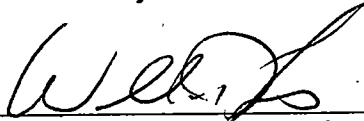
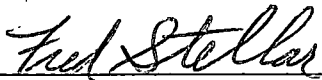


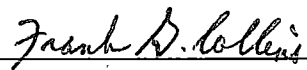
To the Graduate Council:

I am submitting herewith a thesis written by James L. Robbins entitled "Air vehicle Diagnostic System Rotor Head Load and Global Fault Evaluation as Installed on the SH-60F Helicopter". I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Masters of Science, with a major in Aviation Systems


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We have read this thesis and
Recommend its acceptance:


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Dean of the Graduate School

**AIR VEHICLE DIAGNOSTIC SYSTEM
ROTOR HEAD LOAD AND GLOBAL
FAULT EVALUATION AS INSTALLED
ON THE SH-60F HELICOPTER**

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

James Lloyd Robbins
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ABSTRACT

This study covers SH-60F rotor head load estimation, track and balance fault detection, and degraded component diagnostics of the Air Vehicle Diagnostic System Advanced Technology Demonstration. The purpose of this study was to acquire load data from selected SH-60F main rotor components; provide "proof of concept" for the use of neural networks in load estimation; acquire SH-60F main rotor track and balance and vibration data in nominal (no known defect) condition and with intentionally induced known out of track and balance conditions in the main rotor; and acquire SH-60F main rotor track and balance and vibration data in nominal condition and with degraded rotor head components at varying flight conditions. The method of data collection was a survey recorder which houses a memory unit, a neural network and the maneuver recognition algorithms. The memory unit was used to record the load estimation, maneuver recognition algorithms, track and balance, and main rotor degraded component detection system results. Data signals are received directly from the aircraft 1553 data bus, existing aircraft systems, and special sensors designed for each task. Strain gauges are placed at four different rotor head locations to provide truth data for comparison with the neural network data. The instrumentation monitors 37 parameters for three testing phases totaling over 50 flight hours. Results indicate that the neural network estimations are within 10 percent of the actual strain gauge loads and that the main rotor track and balance capability of the system was able to quickly and accurately track the rotor system

and lower vibrations to acceptable levels. The main rotor head fault detection data provided further support for the use of neural networks in vibration analysis.

PREFACE

The Office of Naval Research sponsored investigations into the use of a neural network for diagnostic purposes. The Air Vehicle Diagnostic System Advanced Technology Demonstration evaluated the usefulness of a neural network in the areas of rotor head load estimation, track and balance and global fault detection.

This thesis evaluated the use of neural network systems in conjunction with a structural data recording system to detect rotor head faults and to estimate rotor head loads in an SH-60F aircraft. Additionally, the main rotor track and balance capabilities of the Air Vehicle Diagnostic System were evaluated. Aircraft integration was completed by System Electronics, Inc. and Signal Processing Systems. Testing was conducted at Naval Air Station Patuxent River, Maryland from February to September 1997.

The neural network used in this program was trained by Naval Surface Warfare Center, Carderock Division personnel. The author was the project officer and the project pilot for the AVDS testing. The opinions of the author in no way reflect the opinions of SEI, SPS or the United States Navy.

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LIST OF ABBREVIATIONS

AID IP	Aircraft Instrumentation Division Instrumentation Package
APC	Acquisition Processing Card
ATD	Advanced Technology Demonstration
AVDS	Air Vehicle Diagnostic System
DAMVIBS	Design Analysis Methods for Vibrations
DEK	Data Entry Keyboard
FEM	Finite Element Model
FMC	Flight Mission Critical
FRAN/Vibration Analysis	Finite-Element Airframe Vibration Analysis Program
GFDS	Global Fault Detection System
HHC	Higher Harmonic Control
IBC	Individual Blade Control
KIAS	Knots Indicated Airspeed
MDPS	Maintenance Data Processing Station
MRA	Maneuver Recognition Algorithm
MU	Memory Unit
NADEP	Naval Aviation Depot
NASTRAN	NASA Structural Analysis System
NN	Neural Network
NR	N Per Rev

NSWC CD	Naval Surface Warfare Center, Carderock Division
PCR	Pitch Control Rod
PHV	Pilot Heal Vertical
PRDL	Princeton Rotorcraft Dynamics Laboratory
R	Linear Correlation
1R	Once Per Revolution
2R	Twice Per Revolution
RIEVA	Rotor Induced Empennage Vibration Analysis
RMS	Root Mean Square
ROTABS	Rotor Trim and Balance System
SDC	Signal Data Converter
SDRS	Structural Data Recording Set
SEI	System Electronics, Inc.
SOFM	Self Organizing Feature Map
SPS	Signal Processing Systems
SR	Survey Recorder
T&B	Track and Balance
TAP	Test Article Preparation
USW	Under Surface Warfare
UTD	Universal Tracking Device
VATS	Vibration Analysis Test Set
VMC	Visual Meteorological Conditions

CHAPTER I

STATEMENT OF THE PROBLEM

Dynamic component degradation, such as wear, spalling and cracks, results in measurable changes to the vibration signals produced by those components. Excessive rotor head induced vibrations can also result in component wear thus reducing the mean time between failure and can be correlated to rotor component degradation. Present helicopter drive train vibrations are normally identified by the aircrew through "feel" or sound or through the use of basic analysis equipment. At best these methods typically offer detection only after significant component degradation.

Recent technological advances have been made in vibration diagnostics. Through the use of current analysis techniques and hardware, the time required for diagnosis of the entire helicopter drive train is lengthy. In order to perform on-board drive train diagnostics with all available sensor inputs including chip detector readings, debris monitoring and vibration sensors, a significant increase in either the speed of processing or the simplification of the method of fault detection is required. Neural networks (NN) have shown promise in pattern matching these inputs and offer the speed of true parallel processing. The Air Vehicle Diagnostic System (AVDS) Advanced Technology Demonstration (ATD) evaluated the usefulness of NN in the areas of rotor head load estimation, rotor head track and balance (T&B) and rotor head fault detection.

CHAPTER II

REVIEW OF LITERATURE

Although much time, energy and money has been devoted to the study of airframe and rotor system vibrations, designers still require more data to effectively manufacture helicopters that will endure the life cycle expected within current industry budget constraints. These shrinking budgets and rising repair costs demand development of options that will extend the life of the airframe and the mean time between component failure. Over the past two decades, research has been conducted into various methods of vibration analysis and reduction. Key studies are outlined herein.

Higher Harmonic Control

Higher Harmonic Control (HHC) reduces helicopter airframe vibration through the exercise of rotor blade pitch control at frequencies that are higher harmonics of rotor rotation. Recent HHC studies show potential for measurable vibration reduction.

Princeton Rotorcraft Dynamics Laboratory (PRDL) designed and tested a helicopter model based on the idea that rotor states can be identified by instrumenting the rotor blades. The test team believed that Individual Blade Control (IBC) or HHC pitch input commands could be used to reduce vibrations. The instrumentation configuration used miniature accelerometers to measure and estimate first and second out-of-plane bending mode positions and velocities. To verify the technique, the model was tested,

and resulting data were used to estimate rotor states as well as flap and bending coefficients. Overall test results validated the development of a cost-effective model for future active control experiments (Marraffa and McKillip, 1989).

Harmonic rotor hub loads and airframe interactions in steady flight were reviewed to achieve lower airframe vibration by modifying blade root loads. Flight test and wind tunnel data were reviewed along with sample fuselage response data. Conclusions indicated that improvement of blade tuning could reduce airframe vibrations. Test-based development of favorable blade configurations was shown not only to be feasible but also to generate data to guide further development of these analytical methods (Neibanck, 1984).

Another rotor hub study involved the 5A349/2 research helicopter. Hub loads, which transmit excitations to the fuselage, were predicted through vibration analysis and then correlated with measured hub loads. The predicted and measured hub loads were then coupled with finite element models (FEM) representing the 5A349/2 helicopter fuselage. The resulting vertical acceleration at the pilot's seat was examined. Adjustments were made to the airframe structural models to examine the sensitivity of predicted vertical acceleration to the model. Changes of a few percent to the damping and frequency of specific models led to large reductions in predicted vibration and major improvements in the correlation with measured pilot-seat vertical acceleration (Hefferman, et. al., 1991).

OH-6A helicopter wind tunnel and flight test analysis using HHC technology demonstrated a reduction of vibration levels greater than 80 percent. Blade feathering

capability at rotor speeds other than the first harmonic showed acoustic improvement, increased aerodynamic efficiency and ground resonance reduction (Gupta, et. al., 1984).

Airframe Stiffening

The Chinook helicopter airframe vibrations and loads are concentrated mainly at 3/rev and 6/rev vibration. Lightweight, local airframe stiffening was analyzed for the ability to reduce vibration and airframe fatigue loads. The 3/rev levels were amplified by two nearby modes of the forward pylon, a longitudinal/pitch mode and a lateral/roll mode. At 6/rev, there were a number of airframe modes which contributed to vibration, mainly two aft pylon-pitching modes. This study showed that moving the modes farther away from the rotor forcing frequencies significantly reduced vibration levels (Gabel and Lang, 1995).

NASA Structural Analysis System

The NASA Structural Analysis System (NASTRAN) computer code embodies mathematical models based on the finite element method of structural analysis. These models are routinely used by the helicopter industry to calculate airframe static internal loads for sizing structural members. Historically, little reliance has been placed on vibration predictions based on these models.

In the early 1980's, NASA's Langley Research Center initiated an industry wide program with the objective of creating trust in vibration predictions using the NASTRAN models. A secondary objective was to establish a body of modeling guides which would

enable confident future prediction of airframe vibration and incorporate these predictions as a regular part of the design process. Emphasis was placed on the successful modeling of the Army/Boeing CH-47D which showed correlation with test data. The FEM of the airframe vibration of the Army/Boeing CH-47D helicopter was conducted with comparisons to experimental data in an effort to improve the design process. A NASTRAN FEM was developed that was fully representative of the test configuration and included the support fixture, shakers, and the aircraft/shaker suspension system. Specific attention was given to the prediction of reasonable forced amplitudes throughout the airframe. Reasonable correlation was noted between the FEM and experimental results, although improved correlation could be obtained by including more accurate damping values and secondary effects such as stringer shear loading. A principal finding indicated that improved dynamic analysis required greater attention to detail and perhaps a finer mesh than the usual stress model, especially in the mass distribution (Gabel, et. al., 1992 and 1993).

Under the NASA Design Analysis Methods for Vibrations (DAMVIBS) research program, Bell Helicopter, Inc. conducted NASTRAN modeling, ground vibration testing, and correlation of both metallic (AH-1G) and composite (ACAP) airframes. There has been low confidence in higher frequency vibration predictions for helicopters that have three or more blades with predominant excitation frequencies above 15 Hz. The objectives of the program were to assess NASTRAN airframe vibration correlation, investigate discontinuities and improve modeling techniques. The results under the DAMVIBS program indicated comparable accuracy of FEMs for both composite and

metallic airframes, significant stiffening of secondary structures not normally included in the FEM and phase responses of higher frequencies under damping conditions (Cronkhite, 1993).

A series of ground vibration tests and NASTRAN FEM correlations was conducted on the Bell AH-1G helicopter gunship to investigate the effects of aircraft components on the vibration response of the airframe. Previous correlations of the AH-1G showed good agreement between NASTRAN and test results through 15 to 20 Hz but poor agreement in the higher frequency range of 20 to 30 Hz. To conduct the investigation, the main rotor pylon, secondary structure, nonstructural doors/panels, landing gear, engine and furl were systematically removed to quantify their effects on the overall vibratory response. Secondary structure and damping had significant effects on the frequency response of the airframe above 15 Hz. Also, the nonlinear effects of thrust stiffening and elastomer mounts were significant on the low frequency pylon modes below main rotor 1 per vibrations (5.4 Hz) (Dompka, 1989).

Using the NASTRAN analysis, design processes with acceptable vibration levels were achieved with minimal modifications during development flight test. The analysis involved tuning the pylon isolation system and fuselage structure to minimize vibrations and position the important, hard-to-move modes away from rotor excitation frequencies. The NASA DAMVIBS research program provided an understanding of higher frequency vibration correlation, identified modeling deficiencies to improve main rotor 4/rev vibration predictions and contributed to the design of the V-22 three-bladed tilt rotor (Cronkhite, 1989).

In another study, level flight airframe vibration at main rotor excitation frequencies was calculated. NASTRAN tailboom analysis was compared with test data to determine effected skin in a semimonocoque sheet-stringer structure. The flight vibration correlation involved comparisons of level flight vibrations for two helicopter configurations: clean wing at light gross weight and wing stores at heavy gross weight. In the tailboom correlation, deflections and internal loads were compared using static test data and NASTRAN analysis. An iterative procedure was used to determine the amount of skin with buckled panels under compression load (Cronkhite, et. al., 1976).

Rotor Trim and Balance System

Conventional rotor trim and balance methods rely to some extent on the optical alignment of the blade tips. The Rotor Trim and Balance System (ROTABS) was a unique system that tackled the problem at its source by making a multi-plane analysis of the airframe vibration characteristics using accelerometers. The system was based on the philosophy that the rotor configuration producing the least vibrations would be the best possible combination for performance, comfort and reduced structural fatigue. Through the use of ROTABS vibration levels were significantly reduced from the levels previously achieved by helicopter operators and manufacturers in the U.S. and Europe using conventional systems (Staub and Ventres, 1992).

Advanced Helicopter Technology

Advancing helicopter technology is not aimed at changing the shape of rotorcraft but at eliminating or overcoming their inherent disadvantages. A novel technical project was recently designed to analyze airframe vibration caused by the rotor, process the information through an adaptive control unit and utilize actuators to introduce a counter vibration into the airframe (Barrie, et. al., 1992).

Finite Element Modeling in Rotor/Body Helicopter Systems

A consistent finite-element formulation was developed for the prediction of vibration in rotor/body helicopter systems in forward flight, taking into account interactional rotor/body aerodynamic loads and dynamic coupling. The rotor blades were assumed to be elastic beams undergoing flap and lag bending, torsion and axial deflections. The coupled analysis was formulated retaining non-linear structural, inertial and aerodynamic terms. Rotor excitation included rotor/body interactional loads and fuselage dynamic coupling. The effects of several parameters on vibratory hub loads and body vibration were investigated including blade stiffness, rotor/body clearance, hub location, fuselage stiffness and advance ratio. A significant influence of body upwash on the rotor disk was found to cause vibratory hub shear, an effect which increased with smaller rotor/body clearance. Tuning of rotor and body natural modes substantially reduced vibration levels. Elastic deflections of the body had considerable influence on vibratory hub loads (Fledel, 1989).

Coupled Rotor/Airframe Vibration Analysis

A coupled rotor/airframe vibration analysis was developed as both a design tool for predicting helicopter vibrations and a research tool to quantify the effects of structural properties, aerodynamic interactions and vibration reduction devices. The analysis consisted of a base program that utilized an impedance matching technique to represent the coupled rotor/airframe dynamics of the system. The program was supported by inputs from several external programs supplying advanced rotor and airframe aerodynamic and structural dynamic representations. Correlation results using scale model wind tunnels showed that the analysis could adequately predict trends of vibration variations with airspeed and HHC effects. Predictions of the absolute values of vibration levels were found to be very sensitive to model characteristics (Sopher, et. al., 1982).

A coupled rotor/airframe vibration analysis was performed by the matching of rotor and fuselage impedances. The rotor impedance for the blades was calculated from the periodic coefficient equations of a single blade in forward flight. Three flapping modes were included, and the equation was solved by harmonic balance. The fuselage impedance, including structural damping, was calculated for three rigid-body and three elastic modes in plunge, roll, and pitch. The results showed that the effect of hub motions on rotor loads was greatest for relatively stiff rotors and was not well approximated by lumped-mass or purely inertial rotor models (Hsu and Peters, 1980).

Rotor Induced Empennage Vibration Analysis

A method was developed to determine the unsteady vibratory airloads produced by the interaction of the main rotor wake with a helicopter empennage. This method was incorporated into a computer program called Rotor Induced Empennage Vibration Analysis (RIEVA). The program required input from the main rotor wake position and was dependent on the strength of the vortices located near the empennage surfaces. A non-linear lifting surface analysis was utilized to predict the aerodynamic loads on the empennage. The analysis included the suction of the interacting vortices and the shed vorticity behind the empennage surfaces; it employed a time domain solution. The output of the program consisted of chordwise and spanwise airload distributions on the empennage surfaces. The airload distributions were harmonically analyzed and formulated for input into the Coupled Rotor/Airframe Vibration Analysis (Gangwani, 1982).

Vibration Isolation Systems

A vibration isolation system was developed for the passenger cabin and the long-range fuel tanks of the commercial Chinook helicopter. The passenger floor was isolated from the airframe on a series of passive isolation units. The fuel tanks were also isolated so that their dynamic mass was effectively nulled at all fuel levels thereby avoiding any harmful effect on airframe natural frequency placement. Analyses, component tests, and aircraft shake tests were conducted to verify the effectiveness of the system. The tests demonstrated that the floor isolation could lower the 0.15-g mid cabin

airframe vibration to an average of 0.05-g on the passenger floor. The fuel isolation was also successful, maintaining an important airframe natural frequency within ± 0.2 Hz of its normal value for any fuel level from 0 to 100 percent (Desjardins and Sankewitsch, 1979).

