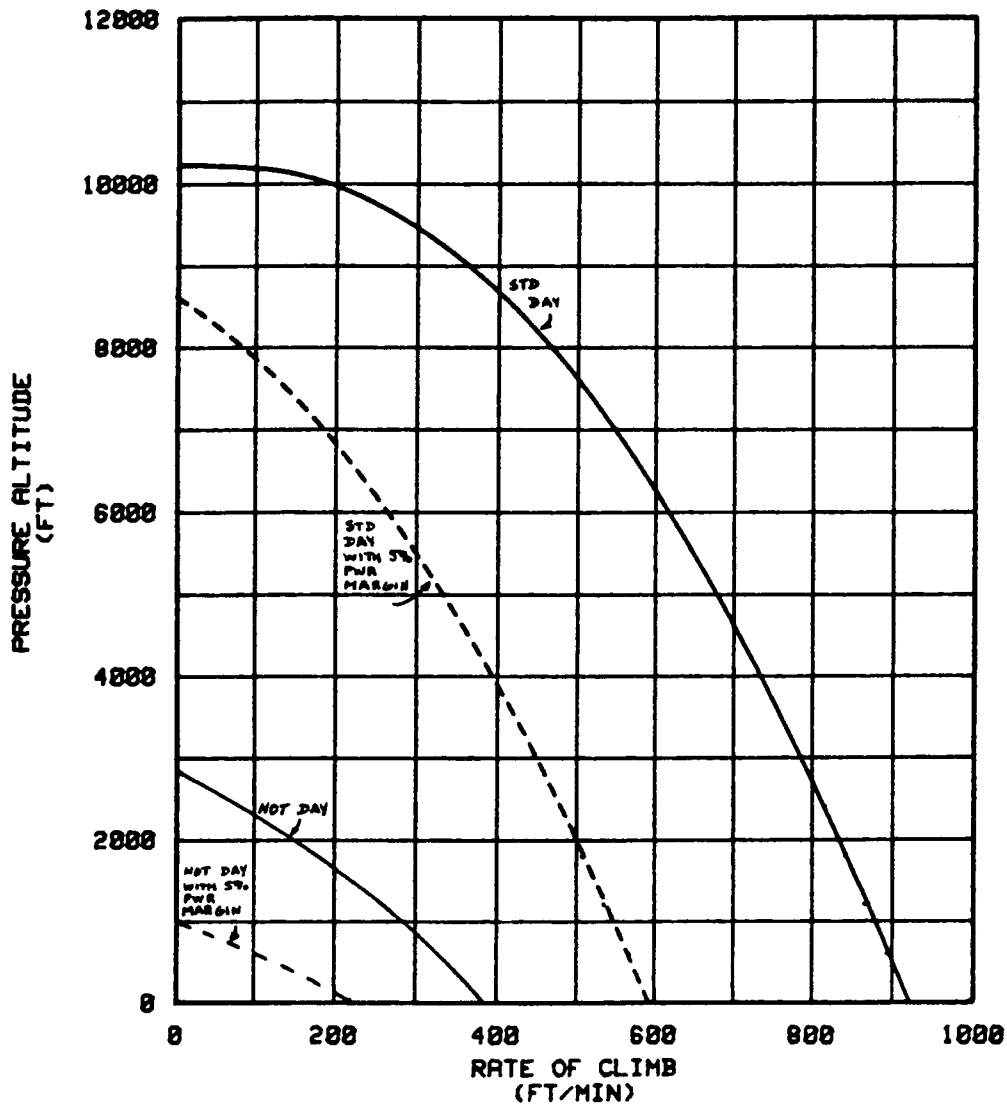


Figure 34
CALCULATED VERTICAL CLIMB PERFORMANCE

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	CGavg:	358.5 IN
FUEL:	JP-5 (6.8 lb/gal)	Nr:	100%
ENGINE BLEED AIR:	OFF	Hp:	SEA LEVEL STD DAY
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	OAT:	STANDARD DAY (15 DEG C)
		METHOD:	CONSTANT REFERRED GW
		TECHNIQUE:	W/a = 20,800 LB

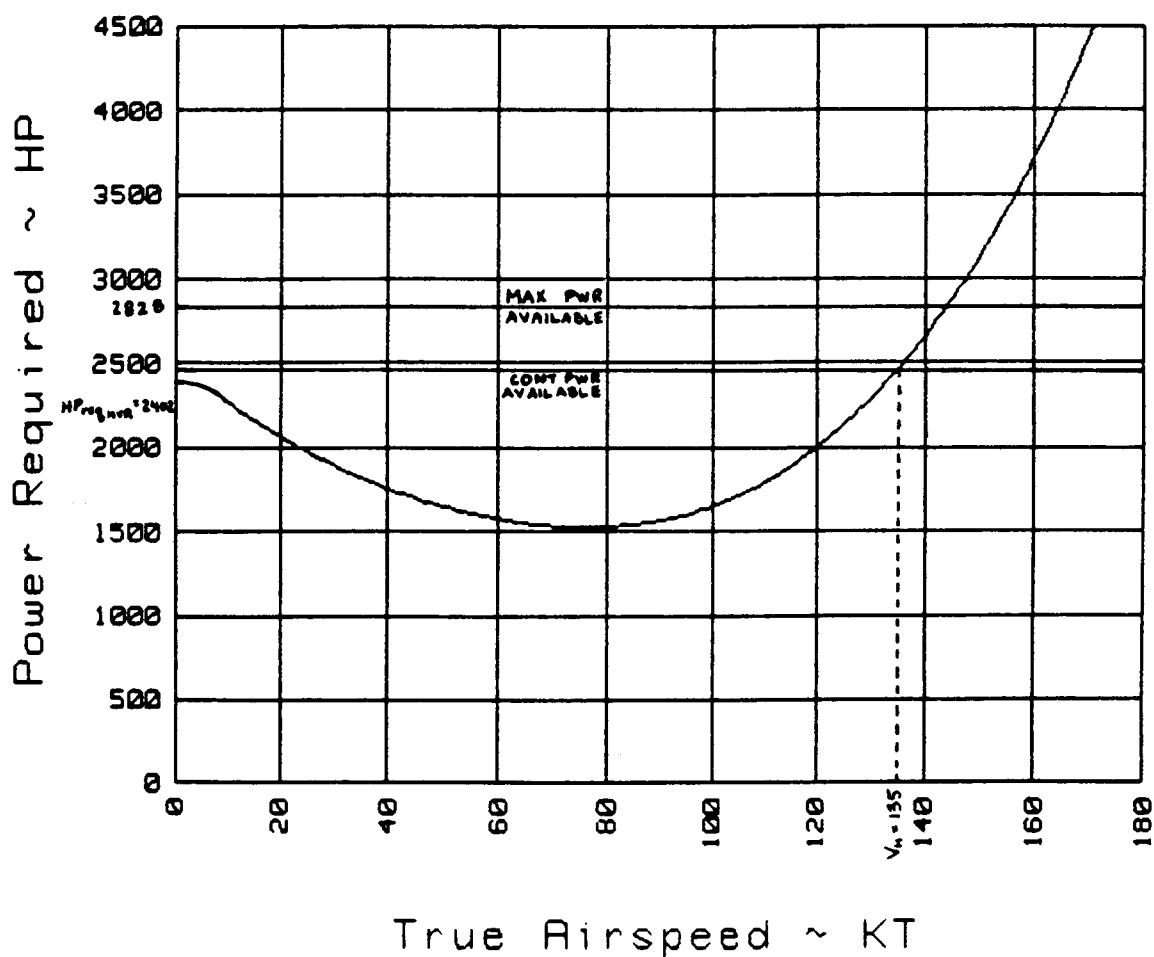


Figure 35
CALCULATED LEVEL FLIGHT PERFORMANCE -
MAXIMUM LEVEL FLIGHT AIRSPEED

APPENDIX F

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	CGavg:	358.5 IN
FUEL:	JP-5 (6.8 lb/gal)	Nr:	100%
ENGINE BLEED AIR:	OFF	Hp:	SEA LEVEL STD DAY
CONFIGURATION:	1 MK-48 TORPEDO (STATION 1)	OAT:	STANDARD DAY (15 DEG C)
		METHOD:	CONSTANT REFERRED GW
		TECHNIQUE:	W/G = 20,800 LB

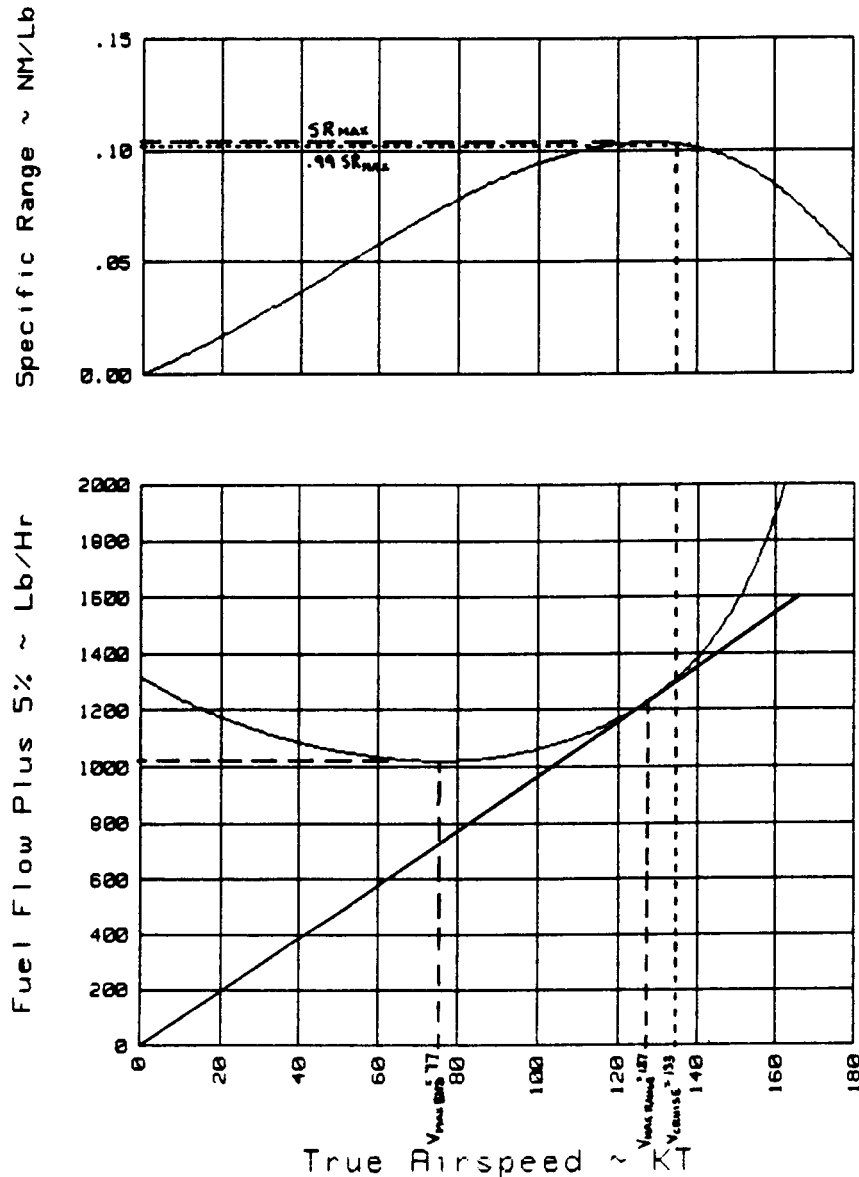


Figure 36
CALCULATED LEVEL FLIGHT PERFORMANCE - MAXIMUM ENDURANCE

MODEL SH-60B HELICOPTER

BuNo 162092

ENGINES:	(2) T700-GE-401	OWavg:	17,400 LB
SERIAL NUMBERS:	#1 212267 #2 212257	CGavg:	355.3 IN
FUEL:	JP-5 (6.8 lb/gal)	Nr:	100%
ENGINE BLEED AIR:	OFF	Hp:	2000 TO 3000 FT
CONFIGURATION:	1 MK-46 TORPEDO (STATION 1)	OAT:	19 DEG C
		METHOD:	SAWTOOTH CLIMB/DESCENTS
		TECHNIQUE:	MAX/MIN PWR CLB/DESC

