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AH-1G DESIGN AND OPERATIONAL FLIGHT
LOADS STUDY

Max E. Glass, et al

Bell Helicopter Company

Prepared for:

Army Air Mobility Research and Development
Laboratory

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13. ABSTRACT This report compares AH-1G helicopter Southeast Asian mission profiles with the original engineering frequency-of-occurrence spectrum and the Navy AR-56 spectrum for attack helicopters. Fatigue lives calculated using the Southeast Asian profile are compared with those determined using the original frequency-of-occurrence spectrum. The development cycle of the Bell Helicopter Company Model 540 rotor system is reviewed, and the fatigue design methods used are presented. Maximum one-time occurrences measured in the Southeast Asian operational survey are compared with those specified in the AH-1G structural design criteria and those measured in structural demonstration flight tests. Recommendations are made regarding future mission surveys, the structural design criteria for attack helicopters, and the upgrading of rotor loads prediction capability.			

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This program was conducted under Contract DAAJ02-71-C-0099 with Bell Helicopter Company.

The information presented herein is the result of an analytical effort to derive improved structural design criteria for gunship-type helicopters based upon flight parameters measured on gunship helicopters operating in Southeast Asia. This is one of four similar efforts being conducted concurrently to develop improved criteria for observation, crane, and transport as well as gunship-type helicopters.

The report has been reviewed by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound. It is published for the exchange of information and the stimulation of future research.

This program was conducted under the technical management of Mr. Herman I. MacDonald, Jr., Technology Applications Division.

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Bell Helicopter Report
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FOREWORD

This report, "AH-1G Design and Operational Flight Loads Study," was prepared by Bell Helicopter Company, Fort Worth, Texas, for the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under Contract DAAJ02-72-C-0099, Task 1F162204A17002. Mr. Herman MacDonald was the contract technical representative.

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- Mr. G. Colvin, Senior Test Pilot
- Mr. D. Bloom, Senior Test Pilot
- Mr. J. Duhon, Group Engineer, Aerodynamics
- Mr. C. Livingston, Group Engineer, Stability and Control

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INTRODUCTION

An essential factor in determining fatigue life is an estimate of the frequency of occurrence of the various flight conditions the helicopter will experience in service. In the preparation of the original engineering development frequency-of-occurrence spectrum for the AH-1G helicopter, as used in Reference 1, full advantage was taken of all information available on this subject. This included considerable past experience in the preparation of similar spectrums for other helicopters, both military and civilian. Data were also available from statistical surveys conducted on helicopters in actual service. Bell Helicopter, under Army funding, has conducted several of these statistical surveys on the YH-40 (HU-1A), HU-1B, and YUH-1D helicopters under normal operating conditions which are reported in References 2, 3, and 4. All of these surveys include VGH (velocity, acceleration, and altitude) data as well as important channels of strain gage data for oscillatory load information. Some other useful statistical publications were available for both civilian and Army helicopters, including the UH-1B, HU-1A, and H-13H as presented in References 5, 6, and 7, the H-13, and UH-1A. A suggested frequency-of-occurrence spectrum for helicopters is also included in Civil Aeronautics Manual 6, "Rotorcraft Airworthiness," Appendix A, "Service Life Determination."

In addition to the background of statistical data and previous experience, discussions with Army helicopter pilots with combat experience and with AH-1G test pilots were considered in evaluating the probable operations of this helicopter. All of the above information was used in deriving the original engineering development frequency-of-occurrence spectrum for the AH-1G shown in Table I. A comparison with the Army logistical evaluation test plan for this helicopter showed good general agreement in proportion of time allotted to cruise and maneuvers, although the fatigue life determination spectrum was weighted more heavily toward the high airspeed capabilities of the helicopter.

The AH-1G helicopter, Figure 1, has undergone heavy usage in the combat zones of Southeast Asia. In an effort to gain information on the operational environment of the AH-1G in combat situations, the flight loads investigation outlined in Reference 8 was conducted. These data present an opportunity to investigate the relationship between the original predicted frequency-of-occurrence spectrum and the actual operational environment of the helicopter. This task is addressed in subsequent sections of this report, with the ultimate purpose being to calculate fatigue lives for various components to study the impact of spectrum variations and modifications.



Figure 1. Bell Model AH-1G Helicopter.

To provide some insight into the fatigue design problem, the development of the 540 rotor system is reviewed and the approach used in sizing the components of this system for fatigue loading is discussed. The fatigue design methods are then evaluated in light of operational experience to determine if weaknesses in the method could be related to a shorter than predicted fatigue life encountered for a major component. The purpose of this section is to show the major variables, component fatigue strength, oscillatory loads, and frequency-of-occurrence spectrum in a total perspective. In this manner, the significance of each variable can be examined and its importance at various stages of the design-development cycle can be evaluated.

In addition to the frequency-of-occurrence spectrum comparison, a study of maximum one-time occurrences was also conducted. The maximum one-time occurrences of the load-significant parameters measured in the mission profile survey are compared with the values specified in the structural design criteria, the aircraft operating limitations, and those measured in the helicopter flight structural demonstration. A discussion of what factors probably caused each maximum one-time value is given, and a method for predicting these limiting parameters in future attack-type helicopters is suggested.

ANALYSIS OF DATA

The operational data presented in Reference 8 were accumulated in two data samples. The first sample represented 201.8 flight hours, and the second sample represented 206.5 flight hours. In order to perform a comparative analysis of these data with the original frequency-of-occurrence spectrum, the mission data were converted to a 100-hour base. This conversion was accomplished by using a weighted percentage for each data increment of each parameter. Having accomplished this conversion, the combat operational data were then compared directly with the data from the original frequency-of-occurrence spectrum.

MISSION SEGMENTS

The operating environment of the AH-1G has been divided into four mission segments for the purpose of comparing original spectrum data with operational data. These segments are: (1) ascent, (2) maneuver, (3) descent, and (4) steady state. The ascent stage is defined as that portion of flight from takeoff to cruise altitude and any ascents to other altitudes. The maneuvering segment includes all attitude and direction changes, power transitions, and any conditions in which weapon firing occurred. The descent stage includes the time in descent, flare, and landing. Steady state includes all stabilized flight such as cruise, hover, and stabilized autorotation.

The data presented in Figure 2 show a significant difference between the maneuver and steady-state time distributions of the original frequency-of-occurrence spectrum and the combat operational data. Some of this difference can possibly be attributed to assigning a few flight conditions from the original spectrum into different segments than was done in Reference 8. However, most of the difference is thought to be the result of a change in helicopter mission description. In preparing the original frequency-of-occurrence spectrum, it was anticipated that the ship would take off, cruise to the target area, accomplish the mission task, and cruise back to its base. The combat operational data indicate frequent excursions during the outgoing and return cruise operation. These excursions were apparently to search for additional activity outside the prescribed mission.

GROSS WEIGHT

The gross weight envelope for the AH-1G helicopter, set forth by Bell Helicopter Company, was from 6500 pounds to a maximum of 9500 pounds. In establishing the fatigue lives for the AH-1G components, the following gross weight distribution was

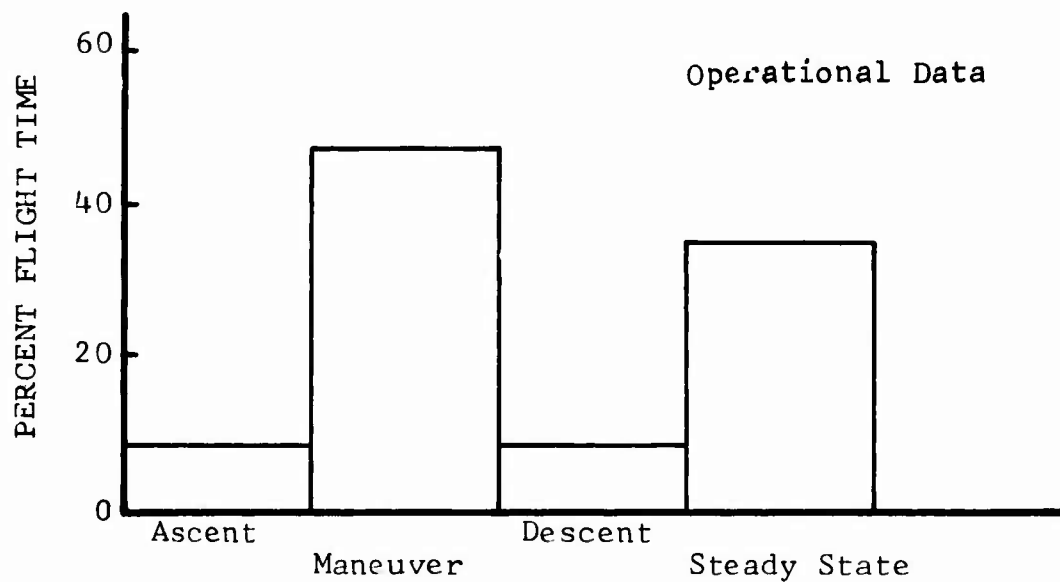
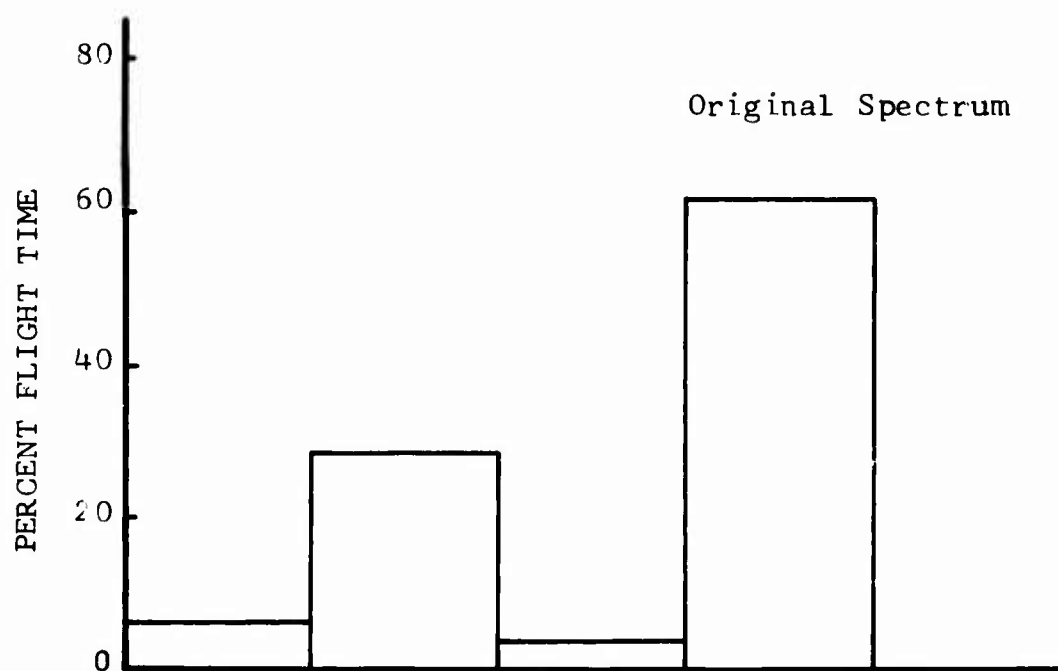


Figure 2. Distribution of Flight Time in Mission Segments for the Original AH-1G Spectrum and the Operational Data.

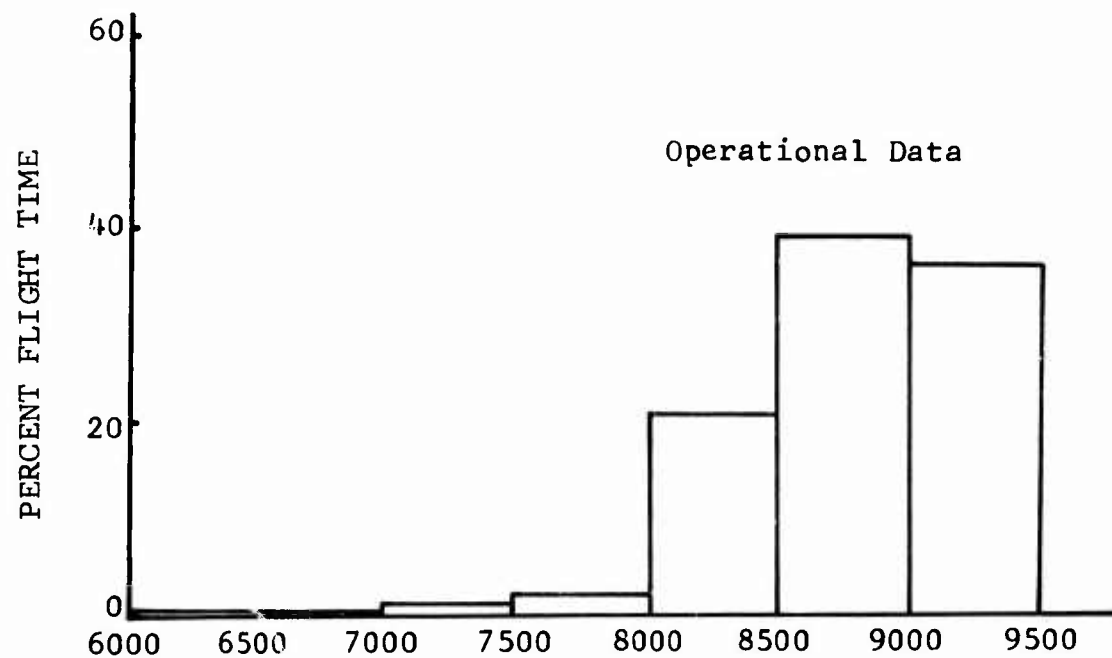


Figure 3. Distribution of Flight Time for Gross Weights for the Original AH-1G Spectrum and the Operational Data.

used: light gross weights to 7000 pounds, 20 percent; medium gross weights from 7001 to 8200 pounds, 50 percent; and heavy gross weights from 8201 to 9500 pounds, 30 percent of total flight time. As shown in Figure 3, the gross weight distribution determined from the operational data indicates that considerably more time is spent in the heavy gross weight conditions.

AIRSPEED

The airspeed distributions presented in Reference 8 were combined into a total combat operational profile shown in Figure 4. These data differ significantly from the airspeed profile for the original spectrum also shown in Figure 4. The combat operational data are distributed about the 90- to 100-knot increment, while the original spectrum data are distributed about the 130- to 140-knot increment. However, the maximum airspeed demonstrated within the original data was 184 knots compared to the combat operations maximum of 186 knots. This indicates that the airspeed range for the original data was sufficient to cover the operational airspeeds of the helicopter, but the airspeed distribution was not accurately predicted.

OTHER PARAMETERS

Operational data for four other parameters - (1) altitude, (2) rotor speed, (3) torque pressure, and (4) vertical load factor - were available for comparison. The original design spectrum did not include distributions of these parameters; therefore, no comparison is shown.