Heavy Lift Helicopter Improvements

Various improvements relating to performance, flying qualities and reliability based on specifications for the Heavy Lift Helicopter and the YUH-61 were developed. Hingeless rotors and improved single rotor yaw control were shown to improve rotor hover performance, reduce airframe vibrations and improve pitch and roll control. Secondly, improvements were seen in aircraft control in a hover for cargo handling, drive system development, reliability, maintainability and improved survivability emphasizing defect tolerant design (Hooper, 1975).

Sikorsky Finite Element Airframe Analysis Program

Static and dynamic degrees of freedom, cockpit structural modeling and the extent of flexible-frame modeling in the transmission support region and in the vicinity of large cutouts were selected as modeling platforms with general application to finite-element dynamic analyses. The testing showed that predicted results were very sensitive to the modeling assumptions (Keningsberg, et. al., 1974).

Sikorsky Finite Element Airframe Analysis Program (FRAN/Vibration Analysis) was found to correlate well with data taken in shake tests of the CH-53A. The

frequencies of fundamental fuselage bending and transmission modes were predicted by the FRAN/Vibration Analyses to an average accuracy of 3 percent. In addition, the mode shapes were defined accurately. The modeling techniques and analysis used in the study were applied during helicopter design and during evaluation of growth versions of current aircraft. These techniques, in combination with an accurate definition of the vehicles structural characteristics, enabled accurate prediction of all transmission and fuselage modes for vehicles without cargo ramps (Keningsberg, 1973).

Drag Effects on Dynamic Blade Loads

Helicopter drag as reflected in rotor propulsive force requirements was found to have a significant effect on dynamic blade loads. These dynamic loads also generated excitation of airframe vibrations. Detailed analysis assessed the impact of helicopter drag on blade and rotor hub loads. Predicted dynamic loads were presented for a 16,000 pound helicopter with a four-bladed stiff-inplane hingeless rotor flying at 150 knots with airframe (including hub and pylon) parasitic drag levels between 0 and 30 square feet of equivalent flat-plate drag area. Dynamic loads were found to have detrimental effects on blade life and airframe vibrations. Reduction of airframe drag decreased vibration excitation and blade loads significantly which resulted in the opportunity for weight reduction and, secondarily, extended blade life (Kerr, 1975).

Absorber Tuning

Absorber tuning was found to help alleviate primary airframe vibrations in helicopters. An absorber with a variable tuning capability was developed, designed, bench tested and flight tested successfully in a CH-47C helicopter. A control system to make this variable tuning absorber self-tuning was incorporated and successfully tested. The CH-47C helicopter operated over a range of rotor speeds with no adverse effect on the acceptability of the primary airframe vibration (Oleary, 1969).

Regime Recognition Algorithms

In order to more accurately assess the fatigue life remaining on flight critical components, the U.S. Navy developed flight regime recognition algorithms for usage monitoring on the AH-1W Helicopter. The system identified the maneuver flown and the duration of the given maneuver to allow for calculation of fatigue damage accumulation on an aircraft-to-aircraft basis. Usage monitoring allowed for less severe calculated fatigue damage accumulation rates. The associated vibratory load was obtained from historical data and was typically the worst-case load applied over the duration of the maneuver (Brandt and Moon, 1994; Moon, et. al., 1996).

Conclusions

Accurate vibration analysis requires simultaneous monitoring of multiple parameters and comprehensive data collection. Small, lightweight systems that provide real-time monitoring of rotor system and airframe vibrations are now feasible and well

within the capabilities of current technology. Although the study of airframe and rotor system vibrations is rapidly advancing, more modeling and flight data are required for further refinement of these systems. In order to produce cost effective, long lasting helicopter components the science of airframe vibration analysis must continue.

CHAPTER III

DESCRIPTION OF TEST SYSTEM

1. TEST EQUIPMENT OVERVIEW

The AVDS used in this testing was created through a joint effort between Naval Surface Warfare Center, Carderock Division (NSWC CD), Signal Processing Systems (SPS) and System Electronics, Inc. (SEI). NSWC CD designed, manufactured, and trained the NN used for the load estimation tests. SPS developed and implemented the T&B hardware and software as well as the Global Fault Detection System (GFDS). SEI was tasked to incorporate the NN, T&B system, and GFDS into an upgraded version of their AN/ASH-37 SDRS system hardware. SEI also provided regime recognition algorithms specific to the SH-60F aircraft by modifying their Maneuver Recognition Algorithms (MRA) currently used on other aircraft. The MRA triggered automatic data collection. The AVDS also allowed data collection via crew input. The final Structural Data Recorder Set (SDRS) consisted of a Survey Recorder (SR), a Signal Data Converter (SDC) and a data entry keyboard (DEK) that had a footprint of approximately two square feet. Figure A-1 shows the components as installed behind the pilot's seat.

SEI provided the equipment and software that allowed for data to be transferred from the aircraft and displayed on the ground station computer. The equipment included a portable Maintenance Data Processing Station (MDPS) and PC-based software for viewing load data. Figure A-2 shows the MDPS. All system components and the

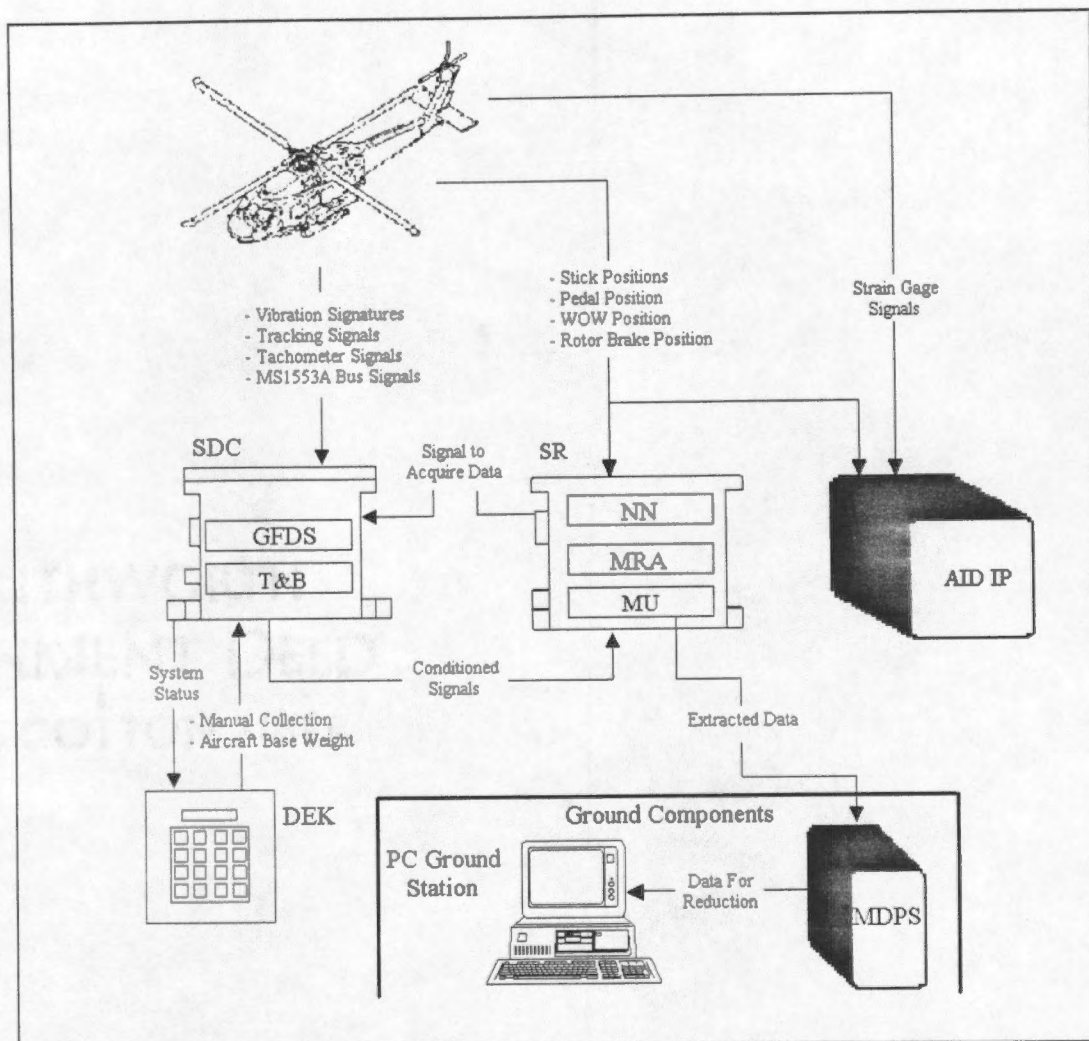


Figure 1. AVDS System components.

associated interactions are shown in Figure 1. The locations of the system's T&B and GFDS sensors are shown in Figures A-3 through A-6. Signals monitored by the AVDS are listed in Table B-1.

The SR housed the Memory Unit (MU), the NN, and the MRA. All data monitored by the system were recorded in the MU at a rate of 8 samples per second. The

MU was also used to record the NN load estimation, MRA, T&B, and GFDS system results. The data in the MU was accessed using the MDPS and then transported to the ground station for analysis. During load estimation flights, the MRA answers were used to classify the NN answers so they could be compared from flight to flight in the ground station. The MRA answers were also used to trigger automatic data acquisition in flight during T&B and GFDS test flights.

The SDC contained either the T&B or GFDS hardware depending on the current flight requirements. It also conditioned certain aircraft measured parameters into a form readable by the SR. The SDC received signals directly from the aircraft 1553 data bus, existing aircraft systems and special sensors designed for the task. The SDC was commanded by the SR when to collect the data and when the required maneuvers ended. The SDC also determined if the maneuver duration was sufficient before sending T&B and GFDS data to the MU for storage.

The SDC controlled the DEK. The DEK was used to input information necessary to accurately calculate gross weight. Store configuration, aircraft base weight, aircraft fuel weight and any external fuel or stores were considered in gross weight calculations. The aircrew could also use the DEK to manually trigger data acquisition and show the AVDS status on its display panel.

An Aircraft Instrumentation Division Instrumentation Package (AID IP) was used to record structural load information, via strain gages, that was used as the truth standard for the NN load estimation tests. The AID IP was a pre-existing package developed by the Test Article Preparation Division (TAP), Patuxent River, MD. The IP had the

capability of recording all relevant load information and aircraft state parameters to magnetic tape during flight. Signals monitored by the AID IP for this test are listed in Table B-2. A vertical acceleration meter was also mounted in the cockpit where the pilot's 8-day clock is normally located.

2. DEVELOPMENT OF LOAD MONITORING ALGORITHMS

Background

The NN mapped a set of inputs to a set of outputs through a series of nonlinear transfer functions. The feed-forward NN consisted of an input layer, one or more hidden layers and an output layer. The hidden layer was composed of individual processing elements. Each processing element contained a transfer function, a bias and a connection weight associated with a specific input. The most distinguishing feature of the NN was the capability to learn the relationship between the inputs and outputs through an iterative process of adjusting connection weights. During the training phase, the connection weights were initially set to a random value and the input data were passed through the network. Initial output values were calculated and compared with desired output values; the error was then calculated and back propagated through the network. The connection weights were then adjusted, and new output values were calculated. This training process continued until the predicted versus actual load error was less than 2%.

A key aspect in successfully training the NN was selecting the proper training data set. The training data set had to include data that were representative of the problem domain. However, in some regions of the input space, data were sparse and training data

selection using a random data selection process resulted in the sparse areas not being represented. In order to ensure that sparse data spaces were represented, a training data selection method using a Self Organizing Feature Map (SOFM) was used. The SOFM was a NN algorithm which used unsupervised learning to develop a map of the input space. The SOFM was used in this analysis to transform a 12-dimensional input space (Table B-3) into a 2-dimensional grid. A 50-by-50 node mapping layer was used to map the data into 2500 bins. Several points from each bin were then selected as training data points. This approach yielded a training data set which closely resembled the data population while also including sparse data regions.

The population of load data was skewed toward low loads since a significant amount of aircraft flight time was spent in steady flight maneuvers such as level flight and hovers. Since the SOFM represented this data population, the training data selected using the SOFM approach was also skewed toward the low loads. To increase the importance of high loads in the training data selected using the SOFM, some data points were weighted such that the high loads held increased importance in the data set. Using this approach, only 7500 data points were required to train the NN.

The type of network architecture used for load prediction was a modular NN. The modular NN (Figure A-7) was a "collection-of-experts" style network using supervised learning. In this collection-of-experts network, several NN functioned in concert with one another. Each network was an expert in some area of the problem domain, and there was a mechanism for determining which expert responded to a specific input. During training, the expert networks learned how to predict the proper response while the gating

layer learned how to partition the data space across the experts. The modular NN used for this study employed three expert networks, each with 65 hidden nodes, and a gating network with nine hidden nodes. The network used 12 input parameters (Table B-3) to predict component loads. These inputs were parameters which were typically measured on a flight data recorder unit. The expert networks were trained with the learning schedule transition point set to 10 times the length of the training data set. The network was trained for approximately 400 passes through the training data set (McCool, 1998).

Baseline Data

The load prediction algorithms were developed using flight data from an SH-60B flight loads survey test program. During the flight loads survey program, a wide variety of SH-60B usage spectrum maneuvers were flown at several different gross weights, altitude and center of gravity combinations. The load survey test aircraft was instrumented with strain gauges and a slip ring assembly to measure loads in the rotor system. Fifty-six spectrum-representative load flights were selected for a baseline data set which was used for NN training and validation. This baseline data consisted of approximately nine flight hours of data corresponding to 255,401 data points sampled at 8Hz. Forty-eight different flight regimes were included in the baseline data. Table B-4 provides a regime description for each regime number.

To better understand the loads associated with each flight regime, the 95th percentile measured pitch control rod (PCR) and damper loads were analyzed for the 48 different regimes. The 95th percentile load was determined by grouping the load

measurements in each regime into 50 lb load intervals. The 95th percentile load corresponded to the load interval below which 95% of the data points in a regime fell. The 95th percentile load provided a better representation of the load variation within the regime (McCool, 1998).

Network Validation on Baseline Data

Once the network was trained to a sufficient level, the connection weights were fixed and the network was assessed on validation data. Validation data were distinct from training data and included data the network had not been exposed to. For the load monitoring algorithms, 7500 points out of the total 255,401 baseline data points were used for training. Since the training data were such a small percentage of the total baseline data set, results were presented for all 255,401 points.

Network load predictions were presented for two components: the main rotor PCR and main rotor lead-lag damper. Results were assessed in several ways. Overall statistical performance is shown in Table 1. Linear correlation, R , indicated how linear the relationship was between the actual and predicted loads. A number close to 1.0 indicated that a linear relationship exists. Root mean squared (RMS) error was shown in normalized form as a percentage of the maximum vibratory load. The maximum load corresponded to the largest anticipated load and was determined from inspection of the baseline data set. The maximum vibratory load was 5000 lb for the PCR and 7000 lb for the damper. Since the baseline data consisted of a variety of high load maneuvers which were performed to determine the boundaries of the aircraft operating envelope, this

Table 1. Statistical summary of load prediction results for the baseline data.

Load	R	Normalized RMS Error (%)
Pitch Control Rod load	0.94	3.65
Damper load	0.78	7.62
Damper > 60 kts	0.88	4.14
Damper < 60 kts	0.61	13.51

number was considered to be a reasonable estimate for the largest anticipated load. For assessment of load predictions, the normalized RMS error was the primary measure of network accuracy.

Analysis of the prediction results showed that between 94 and 96% of the data point errors fell within a range equal to twice the RMS error for a given load. For this reason, there was a 95% confidence that the load predictions would be accurate to within twice the RMS error value. Using this approach, there was a 95% confidence that the PCR load predictions were within $\pm 7.3\%$ of the load range for the baseline data.

Damper load prediction results were found to be significantly different when the baseline data were separated into prediction above and below 60 kts of forward airspeed. Above 60 kts, the normalized RMS error was 4.1% of the load range, while at airspeeds below 60 kts, the RMS error was 13.5% of the load range. This significant change in prediction accuracy may have been due to the damping characteristics of the SH-60 lead-lag damper which operated in two separate modes depending on blade lead-lag velocity. At small lead-lag velocities, the slope of the force-velocity curve was rather high, while at higher lead-lag velocities, the damper entered a bypass mode and the slope of the force-velocity curve was much less. This transition into bypass mode occurred at

airspeeds between 60 and 70 kts. Below 60 kts, the damper load was quite sensitive to the blade lead-lag velocity which was not a NN input. This phenomenon was likely to be a reason for decreased prediction accuracy for data below 60 kts. In addition, blade lead-lag velocity was a function of aircraft airspeed which was not reliable below 40 kts. This may have also contributed to damper prediction inaccuracy in the low speed flight regimes, however only 12.5% of the flight maneuvers were flown at less than 60 kts.