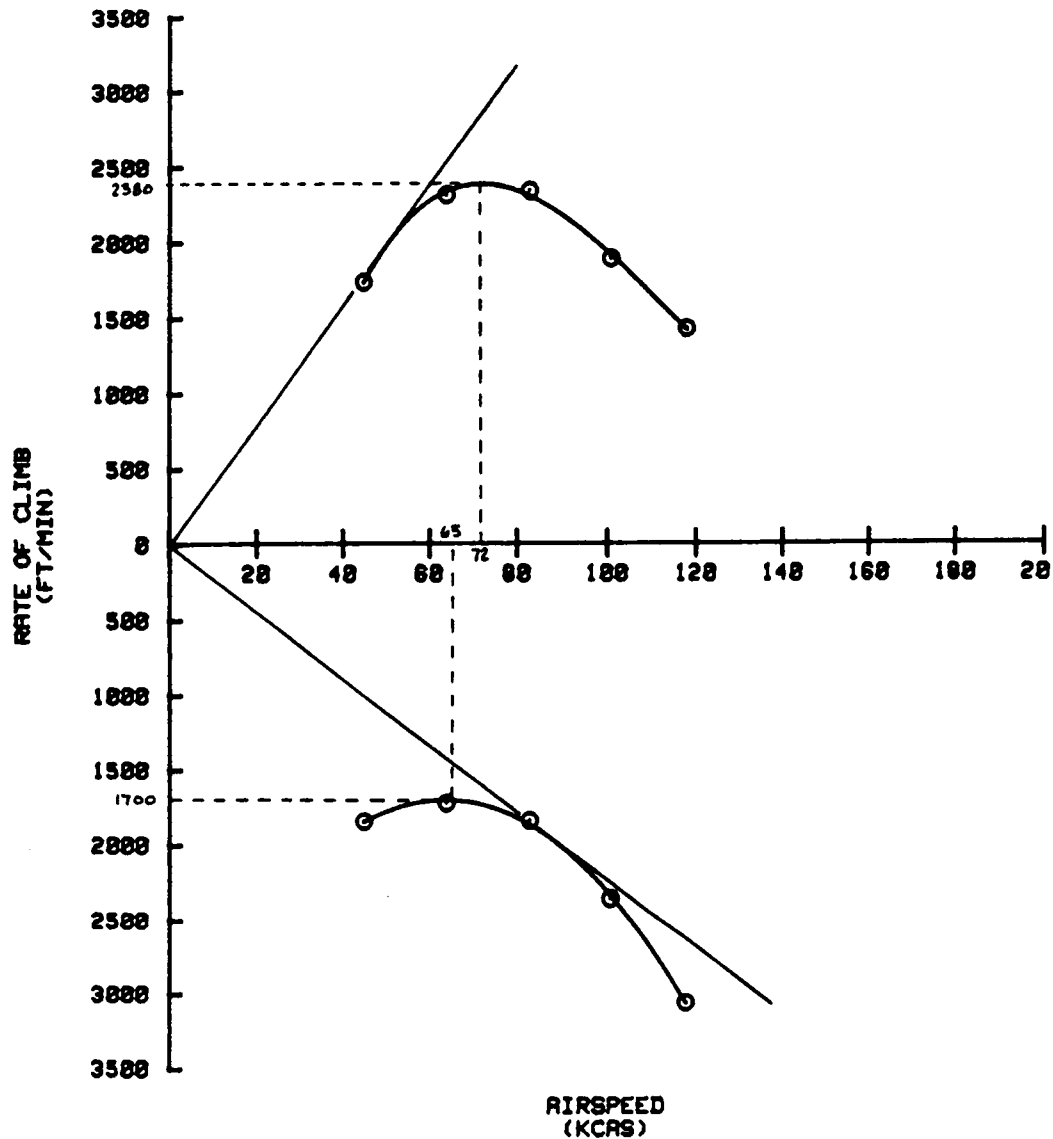


Figure 37
RATE OF CLIMB AND RATE OF DESCENT PERFORMANCE

MODEL SH-60B HELICOPTER

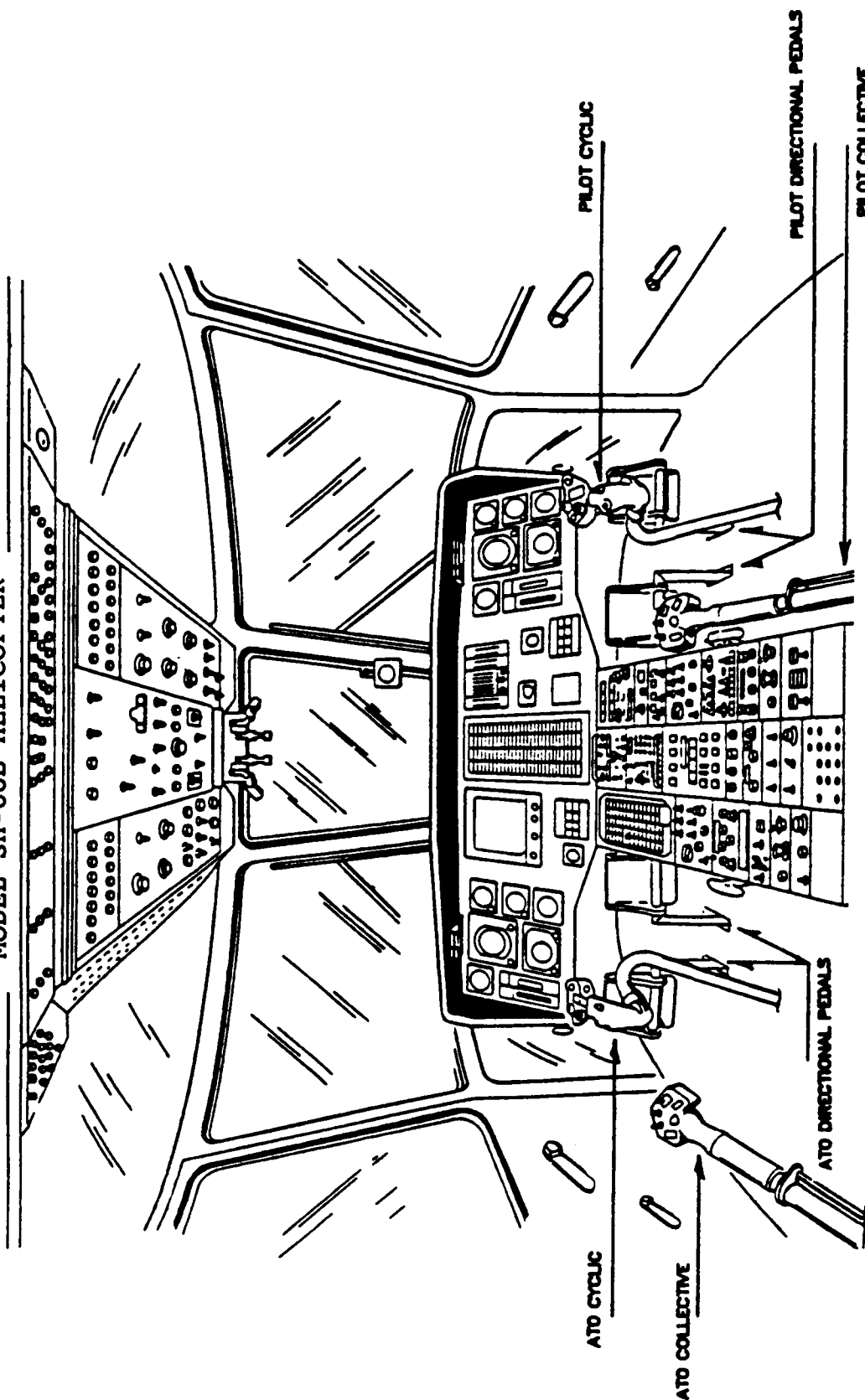


Figure 38
COCKPIT CONTROL LAYOUT

MODEL SH-60B HELICOPTER

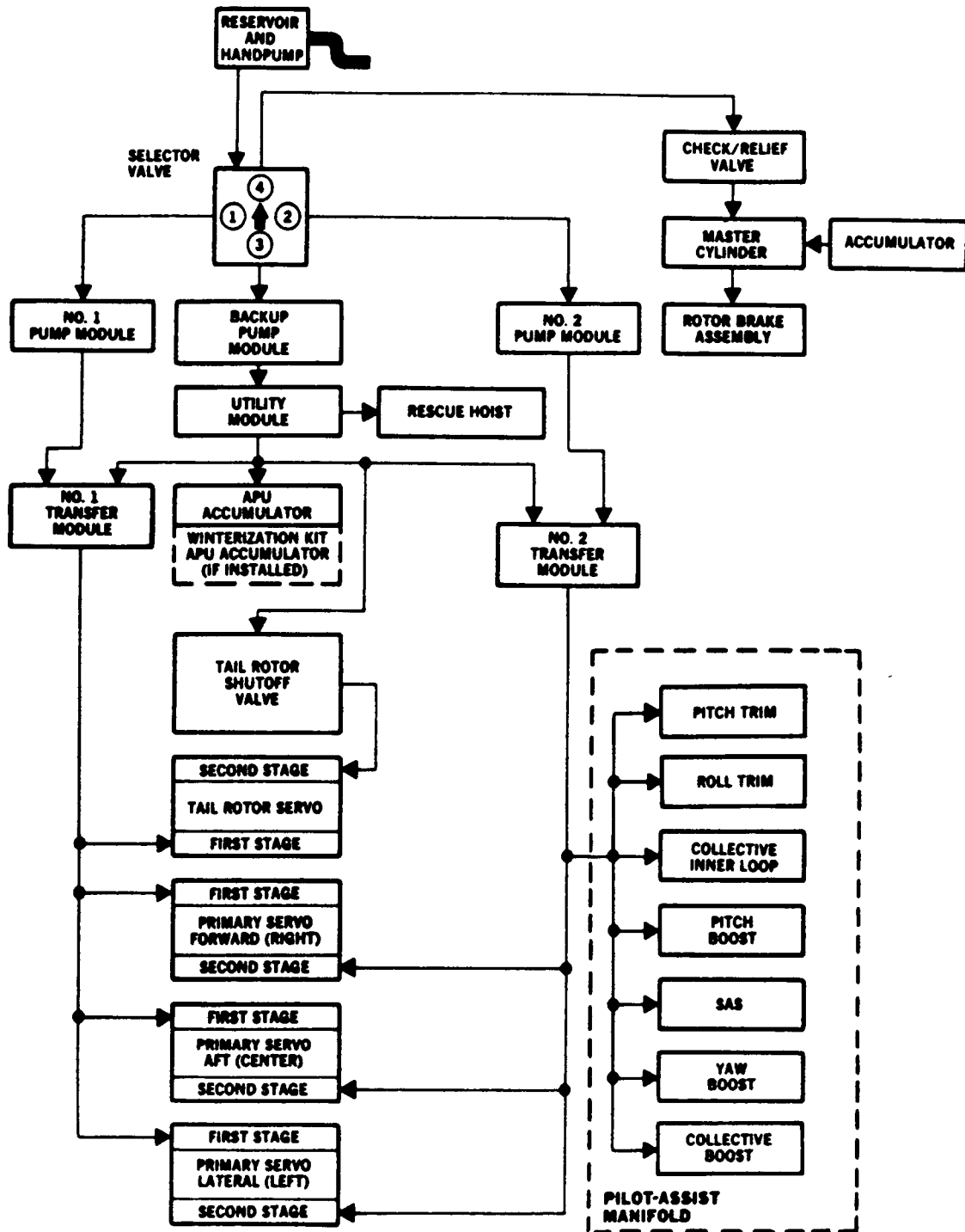


Figure 39
HYDRAULIC SYSTEM

MODEL SH-60B HELICOPTER