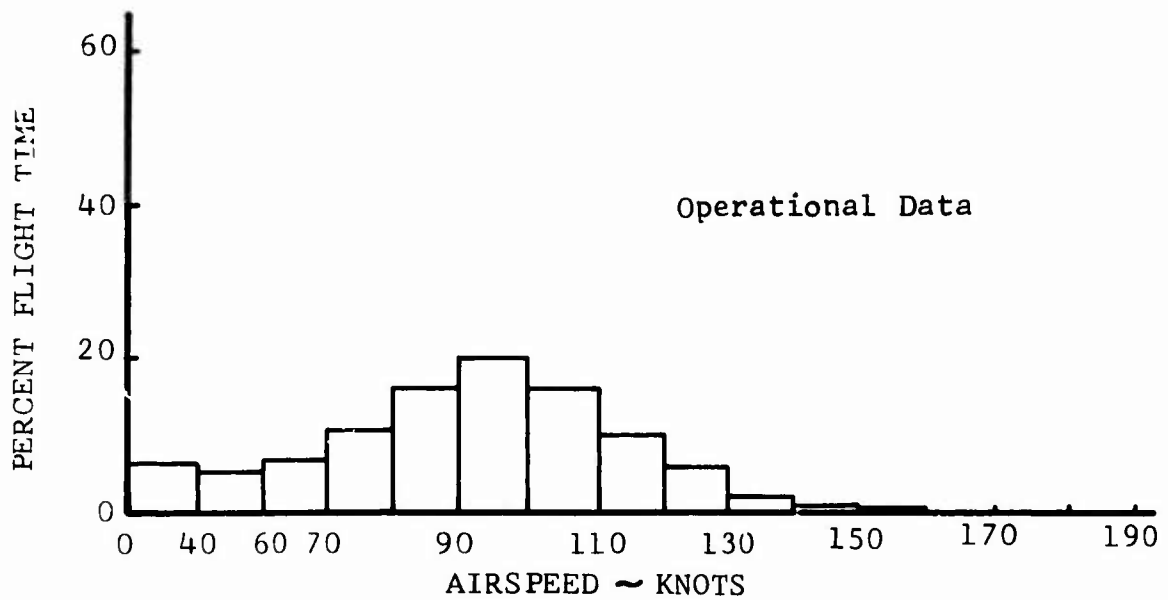
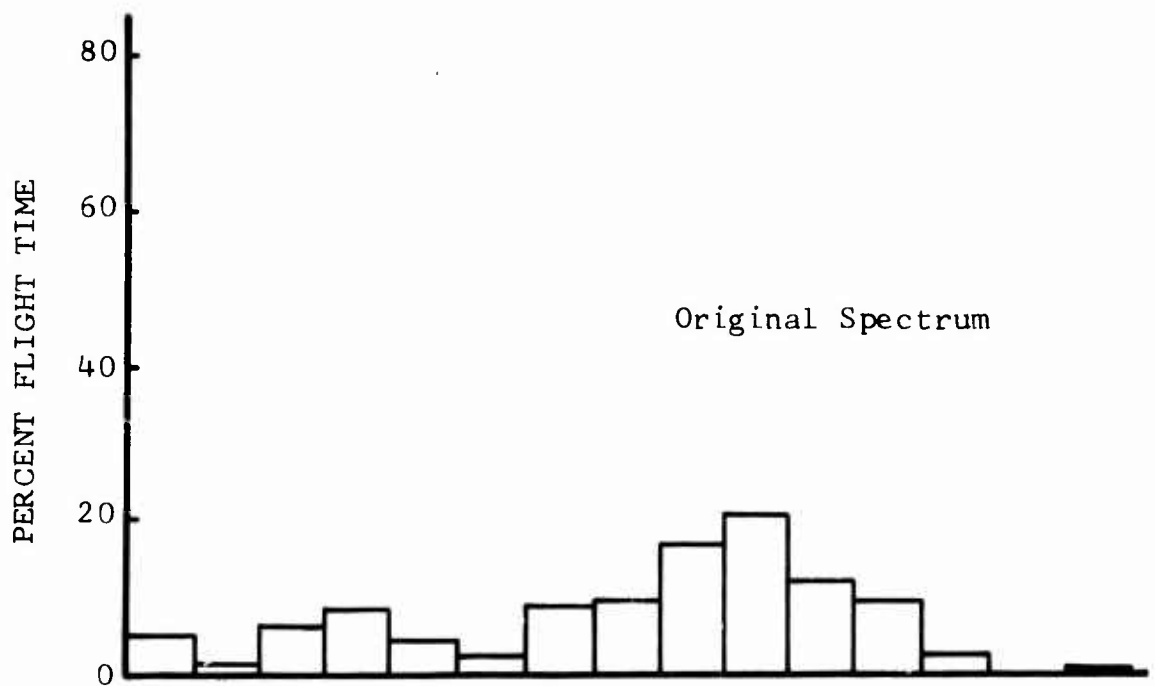


Figure 4. Comparison of Airspeed Distribution for the Original AH-1G Spectrum and the Operational Data.

PROBLEMS ENCOUNTERED DURING ANALYSIS

Since the original design spectrum did not include distributions of all parameters, an effort was made to compare the operational data to the developmental load level survey data of Reference 9. Several problems arose in comparing those data since the load level survey data were obtained at discrete representative values of some parameters rather than at several values distributed to represent the predicted use of the aircraft. For example, the operational data increments for altitude in Reference 8 were:

- 0 to 1000 feet
- 1001 to 2000 feet
- 2001 to 5000 feet
- 5001 to 10,000 feet
- 10,001 to 15,000 feet

This format may have been best for the mission data. However, altitudes in Reference 9 were condition-entry altitudes only: IGE, 1500, 5000, and 10,000 feet, with no continuous readings. Therefore, comparison was virtually impossible and undoubtedly inconclusive.

A similar problem arose when considering vertical load factor. The mission data were continuously reduced, with all excursions outside the 0.8 - 1.2 g range measured and recorded. However, the original spectrum load factor data were reduced only at the point of maximum g's and therefore do not represent a total data profile. For example, consider a diving maneuver. At the pushover, the load factor drops below 1.0 g. During the dive, the g level returns to the 1.0 g level, and when control inputs are made to pull out, a buildup in excess of 1.0 g will occur. Although the load factor has experienced a range of values from less than 1.0 g to a maximum greater than 1.0 g, the original spectrum data would list only the maximum value. The omission of the remaining data prohibited presentation of a load factor distribution of the original spectrum data.

Another problem area was found in the presentation of the rotor speed data. The mission data increments of Reference 8 are listed below.

- Up to 295 rpm
- 296 to 310 rpm

- 311 to 325 rpm
- 326 to 330 rpm
- 331 to 340 rpm
- 341 to 355 rpm
- 356 rpm and above

This arrangement prevented a detailed comparison since the normal operating range of rotor speed for the AH-1G is from 314 to 324 rpm. Due to the unfortunate choice of increments, all of the normal operation time falls into the third increment, leaving less than 5 percent of the total time to be distributed among the remaining increments.

OPERATIONAL SPECTRUM DERIVATION

Having analyzed the available data, the original spectrum was used as a baseline and was modified based on the operational data to derive an operational spectrum. The first step in modification was to redistribute the flight time to agree with the mission segment data from Reference 8. When this was accomplished, the airspeed distribution was then investigated. Due to interaction between the mission segments and airspeed, it was not possible to bring the modified spectrum into exact agreement with the operational data. However, the difference is considered insignificant. Table I shows the original engineering frequency-of-occurrence spectrum, Reference 1, and the derived operational frequency-of-occurrence spectrum to be used for reevaluating fatigue lives of selected AH-1G components.

Figures 5 through 9 contain histograms of five data parameters for the purpose of comparing the profiles of the operational data and data from the derived operational frequency-of-occurrence spectrum. Figure 5 shows the new distribution of flight time in the various mission segments. There are slight differences between the operational data of Reference 8 and the derived operational spectrum. These differences could easily be attributed to variations in the flight conditions as assigned to the various mission segments. Therefore, the extent of agreement is considered satisfactory.

Airspeed distributions are presented in Figure 6. Both the derived operational frequency-of-occurrence spectrum and the operational data are distributed about the 90- to 100-knot increment. Also, the relative distributions of both data sets are quite comparable. There is a very small amount of time remaining in the modified spectrum distribution for airspeeds between 180 and 190 knots. This is necessary since the maximum airspeed for the helicopter is in this range and cannot be completely ignored.

Engine delta torque distributions are presented in Figure 7. The only significant difference in these distributions is the larger amount of time in the derived operational spectrum for the 0- to 10-psi increment. This difference is due to the time allotted for autorotation flight and transitions in the derived operational frequency-of-occurrence spectrum. Autorotation is normally an emergency procedure; therefore, little time is spent in this flight mode during normal combat operations. However, the amount of time allocated to autorotation flight in the derived operational spectrum seems necessary to account for all operations including training and emergency procedures practice.

TABLE I. ORIGINAL AND OPERATIONAL AH-1G
FREQUENCY-OF-OCCURRENCE SPECTRUMS

Condition	Percent Time	
	Original	Operational
Ground conditions		
Normal start	0.5000	0.4000
Shutdown w/coll	0.5000	0.4000
IGE Maneuvers		
Takeoff		
Normal	0.9000	1.2780
Jump	0.1000	0.1420
Hovering		
Steady	2.1700	2.0000
Right turn	0.1000	0.1670
Left turn	0.1000	0.1670
Control reversal		
Longitudinal	0.0100	0.0167
Lateral	0.0100	0.0167
Rudder	0.0100	0.0167
Sideward Flight		
To the right	0.2500	0.2404
To the left	0.2500	0.2404
Rearward flight	0.2500	0.2404
Acceleration		
Hover to climb A/S	0.5000	0.5000
Deceleration		
Normal	0.7000	0.5000
Quick stop	0.3000	0.1000
Approach and landing	1.0000	5.5095
Forward level flight		
Airspeed	RPM	
0.50 VH	314	0.5000
	324	0.2605
0.60 VH	314	4.5000
	324	2.3450
0.70 VH	314	0.2000
	324	0.7723
0.80 VH	314	1.8000
	324	6.9508
0.90 VH	314	0.3000
	324	0.8551
VH	314	2.7000
	324	7.6963
	314	1.5000
	324	1.3773
	314	13.5000
	324	12.3961
	314	2.5000
	324	0.3990
	314	22.5000
	324	3.5910
	314	1.0000
	324	0.3460
	324	9.0000
		3.1140

TABLE I - Continued		
Condition	Percent Time	
	Original	Operational
Nonfiring maneuvers		
Full power climb		
Normal	4.0000	2.5000
High-speed	1.0000	0.0426
Maximum rate accel		
Climb - cruise A/S	2.8000	4.6760
Normal turns		
To the right		
0.5 VH	1.10000	1.6700
0.7 VH	1.0000	1.6700
0.9 VH	2.0000	0.1086
To the left		
0.5 VH	1.0000	1.6700
0.7 VH	1.0000	1.6700
0.9 VH	2.0000	0.1086
0.9 VH control reversal		
Longitudinal	0.5000	0.0835
Lateral	0.0500	0.0835
Rudder	0.0500	0.0141
Sideslip	0.5000	0.2000
Part power descent	2.5500	0.1000
Gunnery maneuvers		
Firing in a hover	0.0750	0.1252
Strafing in accel. from a hover	0.0500	0.0835
Gunnery runs		
PT. Target dives		
To 0.6 VL	0.2800	0.4676
To 0.8 VL	0.8400	2.6003
To 0.9 VL	1.4000	8.0508
To VL	0.2800	0.0200
Spray fire dives		
To 0.6 VL	0.1200	0.2004
To 0.8 VL	0.3600	2.6974
To 0.9 VL	0.6000	5.7264
To VL	0.1200	0.1000
Gunnery run pullup		
To the right		
0.6 VL	0.1000	0.0500
0.8 VL	0.3000	0.1000
0.9 VL	0.5000	0.2500
To the left		
0.6 VL	0.1000	0.0500
0.8 VL	0.3000	0.1000

TABLE I - Continued		
Condition	Percent Time	
	Original	Operational
To the left (Cont'd)		
0.9 VL	0.5000	0.2500
VL	0.1000	0.1670
Symmetrical		
0.6 VL	0.0100	0.0050
0.8 VL	0.0300	0.0501
0.9 VL	0.0500	0.0835
VL	0.0100	0.0167
Gunnery Turns		
To the right		
0.5 VH	0.3750	0.6262
0.7 VH	0.3750	1.8236
0.9 VH	0.7500	0.0400
To the left		
0.5 VH	0.3750	0.6262
0.7 VH	0.3750	1.8236
0.9 VH	0.7500	0.0400
S-Turns		
At 0.8 VH	0.2000	0.1000
At VH	0.0750	0.1282
Power transitions		
Power to auto		
0.5 VH	0.0500	0.0835
0.7 VH	0.1250	0.2087
0.9 VH	0.1750	0.1500
Auto to power		
In ground effect	0.1500	0.1000
0.4 VH	0.1000	0.1670
0.6 VH	0.0750	0.8104
Max auto A/S	0.0250	0.0300
Autorotation		
Stabilized flight		
0.4 VH	0.2000	0.1086
0.6 VH	1.4000	2.0236
Max auto A/S	0.3000	0.1000
Auto turns		
To the right		
0.4 VH	0.0500	0.0835
0.6 VH	0.4000	1.3532
Max auto A/S	0.0500	0.0500

TABLE I - Continued		
Condition	Percent Time	
	Original	Operational
To the left		
0.4 VH	0.0500	0.0835
0.6 VH	0.4000	1.3532
Max auto A/S	0.0500	0.0500
Auto Landing	0.2500	0.1000

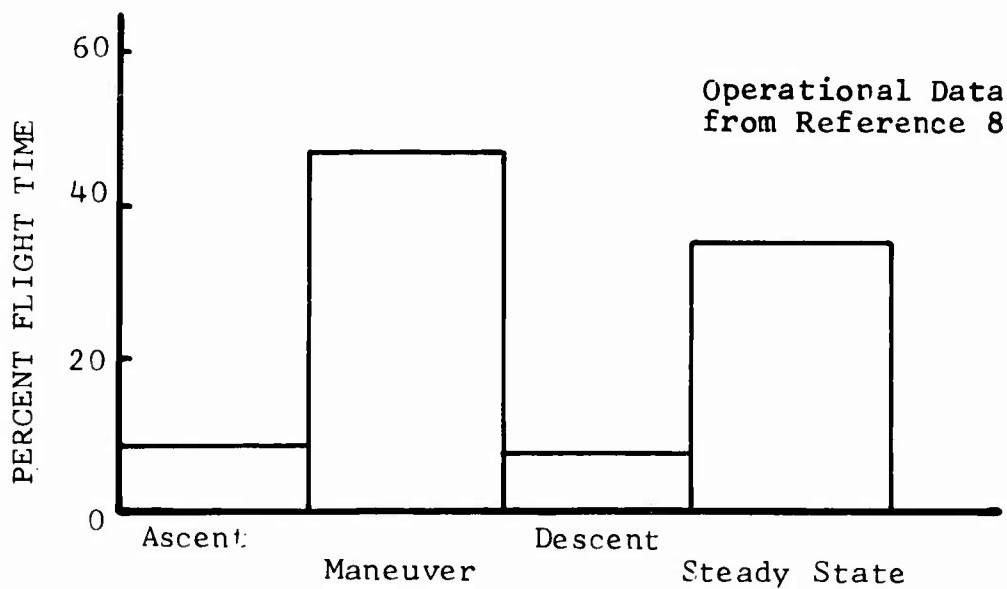
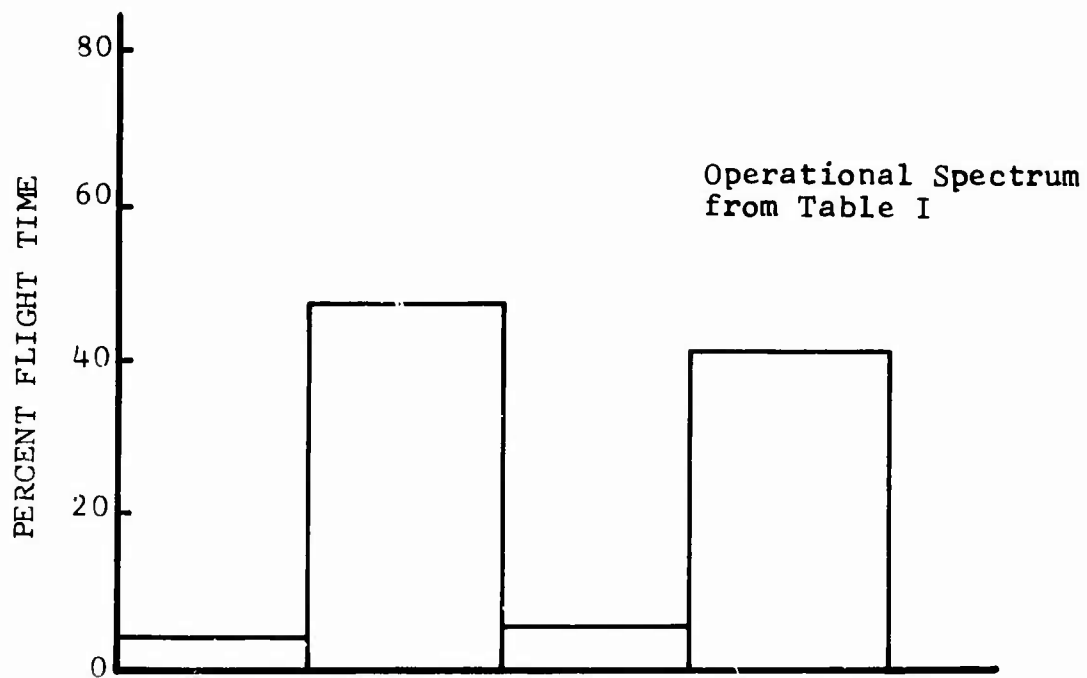


Figure 5. Distribution of Flight Time in Mission Segments for the Operational AH-1G Spectrum and the Operational Data.