Network prediction results were also assessed based on the maneuver flown. RMS error normalized by maximum load was calculated for PCR and damper predictions on a flight regime basis. PCR load predictions were the most accurate with RMS error less than 10% for all regimes. Only four regimes had RMS errors exceeding 5% of the PCR load range: rearward flight, 45 deg angle of bank left turns and 60 deg angle of bank right and left turns where the 60 deg turns represented the worst case scenario. PCR RMS error for the turns was further analyzed, and it was concluded that data points with high gross weight values contributed to the higher RMS error levels. Damper load predictions had a wider range of prediction accuracy. All regimes with RMS error exceeding 10% of the load range were low speed regimes which were consistent with the statistics presented in Table 1. Figures A-8 and A-9 show time-history traces of the predicted versus actual PCR and damper loads for a 60 deg angle of bank right turn. While the PCR RMS error was highest for 60 deg angle of bank turns, results indicated (Figure A-8) that the network predictions captured the change in PCR load levels during this dynamic maneuver (McCool, 1998).

3. TRACK AND BALANCE SYSTEM

Tracker

The Universal Tracking Device (UTD) provided blade track and lead-lag information to the diagnostic algorithms. The tracker detected the dark-to-light and light-to-dark transitions which resulted when a rotor blade shadowed one of two projected beams from the tracker photo diodes (Figure A-10). A pulse was generated when the light transition was detected. As the distance between the optical blade tracker and the rotor blade increased, the period between the pulses detected by each photo diode increased. Accurate pulse time measurements were made by the acquisition processing card (APC). This time pulse information, along with aircraft configuration information, was used by the APC tracker algorithms to compute accurate track and lead-lag information.

Accelerometers

Three accelerometers, mounted in the cockpit (figures A-3 and A-4), measured airframe vibration levels.. The signals from the accelerometers were processed through the SDC to produce the magnitude of vertical vibration in g's, the vibration phase in radians and the vibration period in seconds. The accelerometers provided vibration signatures used for main rotor balancing adjustment calculations.

4. GLOBAL FAULT DETECTION SYSTEM

Global rotor system faults refer to faults that can be detected using remote measurements of "global" system parameters, for example, fuselage vibrations. Typically, a global fault is characterized by a gradual degradation in a component and does not result in sudden catastrophic failure. Current rotor T&B systems provide some indication of when a fault or damage exists, i.e., when the recommended adjustments do not have the intended effect. For example, high vibrations that cannot be alleviated with T&B adjustments may indicate a faulty component. Availability of rotor fault data is very limited due to safety of flight concerns when flying with degraded components; the degradation must remain within the allowable margins of the part. Ideally, data is needed for different levels of degradation ranging from good to near failure.

The rotor T&B and global fault detection data acquisition system for AVDS involved a hardware and software upgrade to the SDRS. Typically, rotor T&B systems record once per rev (1R) and N-per rev (NR) vibration data (where N is the number of rotor blades). SPS developed an APC for vibration and rotor track measurements for insertion into the upgraded SDRS. The APC utilized dual-sided surface mount technology and measured approximately 3.5 by 5.1 inches. It included 16 vibration channels, two tachometer signals and one blade tracker input. The vibration data were processed on-board to extract synchronous once per rev through 12 per rev amplitude and phase data. Forty revolutions of data were averaged to produce the vibration and track results. Three accelerometers were installed: co-pilot vertical (A), pilot vertical (B) and pilot heel vertical (PHV). The aircraft vertical vibration was obtained from vector

addition (in phase component) of the pilot and copilot sensors ($A+B$). The lateral/roll vibration was obtained from vector subtraction (out of phase) of the pilot and copilot sensors ($A-B$) (Haas, 1998).

CHAPTER IV

METHODS

AVDS testing consisted of 64 flights totaling approximately 78 flight hours on a U.S. Navy SH-60F helicopter BuNo 163283. A brief description of the test aircraft is presented in Appendix C. Three test phases were conducted. Phase I consisted of evaluation of NN loads estimation; Phase II evaluated rotor T&B fault detection; and Phase III evaluated the rotor head GFDS monitoring capability. All flights were conducted under day visual meteorological conditions (VMC).

During Phase II and III flights, the Vibration Analysis Test Set (VATS) system was used to collect truth data to ensure that the aircraft did not exceed flight clearance limits during fault flights and that the aircraft was within the Flight Mission Critical (FMC) limits during baseline flights. The VATS system measured main rotor track spread and airframe vibration levels through the use of a hand held video camera and accelerometers, respectively. The VATS system presented the track spread data in inches and the accelerometer data in in/sec (VATS system automatically converted accelerometer data to a velocity). The flight clearance specified that vibration levels could not exceed 0.8 ips and the main rotor track spread could not be 6 inches or more.

Thirty-eight load estimation flights totaling 50 flight hours were flown during Phase I. These flights evaluated the NN load estimation capabilities using the strain gage data as truth data. Strain gauges were installed on several rotor system components including the

main rotor PCR, damper, and lateral and forward longitudinal servos (Figures A-11 through A-14). Analog strain gauge measurements were recorded on a flight data recorder and then digitized and post processed into steady and vibratory components. The strain gauge instrumentation provided a reference strain measurement to compare with the NN load predictions. The maneuvers for these flights are listed in Table B-5. The maneuver matrix was flown in two flight regimes. The first regime included all of the low airspeed maneuvers, and the second two included all other maneuvers. The entire maneuver matrix was performed at two different take-off gross weights: 19000 lb. and 20600 lb. The 20600 lb. gross weight flights required that the aircraft carry two auxiliary external fuel tanks.

Thirteen T&B flights which totaled 15 flight hours were flown during Phase II and followed a standard SH-60F main rotor T&B profile as outlined in the SH-60F Functional Check Flight Checklist (reference 27). These flights were designed to evaluate the system's main rotor smoothing and tail rotor balancing capabilities. Testing was conducted by flying a baseline flight and then inducing a known fault into the rotor by adjusting the main rotor pitch rods, hub weights, or trim tabs. After the fault flight, the T&B coefficients relating rotor adjustments to vibration and track condition were verified with previous test results. AVDS software then provided a maintenance action recommendation to bring the aircraft back within T&B limits (reference 24) to a FMC status. The FMC limits are 4 inches of track spread (2 inches or less optimal) and maximum vibrations of 0.3 in/sec (ips). The recommended maintenance action was performed, and another flight was flown to evaluate the AVDS T&B solution. Table B-6 summarizes all of the aircraft maintenance adjustments

made for each flight during this phase of testing while Table B-7 presents the flight maneuvers.

During Phase III testing, three non-adjustable rotor faults were examined during 13 separate flights. The three faults included a worn upper PCR bearing (Figure A-15), a worn inboard damper bearing (Figure A-15) and a lead-lag damper with an internal hydraulic leak (Figure A-16). The maneuvers flown and the inserted faults for these flights are presented in Tables B-7 and B-8. The same rod end bearing was used for both the upper pitch rod and the inboard damper bearing faults. The worn bearing was at the fleet removal limit of 0.015 inch for the ball-race gap. The gap was verified throughout testing and did not deviate from 0.015 inch. In each case, a baseline flight was flown, then the fault was inserted, and data was collected. After the seeded fault flight, the rotor system was returned to the baseline configuration and re-tested. Flight clearance restrictions required that the worn bearing remained within allowable flight limits and the number of flights and flight hours with faults inserted was restricted to 6 flights of no more than 2 hours each for a total of 12 flight hours. After each flight, the bearing was inspected. The faulty lag damper was bench tested prior to flight to establish that the force velocity characteristics were safe. Flight time with the faulty lag damper was restricted to 4 hours.

Data were reduced at the Naval Surface Warfare Center in Bethesda, Maryland. Computer programs used for data reduction were developed by Kelly M. McCool and David J. Haas.

CHAPTER V

RESULTS

1. PHASE I

Load Estimation Capabilities

During the flight test demonstration, a total of approximately 50 flight hours of data were collected. These data covered a wide variety of flight regimes representing a significant portion of the SH-60F usage spectrum. NN predictions were performed on-board the flight test aircraft in real time. Table 2 shows the overall prediction statistics for the instrumented components. To highlight the extent to which the NN predictions followed the trends of the strain gauge measurements during the flight test, a typical time history trace of predicted versus actual load is presented in Figures A-17 and A-18. These figures show the PCR and damper loads respectively for a high-g symmetric pullout maneuver. Vibratory PCR and damper load predictions were comparable with actual strain gauge measurements. The overall trend, as well as the increase in load level, was captured in both cases. The predicted versus actual loads for the lateral and forward longitudinal servos were equally as successful.

Data gathered from the NN can be used to determine actual vice predicted loads. Currently, component removal and replacement schedules are determined by flight hour accumulation intervals and are based on worst case estimations. NN data can be analyzed to best determine actual loads experienced by components resulting in more accurate

Table 2. Statistical summary of load prediction results on flight test data.

Load	R	Normalized RMS Error (%)
Pitch Control Rod load	0.91	2.11
Damper load	0.84	7.24
Damper > 60 kts	0.83	3.22
Damper < 60 kts	0.49	16.06

maintenance inspection and replacement schedules.

2. PHASE II

General

Phase II tests were designed to demonstrate the system's main and tail rotor balancing and main rotor tracking capabilities. These T&B results were recorded in the MU and then downloaded to the ground station for review. The data were analyzed using software designed by SPS specifically for this purpose.

Automatic Acquisition Capabilities

The automatic acquisition capabilities of the system were also evaluated during Phase II testing. The system was programmed to automatically acquire data when the Phase II MRA algorithms recognized one of the desired flight regimes shown in Table B-8. Prior to the start of Phase II testing, some flights were flown concurrently in Phase I to test the system's automatic triggering software. During the initial flights, the automatic triggering only recognized the ground run and hover points. Further review of the data showed that MRA limits for the straight and level flight maneuvers were incorrect. The

limits were hard coded into an EPROM chip located within the SDC. New limits were developed, and SEI created and installed a new chip into the SDC. The new chip corrected the problem, and no further problems arose with regime recognition during Phase II tests.

Main Rotor Track Capability

The system's capability to track the main rotor was evaluated during 13 dedicated flights. The maneuvers and adjustments to the rotor head are described in Table B-6. The data presented in Table B-9 are for the PCR fault flight. First, a baseline flight was flown, then the fault flight was flown, and finally the evaluation flight using the AVDS solutions. The system failed to record tracker information during the PCR fault flight. This was not apparent until the data analysis phase conducted after each flight. Accelerometer data were analyzed in the absence of main rotor track data. This premise is viable because track directly effects balance, but balance does not effect track. The vibrations averaged 0.01 g's during the baseline flight and increased to 0.04 g's during the fault flight. The vibration value returned to 0.01 g's after the AVDS solution was incorporated. VATS truth data confirmed that the T&B was out of limits (4 to 6 inches and 0.3 to 0.5 ips, respectively) for the fault flight and within limits during the baseline and evaluation flight (2 inch track spread and vibrations of 0.2 ips). The AVDS solution was equally as successful for the trim tab fault flight. The AVDS-suggested adjustments made for the PCR and trim tab faults resulted in vibration levels that returned to the those demonstrated during the baseline flight prior to the fault insertion.

With respect to tracker troubleshooting, the SPS system provided an error code to assist in repairing a malfunctioning tracker; however, SEI's system was not designed to record this code to the MU so the maintenance team was unable to retrieve it.

The nose tracker (Figure A-6), as installed, will interfere with current H-60 wire strike protection systems. Recommend recessing the optical blade tracker into the nose section so that the lens will be flush with the aircraft skin.

Main Rotor Balance Capability

The system's capability to balance the main rotor was evaluated concurrently with main rotor tracking flights. The maneuvers and adjustments to the rotor head are described in Table B-6. The AVDS solutions calculated for the hub weight fault insertions resulted in lower vibration levels in flight, however the ground vibration levels consistently exceeded allowable limits as verified by the VATS (0.7 ips). The AVDS accelerometer data are presented in Table B-10. Investigation revealed that the AVDS accelerometers were consistently reading ground vibrations at one half the amplitude of the VATS accelerometer readings. In-flight readings between the AVDS and VATS accelerometers were comparable; however, AVDS data showed the aircraft vibrations to be within limits whereas VATS truth data showed vibration levels to be slightly greater than allowable limits (0.45 ips). AVDS used only the ground vibration signatures to provide required adjustments of the hub weights. The accelerometers used to record these measurements for each system were mounted approximately one inch apart on the door frame behind the pilot's and co-pilot's seats (Figure A-3). Verification was performed to ensure that the

VATS system was working properly, and no problems were found. SPS never offered troubleshooting assistance for the AVDS accelerometers to determine whether or not the system could be at fault. The AVDS system was unable to bring the hub balance back to FMC limits.

3. PHASE III

General

Prior to Phase III testing the T&B hardware had to be removed from the SDC and replaced with the GFDS hardware.

Bearing Fault Detection Capability

The system's bearing fault detection capability was evaluated during Phase III testing during 10 dedicated flights. The maneuvers flown and the inserted faults for these flights are presented in Tables B-7 and B-8. Only one faulted bearing was used for these tests (Figure A-15). The bearing was rotated between upper pitch rod and inboard damper locations in the main rotor head red blade assembly. Data were first taken from a baseline flight with a PCR bearing clearance of less than 0.005 inches. The worn PCR bearing (0.015 clearance) was then installed on the red blade, and data were collected. Then, the original PCR bearing was reinstalled, and the baseline flight was repeated. The vertical (A+B) vibrations at 1R increased as a result of the installation of the worn bearing (Figure A-19); however, the resultant levels were still generally below flight limits (typically 0.3 ips). Thus, the 1R vibration levels alone cannot be used as a single

indicator of a worn PCR bearing because many other typical rotor T&B adjustments could induce this type of vibration. Analysis of the 1-12R harmonics of the lateral/roll vibrations (A-B) and PHV vibrations showed no significant changes with the worn bearing. In addition, the relative blade track height (at the blade tip) did not appear to be a significant indicator of the worn bearing. The track height standard deviation for each blade was calculated over 40 rotor revolutions to indicate the validity of the data and also to determine the vertical excursions of each blade. With the PCR bearing fault, the track standard deviation increased significantly in the climb and 145 knot regimes. For the 145 knot flight regime, the track standard deviation increased from 0.5 inches with no fault to almost 2 inches with the fault installed. Based on these results, a combination of both track and vibration data could offer an approach to detect worn PCR bearings.

Damper Fault Detection Capability

The system's lead-lag damper fault detection capability was evaluated for 3 dedicated flights during Phase III testing. The maneuvers flown and the inserted faults for these flights are presented in Tables B-7 and B-8. One flight was flown with the modified lead-lag damper installed using the modified flight profile for bearing fault tests with some low airspeed maneuvers added (Table B-8). To achieve the damper faults, a worn PCR end bearing was installed on the inboard end of the damper. The vibration amplitudes and phase showed no significant changes when the worn damper bearing was installed, but the blade lag standard deviation (measured in inches at the blade tip) increased dramatically (from 100 percent to as much as 400 percent) in the climb and

descent flight regimes when the faulty lag damper bearing was installed. These results also suggested that blade track/lag data provided an indication of bearing wear.

To simulate a damper hydraulic leak, a new damper that had the inner seal removed (Figure A-16) by NADEP Corpus Christi was installed on the blue blade assembly. The damper's damping characteristics were documented prior to installation in the test aircraft. The damper fault vibration data are presented in Figure A-20. The fault affected vibrations most noticeably at the hover and 60 knot flight regimes. The (A+B) 2R level was more than double in the hover and 60 knot regimes. The lateral/roll (A-B) 1R vibrations also changed significantly in the hover and 60 knot flight regimes as a result of the faulty lag damper. At higher speeds, the damper fault had little effect on vibrations.

In addition to vibration data, blade lag/track data were also analyzed. The lag angle of each blade with respect to the mean lag angle was calculated for each blade. Examination of the lag tip displacements showed a significant shift in lag displacement for the blade with the faulty damper (approximately 0.5 inches at the blade tip) in the hover, 60 knots, climb and descent flight regimes as compared to when the good damper was installed.

System Acquisition Flexibility

The flexibility of the system acquisition mode was also evaluated in Phase III testing. The AVDS system was capable of collecting data in a total of ten unique regimes. For these tests, the system was programmed to collect data automatically in ten pre-defined aircraft regimes (Table B-7). This automatic data acquisition mode was active whenever the system was powered and could not be turned off. When the data from the first part of Phase III were analyzed, it showed that the ten pre-defined regimes were not adequate for fault detection. Due to time and cost constraints, this problem was not fixed, and data were collected using manual acquisitions while circumventing the automatic acquisition mode. Circumvention of the automatic acquisition mode was accomplished by avoiding those flight regimes during testing.

CHAPTER VI

CONCLUSIONS

The AVDS exhibited excellent potential for use as a load monitoring tool in the fleet. The load estimations provided by the NN were comparable to the actual loads measured for the PCR and damper. The main rotor T&B portion of the system also showed excellent potential, however, tracker operation was intermittent. Troubleshooting codes were generated by the system, but software limitations prevented access to these codes. These problems were never uncovered. The main rotor PCR and trim tab maintenance action suggestions generated by AVDS to smooth the main rotor consistently brought the aircraft within allowable vibration limits in one flight. The AVDS ability to track the main rotor to lower vibration levels in a shorter amount of time than the traditional VATS could significantly reduce the maintenance effort at the squadron level. Ground vibration algorithms for the system require modification to allow for accurate hub weight adjustments. The GFDS demonstrated excellent potential for detecting PCR and damper bearing faults and internal leakage in a main rotor lead-lag damper. The GFDS gives credence for the use of a NN to monitor multiple rotor head parameters making it possible to accurately pinpoint rotor component wear.