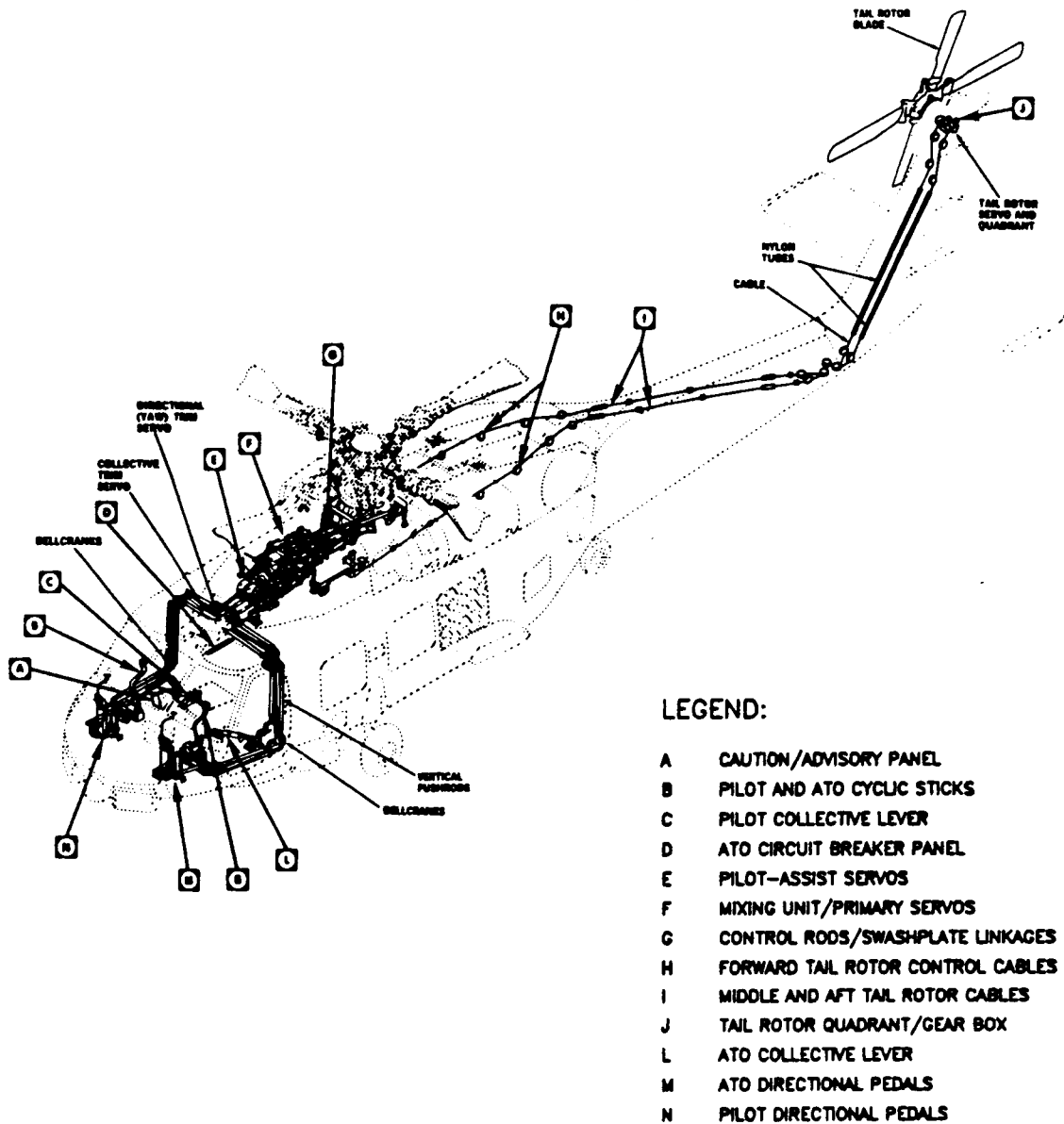


Figure 40
FLIGHT CONTROL COMPONENT LOCATIONS

MODEL SH-60B HELICOPTER

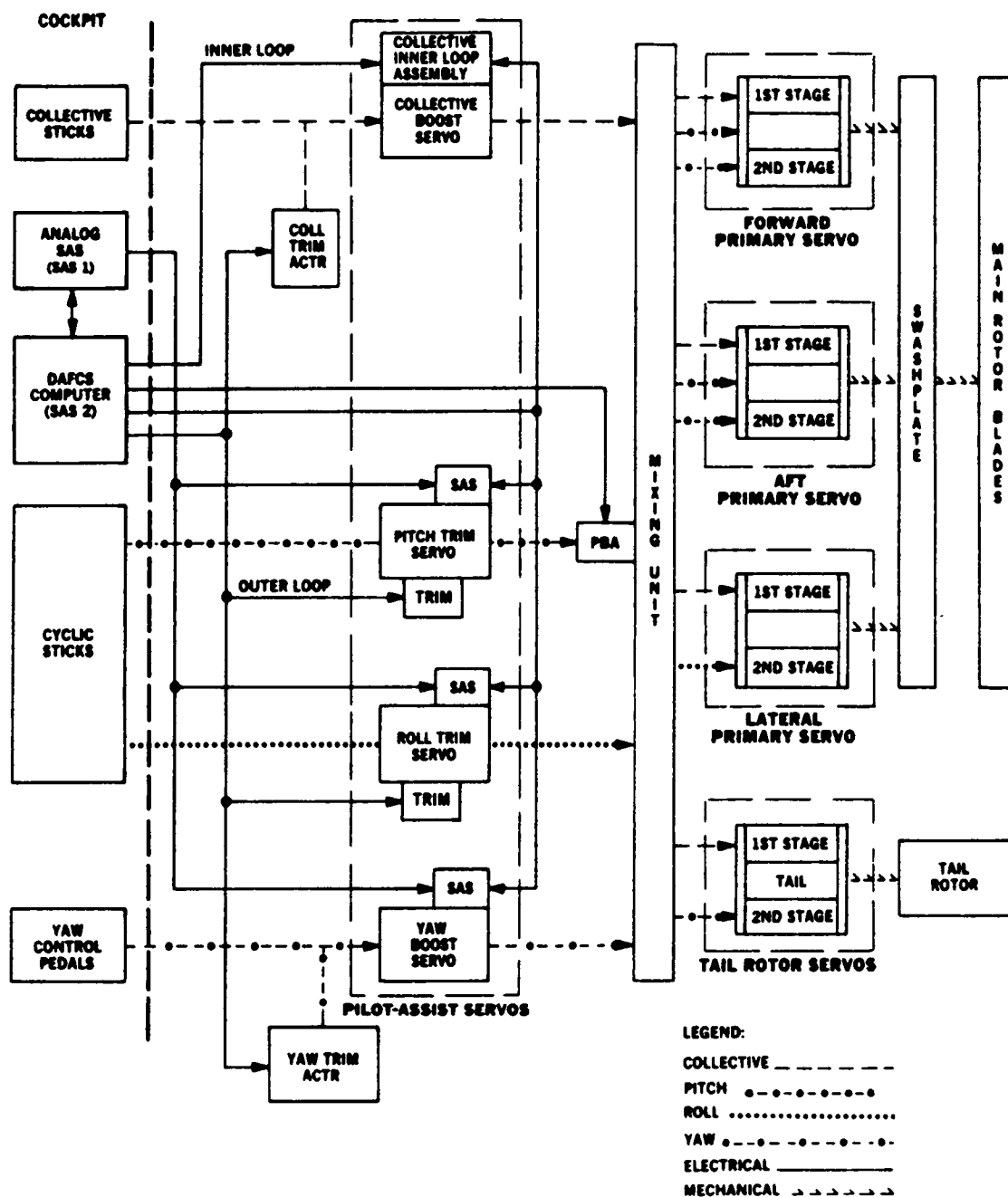


Figure 41
FLIGHT CONTROL SYSTEM

MODEL SH-60B HELICOPTER

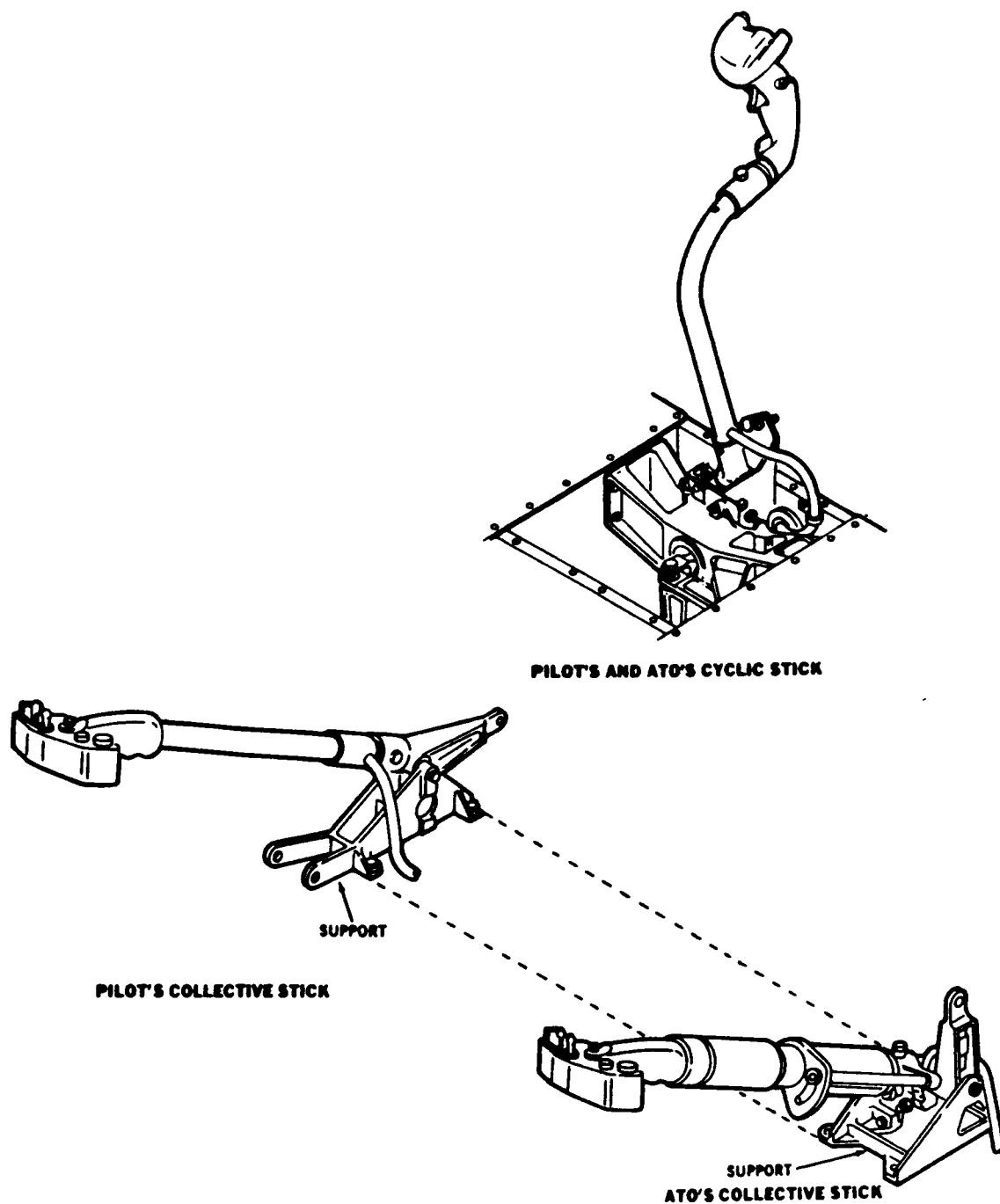


Figure 42
COCKPIT CONTROLS - COLLECTIVE AND CYCLIC

MODEL SH-60B HELICOPTER