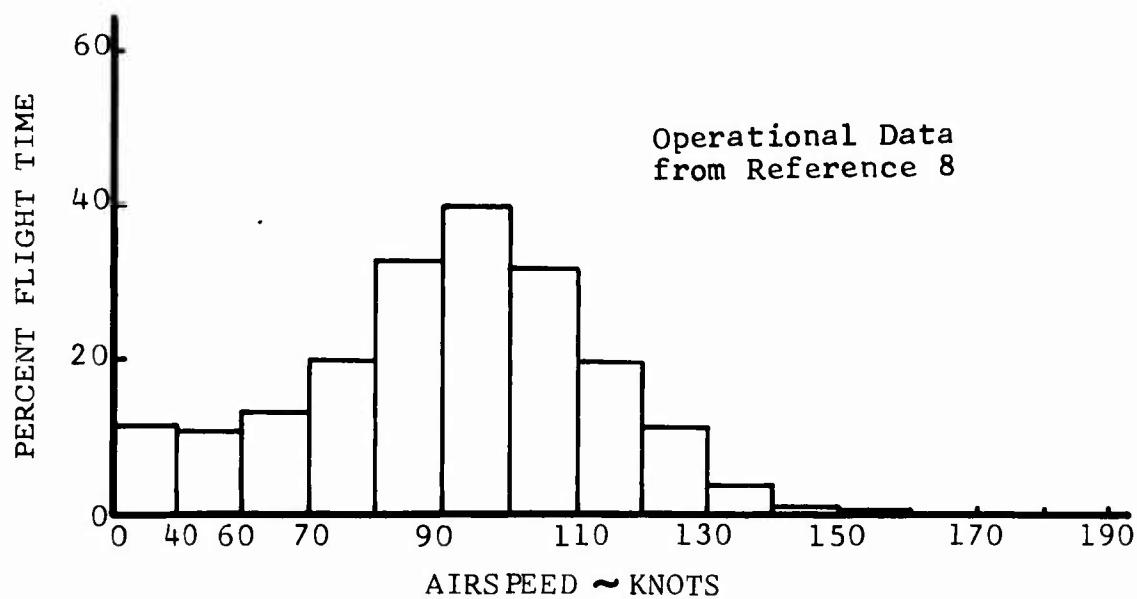
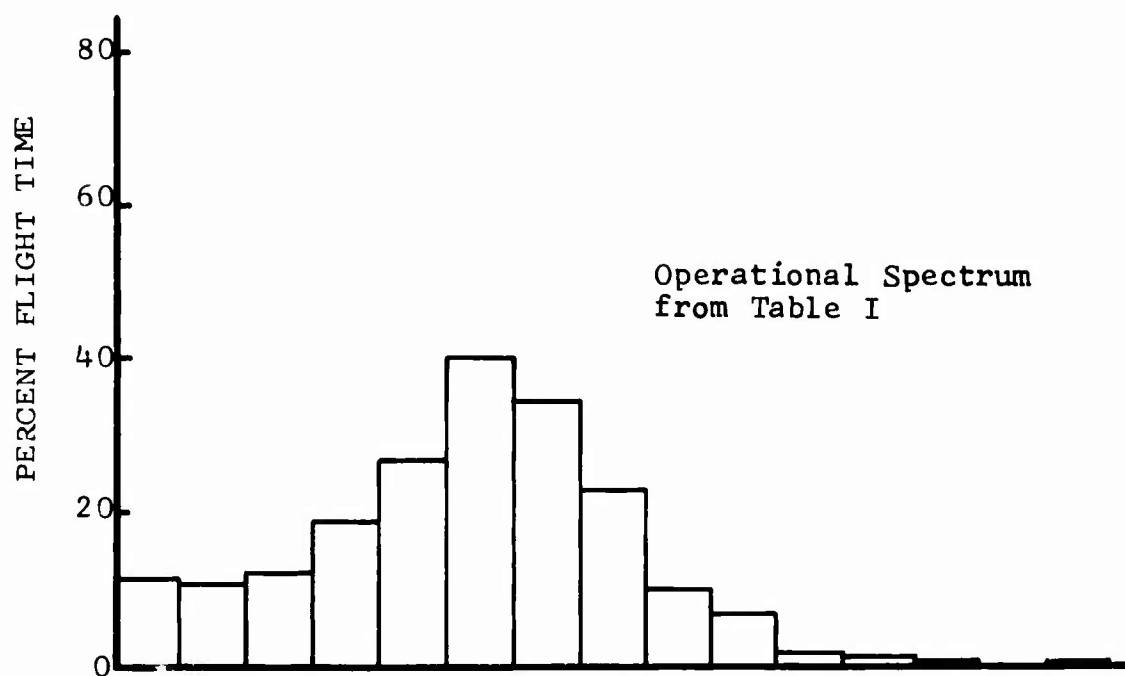


Figure 6. Comparison of Airspeed Distribution for the Operational AH-1G Spectrum and the Operational Data.

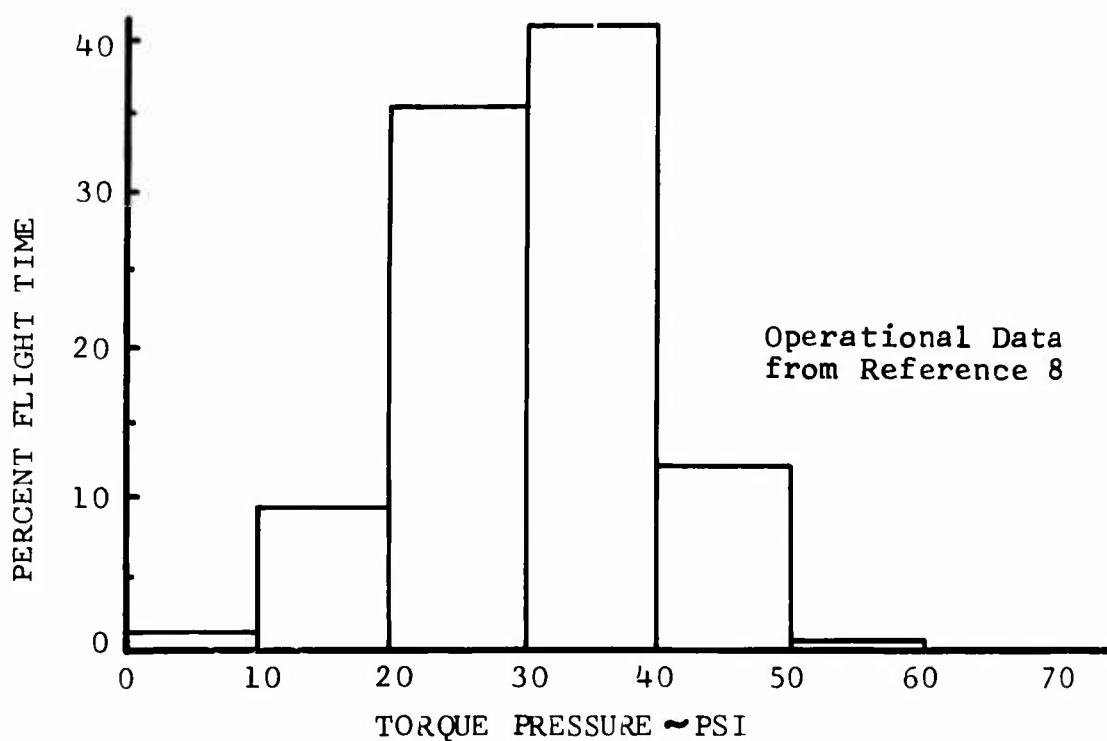
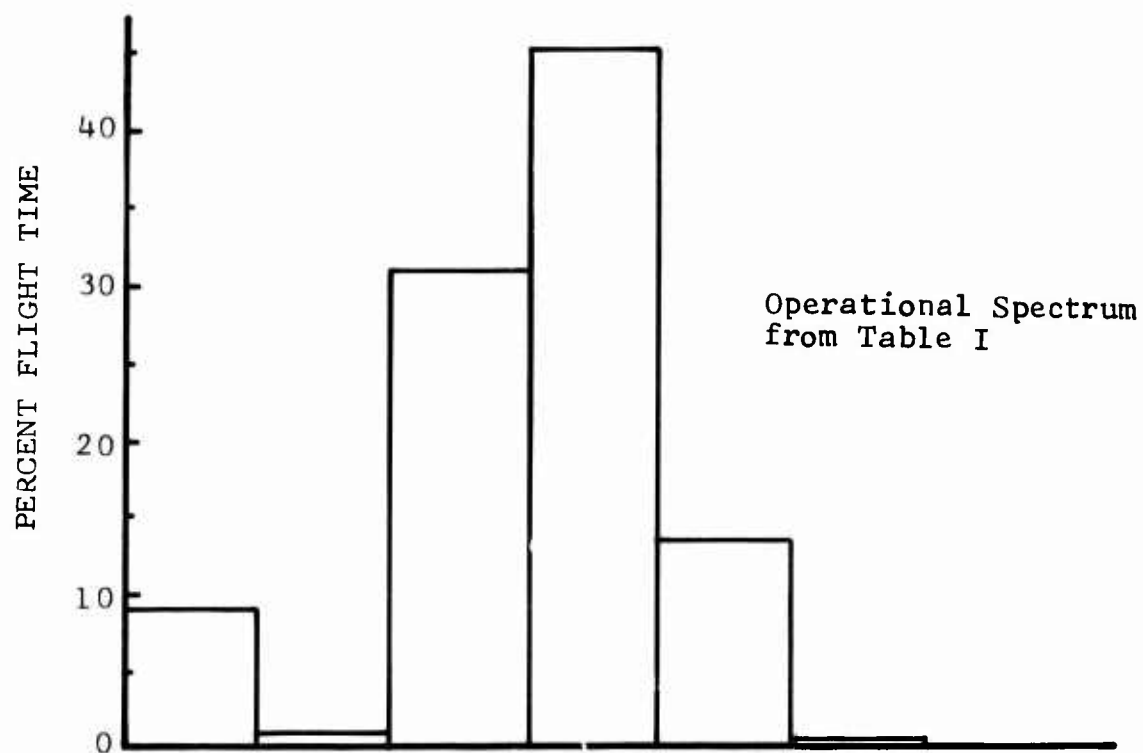


Figure 7. Comparison of Torque Pressure Distribution of the Operational Spectrum and the Operational Data.

Rotor speed data are presented in Figure 8. However, from Figure 8 it can be seen that the derived operational spectrum data show similar distributions compared to the operational data in the intervals outside the normal operating range. Therefore, it is felt that the derived operational spectrum adequately represents the operational data with respect to rotor speed.

Figure 9 is included in this report to show the range of vertical load factors for both the derived operational spectrum and the operational data. It is noted that the ordinate for the graph of operational spectrum data from Table I is expressed in percentage of maneuver time while the operational data from Reference 8 is presented in terms of percentage of total occurrences. This difference in presentation was necessary due to the "one measurement per flight condition" procedure followed in reducing the data for the operational spectrum. This procedure precluded any distribution based on occurrences. The data shown for the operational data was obtained by converting "time to reach or exceed" data from Reference 2 into occurrences per 100 flight hours from which the included distribution was calculated. Therefore, this data cannot be considered conclusive. The figure does show that the extreme g levels are adequately represented in the operational frequency-of-occurrence spectrum.

In view of these comparisons of pertinent parameters, the operational frequency-of-occurrence spectrum is considered to represent the AH-1G combat operational data with reasonable accuracy.

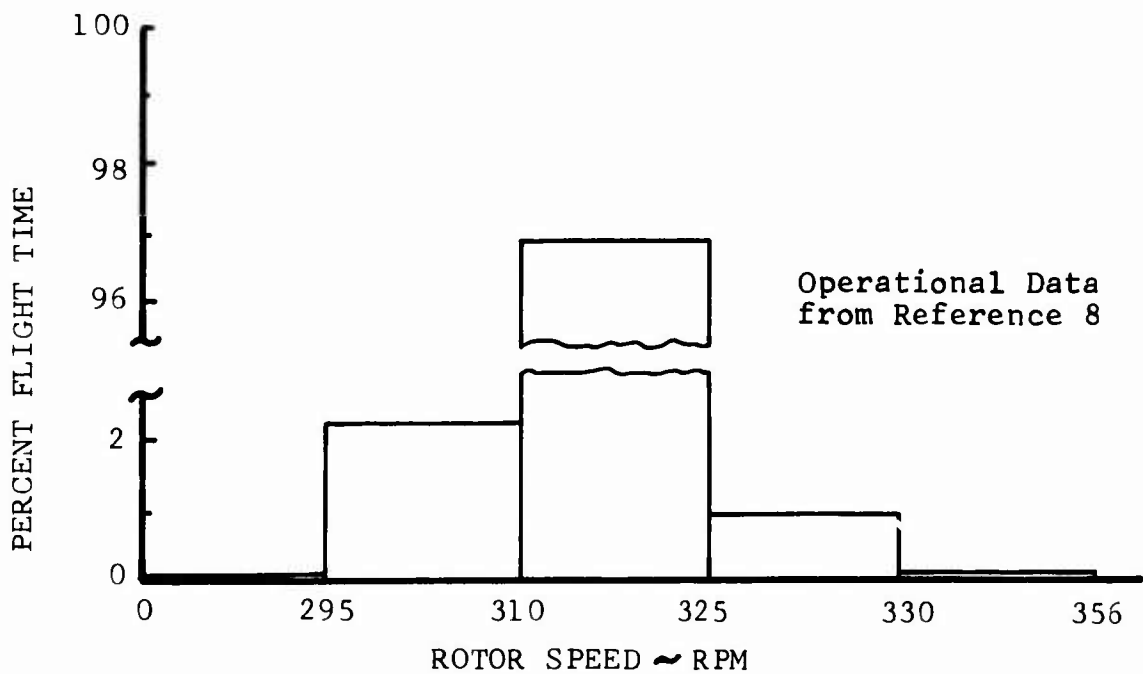
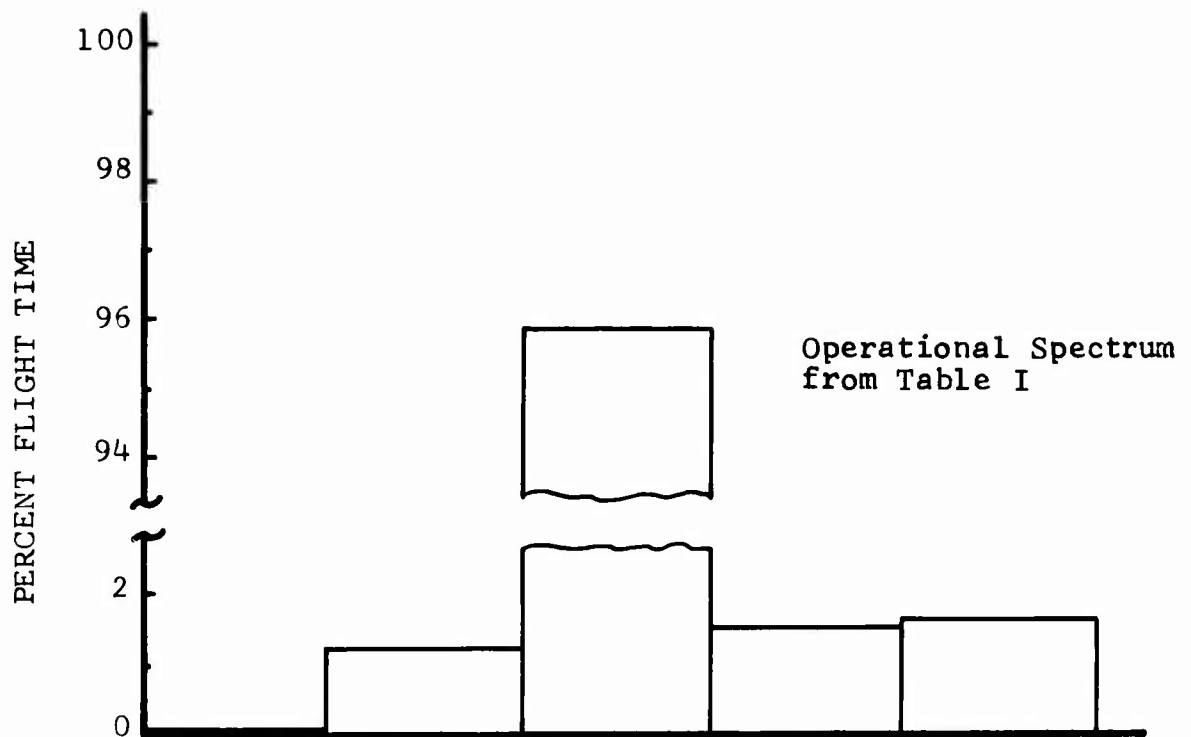


Figure 8. Comparison of Rotor Speed Distribution for the Operational Spectrum and the Operational Data.