CHAPTER VII

RECOMMENDATIONS

The AVDS shows excellent potential for load prediction, accurate T&B functions, and fault detection. The following recommendations are presented:

- Use neural network load data vice flight hour accumulation to determine component inspection and replacement cycles.
- Investigate and correct the AVDS accelerometer discrepancy for rotor hub balancing.
- Modify system software so that the system error codes from the tracker are available for troubleshooting.
- Recessing the optical blade tracker into the nose section so that the lens will be flush with the aircraft skin.
- Research the development of bearing fault detection capabilities to determine if the system can conclusively detect bearing faults.
- Verify the damper fault indications to determine the size of internal damper leaks that the system can detect.
- Include an automatic acquisition mode that can be disabled or the ability to adjust regime limits through software.

- Redesign the system with hardware that allows T&B and GFDS testing to be conducted concurrently and as mutually exclusive data sets instead of requiring hardware changes between flights.

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APPENDICES

APPENDIX A

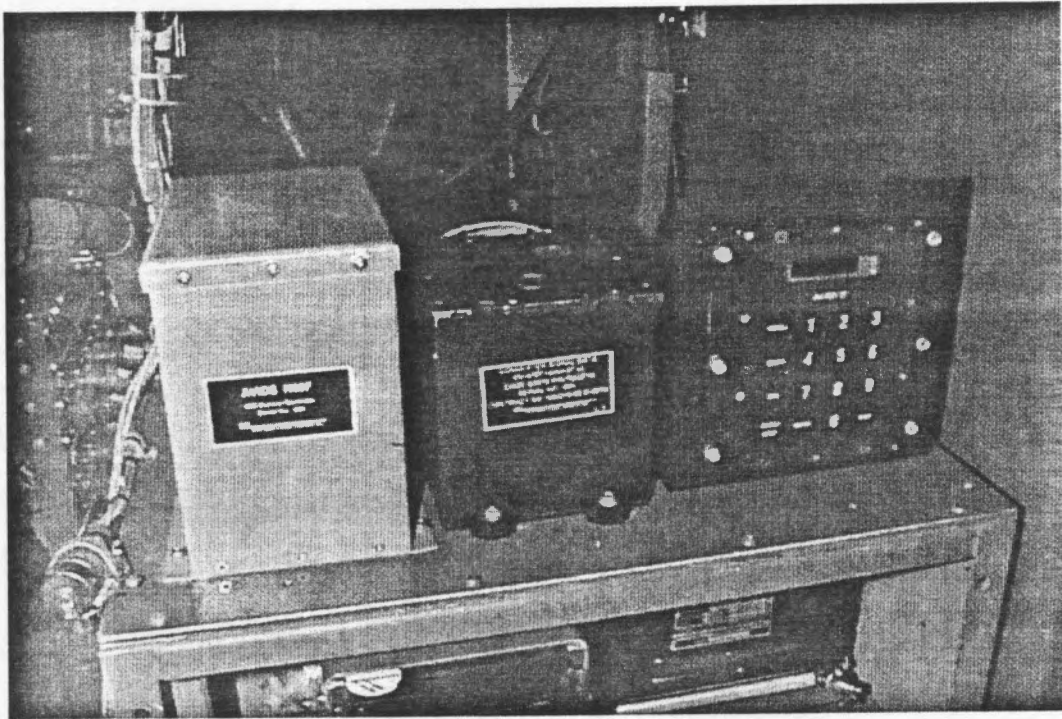


Figure 1. (Left to Right) AVDS survey recorder, signal data converter, and data entry keyboard installed behind pilot's seat.

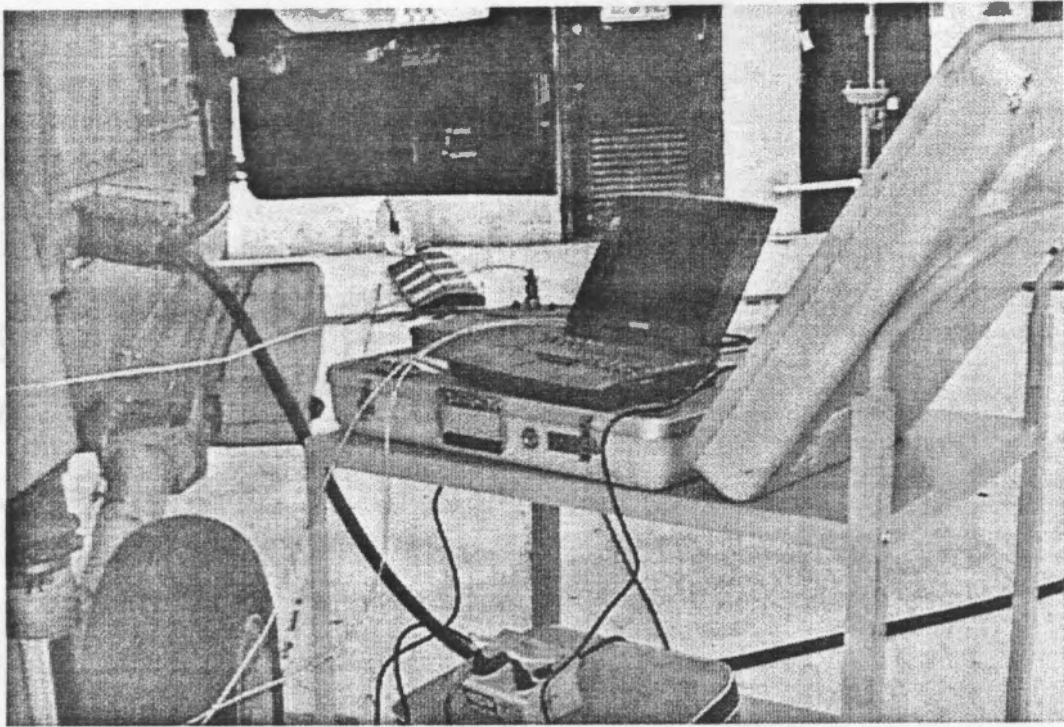


Figure 2. Maintenance data processing station, downloading data from SDC after flight.

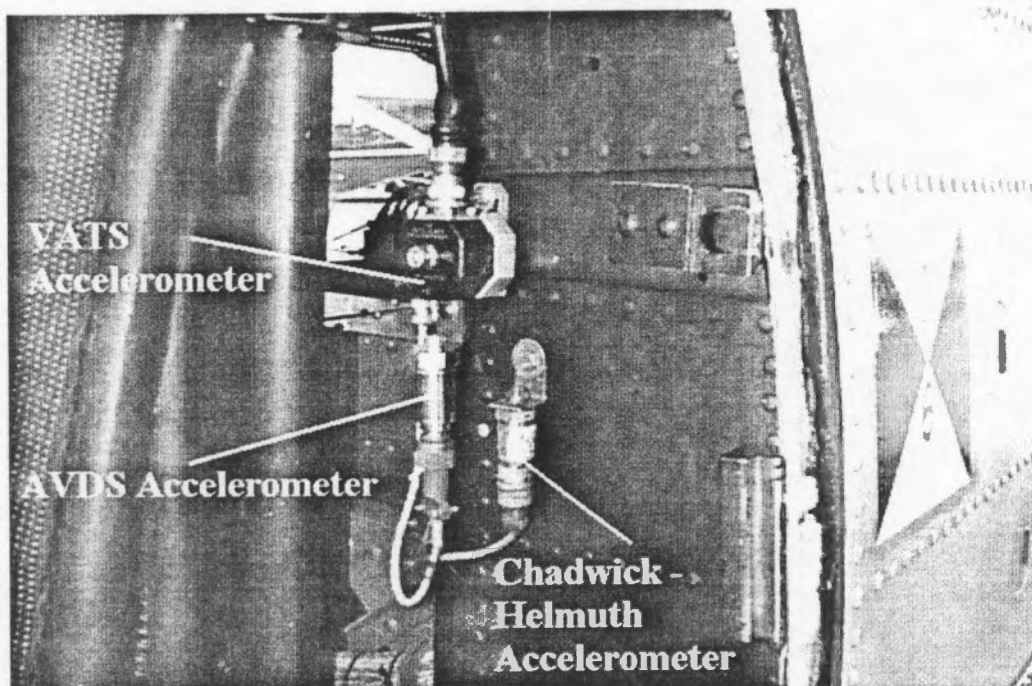
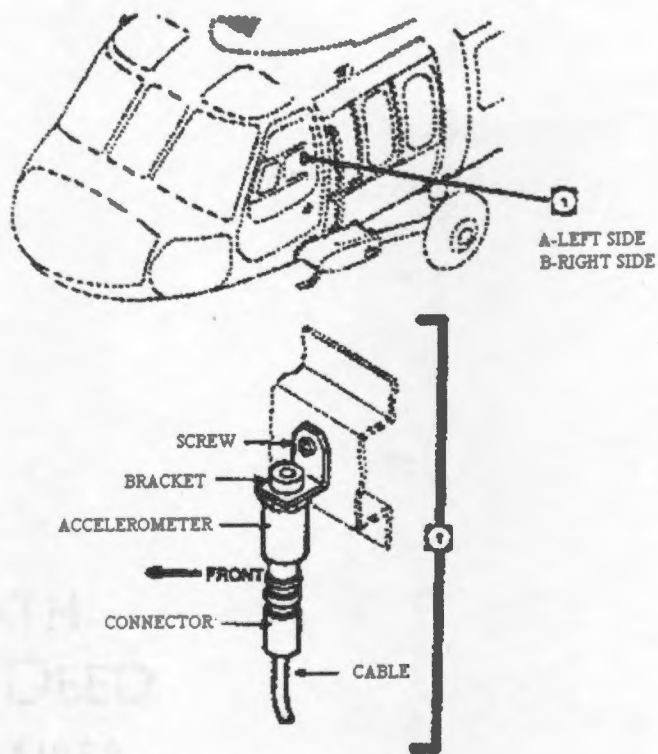


Figure 3. Installation diagram for pilot and co-pilot accelerometers (co-pilot's shown).

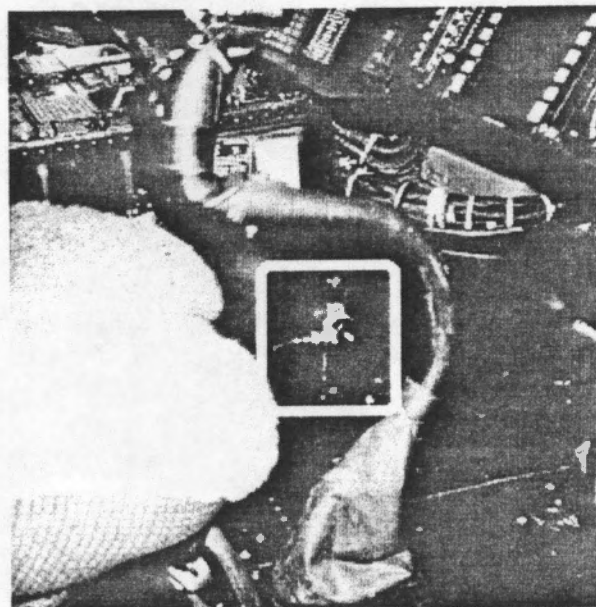
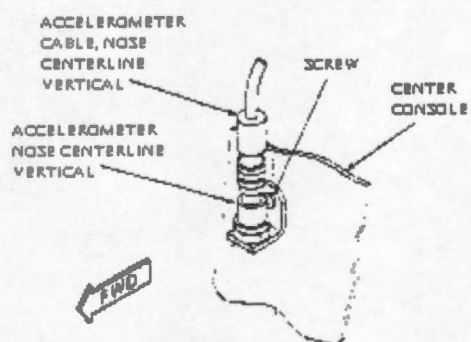


Figure 4. Location of the Pilot's Absorber Accelerometer in Cockpit.

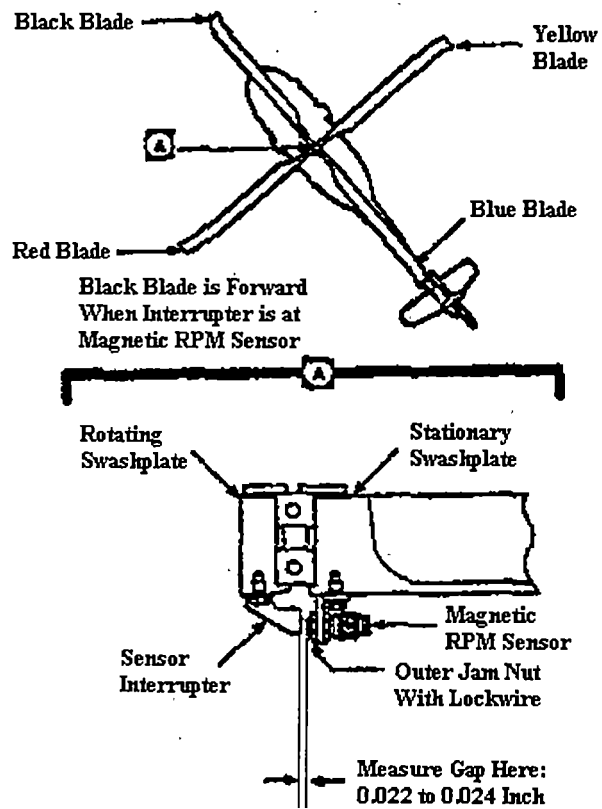


Figure 5. Installation Diagram for Magnetic RPM Sensor.

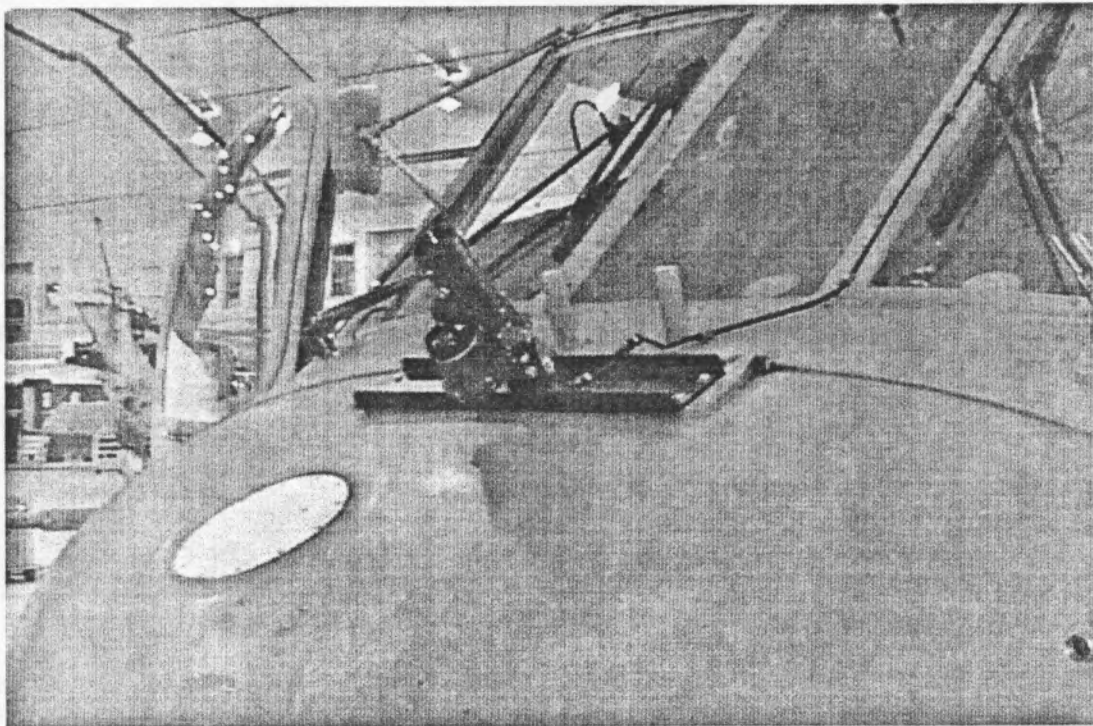
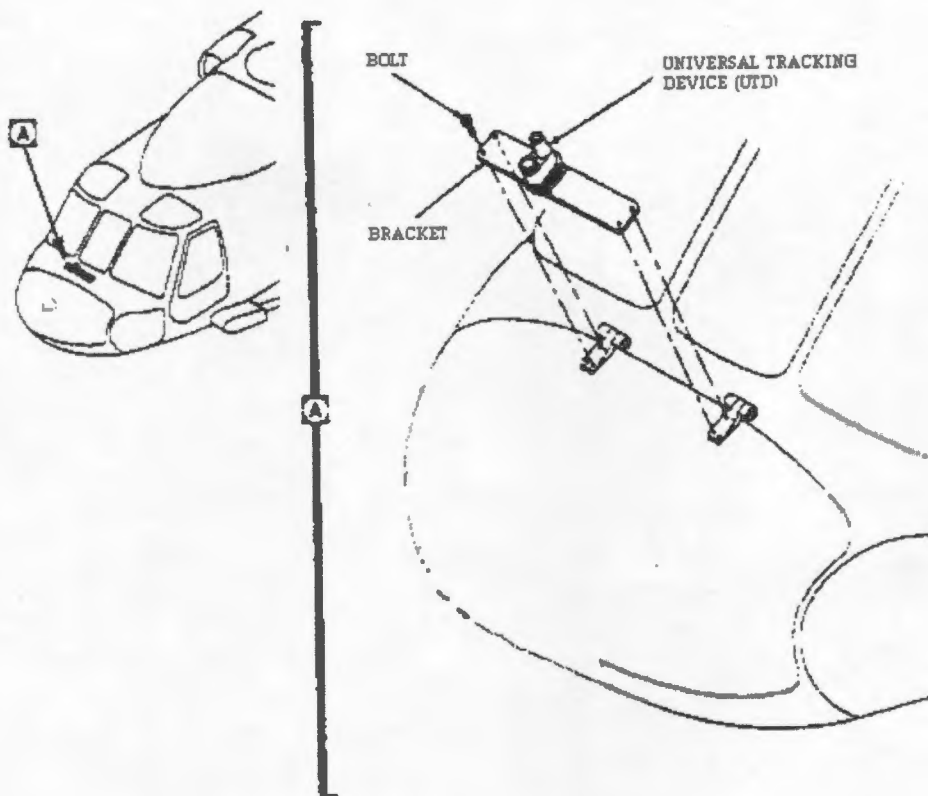


Figure 6. Installation Location Diagram for Main Rotor Optical Tracker.