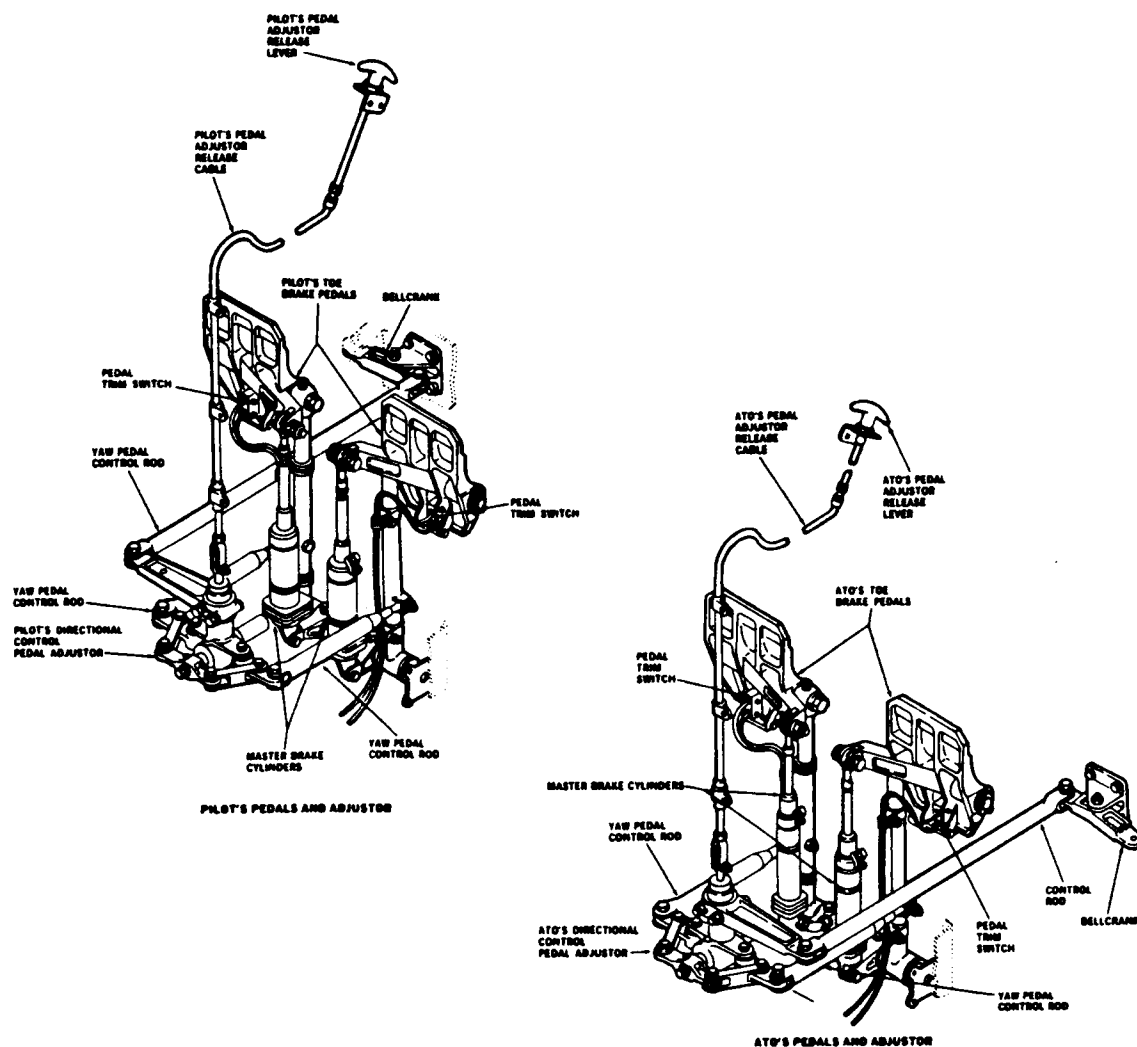


Figure 43
COCKPIT CONTROLS - DIRECTIONAL PEDALS

MODEL SH-60B HELICOPTER

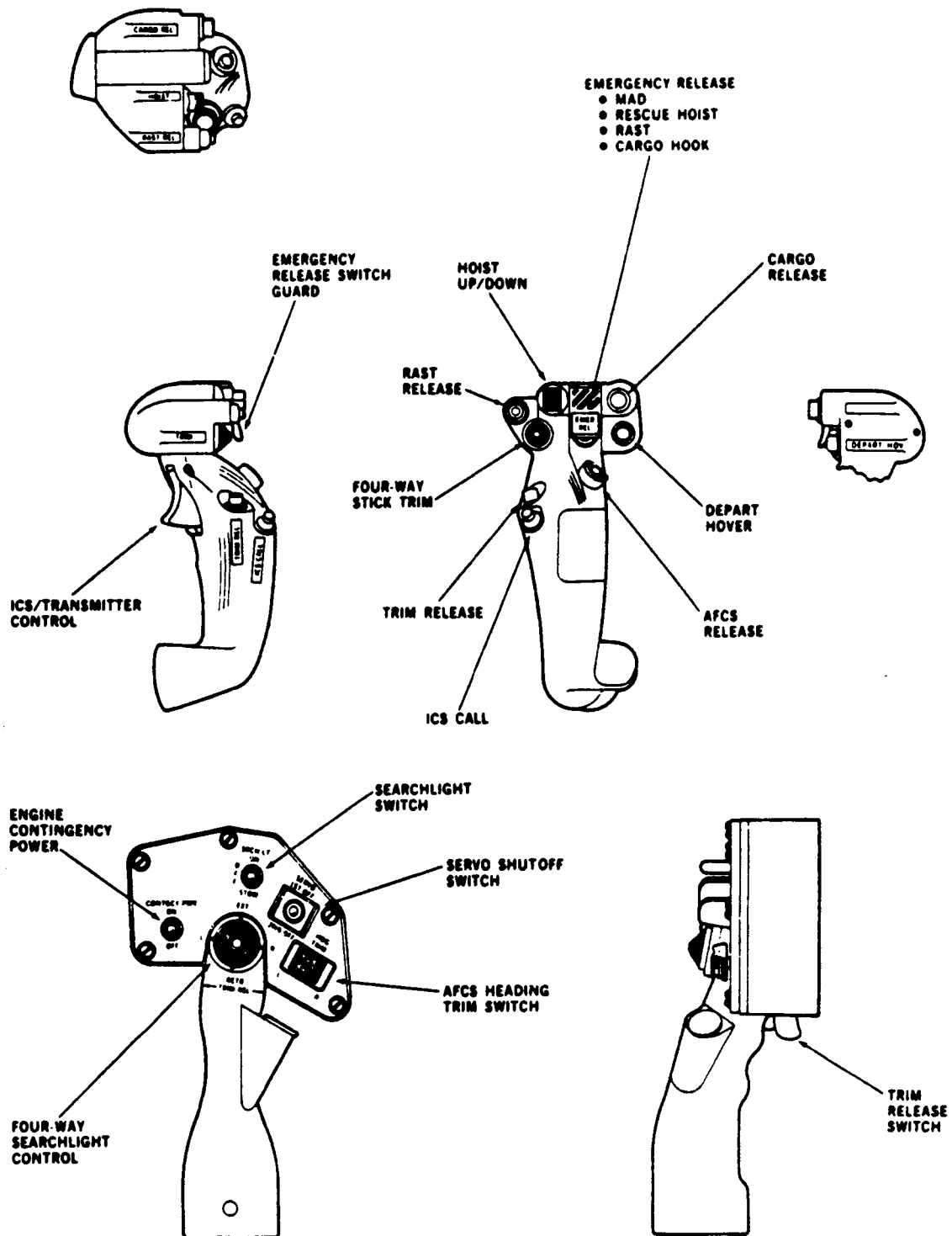


Figure 44
COCKPIT CONTROLS - COLLECTIVE AND CYCLIC GRIPS

MODEL SH-60B HELICOPTER

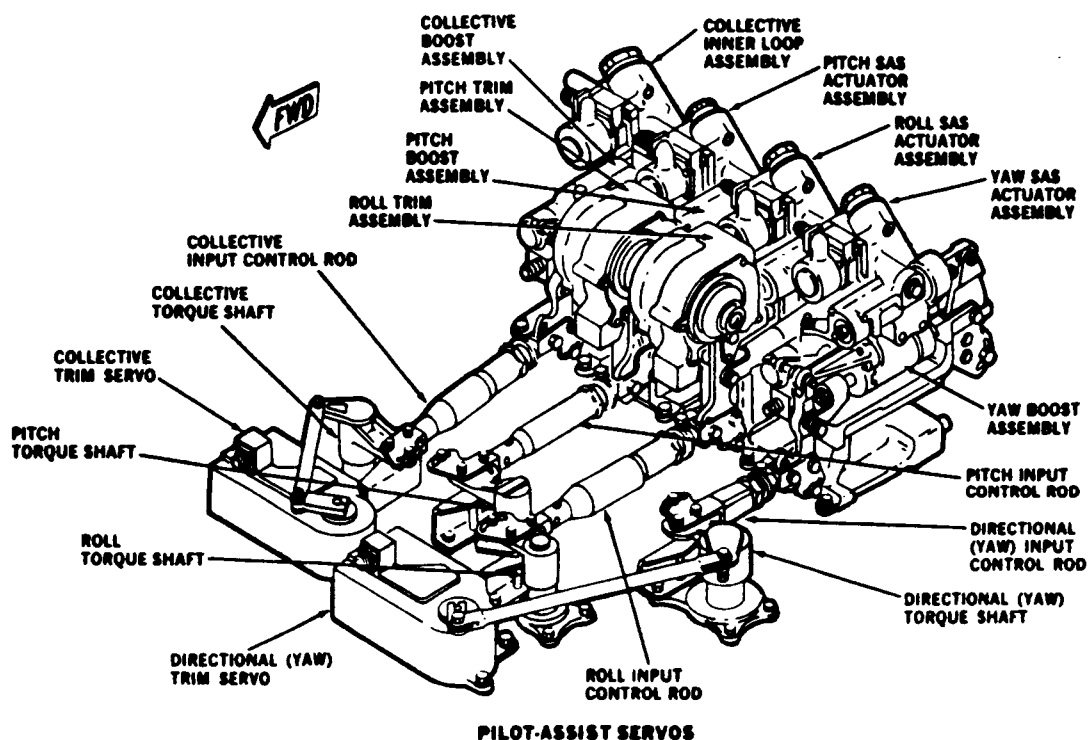
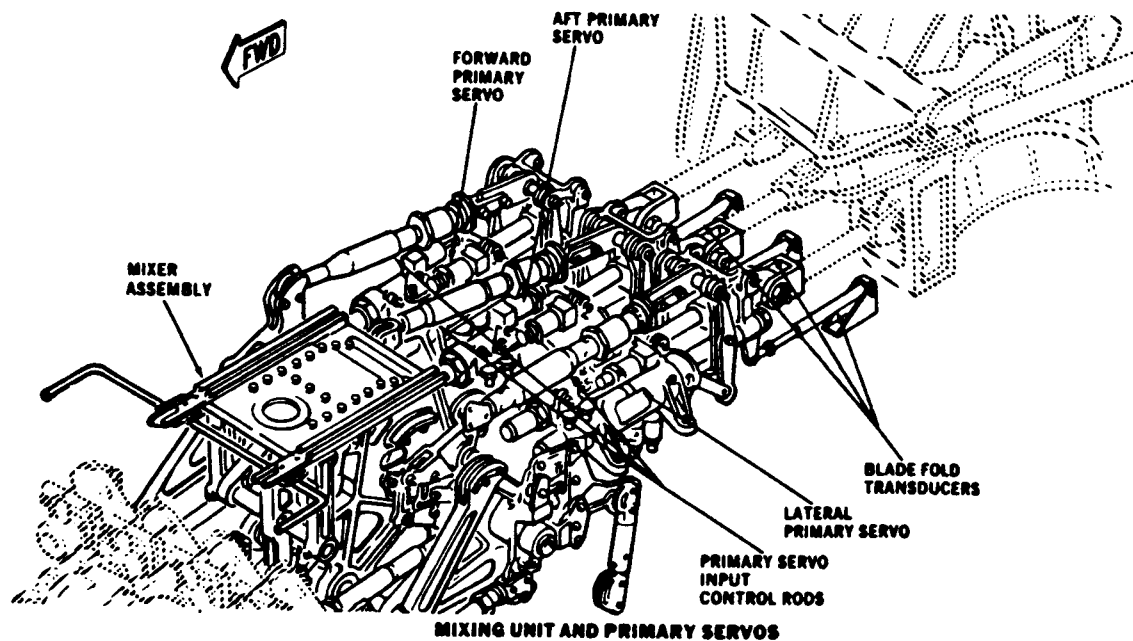