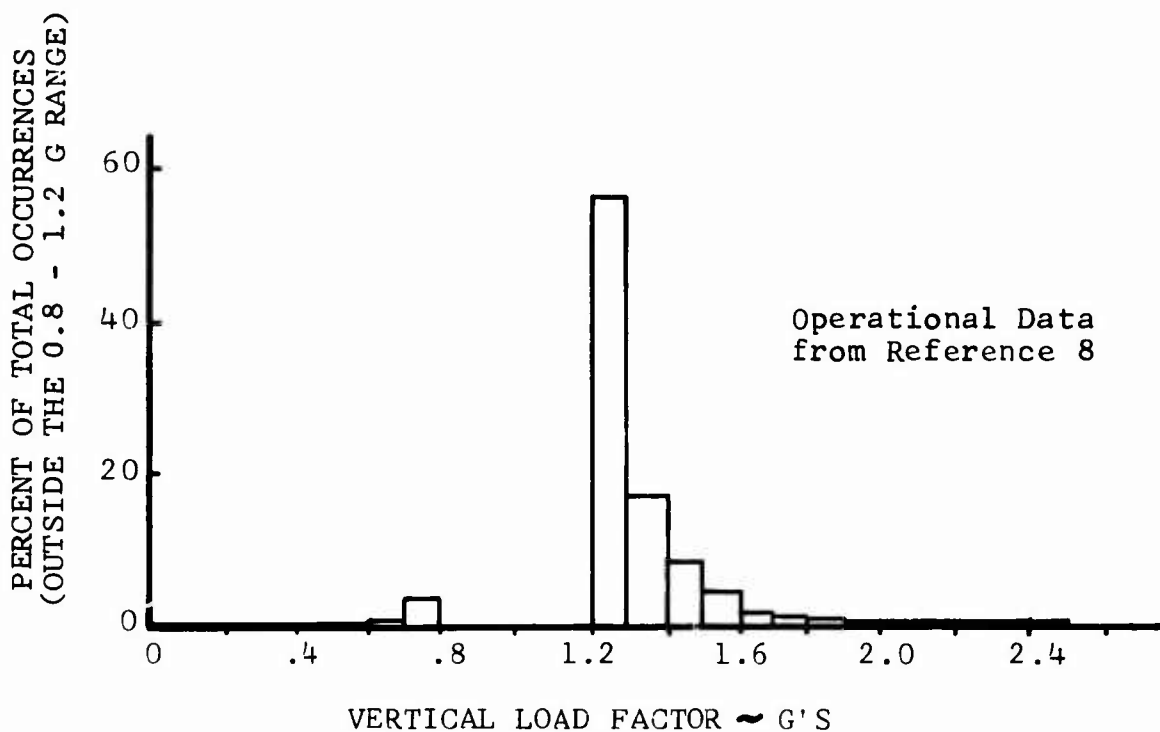
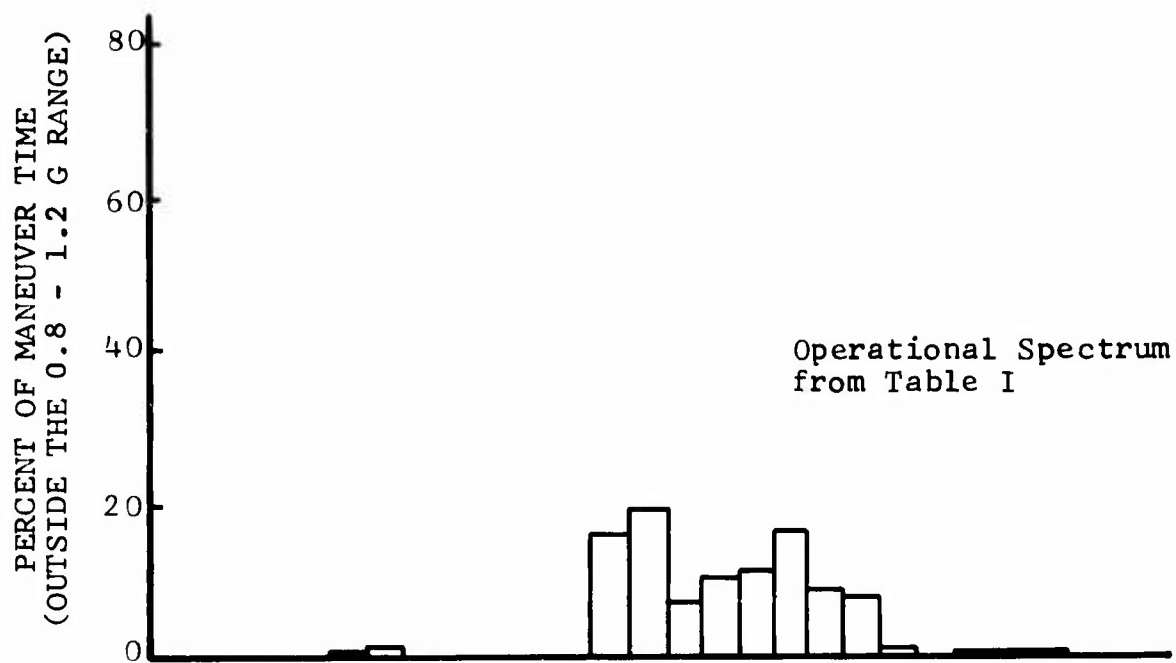


Figure 9. Comparison of Vertical Load Factor Distribution for the Operational Spectrum and the Operational Data.

ADDITIONAL SPECTRUM EVALUATION

The helicopter structural design requirements outlined by the United States Navy are presented in AR-56 (Reference 10). One of the guidelines contained in these requirements is the table of mission profiles for each category of helicopter in the Navy's inventory. The mission profile for attack helicopters, presented in AR-56, has been compared with the operational AH-1G spectrum presented in Table I.

Table II shows the AR-56 spectrum. The frequency of occurrence for most conditions was shown in percentage of total flight time. However, several flight conditions were shown in terms of the number of occurrences per 100 flight hours instead of percentage of time. These occurrences are in addition to the other conditions which already total 100 percent of the flight time. Before the AR-56 spectrum could be compared with the modified AH-1G spectrum, these occurrences had to be converted into percentages of total time. To achieve this conversion, appropriate elapsed time in seconds was assigned to each maneuver condition to be changed; see Table III. Using these elapsed maneuver times, the occurrences were converted into total time allotted per 100 hours. These times were then converted into percentage of total time on a 100-hour basis and incorporated into the AR-56 spectrum with corresponding reductions made in the appropriate steady-state condition times so that the summed frequency of occurrence would total 100 percent flight time. Two other adjustments were also made. The 115 percent V_H forward level flight is considered incorrect by the definition of the term V_H . Considering V_H to be the forward level-flight airspeed at rated engine power, any airspeed greater than V_H would by necessity be in a dive attitude. Therefore, the time allowed to 115 percent V_H forward level flight was added to the gunnery dive condition. Also, the ground-air-ground cycles were omitted from consideration. They are of a low-cycle nature and are generally accounted for by fatigue test methods and/or analysis during fatigue life substantiation.

Table IV shows the resulting AR-56 spectrum. Two areas of comparison are available between the AR-56 spectrum and the operational AH-1G spectrum shown in Table I. These are distribution of flight time into the four mission segments, and the distribution of forward level-flight time between the various airspeed ranges. Figure 10 shows the mission segment distribution. The obvious difference seen in this distribution is the time allocated to steady-state and maneuver segments. The AR-56 spectrum shows more time in steady state and less time in maneuvers than does the operational AH-1G spectrum. The AR-56 distribution is similar to the original AH-1G spectrum.

TABLE II. ORIGINAL AR-56 SPECTRUM (FROM REFERENCE 10)	
Condition	Percentage of Flight Time
On ground	1.0
Takeoff	(400)
Steady hovering	5.0
Turns hovering	(400)
Control reversals hovering	(400)
Sideward flight	1.0
Rearward flight	0.5
Landing approach	(500)
Forward level flight:	
20% VH	2.5
40% VH	4.0
50% VH	4.0
60% VH	8.0
70% VH	8.0
80% VH	15.0
90% VH	15.0
VH	15.0
115% VH	1.0
Takeoff power climb	1.0
Full power climb	3.0
Partial power descents	(500)
Power dives	1.0
Right turns	3.5
Left turns	3.5
Control reversals	(2000)

TABLE II - Continued	
Condition	Percentage of Flight Time
Pullups	(500)
Power to autorotation	(100)
Autorotation to power	(100)
Autorotation - steady	2.0
Autorotation - left turn	0.5
Autorotation - right turn	0.5
Autorotation - control reversals	0.3
Autorotation - landing	0.3
Autorotation - pullups	(100)
Ground air-ground cycles	(100)
Gunnery maneuvers:	
Hovering	0.1
Dives	1.5
Dive pull-outs	0.65
Turns	
Right	1.0
Left	1.0
S	0.15

TABLE III. ADJUSTMENTS TO AR-56 SPECTRUM				
Flight Condition	No. of Occurrences per 100 Hours	Elapsed Time (sec)	Time per 100 Hours (sec)	Percentage of Flight Time
Takeoff	400	8	3200	.89
Hover turns	400	8	3200	.89
Hover cont. rev.	400	3	1200	.34
Landing approach	500	15	7500	2.08
Partial power descent	500	15	7500	2.08
Control reversals	2000	3	6000	1.67
Pullups	500	7	3500	.97
Power to autorotation	100	2	200	.06
Autorotation to power	100	5	500	.14
Autorotation pullups	100	5	500	.14

TABLE IV. ADJUSTED AR-56 SPECTRUM	
Condition	Percentage of Flight Time
On ground	1.0
Takeoff	.89
Steady hovering	3.77
Turns hovering	.89
Control reversals hovering	.34
Sideward flight	1.0
Rearward flight	0.5
Landing approach	2.08
Forward level flight:	
20% VH	1.61
40% VH	1.92
50% VH	1.92
60% VH	8.0
70% VH	8.0
80% VH	12.36
90% VH	15.0
VH	15.0
Takeoff power climb	1.0
Full power climb	3.0
Partial power descents	2.08
Power dives	1.0
Right turns	3.5
Left turns	3.5
Control reversals	1.67

TABLE IV - Continued

Condition	Percentage of Flight Time
Pullups	.97
Power to autorotation	.06
Autorotation to power	.14
Autorotation - steady	1.66
Autorotation - left turn	0.5
Autorotation - right turn	0.5
Autorotation - control reversals	0.3
Autorotation - landing	0.3
Autorotation - pullups	.14
Gunnery Maneuvers:	
Hovering	0.1
Dives	2.5
Dive pull-outs	0.65
Turns	
Right	1.0
Left	1.0
S	0.15

Figure 11 shows the forward level-flight airspeed distributions for both spectrums. The AR-56 spectrum shows a larger amount of time at the higher airspeeds and less in the mid-range airspeeds. Again, this is similar to the original AH-1G spectrum.

In general, the AR-56 spectrum is much closer to the original AH-1G spectrum than to the operational spectrum. This is very likely a result of the Navy's using the frequency-of-occurrence spectrums for the UH-1E and AH-1J, which are very similar to the original AH-1G spectrum, as a basis for the preparation of the spectrum contained in AR-56.

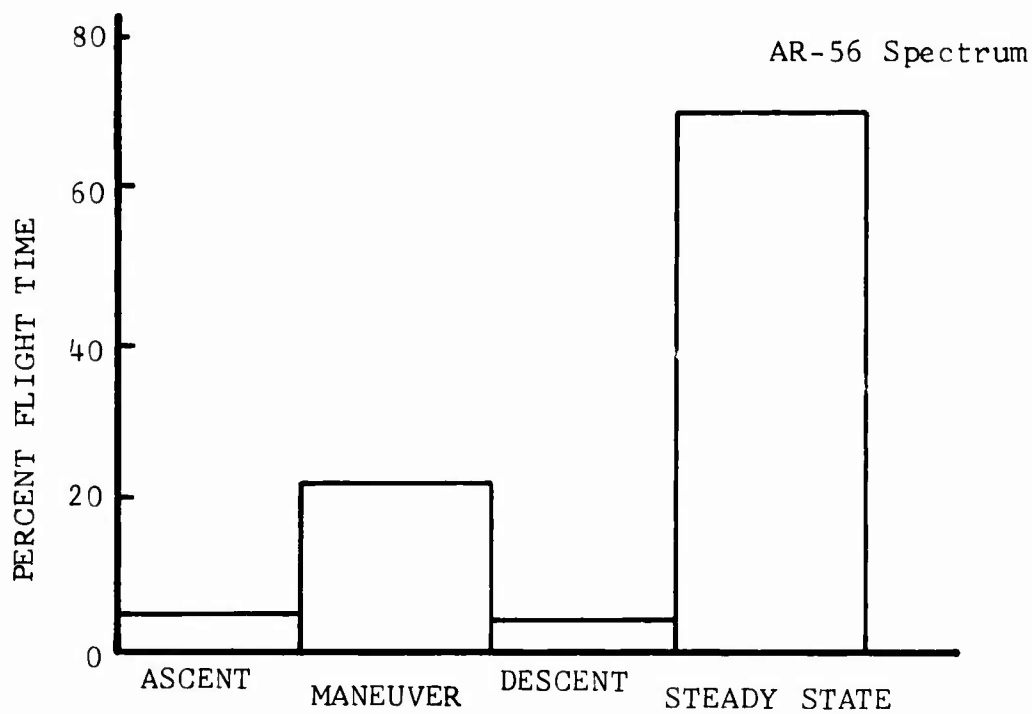
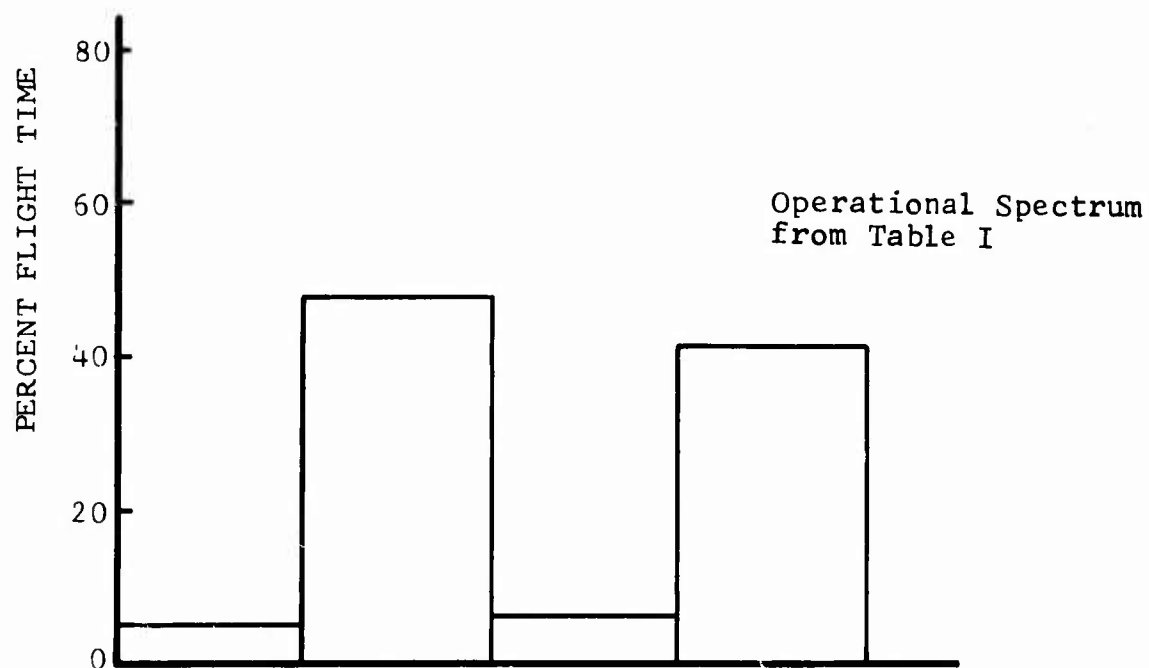


Figure 10. Comparison of Mission Segment Distribution of the Operational Spectrum and the AR-56 Spectrum.