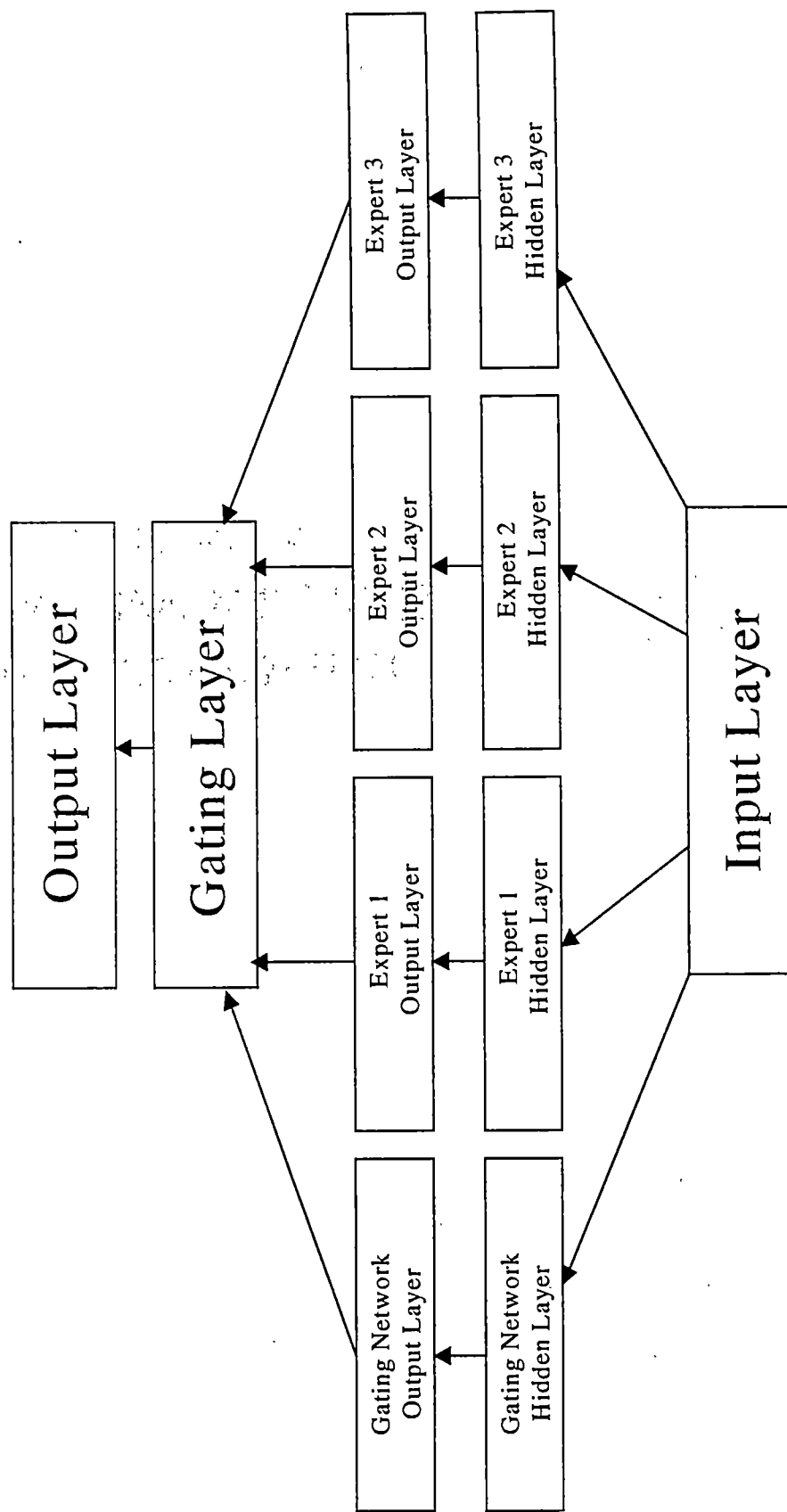


Figure 7. Modular Neural Network Architecture.

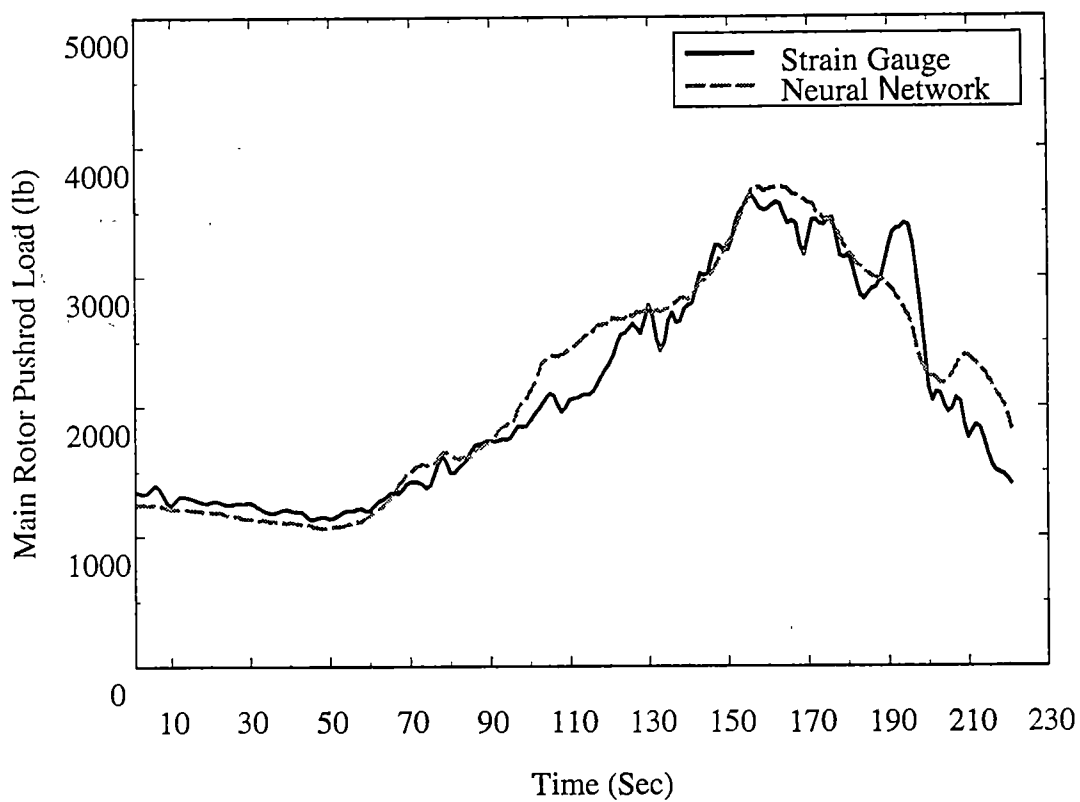


Figure 8. Pitch control rod load predictions for a 60 degree angle of bank right turn in the baseline data.

Source: McCool, Kelly M., Working Notes, Naval Surface Warfare Center, Bethesda, MD, 1998.

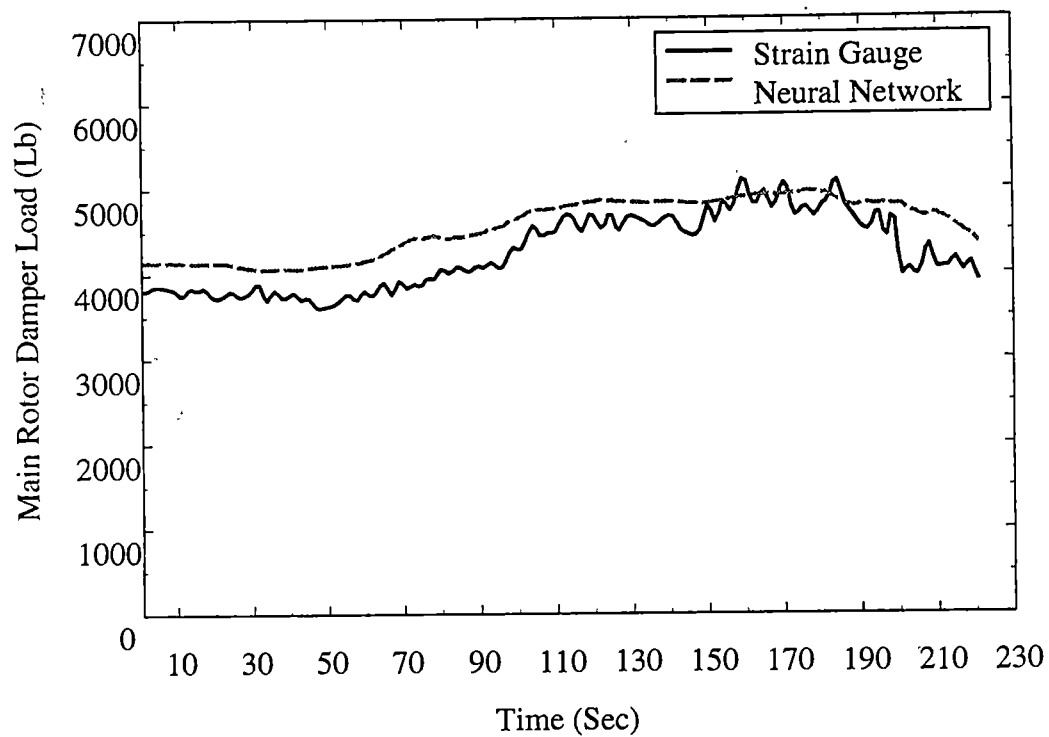


Figure 9. Damper load predictions for a 60 degree angle of bank right turn in the baseline data.

Source: McCool, Kelly M., Working Notes, Naval Surface Warfare Center, Bethesda, MD, 1998.

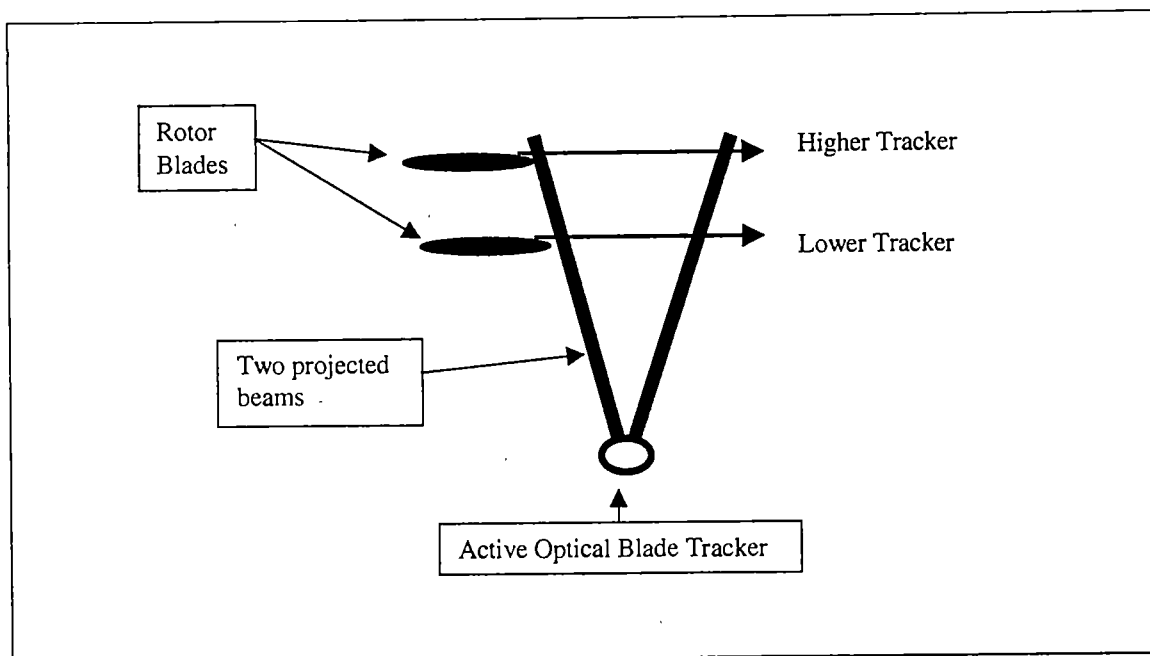


Figure 10. Universal Tracking Device Function.

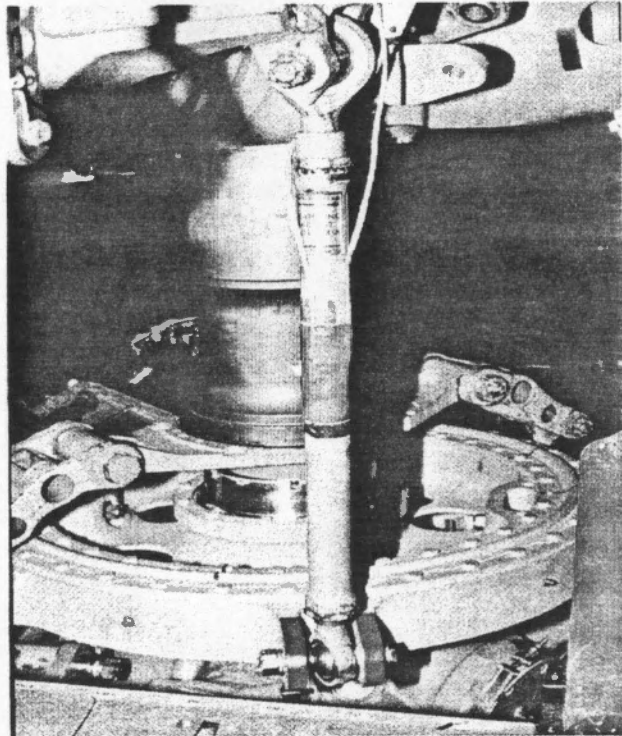
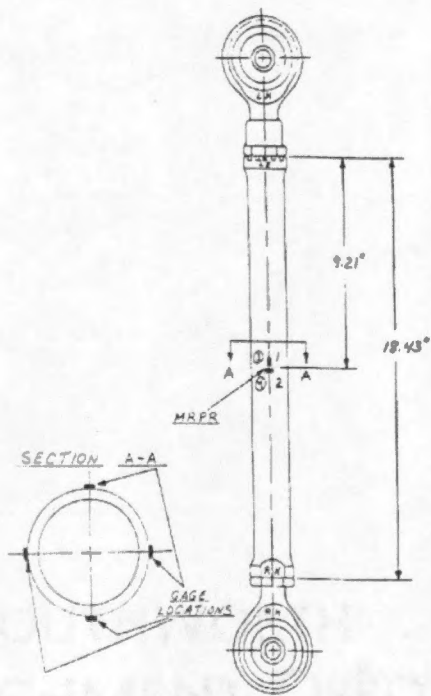


Figure 11. Main Rotor Pitch Control Rod Strain Gage Location Diagram.