Figure 45
FLIGHT CONTROL SERVOS AND MIXING UNIT

MODEL SH-60B HELICOPTER

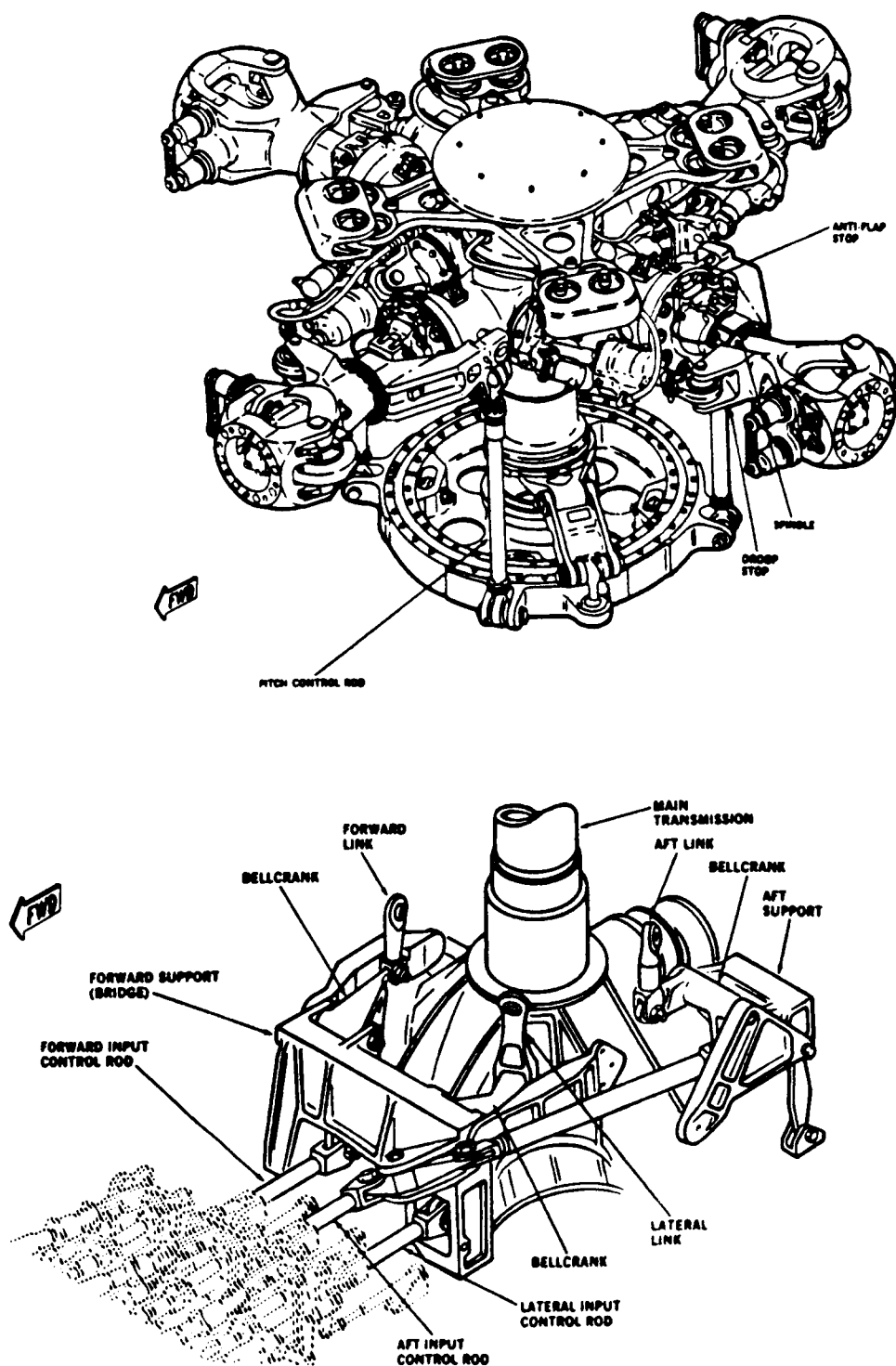


Figure 46
MAIN ROTOR HEAD ASSEMBLY

MODEL SH-60B HELICOPTER

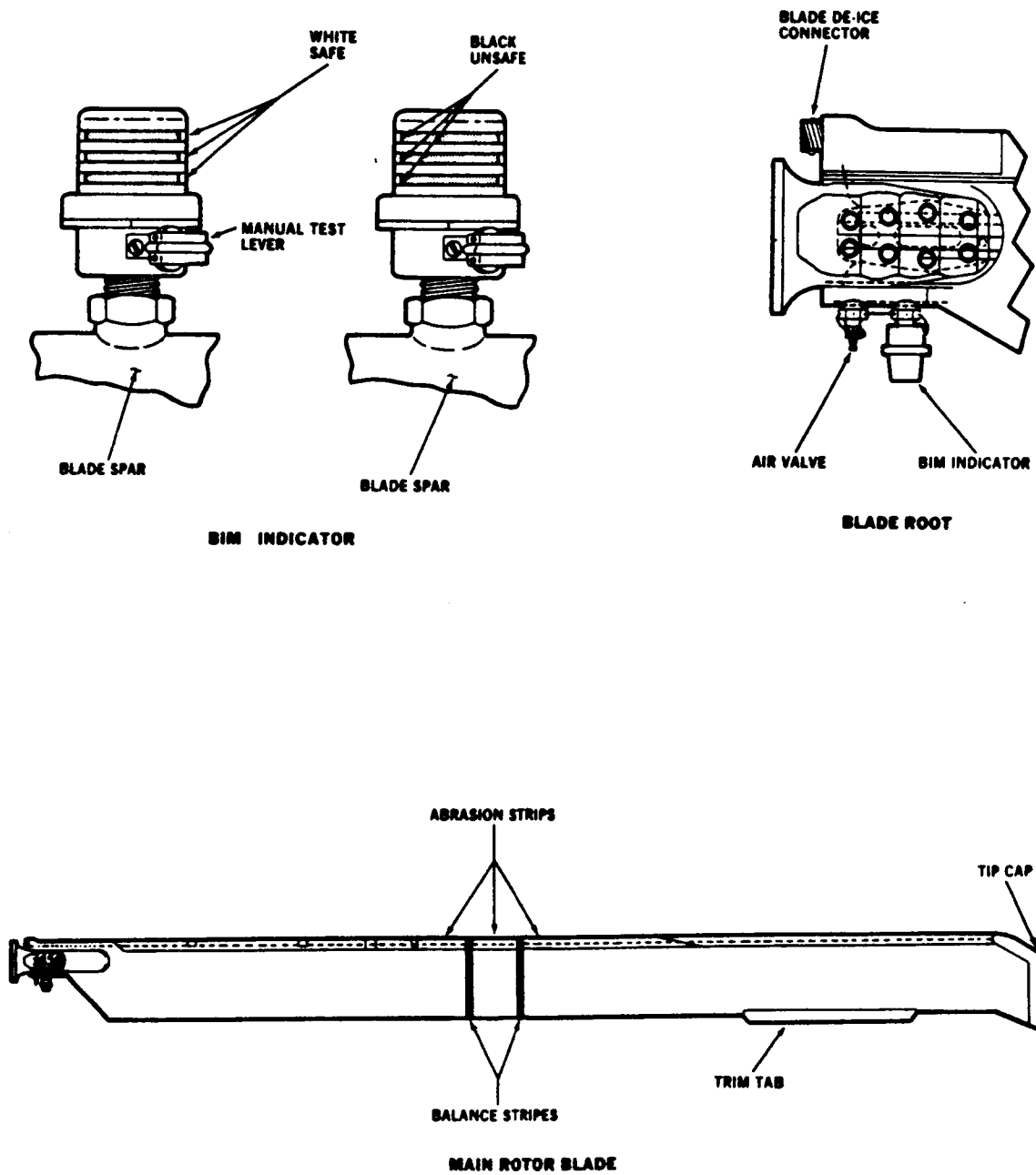


Figure 47
MAIN ROTOR BLADE AND BLADE INDICATOR

MODEL SH-60B HELICOPTER

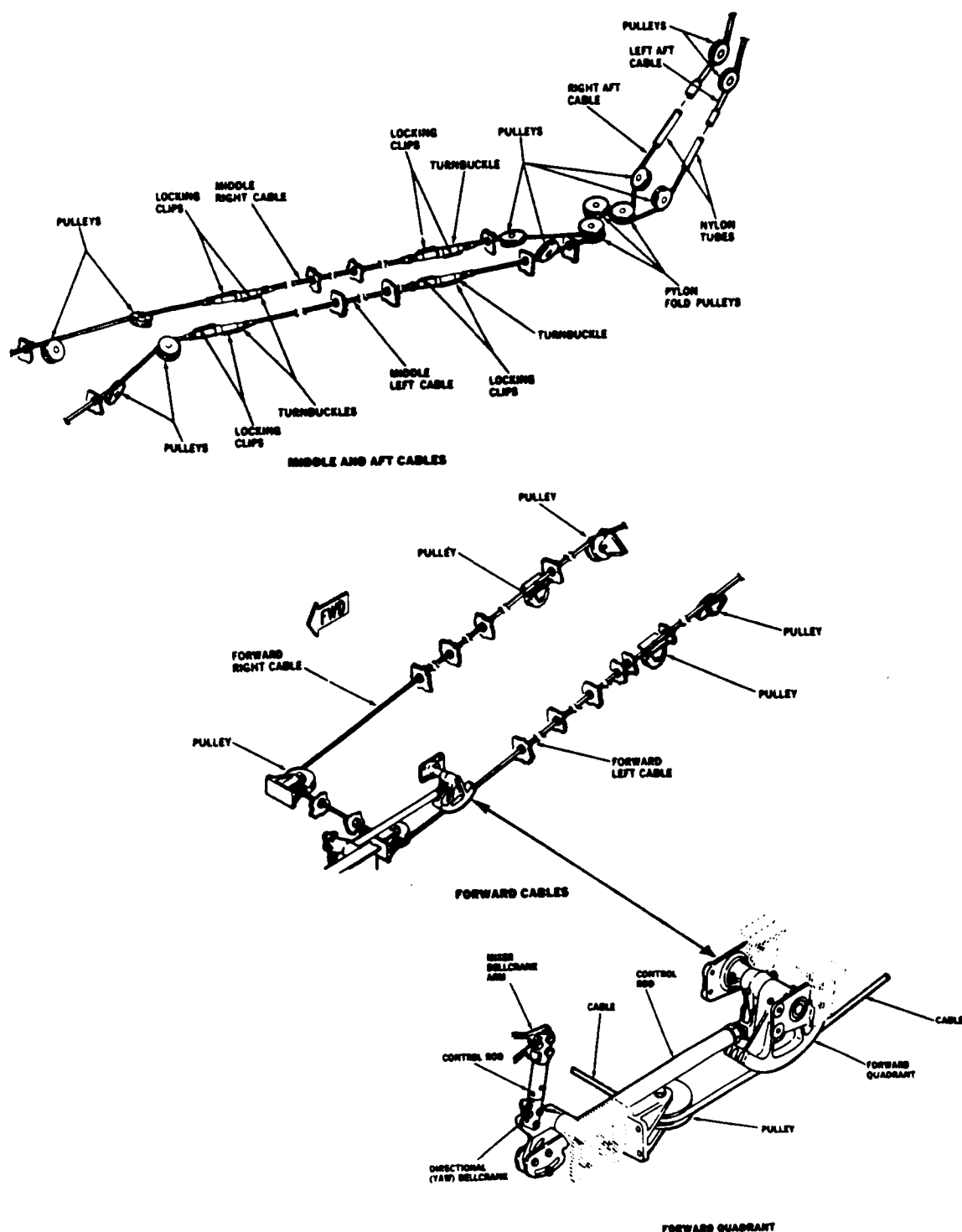


Figure 48
TAIL ROTOR CONTROL CABLES

MODEL SH-60B HELICOPTER

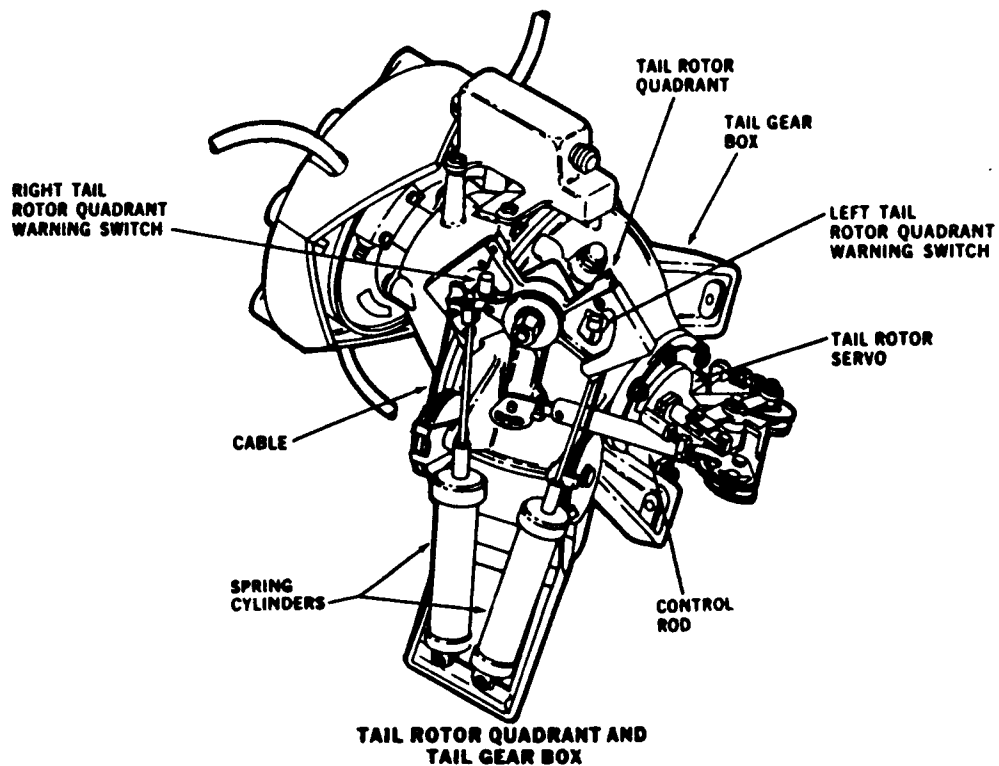
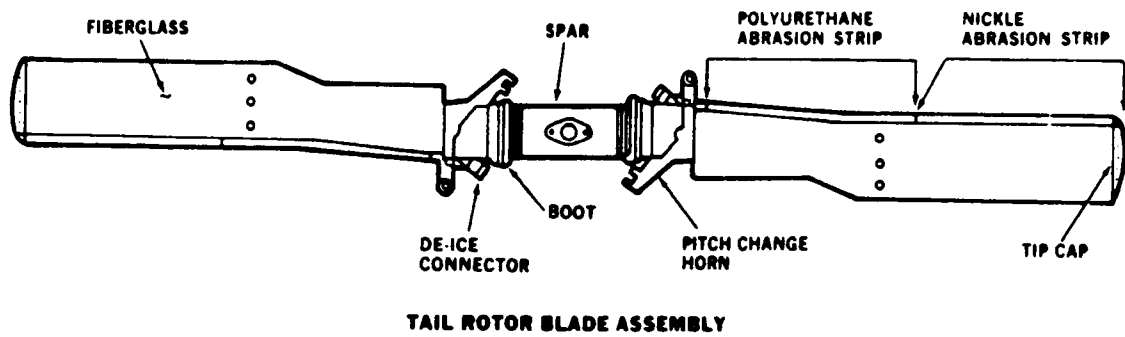