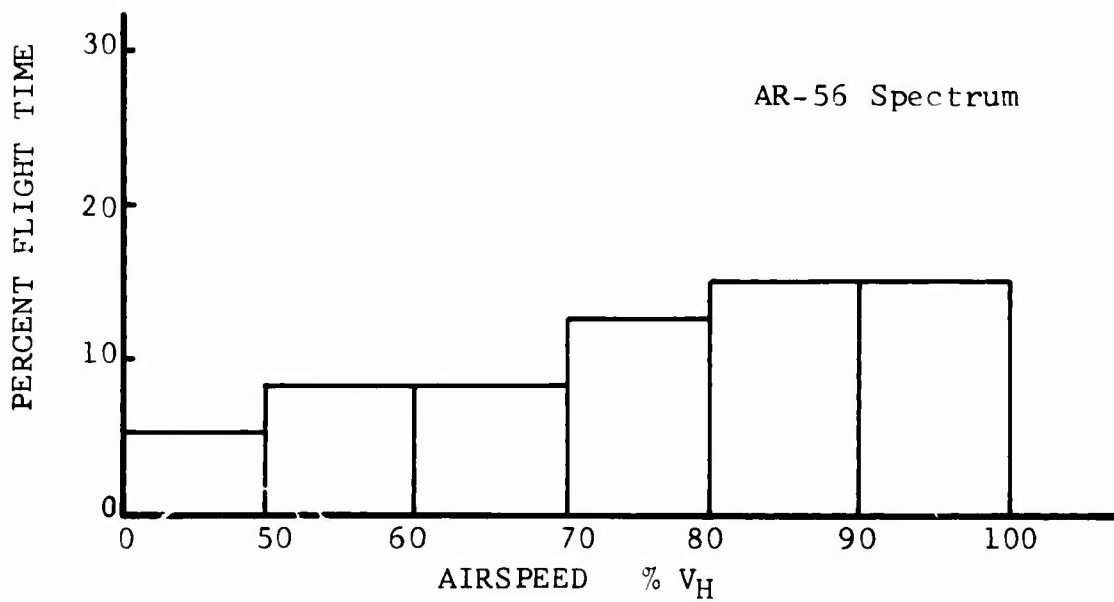
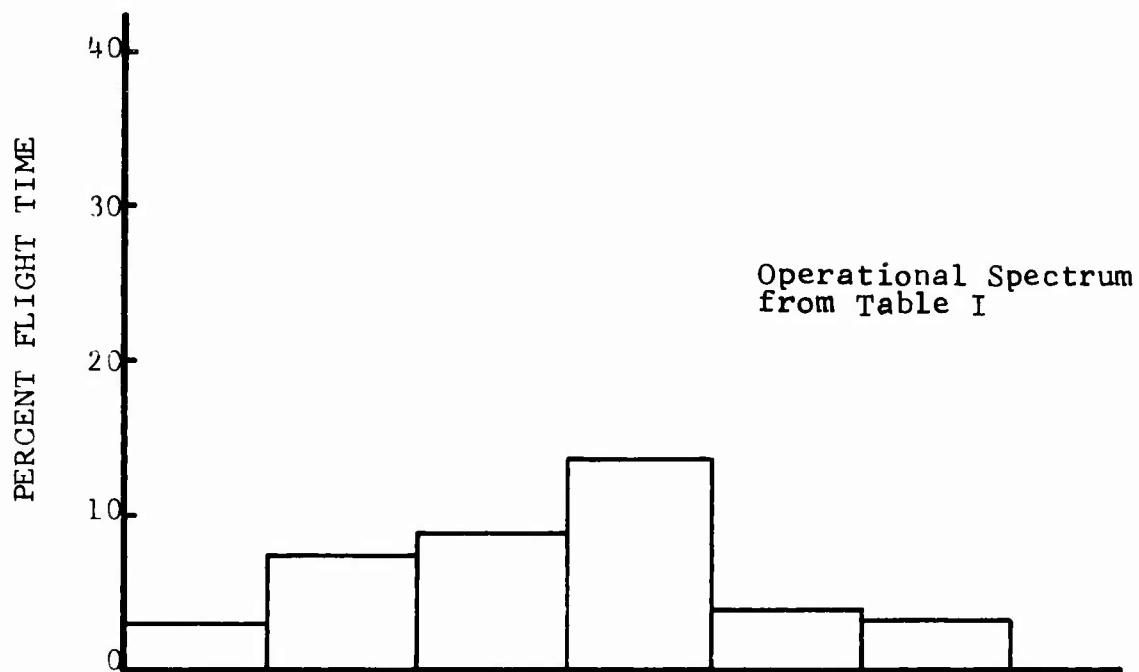


Figure 11. Comparison of Forward Level-Flight Airspeed Distribution for the Operational Spectrum and AR-56 Spectrum.

FATIGUE LIFE CALCULATIONS

Using the operational frequency-of-occurrence spectrum, fatigue lives were calculated for the following AH-1G dynamic components: main rotor blade, main rotor grip, main rotor yoke extension, swashplate outer ring, tail rotor blade, and tail rotor grip. These components were chosen since they are representative of the component categories of main rotor hub and blade, tail rotor hub and blade, and main rotor controls. Also, there are parts of both ferrous and nonferrous materials within this group. For these reasons, any significant effects caused by the spectrum modification should be evident in the fatigue life calculations. The fatigue life calculations shown in the Appendix are summarized in Table V.

Table V also lists the fatigue lives based on the original frequency-of-occurrence spectrum. The effects of the spectrum modification were neither totally detrimental nor totally advantageous. The component lives which showed the most change were those with only a few damaging conditions such as the main rotor yoke extension and the swashplate outer ring. The other component lives showed very little change due to the spectrum modification.

Although there were some relatively large modifications in some areas of the spectrum, the net effect on the resulting fatigue life was not significant. This is attributed to the fact that even though some of the individual changes seemed rather drastic, they tended to compensate for each other. The change in mission segments, wherein the time spent in level flight was reduced and the time spent in maneuver was increased, would lead one to expect a reduction in fatigue life. However, this was apparently compensated for by the reduction in severity of the airspeed distribution, since the oscillatory loads are strongly dependent upon airspeed. The resulting differences in fatigue lives do not appear to be sufficient to warrant a change in recommended retirement intervals.

TABLE V. SUMMARY OF AH-1G FATIGUE LIVES		
Component	Original Spectrum (hour)	Operational Spectrum (hour)
Main rotor blade	2,792	2,476
Main rotor grip	95,057	45,454
Main rotor yoke extension	10,633	9,334
Swashplate outer ring	9,806	19,443
Tail rotor blade	3,764	4,827
Tail rotor grip	8,103	7,587

540 ROTOR SYSTEM DEVELOPMENT AND FATIGUE DESIGN METHODOLOGY

This section reviews the 540 rotor system development from inception to present use on the AH-1G helicopter. The fatigue design methods employed in this time period are reviewed, and where service experience indicated that a fatigue design objective was not met, the reason for the design shortcoming is discussed.

DEVELOPMENT CYCLE

A chronological listing of the significant events of the 540 rotor system design-development-use cycle is shown in Table VI. The objective of the initial design was to develop a rotor system which would give a smoother ride at high speeds, require less maintenance, and have a higher rotational inertia than the UH-1 rotor then in production. The rotational inertia requirement was met by using large tip weights in the blade. Since dynamic considerations in two-bladed semirigid rotor systems require that the first in-plane frequency be well above the rotor operating speed, the increased tip weights made it necessary to provide a corresponding increase in the in-plane stiffness of the rotor system. This was accomplished by using an increased chord blade and by employing a unique door-hinge concept in the feathering bearing region of the rotor hub. Compared with the more conventional spindle-housing bearing arrangement, the fore and aft placement of the feathering and nonfeathering elements of the door-hinge hub provide a more effective utilization of material in obtaining an in-plane stiff structural arrangement. To further stiffen the feathering bearing region, the spanwise spacing of the bearings was increased to approximately twice that used in previous designs. This increased spacing also served to reduce bearing loads and thus increase bearing life. Inboard of the hub feathering bearings, a wide flat-plate structure was used to provide high in-plane stiffness and vertical softness. This flexural plate element served to reduce vertical bending moments in this area by acting as a virtual hinge.

The reduced vibration levels projected for the design were confirmed in an experimental flight test program. In this program, the prototype 540 rotor system was flown on a Model 204B helicopter. Following an evaluation by the Army Test and Evaluation Command, the 540 rotor system was considered for application to the UH-1 series helicopters.

A production design for the UH-1B was initiated. This design retained the basic rotor system geometry of the prototype, and

TABLE VI. 540 ROTOR SYSTEM DESIGN-
DEVELOPMENT-USE CYCLE

1962	Initial design
1963	Experimental flight test UH-1B/1C design
1964	Evaluation by U.S. Army Test and Evaluation Command UH-1C production design Flight test and fatigue test
1965	Production delivery UH-1C AH-1G design
1966	-
1967	Production delivery AH-1G
1968	Initiate operational flight surveys
1969	Main rotor blade modification
1970	Complete operational flight surveys

changes were limited to those typically involved in going from experimental-type hardware to production fabrication. Following flight test and fatigue test, the first production delivery of a 540 rotor system on a UH-1B airframe, now denoted as the UH-1C, was made in June 1965.

In 1965, a program was initiated to develop a high-speed attack-type helicopter. This helicopter, denoted as the Model 209, would use the 540 rotor system. The helicopter features a slim fuselage design; and as a part of the drag reduction program, the stabilizer bar used with the UH-1C control system was discarded in favor of an electronic stabilization system. This resulted in some minor revisions to the components in the dynamic control system. Production delivery of the AH-1G (formerly the Model 209) started in May 1967. One fatigue-oriented structural modification of the rotor system was required subsequent to delivery. This involved a main rotor blade modification and is discussed in more detail later in this section.

Two operational flight surveys of helicopters using the 540 rotor system were conducted from 1968 to 1970. One of those, Reference 11, provided data on the UH-1C and AH-1G operations in Southeast Asia. The second, Reference 8, provides operational data on the AH-1G and is the source of information on which this report is based.

FATIGUE DESIGN METHODS

In the design of fatigue-loaded components, a relationship between the fatigue strength and the magnitude of the flight loads must be established. At different stages in the design-development cycle, the manner in which a fatigue analysis is performed will vary. During the design stages, fatigue strength data from previous component tests and calculated flight loads are used. During the development phase, components are fabricated and flown, flight loads are measured, and components are fatigue tested. At this time, flight loads and fatigue test data on the actual components are used to estimate the relationship between fatigue strength and flight loads. Finally, the component enters service and demonstrates through satisfactory operation or premature failures the true relationship between fatigue strength and flight loads. The common measure of this relationship is fatigue life.

In the preliminary design stage, three methods are commonly used to size components for fatigue loading. The first method is to calculate flight loads for each discrete flight condition, to estimate the frequency of occurrence for each condition, to estimate the component's endurance limit, and

to calculate a fatigue life for the part. Several problems arise when this approach is applied to a practical situation: (1) accuracy is poor in calculating oscillatory loads in maneuvers, (2) the frequency-of-occurrence estimate has to be made before adequate mission and performance data have been obtained, and (3) the calculation process is lengthy and must be repeated each time the structural configuration is changed. Since a considerable number of such changes are usually made during the design stages, this type of analysis can become inefficient and impractical.

A second, far less sophisticated method involves the use of an additional factor applied to the static design loads to ensure low working stresses under actual loading conditions. In this method, a static analysis only is performed and fatigue considerations are included by using a factor which in effect is the ratio of the ultimate or yield strength to the endurance limit. While this method is sometimes used in the analysis of secondary structure or in redundant airframe structure, it is not considered adequate for application to dynamic components.

The third method, developed from experience, utilizes dynamic loads for a single key flight condition as the basis for the fatigue evaluation. Prior to the initial 540 rotor system design effort, it had been shown that for a typical helicopter loading spectrum, a satisfactory fatigue life would be obtained for an aluminum part when the endurance limit of the part was at or just above the alternating stresses for maximum level-flight airspeed (V_H). For steel components, it was shown that the flatter shape of the S-N curve made it necessary to keep the endurance limit approximately 40 percent above the V_H level-flight stresses to obtain a satisfactory life. In this method, the need for accurate maneuver loads calculation is thus circumvented by the use of these empirical relationships. This method was used in the preliminary design of the 540 rotor system.

FATIGUE STRENGTH ALLOWABLES

When designing a new component, the fatigue strength is usually determined by using the results of prior fatigue tests of full-scale, geometrically-similar parts and material. In the case of the 540 rotor system, fatigue data of similar parts were available for most major components. The fatigue strength of the components was, therefore, predicted with a relatively high degree of confidence at the preliminary design level. The same fatigue strength allowables were used later for sizing the production-design rotor components.

Since fatigue tests of all critical rotor components were to be conducted prior to production delivery of the UH-1C, the philosophy used was to design for light weight and modify later as indicated by fatigue test results. With the exception of two joint areas, all components exhibited fatigue strengths equal to or greater than the predicted values used in design. It was only necessary to provide moderate local increases in some sections to obtain the desired fatigue strengths.

When the 540 rotor was considered for use on the Model 209 Cobra, the fatigue strengths established in the UH-1C tests were used in the fatigue analysis. As mentioned previously, the Model 209 dynamic control system was newly designed to eliminate the stabilizer bar. In the design stage, fatigue strengths of these new components were determined from prior fatigue tests conducted on parts of similar geometry and material. Subsequent testing of the new control system verified that the predicted component strengths were equal to or greater than the predicted values used in the design phase.

FATIGUE DESIGN LOADS

At the time of the preliminary design of the 540 rotor system, the available analytical tools for load calculations were far less sophisticated than those in use today. Oscillatory rotor loads for only the steady-state level flight could be calculated with reasonable accuracy. At this point in time, it was, therefore, customary to place more confidence in empirical methods to determine the steady-state level-flight fatigue design loads and to use the analytical methods to study the effects of design variations on rotor loads.

To establish the fatigue design loads for the 540 rotor in the preliminary design phase, flight loads measured in test programs of other two-bladed rotor systems were used. It was found that reasonable correlation of the oscillatory rotor loads of different helicopter models could be achieved when they were compared on a thrust-coefficient and advance-ratio basis. While this method, which considers only aerodynamic parameters, did not account for the differences in dynamic characteristics of the rotor systems, it was found that for the two-bladed, semirigid, teetering rotor family, these differences were small. This empirical method was used in establishing the V_H level-flight fatigue design loads for the initial 540 rotor design.

For the 540 production design, the V_H level-flight loads measured in the flight testing of the prototype rotor were used. Since the complete envelope of steady-state and

maneuver flight had been investigated in the flight test program, these design loads were used with a high degree of confidence.

By the time the 540 rotor was considered for the AH-1G, a considerable amount of testing and service experience with the UH-1C had been accumulated. While the airspeed of the AH-1G was to be higher than that of the UH-1C, loads calculations showed the main rotor fatigue loads for the V_H level-flight condition to be about the same, because the airspeed increase was made by a reduction in fuselage drag and download. The power available, a parameter which is strongly related to oscillatory rotor loads, was the same for both the AH-1G and the UH-1C.

EVALUATION OF FATIGUE DESIGN/ANALYSIS METHODS

The true fatigue life of helicopter dynamic components is proven only after the aircraft has been in service and has accumulated a significant number of hours operating under the various actual mission conditions. In the case of the 540 rotor, a considerable history of operation is available for this system on two different helicopter models: the UH-1C and the AH-1G. A review of this record shows that of the 25 fatigue-loaded components of the main rotor and control system, only one significant design modification was required due to fatigue problems which were encountered in service. This change involved a redesign of the main rotor blade which was incurring premature fatigue failures of the blade skin at approximately mid-span. While the failures were demonstrated to be of a fail-safe mode, it was felt that the problem could be eliminated by a relatively small design change which would be economically attractive. In evaluating the fatigue design methodology, it is pertinent to examine this main rotor blade failure to determine the cause and to identify any shortcomings in the analysis which may have prevented the original design from reaching the fatigue design goal.