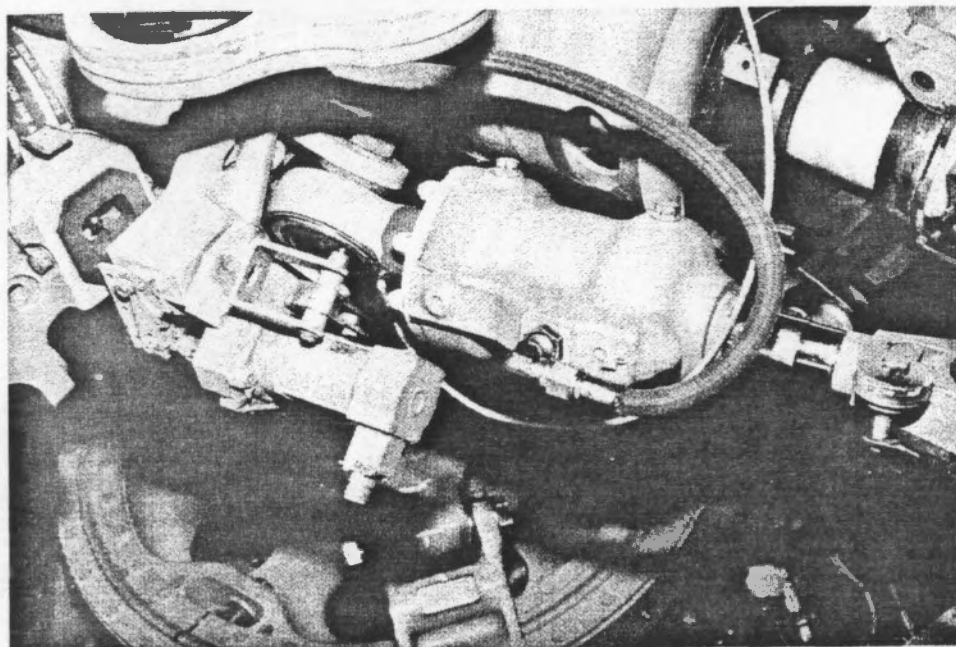
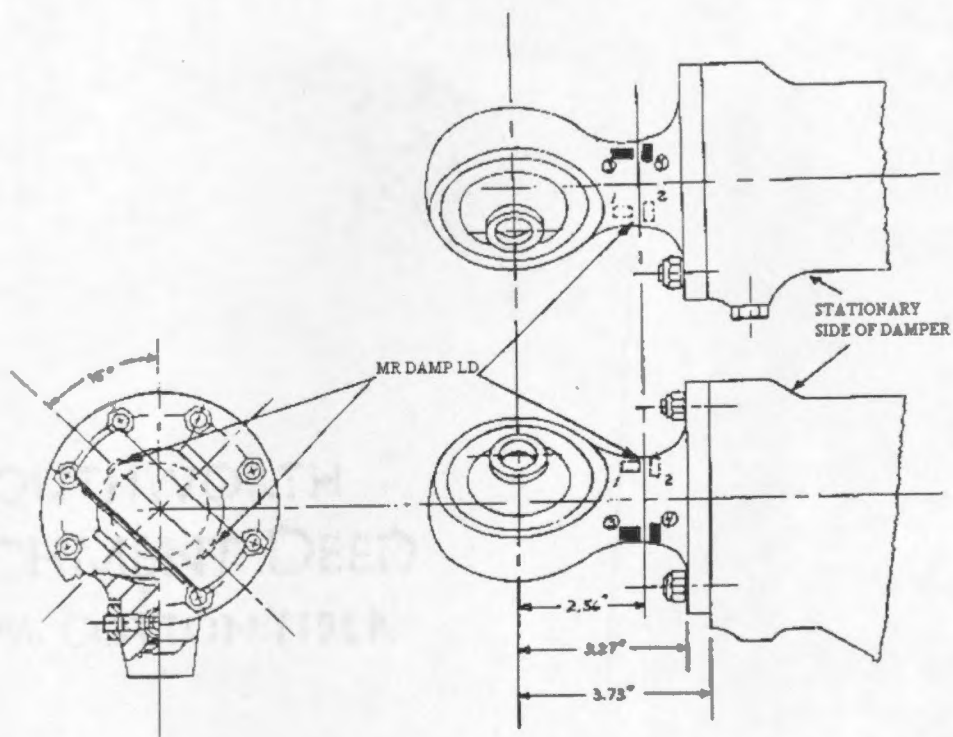


Figure 12. Main Rotor Damper Strain Gage Location Diagram.

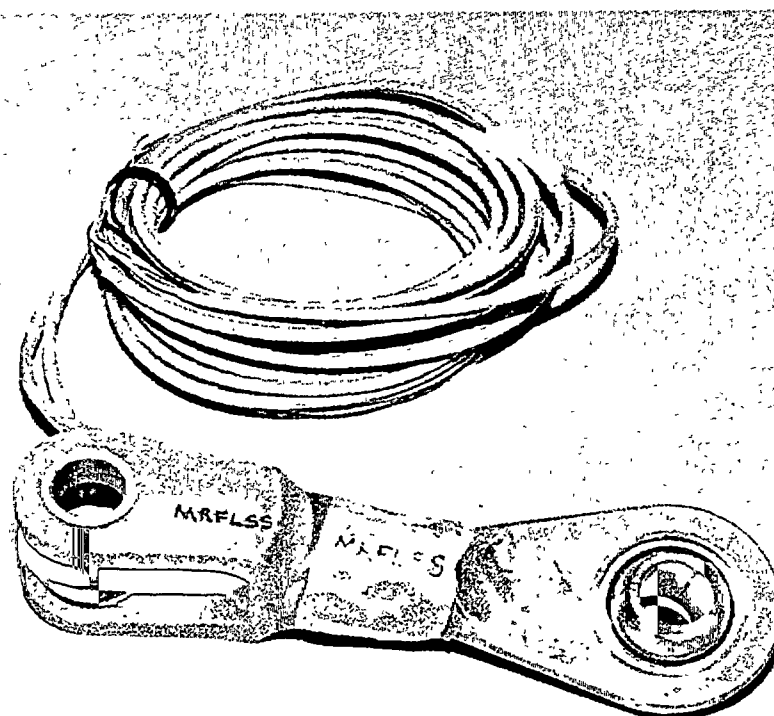
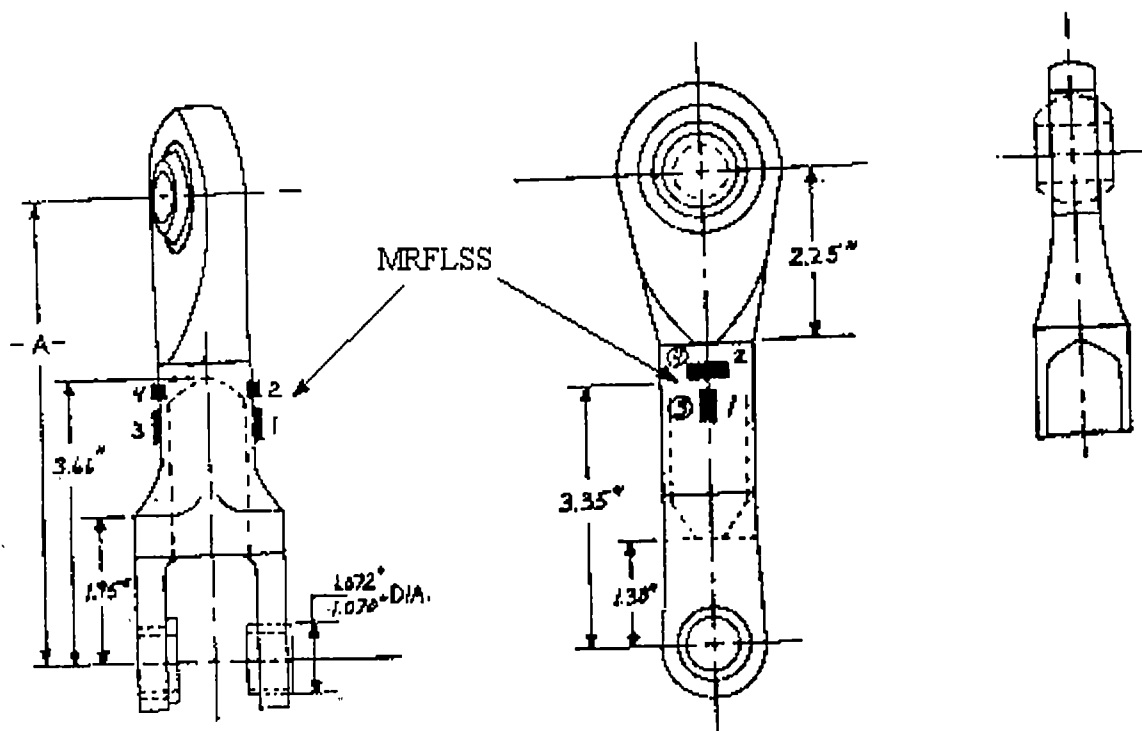


Figure 13. Main Rotor Longitudinal Servo Link Assembly Strain Gage Location Diagram.

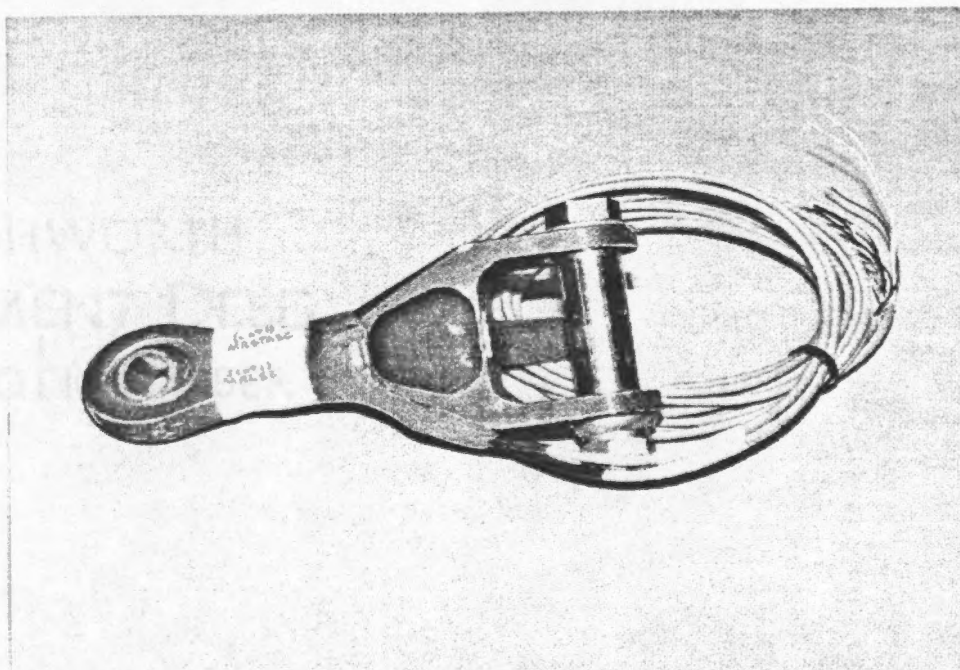
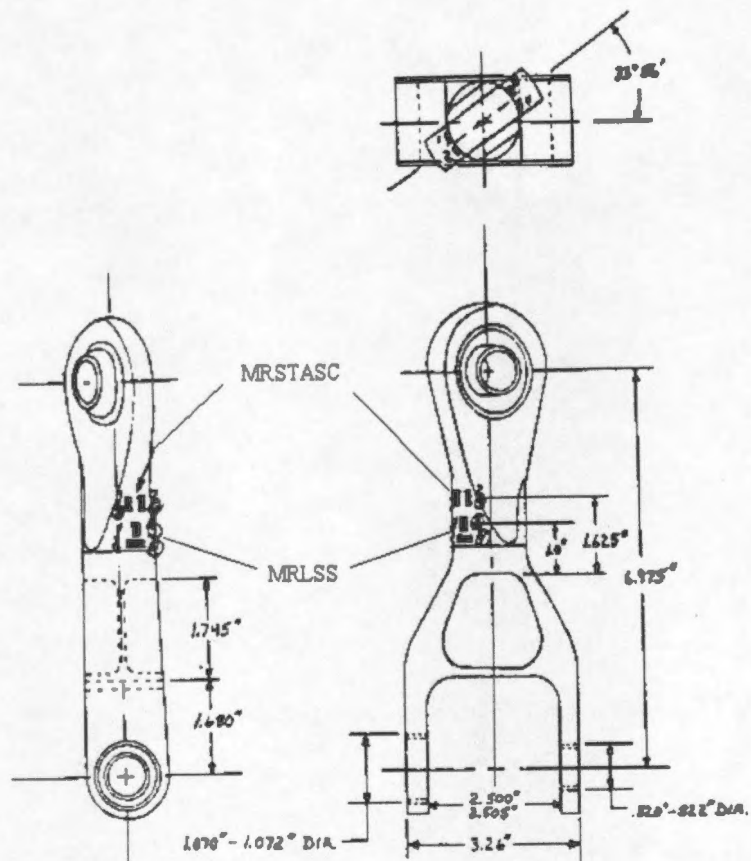


Figure 14. Main Rotor Lateral Servo Link Assembly Strain Gage Location Diagram.

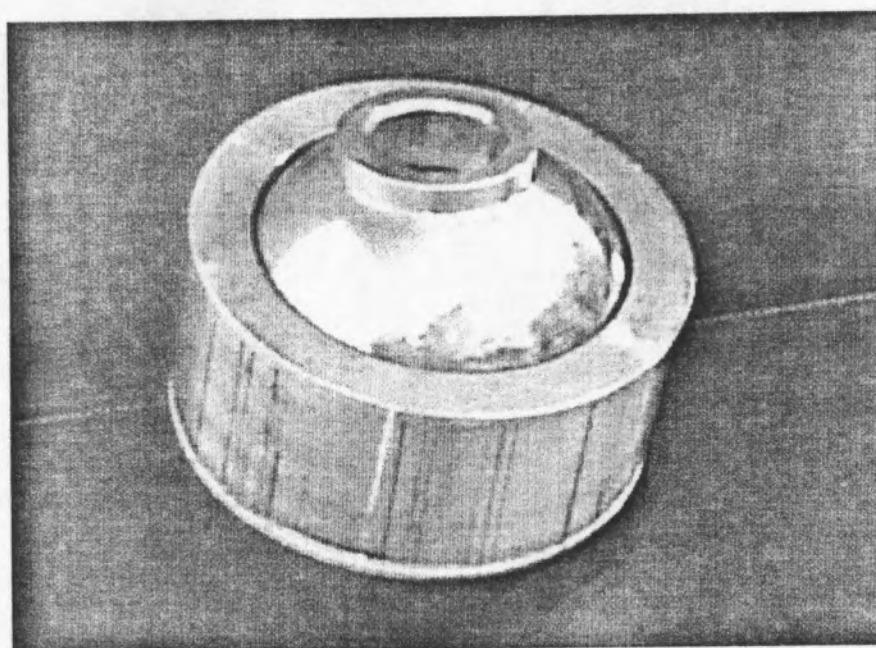
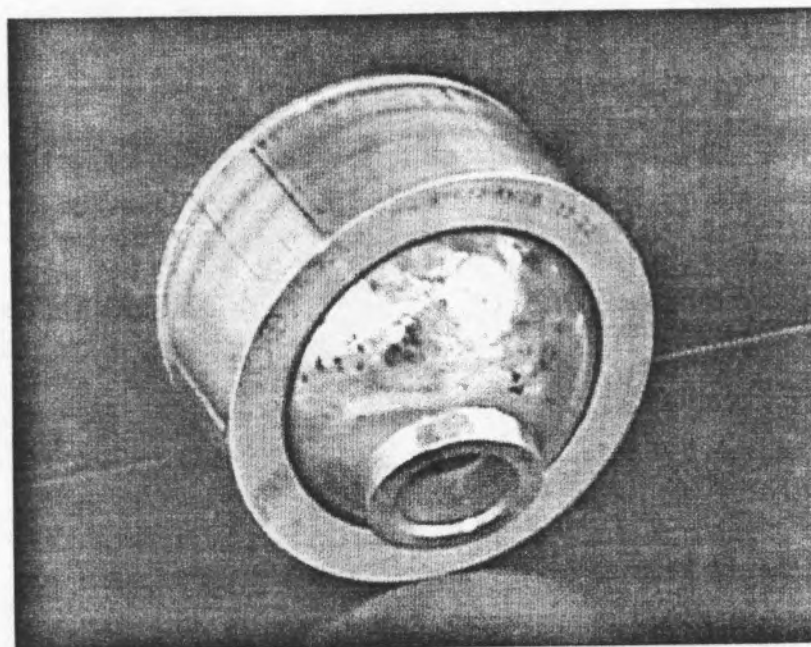


Figure 15. Bearing (Ser. No. 72206) used in AVDS Testing.

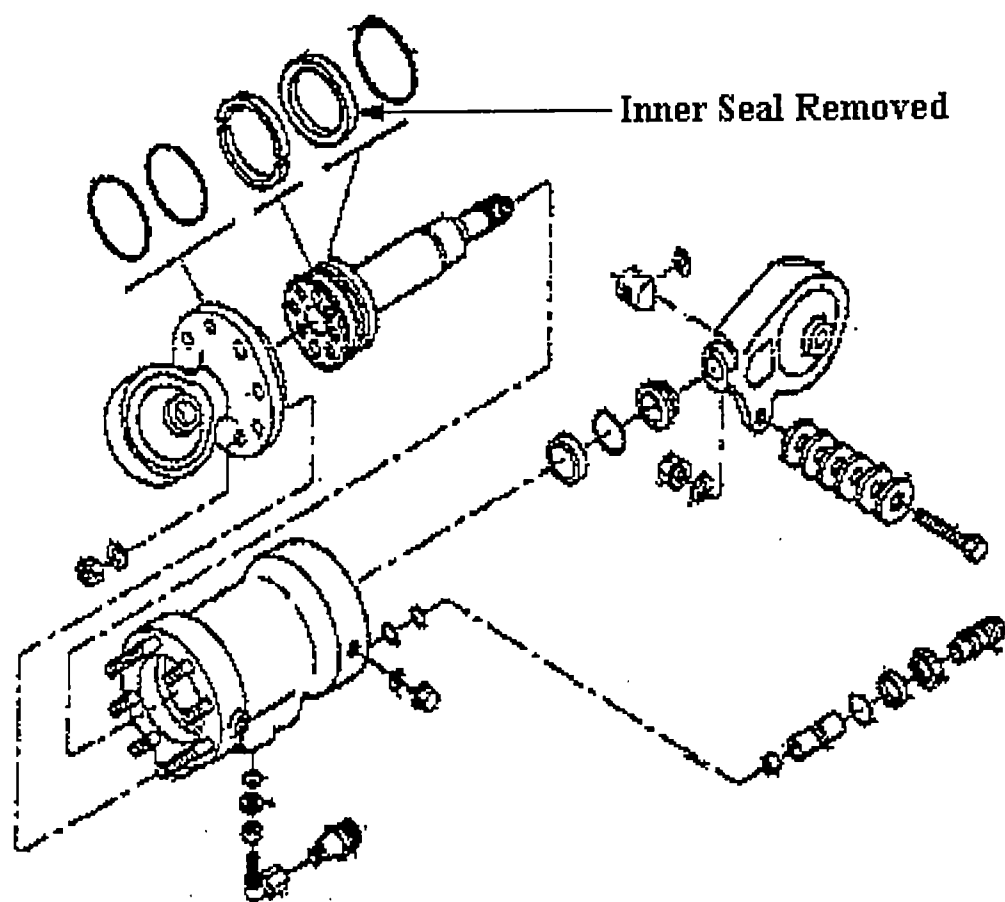


Figure 16. Exploded view of main rotor lead-lag damper highlighting inner seal that was removed for AVDS testing.