Figure 49
TAIL ROTOR COMPONENTS

MODEL SH-60B HELICOPTER

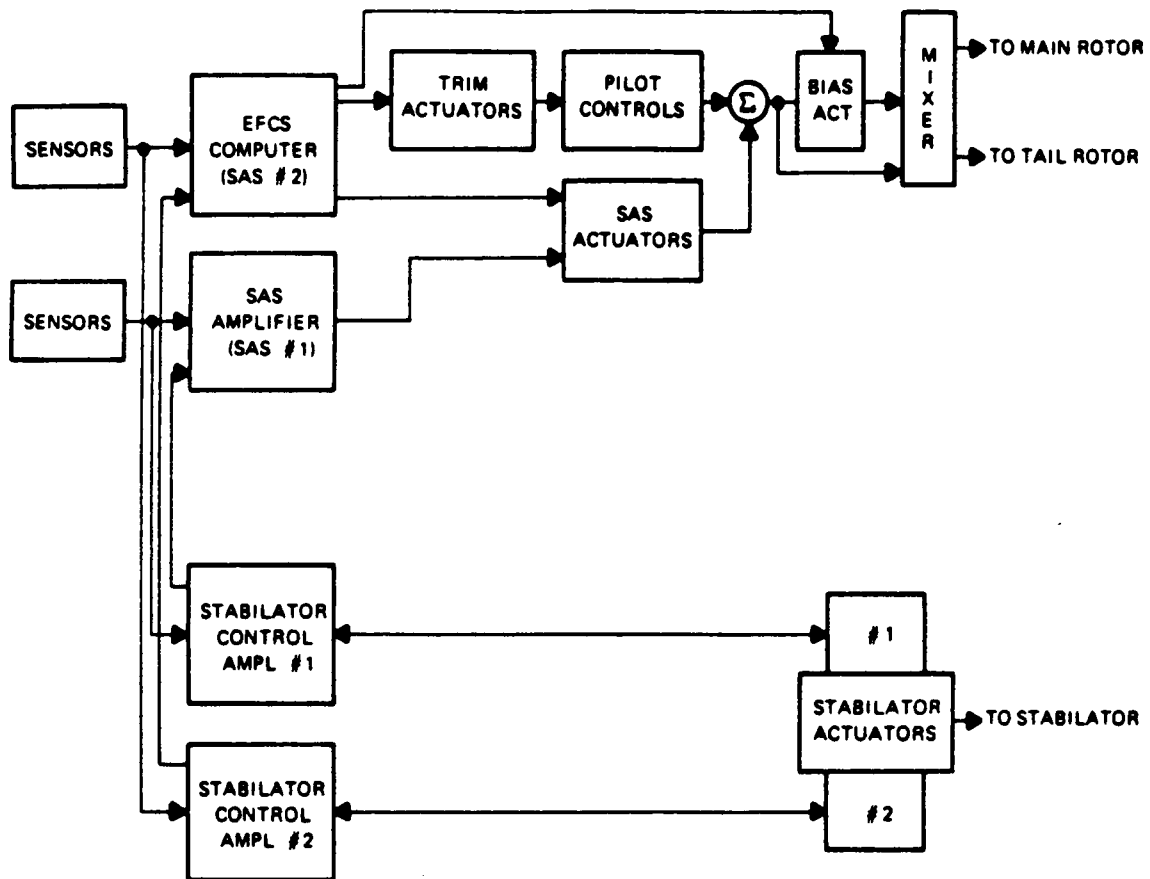


Figure 50
AUTOMATIC FLIGHT CONTROL SYSTEM

MODEL SH-60B HELICOPTER

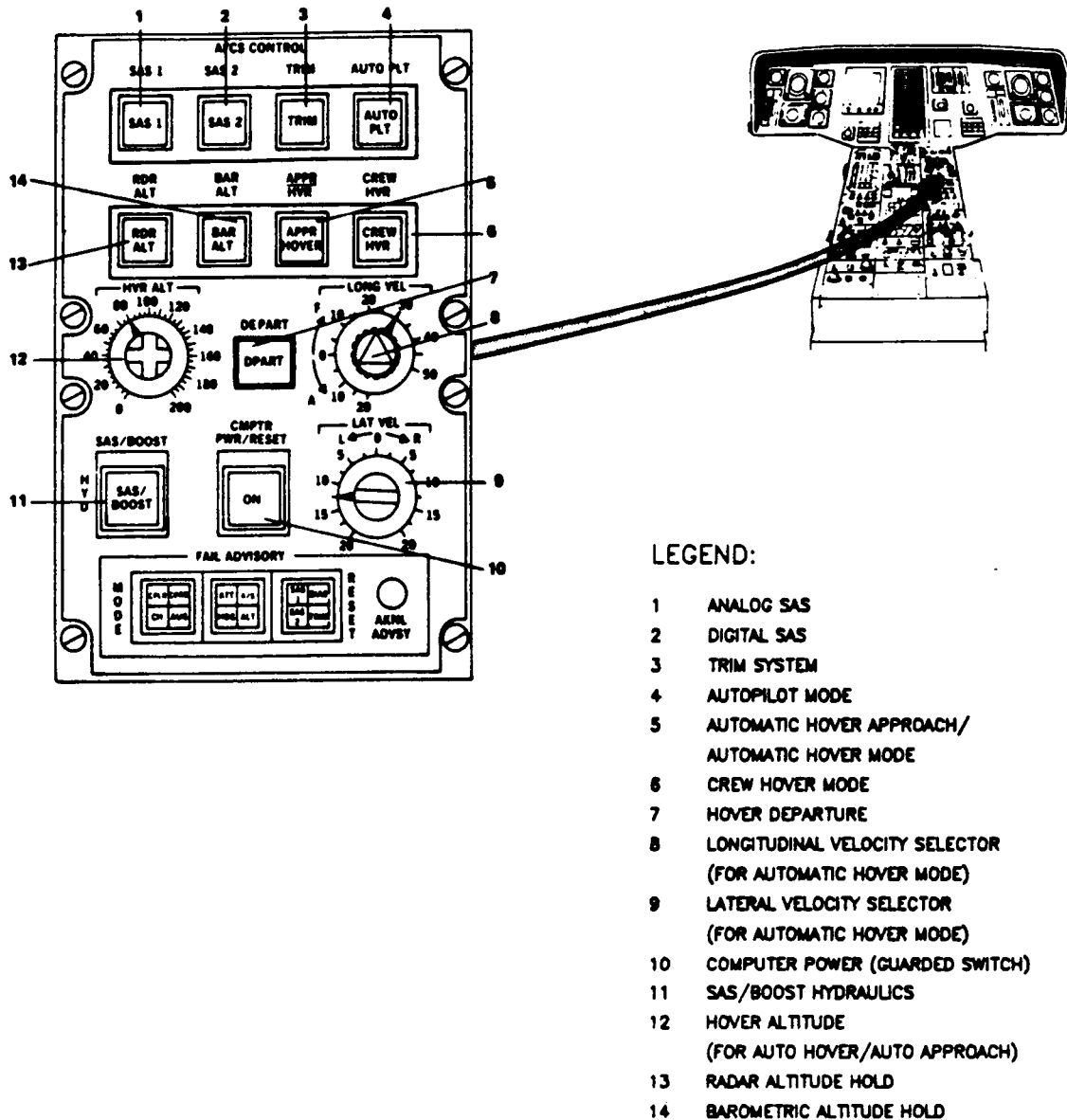


Figure 51
AUTOMATIC FLIGHT CONTROL SYSTEM CONTROL PANEL

MODEL SH-60B HELICOPTER

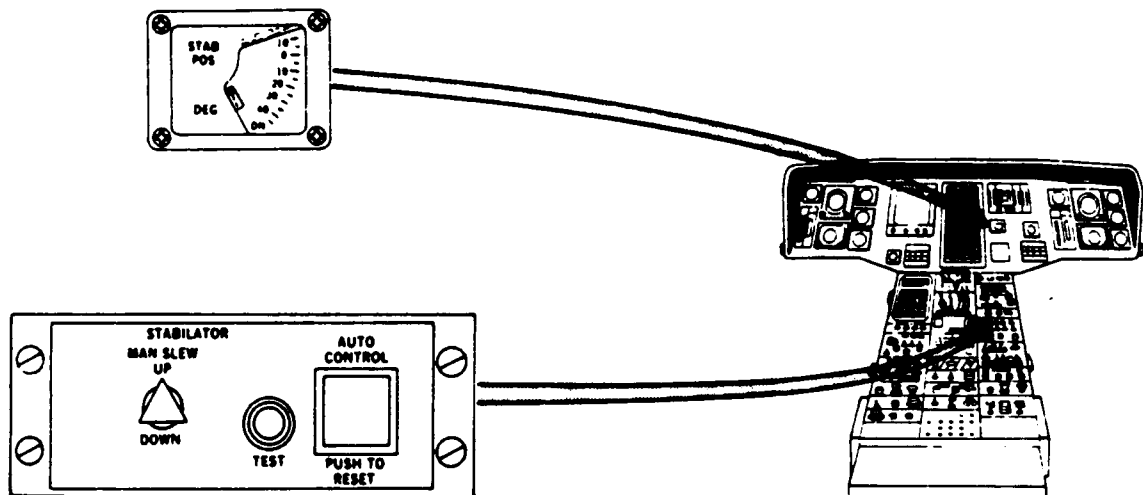


Figure 52
STABILATOR CONTROL PANEL AND POSITION INDICATOR

MODEL SH-60B HELICOPTER

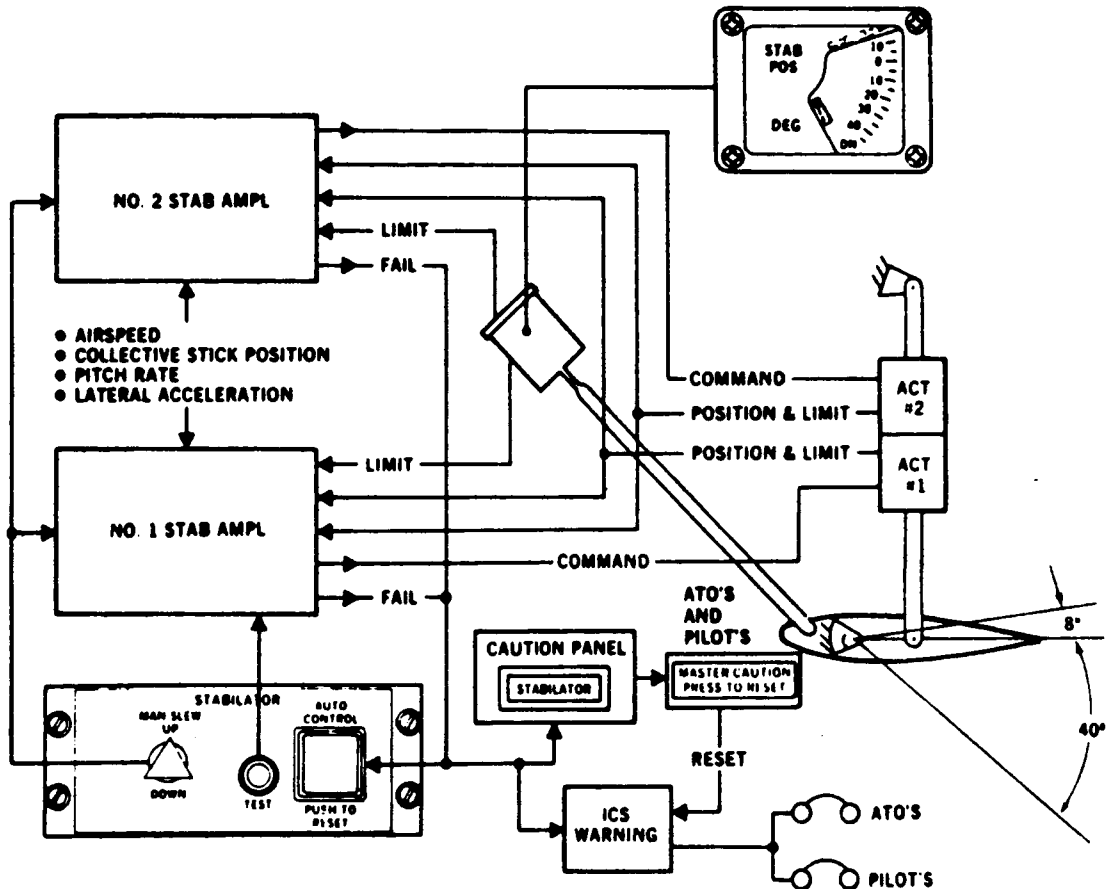


Figure 53
STABILATOR CONTROL SYSTEM

MODEL SH-60B HELICOPTER

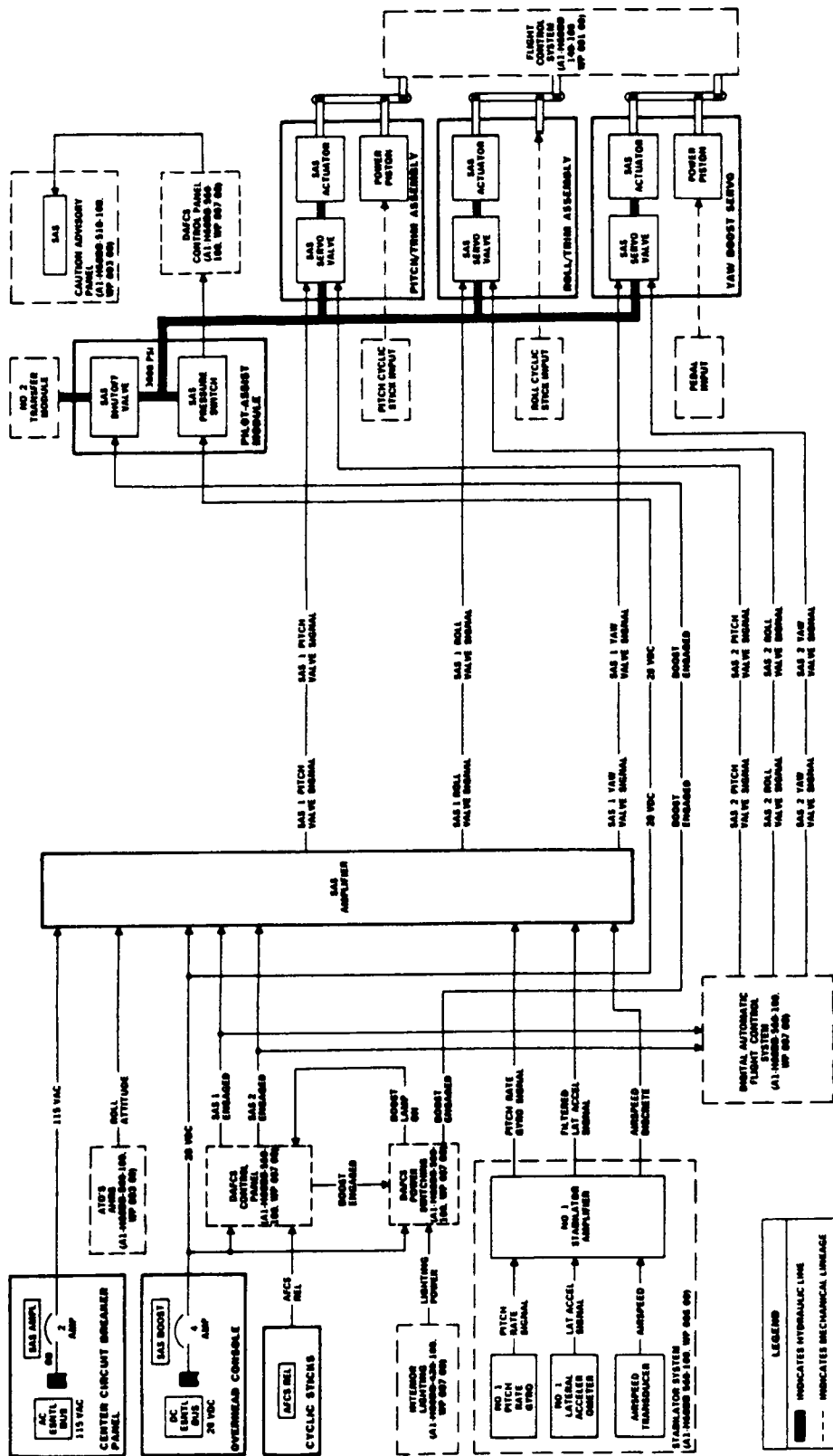


Figure 54
STABILITY AUGMENTATION SYSTEM

MODEL SH-60B HELICOPTER

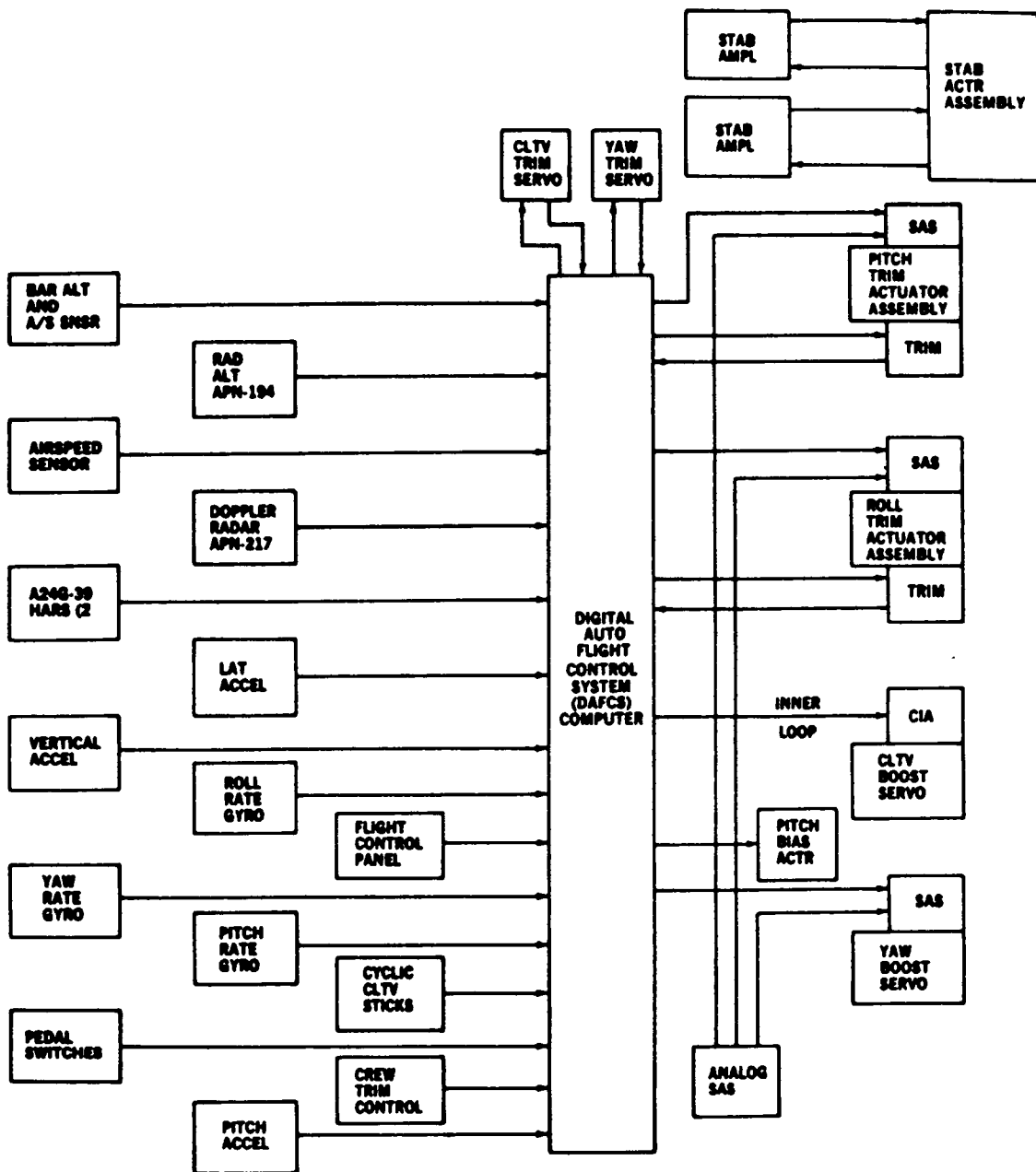


Figure 55
ELECTRONIC FLIGHT CONTROL SYSTEM

MODEL SH-60B HELICOPTER

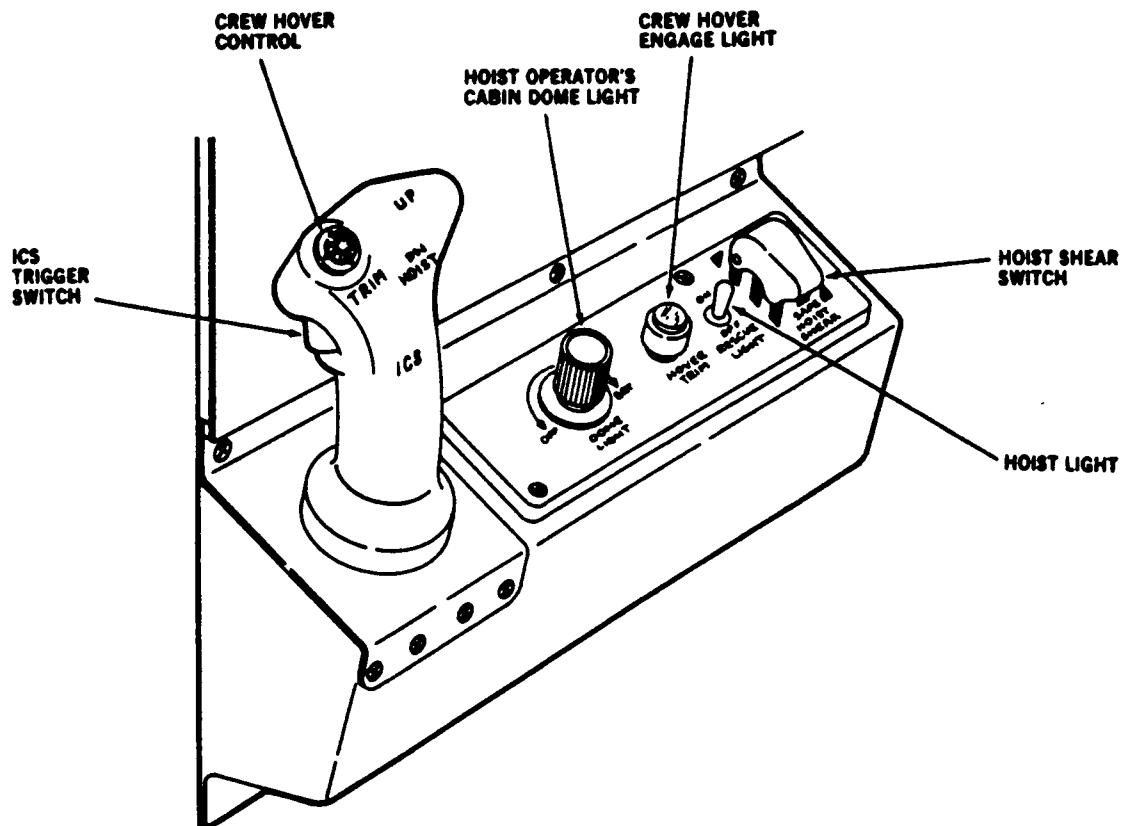


Figure 56
CREW HOVER-TRIM PANEL

APPENDIX G

LIST OF ABBREVIATIONS

LIST OF ABBREVIATIONS

AC	Alternating current
A/C	Aircraft
AFCS	Automatic flight control system
AGL	Above ground level
AHRS	Attitude heading reference system
AOB	Angle of bank
APU	Auxiliary power unit
ASST	Anti-ship surveillance and targeting
ASW	Antisubmarine warfare
ATO	Airborne Tactical Officer
BIM	Blade inspection method
CG	Center of gravity
CPU	Central processing unit
DEP	Design eye position
DN	Down
DOD	Department of Defense
ECS	Environmental control system
EFCS	Electronic flight control system
FWD	Forward
GCA	Ground controlled approach
GW	Gross weight
Hg	Mercury (Chemical symbol)
HGT	Height
Hp	Pressure altitude
HQR	Handling qualities rating
HSI	Horizontal situation indicator
ICAO	International Civil Aviation Organization
IGE	In ground effect
IMC	Instrument meteorological conditions
KIAS	Knots indicated airspeed
KTAS	Knots true airspeed
LAMPS	Light Airborne Multi-Purpose System
LDO	Lateral-directional oscillation
LSO	Landing Signal Officer
LT	Left
MAN SLEW	Manual slew
MEDEVAC	Medical evacuation
NASA	National Aeronautics and Space Administration
NATC	Naval Air Test Center
NATOPS	Naval Air Training & Operating Procedure Standardization
ND	Nose down