The cause of the blade failures was traced to a high-frequency oscillatory load which was present over a rather narrow band of the level-flight airspeed envelope. A combination of blade frequency placement and airloads due to trailing tip vortices caused the blade to be excited at 6/rev at a level-flight airspeed of approximately 90 knots. Since the problem was associated with the blade passage frequency through the shed vortices, the excitation was airspeed-sensitive, and the resonant condition disappeared at speeds above or below the critical speed. Because of the relatively small size of this critical band, the significance of its presence and magnitude was not fully realized during the flight-test program. Subsequent analyses conducted to establish the retirement life of the blade showed the main blade retention joint rather than the mid-span station to be the most fatigue-critical area. It is interesting to note that this condition was present at an airspeed which was assigned a relatively small percentage of time in the original engineering frequency-of-occurrence spectrum. As shown in Figure 4, the larger percentages of time in level flight are assigned to the higher airspeeds. While this would usually be conservative, it was not the case in this particular instance. It will also be noted in Figure 4 that the mission profile data show a much larger percentage of time assigned to the 90-knot level-flight condition. While these data were not available at the time the problem was being investigated, they do, in retrospect, help to explain the shorter-than-predicted fatigue life.

The modifications in the blade redesign were directed toward eliminating the high-frequency resonance problem and providing moderate increase in structural capacity of the out-board blade section. This was accomplished by changing the taper of the trailing-edge strip to increase the chordwise section stiffness. In reviewing the fatigue design methods employed in the design of the original 540 rotor system, it is immediately obvious that the method does not anticipate problems of the type encountered in the 540 blade service failures. In the original design, level-flight loads from previously flown two-bladed rotor systems were used to establish fatigue design loads. While these loads were adjusted to account for the differences in aerodynamic parameters, it was assumed that the dynamic characteristics would be similar. This assumption proved reasonable for the lower frequency loadings but was not valid for the particular problem encountered with the high-frequency loads exhibited in the 540 blade.

With the development of more sophisticated rotor loads calculation programs, improvements in dynamic rotor loads prediction have been achieved for the lower frequency excitations. Although programs for predicting the presence and location of resonant frequencies for a particular rotor design are quite accurate, the ability to predict the magnitude of the higher frequency loads accurately is not currently within the state of the art.

Of the three primary variables used to establish rotor component fatigue lives in the preliminary design stage - (1) component fatigue strength, (2) calculated oscillatory loads, and (3) frequency of occurrence - the area of rotor loads calculation appears most deficient. The estimation of component fatigue strength can be made with a relatively high degree of confidence due to the large and ever-growing data base established by full-scale component fatigue testing. The frequency-of-occurrence spectrum takes on major significance in the preliminary design phase only after the ability to calculate accurate maneuver loads is achieved.

Later in the development cycle, when fatigue testing has been completed and measured flight loads are available, the frequency-of-occurrence spectrum becomes the key variable in establishing fatigue lives. It is important, therefore, that the collection of mission profile data be continued so that the most realistic frequency-of-occurrence information can be used in establishing component retirement intervals.

In recent years, the rapid advances in computer technology have been nearly paralleled by rotor loads prediction capability. Refinements, particularly in the calculation of

oscillatory loads in maneuver conditions, now make it practical to compute fatigue loads of the total flight spectrum. This method is still time consuming, however, and the empirical fatigue design methods are more economical and convenient to use in the early stages of design iteration. When the design is reasonably firm, the fatigue life is calculated using calculated oscillatory loads for each maneuver condition of the total flight spectrum. The empirical methods then provide a check of this calculation. For this reason, a study was conducted in 1967 to review and update the empirical method which was then in use. The method which evolved from this study, Reference 12, utilizes the previous approach of designing with V_H level-flight stress and the component endurance limit to meet a given fatigue life objective. Refinements were made to consider various materials and the differences in sensitivity of the main rotor, tail rotor, and rotating controls to the maneuver loading spectrum. Also, additional flexibility was added to the method by incorporating a spectrum severity index factor which is dependent on the helicopter mission.

MAXIMUM ONE-TIME OCCURRENCES AND LIMITATIONS

The purpose of the work in this section is to compare the values of maximum one-time occurrences measured in Reference 8 with those specified in the structural design criteria, measured in the helicopter flight structural demonstration, or contained in the aircraft operating limitations. A study of the various parameters was made in an attempt to determine what limiting factor caused each maximum one-time value.

COMPARISON OF ONE-TIME VALUES

The maximum one-time values recorded in the Southeast Asia survey for airspeed, normal acceleration, rotor speed, engine torque, and gross weight are shown in Table VII. These are compared with values contained in the aircraft operating limitations, specified in the structural design criteria, and measured in the flight structural demonstration.

A maximum airspeed of 185 knots was measured in the Southeast Asia data. This occurred at an altitude of 5000 feet and at a gross weight of 7546 pounds as the aircraft was pulling out of a descent at 5000 feet per minute. In this altitude range, 3000 feet to 6000 feet, the aircraft is redlined at 174 knots (approximately 1.20 times V_H). The pilot was, therefore, exceeding the operating limits of the aircraft. The airspeed was still well below the 222-knot speed used for structural design and the 210-knot dive recorded in the structural demonstration flight test.

The maximum operational rotor speed of 351 rpm, which occurred in autorotation, was also above the aircraft redline value of 339 rpm. This value is lower than the 356 rpm specified in the structural design criteria but higher than the 342 rpm recorded during the structural demonstration. In establishing the 356 rpm as a design limit in the structural design criteria, a deviation from the MIL-S-8609 specification factor of 1.25 on rotor speed was granted. The factor of 1.05, used for the AH-1G and previous designs of the UH-1 series, appears to be realistic for rotor overspeed considerations for the attack helicopter. Although no factor on the low range of rotor speed was included in the structural design criteria, a 1.05 factor was used in the structural analysis of the rotor, Reference 16, to establish a minimum design limit rotor speed of 280 rpm. No discrete value for extreme minimum rotor speed was reported in Reference 8, only that it was less than 295 rpm. The aircraft operating limit is 294 rpm minimum.

No discrete value for maximum engine torque was given in Reference 8. The maximum value recorded was greater than 60 psi and less than 70 psi. This is in excess of the aircraft

TABLE VII. COMPARISON OF AH-1G MAXIMUM ONE-TIME OCCURRENCES

Parameter	Aircraft Operating Limitations (Ref. 13)	Structural Design Criteria (Ref. 14)	Structural Demonstration Flight Test (Ref. 15)	Southeast Asia Operational Flight Survey (Ref. 8)
Airspeed (kt)	SL-3000 ft, 190 6000 ft, 174	222	210	185
Rotor speed (rpm)	Power on, 294-324 Power off, 294-339	Power on, 294-356 Power off, 294-356	284-342	351
Engine torque (psi)	50	62.5	59	>60
Gross weight (lb)	9500	6600 9500	7500 9400	9522
Normal acceleration (g)	Flight at or near zero g prohibited	3.50 2.43 -.50 -.50	2.73 2.10	2.4 0.1

redline of 50 psi and may be greater than the 62.5 psi used in the structural design criteria. The maximum value recorded in structural demonstration maneuvers was 59 psi.

The gross weight of 9522 pounds reported in the Southeast Asia survey is slightly above the aircraft operating limitation of 9500 pounds. A maximum gross weight of 9500 pounds is specified in the structural design criteria. The structural flight demonstration was conducted at a maximum gross weight of 9400 pounds. The extreme values of normal acceleration reported in Reference 2 were +2.4g's and +0.1g. The aircraft operating limits do not specify any maximum g limitation, and there is no g-meter in the helicopter instrumentation. The Operator's Manual does specify that aerobatic maneuvers and flight at or below 0g are prohibited. The normal acceleration specified in the structural design criteria ranges from +3.50g's to -.50g. In the structural demonstration flight, a maximum value of 2.75g's was measured.

OTHER ONE-TIME MEASURED VALUES

In addition to the maximum one-time values of the parameters listed in Table VII, Reference 8 presents measured values for longitudinal and lateral accelerations, collective and cyclic stick positions, density altitude, and outside air temperature. Although limits for these parameters are not listed in the operating limits or structural design criteria, a brief discussion of each parameter is presented.

The maximum lateral acceleration was .45g left and right. The maximum longitudinal acceleration was .25g forward and .30g aft. No lateral or longitudinal accelerations are specified in the design criteria. In putting the helicopter in equilibrium for the various flight conditions, maximum accelerations of 1.33g lateral and .37g longitudinal were determined. These accelerations were used for structural analysis of the airframe.

The extreme forward longitudinal cyclic stick was in the 0 to 10 percent increment. The extreme aft position was in the 70 percent to 80 percent increment. The extremes recorded for collective stick were in the 0 to 10 percent increment for the most down position and in the 70 percent to 80 percent increment for the most up position.

The maximum density altitude measured was above 10,000 feet and less than 15,000 feet. The outside air temperature ranged from a minimum of 40°F to a maximum of 100°F.

PROBABLE CAUSE OF LIMITING FACTORS

To determine the probable cause of the limits reached in Reference 8, discussions were conducted with people having expertise in various helicopter design disciplines, with test pilots, and with pilots with AH-1G combat experience. Additionally, pertinent literature was surveyed. Limits of the vehicle as seen from the perspective of design, test, and operational personnel provided the composite view necessary to make a balanced evaluation.

These discussions and a review of pertinent literature brought out a point which must be considered. Although the two data samples are an important contribution to the literature and will be used at face value, there is considerable evidence that the pilots flew by the book since they knew they were being monitored. An unbiased sample, if such were possible, would be of even greater value. Since the present report compiles best guesses as to the reasons for the maximum one-time occurrences, comments which indicate data biasing seem important to the explanation and will be included.

Difficulties were encountered in some cases due to the limited presentation of data in Reference 8. When evaluating the cause for the limit of a particular parameter, it would be desirable to have the instantaneous value of all other measured parameters, or even better, a short time history of the event such as presented in Reference 17. This would aid in reconstructing the condition and thus give more credibility to the argument for the cause.

AIRSPEED

The maximum airspeed of 185 knots was measured at 5300 feet altitude with the aircraft at 7546 pounds gross weight pulling out of a 5000-feet-per-minute descent. This maneuver was not reported as a gunnery run. This odd point can be put in perspective by noting a couple of statistics: (a) less than one minute in 202 hours (<.008%) was spent above 170 knots, and (b) only 60 minutes (.5%) was spent above 140 knots. From a gross perspective, the airspeed data peaked in the 80- to 100-knot range, which is considerably below V_H --140 to 150 knots for clean configuration and 130 to 140 knots for HOG (19 tube rocket pods inboard and outboard) configuration.

Vibration and handling qualities are good enough to permit the AH-1G to spend a high percentage of time at V_H ; thus the peaking of the speed spectrum at a lower value-- $0.5V_H$ to $0.75V_H$ --is due to factors other than flight characteristics, probably theater tactics.

Further, the AH-1G can extend its speed from the 140-150 knot level-flight range to the redline speed of 190 knots by diving. The data show that high-speed dives were not frequently made. Reasons for this probably include both ship's characteristics and theater tactics.

High-speed dive characteristics vary from mild to severe depending on how the maneuver is executed. Consider the following excerpt which contains most of what is presented in the Operator's Manual, Reference 13.

"Diving flight presents no particular problems in the AH-1G; however, the pilot should have a good understanding of such things as rate of descent versus airspeed, rate of closure, and rates of descent versus power. Because of relatively low drag, the aircraft gains airspeed quite rapidly in a dive, and it is fairly easy to exceed the redline. Rates of descent of 3500 feet per minute to 4800 feet per minute at full power are not uncommon during high speed dives. These high rates of descent coupled with the high flight path speeds (320 feet per second at 190 knots true airspeed) require that the pilot monitor both rate of closure and terrain features very closely and plan his dive recovery in time to avoid having to make an abrupt recovery. If an abrupt recovery is attempted at airspeeds near redline airspeed, "mushing" of the aircraft can occur. If mushing is experienced, do not increase collective. Application of increased collective will aggravate the condition.

At speeds above the maximum level flight speed, the rate of descent will increase approximately 1000 feet per minute for every 10 knots increase in airspeed for the full power condition. At redline airspeed, the rate of descent will not change appreciably for any torque pressure between 40 and 50 psi."

The excerpt above does not tell all there is to know about a dive and recovery. There is a problem in determining how much information to put in the manual and how much to leave to flight training. Characterizing a dive is difficult since there are so many variables involved: entry speed, power, and altitude; dive angle and power; recovery method and constraints; etc. Most of the limitations are on the recovery, not on the dive itself.

Flight characteristics which deter extreme dives are steep dive angles, reaction time for recovery from engine failure, closure speed, vibration, and recovery constraints. For

example, there is little deterrence to a steep dive from 7000 to 3500 feet ground height, but there is extreme deterrence to the same maneuver executed between 4500 and 1000 feet. Vibration is not the principal or even a serious deterrent, although it is a consideration and deserves some discussion. The main fuselage vibrations are at frequencies which are multiples of rotor speed, namely, two-, four-, and six-per-rev. Since lateral vibration is never a problem, only the vertical will be discussed. Cabin vibration comfort at V_H is good but degrades with increasing dive speed, reaching the following maximum values near 180 knots: 2/rev, $\pm 0.3g$; 4/rev, $\pm 0.5g$; 6/rev, $\pm 0.5g$ (Reference 18). These are not acceptable levels for steady-state flight, but they are not severe enough to seriously deter high-speed transients which last only a few seconds in a regime where pilots expect high vibration; however, there is a learning curve involved. Pilots accept increased vibration in transients such as high g turns and landing flares if they repeat the maneuver often enough to learn what is normal. A pilot who does not regularly execute high-speed dives will have trouble learning the normal vibration characteristics of the AH-1G since the harmonic content varies with several parameters: loading configuration (gross weight, cg, and stores), airspeed, density altitude, power, and rpm. Additionally, the pilot and gunner will frequently disagree on the effect of a given parameter since the variation in a given harmonic at the two seats is often opposite. These variations are readily explainable in terms of the response mode shapes; however, no conscious effort has been made to teach this pattern to AH-1G flight crews.

The dive recovery varies from routine to severe depending on dive speed and angle, and recovery constraints such as altitude, obstacles, air traffic, and enemy fire. If it is assumed that the high-speed dive is an indispensable tactic, that it will be used frequently, and that recoveries will have to be made under the most extreme conditions, then there is no substitute for rigorous instruction and drill in making high-speed dives and recoveries during flight training. In the field, practice should be encouraged to continue until each pilot knows how to manage all of the parameters in all of their combinations. It is not known to the authors of this report what the Vietnam pilots know and believe about the real limits of high-speed recoveries and how they developed their knowledge, skill, and feelings. The results of querying one Vietnam pilot are included in this discussion. This is a very small sample, and the querying method was not at all sophisticated; yet the perspective and insight gained are very valuable. It is beyond the scope of this study to develop a sophisticated questionnaire for a larger sample of pilots who flew the AH-1G in Vietnam; however, this is a task

which should be done as quickly as possible before these pilots are dispersed or their recollections dimmed.

Such information from a larger sample could add an important dimension to the probing for the real limits and desirable characteristics of new-generation armed helicopters. The results of this larger poll would answer the question at hand concerning the maximum dive speed and the low frequency of occurrence of speeds in the 160- to 190-knot range reported in Reference 8. Was the frequency of occurrence due to (a) the fact that it was not a useful tactic, (b) real or imagined physical abuse of the helicopter, (c) degradation of the gun platform, or (d) physical discomfort, unmanageable workload, or apprehension during dive recovery? Factors affecting the smartness of dive recovery are enumerated in the following paragraphs.