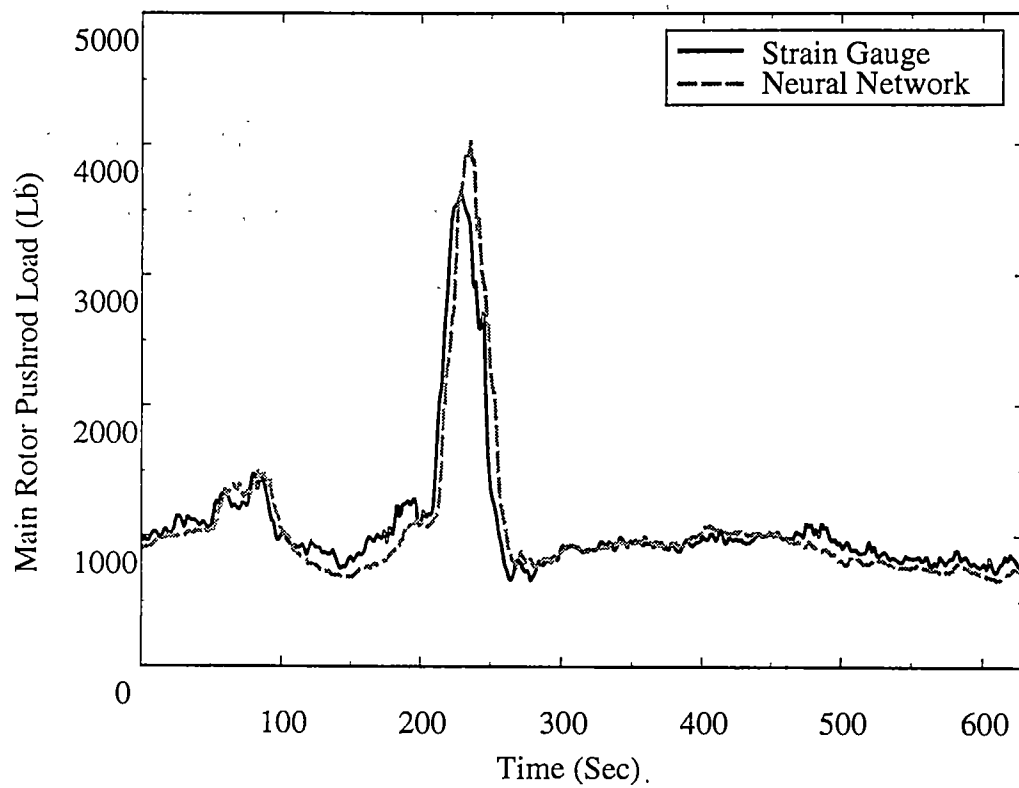


Figure 17. Pitch control rod load predictions for a high-g symmetric pullout maneuver from the flight test data.

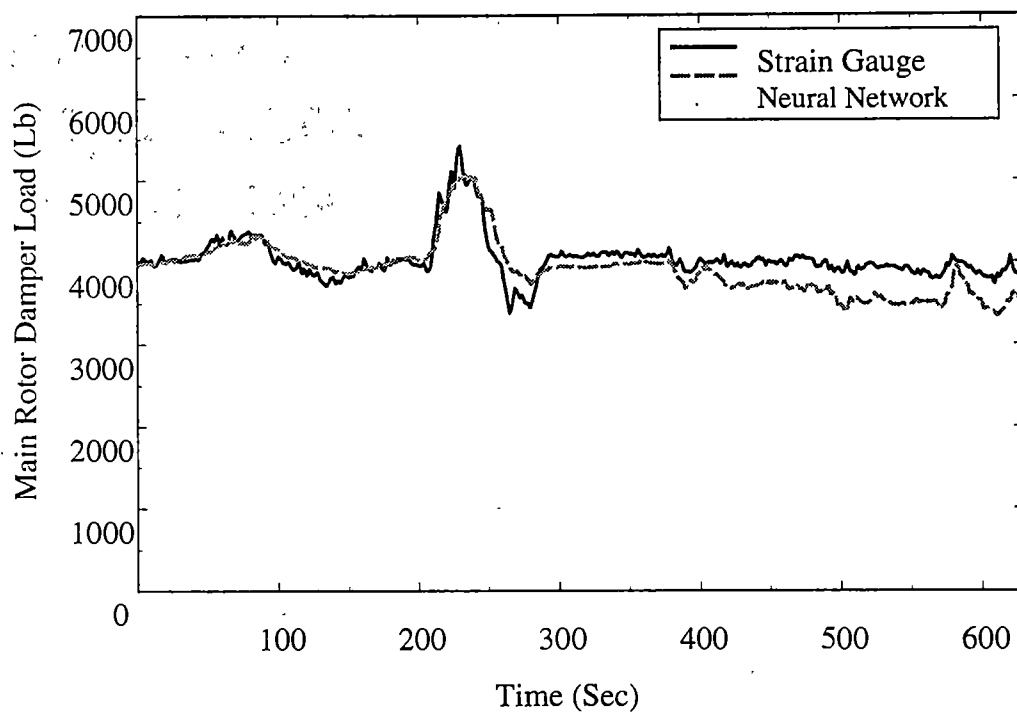


Figure 18. Damper load predictions for a high-g symmetric pullout maneuver from the flight test data.

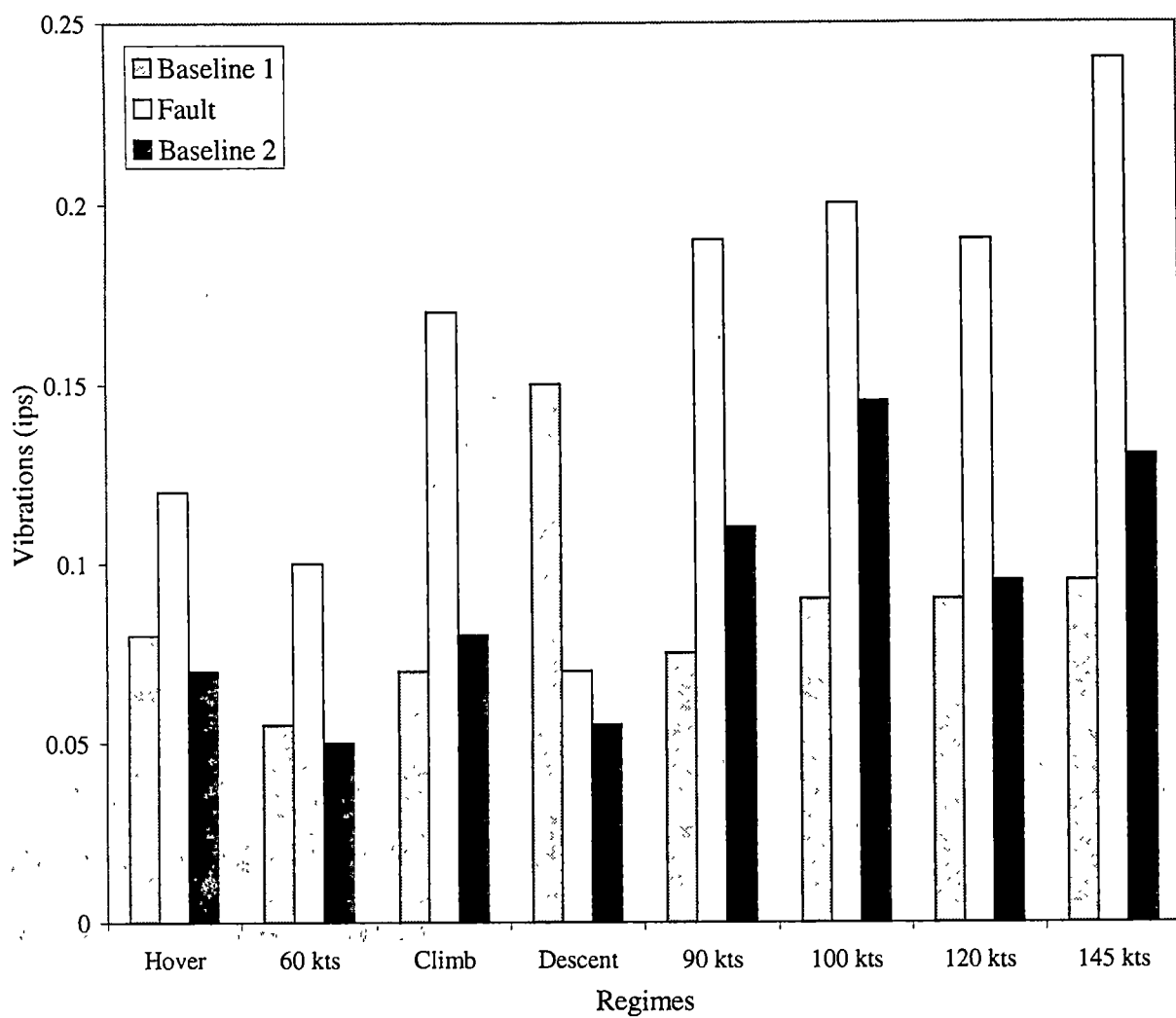


Figure 19. Pitch Control Rod Bearing Fault (A+B)1R.

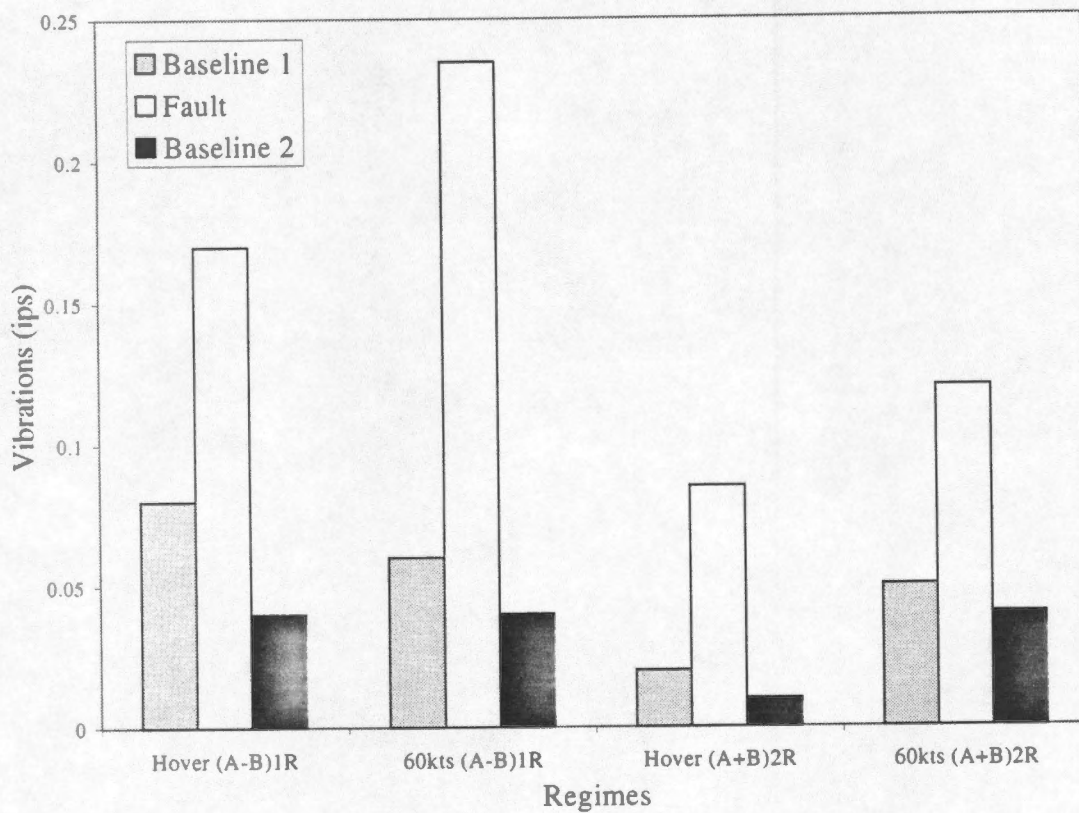


Figure 20. Lag Damper Fault, (A-B)1R and (A+B)2R.

APPENDIX B

Table 1. SEI SDRS Instrumented Parameters.

Parameter	Signal Source
Co-pilot's Door Vertical Vibration	Instrumentation Tri-Axial Accelerometer
Pilot's Door Plate Lateral Vibration	Instrumentation Tri-Axial Accelerometer
Tail Rotor Drive Shaft Acceleration	Instrumentation Tri-Axial Accelerometer
Main Rotor Optical Tracking	Optical Tracker
Main Rotor Tach Signal	Magnetic Pick-up
Tail Rotor Tach Signal	Optical Tracker
Pitch Attitude	MS1553A Data Bus
Roll Attitude	MS1553A Data Bus
Pitch Rate	Instrumentation Three Axis Rate Gyro
Roll Rate	Instrumentation Three Axis Rate Gyro
Yaw Rate	Instrumentation Three Axis Rate Gyro
Nz (Load Factor)	Instrumentation Accelerometer, Center Line of Aircraft
Indicated Air Speed	MS1553A Data Bus
Outside Air Temperature	MS1553A Data Bus
Pressure Altitude	MS1553A Data Bus
Rate of Climb	MS1553A Data Bus
Total Fuel	MS1553A Data Bus
Engine Torque 1	Aircraft Signal Data Converter #1
Engine Torque 2	Aircraft Signal Data Converter #2
Collective Stick Position	Automatic Flight Control System Test Panel
Lateral Stick	Automatic Flight Control System Test Panel
Longitudinal Stick	Automatic Flight Control System Test Panel
Pedal Position	LVDT Position Pot on Linkage
Main Rotor RPM	Aircraft Signal Data Converter #2
Weight on Wheels	Main Landing Gear WOW Switch
Rotor Brake	Caution/Advisory Panel

Table 2. Parameters Monitored by AID IP Package.

Parameter	Signal Source
Main Rotor Push Rod Load	Instrumentation Strain Gage
Main Rotor Damper Load	Instrumentation Strain Gage
Main Rotor Longitudinal Servo Link Load	Instrumentation Strain Gage
Main Rotor Lateral Servo Link Bending Load	Instrumentation Strain Gage
Main Rotor Lateral Servo Link Axial Load	Instrumentation Strain Gage
Doppler Drift	Aircraft Doppler
Doppler Ground Speed	Aircraft Doppler
Indicated Airspeed	Pilot's Pitot Probe
Pitch Attitude	Pilot's Gyro Amplifier
Roll Attitude	Pilot's Gyro Amplifier
Pressure Altitude	Pilot's Static Line
Rate of Climb	Pilot's Static Line
Radar Altitude (Low)	Aircraft Radar
Radar Altitude (High)	Aircraft Radar
Pitch Rate	Instrumentation Three Axis Rate Gyro
Roll Rate	Instrumentation Three Axis Rate Gyro
Yaw Rate	Instrumentation Three Axis Rate Gyro
Main Rotor RPM	Aircraft Signal Data Converter #2
#1 Engine Torque	Aircraft Signal Data Converter #1
#2 Engine Torque	Aircraft Signal Data Converter #2
Longitudinal Stick Position	Automatic Flight Control System Test Panel
Lateral Stick Position	Automatic Flight Control System Test Panel
Pedal Position	LVDT Position Pot on Linkage
Collective Stick Position	Automatic Flight Control System Test Panel
Load Factor	Instrumentation Accelerometer, Center Line of Aircraft
Event Switch	Instrumentation Rack Control Panel

Table 3. Neural Network Inputs.

Dimensional Input Space	Input Parameter
1	Airspeed (kts)
2	Main rotor speed (%)
3	Engine torque (%)
4	Load factor (g's)
5	Pitch attitude (deg)
6	Roll attitude (deg)
7	Lateral stick position (%)
8	Longitudinal stick position (%)
9	Collective stick position (5)
10	Pitch rate (deg/sec)
11	Rate of climb (ft/min)
12	Gross weight (lb)

Table 4. H-60 Regimes Included in Baseline Data.

Regime Number	Regime Description
1	Taxi
2	Hover
3	Left hover turn
4	Right hover turn
5	Left sideward flight
6	Right sideward flight
7	Rearward flight
8	Level flight 0.2Vh
9	Level flight 0.4 Vh
10	Level flight 0.5 Vh
11	Level flight 0.6 Vh
12	Level flight 0.7 Vh
13	Level flight 0.8 Vh
14	Level flight 0.9 Vh
15	Level flight 1.0 Vh
16	Partial power descent
17	Left sideslip
18	Right sideslip
19	Jump takeoff
20	Rolling takeoff
21	Vertical takeoff
22	Takeoff power climb
23	Steady climb
24	Approach
25	Hover landing
26	Run on landing
27	30° angle of bank left turn
28	30° angle of bank right turn
29	45° angle of bank left turn
30	45° angle of bank right turn
31	60° angle of bank left turn
32	60° angle of bank right turn
33	Longitudinal reversal hover
34	Longitudinal reversal level flight
35	Lateral reversal hover
36	Lateral reversal level flight

Table 4 (continued).