Nr	Main rotor speed in revolutions per minute
NU	Nose up
OAT	Outside air temperature
OGE	Out of ground effect

PBA	Pitch bias actuator
PCL	Power condition lever
PIO	Pilot induced oscillation
RADALT	Radar altimeter
RAST	Recovery assist, securing and traversing
RWATD	Rotary Wing Aircraft Test Directorate
ROC	Rate of climb
ROD	Rate of descent
RPM	Revolutions per minute
RT	Right
SAR	Search and rescue
SAS	Stability augmentation system
SENSO	Sensor Operator
SHP	Shaft horsepower
TFCP	Trimmed flight control position
USNTPS	United States Naval Test Pilot School
VMC	Visual meteorological conditions
Vh	Maximum level flight airspeed
Vne	Never exceed airspeed
deg C	Degrees Celsius
deg F	Degrees Fahrenheit
g	Normal acceleration due to gravity
nmi	Nautical miles
ΔH_p	Change in pressure altitude
W/α	Weight to density ratio

APPENDIX H

GLOSSARY

GLOSSARY

ADVERSE YAW. An initial tendency for the aircraft to yaw opposite the direction of cyclic input.

AIRBORNE TACTICAL OFFICER (ATO). The officer who occupies the copilot seat and conducts the airborne tactical portion of the ASW mission. The ATO decides which detection devices will be deployed and monitored, and coordinates attacks and re-attacks on hostile targets.

ANTHROPOMETRIC MEASUREMENTS. A person's physical measurements (i.e. total sitting height, sitting eye height, functional reach, etc.) as compared to a data base. For the purpose of this thesis, the data base was a sampling of Naval Aviators compiled by the US Navy Aviation Physiology Department.

AUXILIARY POWER UNIT (APU). The aircraft system designed to provide hydraulic and electrical power while on the ground with rotors static.

BLADE FOLD SYSTEM. The system designed to automatically position and fold the main rotor blades in order to facilitate maintenance and hangaring of the aircraft. The tail pylon folds manually for the same purpose.

BLOWBACK. The tendency of the rotor to flap away from the relative direction of aircraft travel (i.e. aft during forward flight, left during right sideward flight, etc.).