The following phenomena tend to limit a symmetrical pullup:

- (a) Overspeeding the rotor. Application of aft cyclic tends to put the rotor in autorotation, cones the rotor, and washes out collective pitch via pitch-cone coupling. Both autorotation and pitch reduction tend to overspeed the rotor; thus cyclic and collective rate and magnitude must be coordinated to limit rotor speed.
- (b) Rapid buildup in pitch rate and attitude causing loss of visual reference, which is the primary cue for exiting the maneuver via a pushover or pedal turn. If the pull-up is held too long, the only coordinated maneuver for recovery is a loop. This has reportedly been accomplished, but it is not recommended since failure to follow through can have disastrous consequences. Uncoordinated exits are not recommended because not enough is known about them; especially worrisome is the potential for exceeding the flapping clearance and striking the flapping stops.
- (c) Unloading the rotor. The pushover which may follow the pullup tends to unload the rotor and decrease control power. Flight at or below 0g is prohibited since, with SCAS off, a divergent right roll can develop which cannot be controlled by the instinctive left cyclic reaction.
- (d) Mushing of the rotor. This is caused by stall or pitch-cone coupling, or both, as discussed in the Operator's Manual, Reference 13. The Vietnam pilot reported this (and the accompanying loss of altitude) as the primary deterrent to a severe pullup. Bell pilots maintain that mushing can be mitigated by relying more heavily on the cyclic flare for building the initial normal load factor.

A coordinated application of cyclic and collective is then used to control rpm and build additional load factor.

- (e) Boost feedback. Cyclic or collective feedback, or both, can occur in high-g maneuvers. The symmetrical pullups of the structural demonstration were limited by collective feedback, while cyclic feedback limited similar maneuvers in Reference 14.

Both cyclic and collective feedback are due to high oscillatory loads with a rational pattern which is explainable but not yet predictable by analysis. There is some controversy as to whether the conditions for occurrence are repeatable. The harmonic content of the loads and the transfer pattern from the rotating to the fixed control system are rationally related to the blade frequency diagram and the transfer trigonometry. The airloads are less understood but are most likely a growth in regular oscillatory loads plus impulsive stall loads and unsteady airloads from vorticity. The rotor modes involved are the first inplane and the first and second out-of-plane asymmetrical modes. Torsional modes are not involved; thus there is no reason to think of this as stall flutter.

There is an abundance of flight test data for constructing a clear and rational picture of the sequence of events during feedback for current design guidance and future analytical guidance; however, this has not been deemed necessary to date for the following reason. During development flight testing, the boost capability is increased to a point where boost feedback does not occur so early as to seriously restrict the flight envelope nor so late as to allow an unannounced buildup of dangerous rotor oscillatory loads. The pilots have a very strong feeling about this iteration. They rely on boost feedback as the primary warning that the rotor is being abused, and it is their signal to back off. An increase in vibration usually accompanies the load buildup on current helicopters, but this will be less true in the future since more effective isolation systems are coming into use. Because of this and the fact that the vibration in the stick commands more attention than vibration in the structure, pilots do not want to do away with boost feedback.

The symmetrical pullup followed by a pedal turn is reported to be easier to execute and less taxing on the helicopter than the pullup and pushover. An important factor is that the pilot does not lose visual reference. The rolling pullup also presents the pilot with an easier work load than the symmetrical pullup and pushover. Loss of visual reference, overspeed, mushing, and Og are mitigated or absent. The

limiting phenomena will likely be boost feedback, vibration, or overtorquing. In the case of a left roll, a 10- to 15-psi torque surge can occur.

From the foregoing, it is concluded that the maximum speed of 185 knots at 7546 pounds was probably limited by vibration in the dive and vibration and boost feedback in the recovery. However, the infrequent use of the high-speed level-flight capability of the AH-1G must be explained in terms of local requirements.

- Low requirement for long, high-speed escort missions
- Infrequent need for long, high-speed dashes from base to scene of action
- Difficulty of locating and attacking mobile enemy troops in protective ground cover when flying at high speed
- Reduced visibility and sighting time because of required pullout height and rapid closure rate
- Absence of concentrated, sophisticated antiaircraft installations, etc.

ROTOR RPM

The rotor rpm distributions show that more than 95 percent of the time in both data samples was flown between 310 and 325 rpm with the normal steady value being between 315 and 318. It appears that rotor rpm was controlled within limits, with very few points falling above 330 rpm. A very small amount of time was recorded in the 340- to 355-rpm range during the maneuver segment. All such recording occurred during descents. The highest value of 351 rpm was recorded during a descent which appeared to approach an autorotation.

During design and development testing, a great deal of attention is paid to rotor rpm since it optimizes and sizes so many details of the helicopter: rotor and engine performance; shafting and bearings; rotor strength and fatigue life; natural frequencies of the rotor, pylon, fuselage, and drive-shafts; rotor noise; etc. The final choice of rpm range is frequently a compromise made during development testing. The band adopted for the AH-1G and listed in the Operator's Manual is 294 to 324 rpm. Not all rpm's in this band are equally desirable; 324 is considered the normal, or preferred, rpm. It is interesting to note that the data peak was in the 315- to 318-rpm band. The Vietnam pilot queried said it was commonly believed that operating at less than design maximum was

less abusive of the helicopter. The AH-1G is not overly sensitive to this rpm spread; however, this is not always the case. Often rotor loads and vibration are quite sensitive to such a spread, being worse at lower rpm. Possibly the Operator's Manual should state the preferred rpm and list the reasons. More information about why 315 to 318 rpm was used plus a finer breakout of rpm would be a useful contribution to the study.

The good rpm control shown by the data suggests that it was a simple task, while several other sources suggest that rpm control is difficult. Consider the following excerpt from Reference 20.

"The maximum angle-of-attack capability of the AH-1G during pullups is severely restricted by a rapid buildup of rotor rpm. Pilot workload to control rpm is frequently excessive. The power-off upper rotor speed limit (339 rpm) is marked with a red line on the aircraft instruments and is interpreted by the pilot as a not-to-exceed rotor speed. The 339-rpm limit allows less than a 5-percent overspeed from the normal operating value of 324. This small margin is considered a design shortcoming. The military specification for structural design requirements for helicopters requires a 25-percent margin between design maximum and power-on limit rotor speed. A minimum margin of 10-percent between design maximum and limit operating rpm should be specified for Army helicopters."

It is important to know if the pilots found it easy or difficult to control rpm. Did they bias the data? Again, a wider poll of Vietnam pilots would help to answer this question. If the control evidenced by the data was relatively easy, then the 5-percent overspeed margin used for the AH-1G is validated by the data.

Concerning the reason for the maximum one-time occurrence and the neighboring values, they all occurred during descents. This is the most likely condition for such overspeeds, especially if the descent started at, say, 5000 feet and terminated in a cyclic pullup, with primary attention of the pilot being on the target.

NORMAL ACCELERATION

A maximum normal acceleration of 2.4g's was measured several times in both samples of data. This maximum was most probably encountered in a turn. At rotor thrusts of 16,500 to 18,000 pounds, control feedback forces are encountered which provide

a warning to the pilot. The BHC Model 540 rotor hub and control geometry also provides a pitch-cone coupling which tends to reduce blade pitch when the rotor cones upward. This has a limiting influence on the load factor which is developed in maneuvering flight. The extreme minimum normal acceleration of 0.1g probably occurred during a pushover. Since there is no g-meter in the helicopter, the pilot must rely on instinct. The helicopter pilot's tolerance to g levels in this range is generally quite low. This tolerance, plus an awareness of the warning in the Operator's Manual, could provide cause for terminating this maneuver.

TORQUE

The torque distributions for the two samples are rational in themselves and compared to each other. The aircraft were probably taking off fully loaded and flying at about the same height above the ground. The higher temperatures of the Sample II data provided high density altitudes, requiring higher power and higher torques than Sample I.

The ratio of the percentage of time above 80°F of Sample II to Sample I was 1.25, and the ratio of percentage of time above 40 psi was 1.35. There was not much chance for over-torquing since a relatively small percentage of time was spent above 40 psi - 12.6 percent for Sample I and 16.1 percent for Sample II. The data indicate that the aircraft were operating within the constraint of available power, which was generally below the torque limit. If greater power had been available, overtorquing might have been more frequent.

Some valuable information which could influence the rationale for the transmission restriction could be derived from the two data samples. This would require a finer breakdown and clearer explanation in terms of the oscillograph trace, and it would be instructive to compare the finer histograms with the oscillograph traces themselves. The limited number of trace samples contained in Reference 17 show that the torque pressure has a complex wave form reflecting the varying torque demand experienced during maneuvering under combat conditions.

The data presented do not produce a strong physical picture of events that were going on due to transient excitation of the first torsional mode, energy maneuverability, power increase in dives (Reference 20), left rolls (Reference 19), etc. Further, it would be useful to know how well the pilot's torque meter followed the oscillograph galvanometer.

The maximum engine torque of greater than 60 psi occurred under conditions of 0 to 1000 feet altitude, 0 to 40 knots airspeed, 8000 to 9000 pounds gross weight, 0 to 300 feet-

per-minute rate of climb, 310 to 325 rotor rpm, and an OAT of 60° to 70°F. The low altitude and airspeed indicate that the overtorque occurred during takeoff. The most probable event was the requirement to take off over an obstacle under conditions requiring high power. The extremes of the above parameter ranges - 1000 feet, 0 knots, 9000 pounds, 300 feet per minute, 310 rpm, 60°F - would be more than sufficient as a condition.

GROSS WEIGHT

The maximum recorded gross weight of 9522 pounds indicates that the pilots were paying close attention to the operating limit of 9500 pounds. This was encouraged by requiring the pilots to fill out supplemental information sheets indicating the fuel and armament loading at takeoff. It is probable that, had this not been required, the helicopters would have been flown more frequently at the maximum overload weight at which they could get airborne. The Vietnam pilot and others who observed Vietnam helicopter operations report that the AH-1G's frequently took off with the maximum liftable weight. Because of the temperature's effect on power, this would constitute an overload only in the early morning; thus, there was little opportunity to seriously overload the AH-1G during the Vietnam operation. However, this could be a problem in a cooler climate.

Under combat conditions, taking off with the maximum liftable load would be a very natural thing to do, and whether this is right or not would have to be judged for each case. For example, a field commander will probably not concern himself with meeting established life and overhaul schedules if the intensity of battle is such that the probable survival life is much shorter than the overhaul life. Under these conditions, resourceful pilots will experiment with overload configurations which meet an immediate need. If no obvious limitation or problem results, the configuration will be adopted. This is not necessarily bad, and it may or may not lead to a problem later. However, such practices should be reported to cognizant Army technical agencies and the contractor for evaluation and information. This situation should be provided for in the maintainability/reliability program and the design/development loop for future helicopters; for existing helicopters, the safety of flight release via the contractor should be more conscientiously applied. The foregoing again points out the need for a wider poll of Vietnam veterans to assess bias of the data.

CONSIDERATIONS FOR CHANGES IN STRUCTURAL DESIGN CRITERIA

Based only on the results of the values of maximum one-time occurrences measured and reported in Reference 8, no changes in the structural design criteria used for the AH-1G can be recommended. Only the gross weight was shown to have exceeded the value specified in the structural design criteria, and this was by an insignificant amount.

Although the 1.05 factor was shown to adequately cover rotor overspeed conditions for the 540 rotor system, the advisability of recommending a change in the structural design criteria to reduce the design limit rotor speed factor from 1.25 to some lesser value could be questioned. As mentioned earlier, one of the initial design objectives in the 540 rotor system was to develop a rotor with high rotational inertia to provide increased safety in emergency autorotational landings. A higher inertia system accelerates less in a cyclic flare and gives the pilot more time to respond and apply corrective control in case of rotor overspeed. This suggests that some flexibility in specifying rotor overspeed factors could be incorporated in future design criteria to account for differences in rotational inertia. For the 540 rotor system, a value of 1.10 would appear to be a realistic factor which still contained a reasonable degree of conservatism.

A study comparing maximum measured rotor speeds from operational surveys of other types and models of helicopters would provide information to establish the validity of some rotor inertia/rotor overspeed relationship.

PREDICTION OF LIMITING PARAMETERS

Because of the large number of parameters and the complex coupling and interactions involved, it would appear that there is no simple method for predicting maximum one-time occurrences. The development of a comprehensive mathematical model capable of handling all of the known significant parameters with provisions for "flying in the computer" would appear to be the best approach. With this tool, the vehicle limits could be explored and maximum values could be determined for specific parameters.

In addition to the limits established for the vehicle, pilot-imposed limits should be considered. The interface of vehicle and pilot could be handled in several different ways. While fixed- or moving-base simulators could be used for pilot input and response, this degree of sophistication might not be required. Since human factors information is available in the form of tolerance to vibration level, pitch rates, control input strength, attitude, etc., these limits could be super-

imposed on plots of vehicle limits to determine the probable maximum values for the parameters of interest. The super-imposed plots of vehicle and pilot limits would then indicate areas where auxiliary systems such as warning devices or control rate limiters should be provided to ensure safe operation of the helicopter.

CONCLUSIONS

Based on the results of this study, it is concluded that:

1. Comparison of the original engineering frequency-of-occurrence spectrum and the Southeast Asia mission profile data showed that the AH-1G was being operated at higher gross weights and lower airspeeds for larger percentages of time than originally estimated.
2. These differences in use had compensating effects which caused the fatigue lives calculated by the original engineering spectrum and the Southeast Asia spectrum to be nearly the same.
3. The attack helicopter spectrum shown in Navy AR-56 is very similar to the original AH-1G engineering frequency-of-occurrence spectrum.
4. Of the maximum one-time values of load-sensitive parameters measured in the Southeast Asia operation survey, only the gross weight was shown to have exceeded the value specified in the structural design criteria, and this was by an insignificant amount.

RECOMMENDATIONS

The following recommendations are based upon the study presented in this report.

1. In presenting data from future operational surveys, an effort should be made to establish histograms with smaller class intervals.
2. The presentation of maximum one-time occurrence data should be in the form of a short-time history of all parameters measured during the event.
3. Consideration should be given to reducing the factor used to establish design limit rotor speed from 1.25 to 1.10 for high-inertia rotor systems.
4. Work in the area of developing analytical methods for predicting helicopter operating limits and loads associated with these limits should be encouraged.
5. A questionnaire regarding vehicle use and limits should be developed to quiz pilots who flew the AH-1G in Vietnam.