Regime Number	Regime Description
37	Rudder reversal hover
38	Rudder reversal Level flight
39	Severe symmetric pullout
40	Moderate symmetric pullout
41	Power dive
42	Autorotation entry
43	Steady autorotation
44	Autorotation recovery
45	Autorotation left turn
46	Autorotation right turn
47	Control reversal autorotation
48	Autorotation pullout

Table 5. Phase 1 Flight Profiles for Load Evaluation Tests.

Flt No.	Maneuver Sequence		Altitude ¹	Airspeed ²	Bank Angle (Deg)
1	Ground Run		0	0	0
1	Taxi		0	0 - 5 KGS	0
1	Taxi Turns		0	0 - 5 KGS	0
1	Normal Takeoff		0 - 200	0 - 70	0
1	Normal Approach to Hover		200 - 10	70 - 0	0
1	Normal Hover Landing		10 - 0	0	0
1	Jump Takeoff		0 - 200	0 - 70	0
1	Steep Approach to Hover		200 - 10	40 - 0 KGS	0
1	HIGE		10	0	0
1	HOGE		100	0	0
1	Lateral Hovering Control Doublet (1 inch, 1 Hz)	HIGE	10	0	0
		HOGE	100	0	0
1	Longitudinal Hovering Control Doublet (1 inch, 1 Hz)	HIGE	10	0	0
		HOGE	100	0	0
1	Collective Hovering Control Doublet (1 inch, 1 Hz)	HIGE	10	0	0
		HOGE	100	0	0
1	Pedal Hovering Control Doublet (1 inch, 1 Hz)	HIGE	10	0	0
		HOGE	100	0	0
1	Left Hovering Turn	HIGE	10	0	0
		HOGE	100	0	0
1	Right Hovering Turn	HIGE	10	0	0
		HOGE	100	0	0
1	Entry & Recovery (E&R) Left Sideward Flight	HIGE	20 $\pm$ 5	25 KGS	<45
		HOGE	100 $\pm$ 5	25 KGS	<45
1	Left Side Flight	HIGE	20 $\pm$ 5	25 KGS	<45
		HOGE	100 $\pm$ 5	25 KGS	<45
1	E&R Right Sideward Flight	HIGE	20 $\pm$ 5	0 - 25 KGS	<45

Table 5 (continued).

Flt No.	Maneuver Sequence		Altitude ¹	Airspeed ²	Bank Angle (Deg)
1	E&R Right Sideward Flight	HOGE	100 $\pm$ 5	0 - 25 KGS	<45
1	Right Side Flight	HIGE	20 $\pm$ 5	0 - 25 KGS	<45
		HOGE	100 $\pm$ 5	0 - 25 KGS	<45
1	E&R Rear Flight	HIGE	20 $\pm$ 5	0 - 25 KGS	0
		HOGE	100 $\pm$ 5	0 - 25 KGS	0
1	Rear Flight	HIGE	20 $\pm$ 5	0 - 25 KGS	0
		HOGE	100 $\pm$ 5	0 - 25 KGS	0
1	Running Landing		500 - 0	45 - 0	0
1	Rolling Takeoff		0 - 500	0 - 35	0
2	Steady Climb		1000 - 3000	80	0
2	E&R Partial Power Descent		3000 - 1000	80	0
2	Steady Partial Power Descent		3000 - 1000	80	0
2	Climb Turn Right		1000 - 3000	80	30/45
2	Partial Power Right Turn Descent		3000 - 1000	80	30
2	Climb Turn Left		1000 - 3000	80	30/45
2	Partial Power Left Turn Descent		3000 - 1000	80	30
2	Level Flight		2000	0.4 Vh	0
				0.5 Vh	
				0.6 Vh	
				0.7 Vh	
				0.8 Vh	
				0.9 Vh	
				Vh	
2	Level Flight Control Doublets (1 inch, 1 Hz)	Lateral	2000	80	0
		Long			
		Coll			
		Pedal			
2	Sideslip (1 ball)	Left	2000	80	0

Table 5 (continued).

Flt No.	Maneuver Sequence		Altitude ¹	Airspeed ²	Bank Angle (Deg)
2	E&R Right Turn		2000	80/120	30/45/60
2	Steady Right Turn		2000	80/120	30/45/60
2	E&R Left Turn		2000	80/120	30/45/60
2	Steady Left Turn		2000	80/120	30/45/60
2	Pedal Only Turns	Left and Right	2000	80	30/45/60
2	E&R Autorotation		3000 - 1000	80	0
2	Steady Autorotation		3000 - 1000	80	0
2	Autorotation Turn Left		3000 - 1000	80	30/45
2	Autorotation Turn Right		3000 - 1000	80	30/45
2	Autorotation Control Doublets (1 inch, 1 Hz)	Lateral	3000 - 1000	80	0
		Long			
		Coll			
		Pedal			
2	Dive		3000 - 1000	1.2 Vh	0
2	Dive		3000 - 1000	1.1 Vh	0
2	Moderate Pullout	1.5 to 2g's	1000 - 2000	40 - 1.2 Vh	0
2	Severe Pullout	>2g's	1000 - 2000	40 - 1.1 Vh	0
2	Pushover	1 to 0g's	2000 - 1000	80	0
3	Low Airspeed Flight (3)		20 $\pm$ 5	10, 15, 25, 35 KGS	
			70 $\pm$ 5	10, 15, 25, 35 KGS	

- Notes:
- (1) Altitudes below 1000 ft are AGL, altitudes at or above 1000 ft are MSL.
 - (2) Airspeeds are in KIAS unless labeled otherwise.
 - (3) Airspeeds were determined using aircraft Doppler radar or by conducting paced flight with a ground vehicle.

Table 6. Flights and Fault Conditions for Phase II Testing.

Date (1997)	Flt No.	Flt Title	Adjustment Made Prior To Flight	Comments
12 May	8	Baseline Flight	-	
13 May	9	Baseline Flight	-	
14 May	10	Pitch Control Rod Fault Flight	- Blue Blade Pitch Control Rod moved down 11 clicks	Adjustment calculated by project personnel. Vibration levels during flight were outside of FMC limits.
19 May	11	Evaluation Flight	- Blue Blade Pitch Control Rod increased 10 clicks - Yellow Blade Pitch Control Rod increased 1 click	Adjustment calculated by SPS ground station. Vibration levels during flight were within FMC limits.
20 May	12	Hub Weight Fault Flight	- 2.5 lbs added to Main Rotor Yellow Blade Hub - 3 lbs removed from Main Rotor Red Blade Hub	Adjustment calculated by project personnel. Vibration levels during flight were outside of FMC limits but not high enough for SPS algorithm verification
21 May	13	Hub Weight Fault Flight #2	- 2.5 lbs added to Main Rotor Yellow Blade Hub	Adjustment calculated by project personnel. Vibration levels during flight were outside of FMC limits and sufficient for SPS algorithm verification.
22 May	14	Evaluation Flight	- 3 lbs, 10 oz removed from Main Rotor Yellow Blade Hub - 11 oz added to Main Rotor Black Blade Hub - Yellow Blade Pitch Control Rod increased 3 clicks - Blue Blade Pitch Control Rod increased 5 clicks	Adjustment calculated by SPS ground station. Vibration levels during flight were outside of FMC limits for ground regime. The SPS ground station incorrectly indicated that the aircraft was within FMC limits and would not provide a solution.

Table 6 (continued).

Date (1997)	Flt No.	Flt Title	Adjustment Made Prior To Flight	Comments
23 May	15	Baseline Flight	- 1 lbs, 6 oz removed from Main Rotor Yellow Blade Hub - 11 oz removed from Main Rotor Black Blade Hub - Yellow Blade Pitch Control Rod decreased 3 clicks - Blue Blade Pitch Control Rod decreased 5 clicks	Adjustment calculated by project personnel. This removed all adjustments made after flight 11 to put the aircraft back into FMC limits. Vibration levels in flight were still outside of FMC limits.
18 June	17	Baseline Flight	- Black Blade Pitch Control Rod decreased 4 clicks	Adjustment calculated by VATS system. Vibration levels were within FMC limits.
19 June	18	Trim Tab Fault Flight	- 25 mils up on Main Rotor Red Blade	Adjustment calculated by project personnel. Vibration levels during flight were outside of flight clearance limits so flight was aborted.
24 June	19	Trim Tab Fault Flight #2	- 10 mils down on Main Rotor Red Blade	Adjustment calculated by project personnel. Vibration levels during flight were outside of FMC limits and were used for SPS algorithm verification.
24 June	20	Evaluation Flight	- 22 mils down on Main Rotor Red Blade - 4 mils up on Main Rotor Black Blade	Adjustment calculated by SPS ground station. Vibration levels during flight were within FMC limits.
8 July	23	Baseline	-	-

Table 7. Phase II and III Flight Test Profiles¹.

Standard Flight Conditions For Fault Detection Testing					
Event No. ²	Maneuver	% Nr	Altitude (Msl)	Airspeed (Kias)	Duration (Min)
1	Ground Run	100	GRND	N/A	4
2	Ground Idle	50 ³	GRND	N/A	1
3	HIGE	100	100 ft AGL	N/A	4
4	Level Flight	100	3000	60	4
5	Level Flight	100	3000	70	4
6	Level Flight	100	3000	80	4
7	Level Flight	100	3000	90	4
8	Level Flight	100	3000	100	4
9	Level Flight	100	3000	120	4
10	Level Flight	100	3000	145	4
11	Level Flight	100	3000	Vh	4
12	Climb	100	2000 - 3000	70	1
13	Descent	100	3000 - 2000	70	1
14	Left Sideward Flight	100	100 ft AGL	25 KGS (in wind direction)	0.5
15	Rearward Flight	100	100 ft AGL	25 KGS (in wind direction)	0.5
16	Right Sideward Flight	100	100 ft AGL	25 KGS (in wind direction)	0.5

- Notes:
- (1) All ground and hover points were conducted with the nose of the aircraft into the wind.
 - (2) Event numbers 2, 12, and 13 were added to the matrix for the second set of bearing fault flights. Event numbers 14, 15, and 16 were added for the lead-lag damper fault flight.
 - (3) Aircraft engines were held at ground idle and aircraft rotor was permitted to spin down to steady state. Rotor brake was not applied.

Table 8. Flights and Fault Conditions for Phase III Testing.

Date (1997)	Flt No.	Flt Title	Change Made Prior to Flight
4 August	32	Baseline	-
5 August	33	Inboard Damper Bearing Fault Flight	Bearing S/N 72206 placed in the inboard spot of the Red Blade Lead-Lag Damper
6 August	34	Baseline	New bearing placed in the inboard spot of the Red Blade Lead-Lag Damper
7 August	35	Upper Pitch Control Rod Bearing Fault Flight	Bearing S/N 72206 placed in the upper spot of the Red Blade Pitch Control Rod
8 August	36	Baseline	New bearing placed in the upper spot of the Red Blade Pitch Control Rod
29 August	51	Baseline	-
29 August	52	Upper Pitch Control Rod Bearing Fault Flight	Bearing S/N 72206 placed in the upper spot of the Red Blade Pitch Control Rod
2 September	53	Baseline	- New bearing placed in the upper spot of the Red Blade Pitch Control Rod - New bearing placed in the outboard spot of the Red Blade Lead-Lag Damper
2 September	54	Inboard Damper Bearing Fault Flight	Bearing S/N 72206 placed in the inboard spot of the Red Blade Lead-Lag Damper
3 September	55	Baseline	New bearing placed in the inboard spot of the Red Blade Lead-Lag Damper
4 September	56	Baseline	-
5 September	57	Lead-Lag Damper Fault Flight	Damper S/N A221-00666 (no internal seal) placed on the Red Blade
5 September	58	Baseline	Previous damper (from flight 56) replaced on Red Blade

Table 9. Main rotor track and balance flight results: baseline flight data (a), pitch control rod fault flight (b) and evaluation flight with AVDS corrections (c).

(a)

Maneuver	Track Spread (cm) ²	Accelerometer No. 1 (g's) ¹	Accelerometer No. 2 (g's) ¹
Ground Run	0.75	0.012	0.011
HIGE	0.66	0.009	0.008
80 kt Level Flight	1.67	0.008	0.003
120 kt Level Flight	3.49	0.018	0.008
145 kt Level Flight	5.18	0.013	0.010

(b)

Maneuver	Track Spread (cm) ²	Accelerometer No. 1 (g's) ¹	Accelerometer No. 2 (g's) ¹
Ground Run	No Input	0.010	0.015
HIGE	No Input	0.044	0.039
80 kt Level Flight	No Input	0.049	0.030
120 kt Level Flight	No input	0.056	0.046
145 kt Level Flight	No Input	0.085	0.038

(c)

Maneuver	Track Spread (cm) ²	Accelerometer No. 1 (g's) ¹	Accelerometer No. 2 (g's) ¹
Ground Run	0.22	0.007	0.019
HIGE	1.39	0.010	0.004
80 kt Level Flight	1.75	0.014	0.012
120 kt Level Flight	3.40	0.008	0.010
145 kt Level Flight	5.82	0.006	0.012

- Notes: (1) 0.019 g's corresponds to 0.3 ips.
 (2) 1 cm = 0.3937 in.

Table 10. Main rotor balance flight results: hub weight fault flight (a) and evaluation flight with AVDS corrections (b).

(a)

Maneuver	Track Spread (cm)	Accelerometer No. 1 (g's)	Accelerometer No. 2 (g's)
Ground Run	0.14	0.020	0.020
HIGE	1.37	0.030	0.030
80 kt Level Flight	1.83	0.035	0.027
120 kt Level Flight	3.38	0.042	0.022
145 kt Level Flight	5.97	0.043	0.030

(b)

Maneuver	Track Spread (cm)	Accelerometer No. 1 (g's)	Accelerometer No. 2 (g's)
Ground Run	1.68	0.018	0.014
HIGE	1.85	0.010	0.004
80 kt Level Flight	3.72	0.001	0.012
120 kt Level Flight	5.07	0.005	0.017
145 kt Level Flight	7.98	0.17	0.14

APPENDIX C

DESCRIPTION OF TEST AIRCRAFT

The SH-60F is a single main rotor, twin engine helicopter manufactured by United Technologies Corporation, Sikorsky Aircraft Division. The helicopter is primarily designed for Under Surface Warfare (USW) defense of the CV inner zone. The helicopter is powered by two T700-GE-401C engines, each capable of developing a continuous SHP rating of 1662 and a maximum of 1940 SHP for 2.5 minutes (sea level/std day). The drivetrain system consists of the main transmission gearbox, two input modules, two accessory modules, tail rotor drive shaft, oil cooler axial fan (integral to the tail rotor drive shaft system), intermediate gearbox, and a tail rotor gearbox. Each accessory module is driven through the main gearbox and provides for mounting and drive of an AC electrical generator and a hydraulic pump package. The fully articulated main rotor incorporates two elastomeric bearings per blade which enable the four blades to independently flap, lead, lag, and change pitch. The four bladed tractor type (bearingless) tail rotor is canted 20 degrees upward providing 2.5% of the total lifting force in a hover. The main rotor provides for track and balance adjustments through ground adjustable pitch change rods, trim tabs and mounting for rotor head balance weights. The tail rotor similarly provides mounting for balance weights. A more detailed description of the SH-60F may be found in the NATOPS Flight Manual, (reference 25) and the Organizational Principles of Operation (reference 26). A more detailed description of the drivetrain may be found in the U. S. Navy Helicopter SH-60F Vibration Analysis Manual, NAVAIR 01-1A-24 (reference 24).

VITA

James L. Robbins was born in Arlington Heights, Illinois on December 2, 1964. He was schooled in the public systems of Marengo, Illinois, where he graduated from Marengo Community High School in June 1983. He entered The University of Texas at Austin during August of 1983 with a four-year scholarship from the U.S. Navy. In December 1987, he earned the Bachelor of Science in Aquatic Biology. After graduation, he earned his commission in the U.S. Navy and proceeded to Pensacola, Florida for flight training. He earned his Navy pilot wings in June 1989 and transferred to Jacksonville, Florida where he served as a fleet pilot for Helicopter Anti-Submarine Squadrons Seventeen and Five and later as a flight instructor for Helicopter Anti-Submarine Squadron One. He was selected to attend the U.S. Naval Test Pilot School in March 1995 where he graduated as a member of Class 109 in June 1996. He served at the Naval Rotary Wing Aircraft Test Squadron as a project officer and pilot from July 1996 to August 1997 when he was selected to return to the Test Pilot School as a flight instructor. In August of 1996, he enrolled in The University of Tennessee Space Institute to pursue the Master's of Science in Aviation Systems. The Master of Science degree was received August 1999.

LCDR Robbins has flown over 2,700 flight hours in rotary and fixed wing aircraft and has held Aircraft Commander certifications in 14 different airframes. He presently teaches Navy pilots to tactically operate the SH-60F and HH-60H helicopters for Helicopter Attack Squadron Ten at Naval Air Station, North Island in San Diego, California.