BREAKOUT PLUS FRICTION. The sum of the pre-loaded flight control force and the friction band. Breakout plus friction is the maximum force exerted by the pilot before movement of the flight control from trim. Excessive breakout plus friction can cause "spiked" inputs, over controlling, and/or aggravated stick jump.

CLOSED LOOP. The flight control system including the pilot as an integral component (i.e. making inputs as necessary to maintain a desired level of system performance).

COLLECTIVE. The cockpit control which provides vertical control of the aircraft by collectively increasing or decreasing the pitch of the main rotor blades. The collective is typically held in the pilot's left hand.

CONTROL CENTERING. The tendency for a flight control to return to a trimmed position following displacement. Control centering is a function of breakout plus friction and force gradient. Poor control centering will contribute to a trimmed control displacement band.

CONTROL COUPLING. When movement of a flight control in a single axis results in flight control and aircraft movement in an additional axis.

CONTROL DOUBLET. A flight control input entered by displacing a control a certain magnitude in one direction, reversing the input the same magnitude in the opposite direction, and then returning the control to the original position.

CONTROL DYNAMICS. The resultant motion of a flight control following a release from a given amount of displacement.

CONTROL FORCE GRADIENT. The force required per increment of control displacement once a flight control has been displaced from trim. The average gradient is the slope of the average variation of force versus displacement.

CONTROL JUMP. Uncommanded movement of a flight control when force trim is released. Control jump is a function of control displacement, force gradient, friction band, and viscous damping.

CONTROL MARGINS. The amount of flight control travel remaining before encountering a mechanical control stop. Control margins are expressed as a percentage of total control travel.

CONTROL PULSE. A flight control input entered by displacing a control a certain magnitude and then returning the control to its original position.

CONTROL RESPONSE. The aircraft response resulting from a cyclic, collective, or pedal input.

CONTROL SENSITIVITY. The rate of aircraft response generated by a specific amount of flight control input.

CRITICAL AZIMUTH. While operating in a hover, the wind direction and velocity relative to the aircraft which results in the most reduced control margin and/or the highest vibration level.

CYCLIC. The cockpit control which provides lateral and longitudinal control of the aircraft by cyclicly altering the pitch of the main rotor blades and thereby tilting the rotor in the direction of desired movement. The cyclic stick is typically held in the pilot's right hand.

CYCLIC ONLY TURNS. Turns performed by displacing the lateral cyclic only (i.e. pedals fixed).

DESIGN EYE POSITION. The point in space located at the sitting eye height dimension of the "average" (50th percentile) aviator. Design eye position is the source of measurement for control reach, display visibility, cockpit space accommodation, etc.

DETECTION ZONE. The area in which a possible enemy vessel has been encountered. This vessel could be a surface or sub-surface ship.

DIRECTIONAL PEDALS. The cockpit control which provides yaw control of the aircraft by increasing or decreasing the pitch of the tail rotor blades. The directional pedals are operated by the pilot's feet.

DIRECTIONAL STABILITY. The natural tendency of the aircraft to align itself with the relative wind (weather vaning).

DOPPLER NAVIGATION SYSTEM. A radar navigation set designed to provide the pilot with ground speed, drift angle, and vertical velocity information.

DROOP STOP. Mechanical stops located on the rotor head designed to prevent excessive flapping of the main rotor blades.

DYNAMIC STABILITY. The overall tendency of the aircraft to return to trim conditions after experiencing an external disturbance.

EFFECTIVE DIHEDRAL. Roll due to sideslip.

ENGINE CROSS BLEED. Pneumatic ducting between the two engines enabling compressor bleed air from an operating engine to be used for starting the second engine.

ENGINE TRIM SYSTEM. Fine adjustment of the engine free power turbine speed which enables the pilot to make small main rotor RPM changes. The engine fuel control system

is designed to maintain the engine trim system settings during changes in engine and rotor loading.

FREE HOVER. Hovering the helicopter without a ground restraint (i.e. unteathered).

FREQUENCY SWEEP. A sinusoidal flight control input maintained for some finite period of time in order to examine aircraft response. The application rate of the input can be incrementally altered and the resulting responses examined as required.

FULLY ARTICULATED. A rotor system designed to provide the rotor blades with four degrees of freedom (rotation, pitch, flapping, and lead/lag).

HOT DAY. In accordance with reference (9), hot day sea level conditions are as follows: temperature = 95 deg F, air pressure = 2116.22 lb/square ft (29.92 in. Hg), and air density = 0.0023769 slugs/cubic ft.

HOVER CEILING. The altitude at which the helicopter has power available to hover, but no excess power with which to climb.

HYDRAULIC DAMPER. A mechanical unit designed to provide an opposing force proportional to rate of motion. Hydraulic dampers can be utilized to resist rapid control inputs.

IN GROUND EFFECT. The band of height above the ground within which the effects of rotor downwash reduce the power required to hover (approximately 1.1 times the rotor diameter and lower).

INDICATED AIRSPEED. The aircraft velocity value corrected for instrument error, but not for position, compressibility, or air density errors.

INLET RECOVERY CHARACTERISTICS. The effect of the engine inlet on the free stream air flow, specifically air pressure and temperature rises at the engine compressor face.

INSTRUMENT METEOROLOGICAL CONDITIONS (IMC). Meteorological conditions under which the pilot must rely solely upon aircraft instruments to maintain aircraft attitude and position.

IRREVERSIBLE FLIGHT CONTROL SYSTEM. A system which transmits pilot commands to the aircraft control surfaces via the flight controls, but which does not allow any control surface loads to be felt by the pilot.

LANDING SIGNAL OFFICER (LSO). A ship-based officer responsible for directing aircraft operations (i.e. takeoff, landing, and RAST) over the flight deck.

LATERAL DIRECTIONAL OSCILLATIONS. Oscillations about the roll and yaw axes (i.e. dutch roll).

LIMIT CONTROL FORCE. The maximum flight control force for any given control displacement against trim.

LOCKOUT. An activation switch on each engine PCL designed to disconnect the rotor speed governing and give the pilot a fuel control system override capability.

LONG TERM LONGITUDINAL DYNAMIC RESPONSE. Aircraft pitch response characteristics over a period of time following an external disturbance.

MANEUVERING STABILITY. An evaluation of static longitudinal stability in other than 1 "g" flight. Maneuvering stability is evaluated by measuring longitudinal cyclic displacement with load factor variations.

MISSED TRAP. When the aircraft is being winched to the deck during a RAST operation and does not align correctly with the deck securing device. The aircraft will likely have to perform a takeoff and be winched in again.

MISSION RADIUS. The total distance that the aircraft can fly and still have adequate fuel to perform the mission upon reaching its destination and adequate fuel reserve upon return to the parent ship.

MULTI-PURPOSE DISPLAY. An electronic cockpit display (usually a cathode ray tube) which has the capability of switching data layouts and thereby providing the pilot with a variety of flight and mission information.

NATURAL FREQUENCY. The frequency at which an object will oscillate if not acted upon by an external force.

OPEN LOOP. The flight control system excluding the pilot (i.e. no corrective inputs).

OUT OF GROUND EFFECT. The height above the ground where rotor downwash no longer effects the power required to hover (approximately 1.1 times the rotor diameter and higher).

PEDAL ONLY TURNS. Turns performed by displacing the directional pedals only (i.e. lateral cyclic fixed).

PRESSURE ALTITUDE. The altitude measured when the barometric altimeter is set to ICAO standard day pressure conditions (29.92 in. Hg).

PROPRIOCEPTIVE CUES. The "seat of the pants" feeling a pilot perceives while operating the aircraft. Proprioceptive cues will likely alert the pilot to off-trim conditions.

PULL UPS. A maneuvering stability test technique designed to momentarily increase aircraft load factor by rapidly applying aft longitudinal cyclic and generating a nose up pitch rate.

PUSH OVERS. A maneuvering stability test technique designed to momentarily decrease aircraft load factor by rapidly applying forward longitudinal cyclic and generating a nose down pitch rate.

RADAR ALTITUDE. Height of the aircraft above the ground as determined by an onboard system which transmits a radar signal towards the ground, measures the time for the signal to return, and thereby computes aircraft altitude.

READY ALERT. A flight quarters condition in which the aircraft must be launched within five minutes.

RECOVERY ASSIST, SECURING, AND TRAVERSING (RAST). The mechanical equipment incorporated into the flight deck of a parent ship which aids in the recovery of the helicopter. The RAST is designed to tether the aircraft while in a low, over-deck hover; winch the aircraft to the deck; secure (i.e. trap) the aircraft; and finally, maneuver the aircraft into the ship's hangar.

REFERRED DATA. Performance data corrected to account for variations in ambient conditions and engine variables.

ROTOR DECAY (Nr DECAY). A decrease in main rotor RPM, usually caused by a sudden increase in rotor loading or decrease in engine power available.

ROTOR SPEED GOVERNING (Nr GOVERNING). Maintaining a constant main rotor RPM during changes in engine and rotor loading. Rotor speed governing is accomplished through the engine fuel control system.

SENSOR OPERATOR (SENSO). The aircrew member responsible for operation of the onboard sensors and data processing of information from the same.

SHORT TERM LONGITUDINAL DYNAMIC RESPONSE. The aircraft response to small, non-accelerated pitch attitude changes generated by small longitudinal cyclic displacements.

SIDEFORCE. Lateral lift and drag on the fuselage and tail rotor while in a sideslip.

SONOBUOYS. Expendable submarine detection devices which relay sonic detection information to an aircraft or surface ship. Sonobuoys can be active (pinging/listening) or passive (listening only) in nature.

SPIRAL STABILITY. The natural tendency of the aircraft to return to wings level following a disturbance in the roll axis.

STABILATOR. The moveable horizontal stabilizer which is capable of pivoting from 8 deg trailing edge up to 40 deg trailing edge down.

STANDARD DAY. In accordance with the International Civil Aviation Organization, standard day sea level conditions are defined as follows: temperature = 59 deg F, air pressure = 2116.22 lb/square ft (29.92 in. Hg), and air density = 0.0023769 slugs/cubic ft.

STATIC DROOP. The difference between the initial main rotor RPM and the final steady state RPM as a result of a change in power demand.

STATIC STABILITY. The initial tendency of the aircraft to return to trim conditions after experiencing an external disturbance.

STEADY HEADING SIDESLIP. Flying the aircraft in a sideslip by displacing the directional pedals while maintaining a steady ground track by displacing lateral cyclic.

STEADY TURNS. A constant angle of bank and airspeed turn performed with the collective fixed (i.e. altitude will not remain constant).

STEP INPUT. A rapid, precise flight control input of a specific magnitude maintained for some finite period of time in order to examine aircraft response.

TOTAL SYSTEM FREEPLAY. The lost motion that exists between a flight control and the corresponding aircraft control surface.

TRANSIENT DROOP. The difference between the minimum main rotor RPM achieved and the final steady state RPM as a result of a change in power demand. Transient droop is a function of the rate of change of power demand.

TRIM CONTROL DISPLACEMENT BAND. The band of flight control displacement about trim within which the force gradient is equal to or less than the friction force. No force cues as to the true location of the trim position are provided within this band.

TRIM SYSTEM FREEPLAY. The lost motion that exists between a flight control and the trim system. Sometimes called hysteresis, the pilot may recognize it as "slop".

TRIMMED FLIGHT CONTROL POSITIONS. Flight control positions measured while in a stabilized flight condition. Trimmed flight control positions are expressed as a percentage of total control travel.

TRUE AIRSPEED. The aircraft velocity value corrected for instrument, position, compressibility, and air density errors.

UNREFERRED DATA. Process by which referred data is applied to a set of specific ambient conditions and engine variables.

VISUAL METEOROLOGICAL CONDITIONS (VMC). Meteorological conditions under which the pilot may rely upon outside references to maintain aircraft attitude and position.

W/SIGMA. A comparison of aircraft weight and air density ratio at sea level. As the aircraft burns off fuel (i.e. decreases weight), altitude can be increased to keep W/sigma constant.

VITA

Neil Berryman was born in Norristown, Pennsylvania on September 14, 1959. He attended Great Valley Senior High School in Malvern, Pennsylvania, where he graduated in 1977. After completing a year at the Embry-Riddle Aeronautical University in Daytona Beach, Florida, he transferred to the Pennsylvania State University where he received a Bachelor of Science Degree in Aerospace Engineering in May 1984.

Following graduation he accepted an aeronautical engineering position at the Rotary Wing Aircraft Test Directorate (RWATD) of the Naval Air Test Center (NATC). After one year in the Directorate he was selected to attend the United States Naval Test Pilot School rotary wing engineering curriculum, from which he was the first to graduate in June of 1986. Since returning to RWATD he has worked on helicopter air-to-air combat tests in conjunction with the United States Army, and is currently serving as the lead engineer on the MH-53E helicopter test program. He enrolled in The Graduate School of the University of Tennessee, Knoxville, in September of 1987.

The author is a member of the American Helicopter Society and the American Institute of Aeronautics and Astronautics. He will continue his work with NATC after graduation.