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APPENDIX
FATIGUE LIFE DETERMINATION

TABLE VIII. 540-011-250-1 MAIN ROTOR BLADE
FATIGUE LIFE DETERMINATION

FLIGHT CONDITION	FREQUENCY OF PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
I. GROUND CONDITIONS				
A. NORMAL START	0.4000	7656	0 FA	0.0
B. SHUTDOWN W/COLL.	0.4000	7656	0 FA	0.0
II. ICE MANEUVERS				
A. TAKE-OFF				
1. NORMAL	0.0511	978	529 AA	0.0
	0.7668	14677	363 BA	0.0
	0.4601	8806	324 CA	0.0
2. JUMP	0.0057	109	299 AA	0.0
	0.0852	1631	391 BA	0.0
	0.0511	978	397 CA	0.0
B. HOVERING				
1. STEADY	0.0800	1531	228 AA	0.0
	1.2000	22968	310 BA	0.0
	0.7200	13781	340 CA	0.0
2. RIGHT TURN	0.0067	128	380 AA	0.0
	0.1002	1918	284 BA	0.0
	0.0601	1151	354 CA	0.0
3. LEFT TURN	0.0067	128	337 AA	0.0
	0.1002	1918	376 BA	0.0
	0.0601	1151	322 CA	0.0
4. CONTROL CORR.				
(A). LONGITUDINAL	0.0007	13	1364 AA	2.365 0.000005
	0.0100	192	1118 BA	13.868 0.000014
	0.0060	115	1073 CA	22.058 0.000005
(B). LATERAL	0.0007	13	939 AA	0.0
	0.0100	192	959 BA	0.0
	0.0060	115	834 CA	0.0
(C). RUDDER	0.0007	13	0 AC	0.0
	0.0100	192	506 BA	0.0
	0.0060	115	491 CA	0.0
C. SIDEWARD FLIGHT				
1. TO THE RIGHT	0.0096	184	487 AA	0.0
	0.1442	2761	591 BA	0.0
	0.0865	1656	636 CA	0.0
2. TO THE LEFT	0.0096	184	312 AA	0.0
	0.1442	2761	402 BA	0.0
	0.0865	1656	471 CA	0.0
D. REARWARD FLIGHT				
	0.0096	184	493 AA	0.0
	0.1442	2761	650 BA	0.0
	0.0865	1656	875 CA	0.0
E. ACCELERATION				
HOVER TO CLIMB A/S	0.0200	383	578 AA	0.0
	0.3000	5742	565 BA	0.0
	0.1800	3445	652 CA	0.0

TABLE VIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE		OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	PCT. TIME	CYCLES IN 100 HRS.			
F. DECELERATION					
1. NORMAL	0.0200	383	695 AA		0.0
	0.3000	5742	848 BA		0.0
	0.1800	3445	900 CA		0.0
2. QUICK STOP	0.0040	77	818 AA		0.0
	0.0600	1148	1015 BA	44.531	0.000026
	0.0360	689	1147 CA	10.686	0.000064
G. APPR. AND LANDING	0.2204	4218	705 AA		0.0
	3.3057	63271	1070 BA	22.708	0.002786
	1.9834	37963	1051 CA	28.202	0.001346
III. FORWARD LEVEL FLIGHT					
AIRSPED	RPM				
A. 0.50 VH	314	0.0104	196	387 AA	0.0
		0.1563	2945	403 BA	0.0
		0.0938	1767	512 CA	0.0
	324	0.0938	1823	327 AA	0.0
		1.4070	27352	442 BA	0.0
		0.8442	16411	444 CA	0.0
B. 0.60 VH	314	0.0309	582	497 AA	0.0
		0.4634	8730	583 BA	0.0
		0.2780	5238	557 CA	0.0
	324	0.2780	5405	468 AA	0.0
		4.1705	81074	483 BA	0.0
		2.5023	48644	580 CA	0.0
C. 0.70 VH	314	0.0342	644	501 AA	0.0
		0.5131	9666	614 BA	0.0
		0.3078	5800	769 CA	0.0
	324	0.3079	5985	512 AA	0.0
		4.6178	89770	598 BA	0.0
		2.7707	53862	662 CA	0.0
D. 0.80 VH	314	0.0551	1038	639 AA	0.0
		0.8264	15569	628 BA	0.0
		0.4958	9341	757 CA	0.0
	324	0.4958	9639	584 AA	0.0
		7.4377	144588	553 BA	0.0
		4.4626	86753	707 CA	0.0
E. 0.90 VH	314	0.0160	301	944 AA	0.0
		0.2394	4510	728 BA	0.0
		0.1436	2706	758 CA	0.0
	324	0.1436	2792	848 AA	0.0
		2.1546	41885	634 BA	0.0
		1.2928	25131	675 CA	0.0
F. VH	314	0.0138	261	1261 AA	4.440
		0.2076	3911	989 BA	65.076
		0.1246	2347	988 CA	65.699

TABLE VIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
324	0.1246	2421	1029 AA	36.977	0.000065
	1.8684	36322	961 BA		0.0
	1.1210	21793	908 CA		0.0
IV. NON-FIRING MANEUVERS					
A. FULL POWER CLIMB					
1. NORMAL	0.1000	1914	428 AA		0.0
	1.5000	28710	545 BA		0.0
	0.9000	17226	465 CA		0.0
2. HIGH-SPEED	0.0017	33	695 AA		0.0
	0.0256	489	890 BA		0.0
	0.0153	294	1098 CA	16.862	0.000017
B. MAXIMUM RATE ACCEL CLIMB - CRUISE A/S					
	0.1870	3580	865 AA		0.0
	2.8056	53699	753 BA		0.0
	1.6834	32219	817 CA		0.0
C. NORMAL TURNS					
1. TO THE RIGHT					
(A) 0.5 VH	0.0668	1279	574 AA		0.0
	1.0020	19178	672 BA		0.0
	0.6012	11507	668 CA		0.0
(B) 0.7 VH	0.0668	1279	674 AA		0.0
	1.0020	19178	968 BA	90.757	0.000211
	0.6012	11507	897 CA		0.0
(C) 0.9 VH	0.0043	83	823 AA		0.0
	0.0652	1247	903 BA		0.0
	0.0391	748	1142 CA	*	0.000007
2. TO THE LEFT					
(A) 0.5 VH	0.0668	1279	564 AA		0.0
	1.0020	19178	592 BA		0.0
	0.6012	11507	695 CA		0.0
(B) 0.7 VH	0.0668	1279	694 AA		0.0
	1.0020	19178	891 BA		0.0
	0.6012	11507	907 CA		0.0
(C) 0.9 VH	0.0043	83	796 AA		0.0
	0.0652	1247	856 BA		0.0
	0.0391	748	940 CA		0.0
D. .9 VH CONTR. CORR					
1. LONGITUDINAL	0.0033	64	1364 AA	2.361	0.000027
	0.0501	959	1429 BA	1.676	0.000572
	0.0301	575	1046 CA	30.052	0.000019
2. LATERAL	0.0033	64	1300 AA	3.437	0.000019
	0.0501	959	1246 BA	4.914	0.000195
	0.0301	575	1308 CA	3.272	0.000176
3. RUDDER	0.0006	11	953 AA		0.0
	0.0085	162	916 BA		0.0

TABLE VIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
E. SIDESLIP	0.0051 97	841 CA		0.0
	0.0080 153	382 AA		0.0
	0.1200 2297	391 BA		0.0
	0.0720 1378	588 CA		0.0
F. PART POWER DESCENT	0.0040 77	637 AA		0.0
	0.0600 1148	548 BA		0.0
	0.0360 689	679 CA		0.0
V. GUNNERY MANEUVERS				
A. FIRING IN A HOVER	0.0050 96	0 AC		0.0
	0.0751 1438	0 BC		0.0
	0.0451 863	0 CC		0.0
B. STRAFING IN ACCEL. FROM A HOVER	0.0033 64	0 AC		0.0
	0.0501 959	0 BC		0.0
	0.0301 575	0 CC		0.0
C. GUNNERY RUNS				
1. PT. TARGET DIVES				
(A) TO 0.6 VL	0.0187 358	324 AA		0.0
	0.2806 5370	393 BA		0.0
	0.1683 3222	679 CA		0.0
(B) TO 0.8 VL	0.1040 1991	619 AA		0.0
	1.5602 29862	861 BA		0.0
	0.9361 17917	989 CA	64.876	0.000276
(C) TO 0.9 VL	0.3220 6164	852 AA		0.0
	4.8305 92455	1106 BA	*	0.000559
	2.8983 55473	1240 CA	*	0.000938
(D) TO VL	0.0008 15	1267 AA	4.272	0.000004
	0.0120 230	1404 BA	*	0.000002
	0.0072 138	1598 CA	*	0.000031
2. SPRAY FIRE DIVES				
(A) TO 0.6 VL	0.0080 153	306 AA		0.0
	0.1202 2301	443 BA		0.0
	0.0721 1381	690 CA		0.0
(B) TO 0.8 VL	0.1079 2065	510 AA		0.0
	1.6184 30977	936 BA		0.0
	0.9711 18586	1030 CA	36.782	0.000505
(C) TO 0.9 VL	0.2291 4384	988 AA	65.556	0.000067
	3.4358 65762	1300 BA	*	0.000928
	2.0615 39457	1337 CA	*	0.002256
(D) TO VL	0.0040 77	1036 AA	33.905	0.000002
	0.0600 1148	846 BA		0.0
	0.0360 689	1754 CA	*	0.000244
D. GUNNERY RUN P/U				
1. TO THE RIGHT				
(A) 0.6 VL	0.0020 38	751 AA		0.0

TABLE VIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	0.0300	574	839 BA	0.0
	0.0180	345	1199 CA	0.000050
(B) 0.8 VL	0.0040	77	1204 AA	0.000011
	0.0600	1148	1251 BA	0.000241
	0.0360	689	1460 CA	0.000168
(C) 0.9 VL	0.0100	191	1449 AA	0.000013
	0.1500	2871	1542 BA	0.000886
	0.0900	1723	1676 CA	0.001068
(D) VL	0.0067	128	1804 AA	0.000139
	0.1002	1918	1925 BA	0.001643
	0.0601	1151	1944 CA	0.002635
2. TO THE LEFT				
(A) 0.6 VL	0.0020	38	649 AA	0.0
	0.0300	574	1067 BA	0.000024
	0.0180	345	1239 CA	0.000067
(B) 0.8 VL	0.0040	77	1131 AA	0.000006
	0.0600	1148	1257 BA	0.000251
	0.0360	689	1579 CA	0.000200
(C) 0.9 VL	0.0100	191	1364 AA	0.000014
	0.1500	2871	1517 BA	0.000689
	0.0900	1723	1745 CA	0.001546
(D) VL	0.0067	128	1724 AA	0.000260
	0.1002	1918	1865 BA	0.001617
	0.0601	1151	2234 CA	0.003525
3. SYMMETRICAL				
(A) 0.6 VL	0.0002	4	871 AA	0.0
	0.0030	57	1071 BA	0.000003
	0.0018	34	1239 CA	0.000007
(B) 0.8 VL	0.0020	38	1045 AA	0.000001
	0.0301	575	1302 BA	0.000169
	0.0180	345	1446 CA	0.000225
(C) 0.9 VL	0.0033	64	1451 AA	0.000043
	0.0501	959	1598 BA	0.001225
	0.0301	575	1779 CA	0.001413
(D) VL	0.0007	13	2071 AA	0.000073
	0.0100	192	1943 BA	0.000778
	0.0060	115	2102 CA	0.000711
E. GUNNERY TURNS				
1. TO THE RIGHT				
(A) 0.5 VH	0.0250	479	803 AA	0.0
	0.3757	7191	1000 BA	0.000131
	0.2254	4315	1158 CA	0.000446
(B) 0.7 VH	0.0729	1396	1031 AA	0.000039
	1.0942	20942	1308 BA	0.000753
	0.6565	12565	1148 CA	0.001187
(C) 0.9 VH	0.0016	31	1870 AA	0.000007

TABLE VIII (Continued)

TABLE VIII (Continued)					
FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION	
	0.0240	459	1841 BA	*	0.000178
	0.0144	276	1766 CA	*	0.000094
2. TO THE LEFT					
(A) 0.5 VH	0.0250	479	930 AA		0.0
	0.3757	7191	918 BA		0.0
	0.2254	4315	1252 CA	4.7'9	0.000915
(B) 0.7 VH	0.0729	1396	952 AA		0.0
	1.0942	20942	1049 BA	28.864	0.000726
	0.6565	12565	1174 CA	8.456	0.001486
(C) 0.9 VH	0.0016	31	1141 AA	11.226	0.000003
	0.0240	459	1339 BA	*	0.000014
	0.0144	276	1578 CA	*	0.000063
F. S-TURNS					
1. AT 0.8 VH	0.0040	77	1205 AA	6.623	0.000012
	0.0600	1148	1847 BA	*	0.000306
	0.0360	689	2108 CA	*	0.000404
2. AT VH	0.0051	98	1522 AA	1.080	0.000091
	0.0769	1472	1849 BA	*	0.000773
	0.0462	883	2034 CA	*	0.000561
VI. POWER TRANSITIONS					
A. POWER TO AUTO					
1. 0.5 VH	0.0033	64	383 AA		0.0
	0.0501	959	453 BA		0.0
	0.0301	575	529 CA		0.0
2. 0.7 VH	0.0083	160	463 AA		0.0
	0.1252	2397	603 BA		0.0
	0.0751	1438	770 CA		0.0
3. 0.9 VH	0.0060	115	809 AA		0.0
	0.0900	1723	742 BA		0.0
	0.0540	1034	765 CA		0.0
B. AUTO TO POWER					
1. IN GROUND-EFFECT	0.0040	77	830 AA		0.0
	0.0600	1148	809 BA		0.0
	0.0360	689	1188 CA	7.596	0.000091
2. 0.4 VH	0.0067	128	541 AA		0.0
	0.1002	1918	560 BA		0.0
	0.0601	1151	824 CA		0.0
3. 0.6 VH	0.0324	620	902 AA		0.0
	0.4862	9307	924 BA		0.0
	0.2917	5584	1080 CA	20.320	0.000275
4. MAX AUTO A/S	0.0012	23	1008 AA	49.085	0.000000
	0.0180	345	964 BA	97.315	0.000004
	0.0108	207	1375 CA	2.224	0.000093
VII. AUTOROTATION					

TABLE VIII (Continued)

TABLE VIII (Continued)					
FLIGHT CONDITION	FREQUENCY OF OCCURRENCE		OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	PCT. TIME	CYCLES IN 100 HRS.			
A. STABILIZED FLIGHT					
1. 0.4 VH	0.0043	83	322 AA		0.0
	0.0652	1247	330 BA		0.0
	0.0391	748	391 CA		0.0
2. 0.6 VH	0.0809	1549	393 AA		0.0
	1.2142	23239	419 BA		0.0
	0.7285	13943	435 CA		0.0
3. MAX AUTO A/S	0.0040	77	553 AA		0.0
	0.0600	1148	624 BA		0.0
	0.0360	689	622 CA		0.0
B. AUTO TURNS					
1. TO THE RIGHT					
(A) 0.4 VH	0.0033	64	526 AA		0.0
	0.0501	959	477 BA		0.0
	0.0301	575	484 CA		0.0
(B) 0.6 VH	0.0541	1036	627 AA		0.0
	0.8119	15540	604 BA		0.0
	0.4872	9324	553 CA		0.0
(C) MAX AUTO A/S	0.0020	38	778 AA		0.0
	0.0300	574	822 BA		0.0
	0.0180	345	625 CA		0.0
2. TO THE LEFT					
(A) 0.4 VH	0.0033	64	429 AA		0.0
	0.0501	959	447 BA		0.0
	0.0301	575	434 CA		0.0
(B) 0.6 VH	0.0541	1036	593 AA		0.0
	0.8119	15540	697 BA		0.0
	0.4872	9324	520 CA		0.0
(C) MAX AUTO A/S	0.0020	38	777 AA		0.0
	0.0300	574	789 BA		0.0
	0.0180	345	703 CA		0.0
C. AUTO LANDING					
	0.0040	77	1424 AA	1.718	0.000045
	0.0600	1148	1293 BA	3.599	0.000319
	0.0360	689	2189 CA	*	0.000142
ENDURANCE LIMIT =		962.0	TOTAL DAMAGE (D) =		0.040377
MATERIAL = ALUM					
FREQUENCY = 1 / REV OF M/R		FATIGUE LIFE = 100/D =		2476 HOURS	
* DAMAGE CALCULATED FROM MEASURED LOAD FREQUENCIES.					

TABLE IX. 540-011-154-5 MAIN ROTOR GRIP
FATIGUE LIFE DETERMINATION

Flight Condition	Frequency of Occurrence % of Total Flt Time	Cycles in 100 Hours (n)	Oscillatory Stress in Lower Grip Tang--PSI	Cycles to Failure (N)x10 ⁻⁶	Damage Fraction (n/N)
V _H Level Flt 314 RPM	0.0138 0.2076 0.1246		4484 AB 2981 DA 3521 FA		
324 RPM	0.1246 1.8684 1.1210		3785 AB 3355 DA 3334 FA		
S-Turn 0.8 V _H	0.0040 0.0600 0.0360		4903 AA 5497 CA 5604 FB		
V _H	0.0051 0.0769 0.0462	883	6202 AA 5485 CA 7037 FA	0.40	0.002200
Total Damage in 100 Hr. (D) =					0.002200
Fatigue Life = 100/D = 45,454 Hours					

TABLE X. 540-011-153-13 MAIN ROTOR YOKE
EXTENSION FATIGUE LIFE DETERMINATION

Flight Condition	Frequency of Occurrence % of Total Flt Time	Cycles in 100 Hours (n)	Oscillatory in T/E Lug (psi)	Cycles to Failure (N) $\times 10^{-6}$	Damage Fraction (n/N)
V _H Level Flt 314 RPM	.0138		12,551 AB		
	.2076		9,556 DA		
	.1246		10,647 FC		
324 RPM	.1246		10,553 AB		
	1.8684		9,234 DA		
	1.1210		10,815 FC		
Gunnery Run Pull-out					
1. To the right					
c. 0.9 V _L	.0100	191	17,287 BA	6.448	0.000030
	.1500	2,871	18,080 CA	2.704	0.001062
	.0900	1,723	17,736 EA	3.747	0.000460
d. V _L	.0067	128	18,851 BA	1.535	0.000083
	.1002	1,918	20,080 CA	0.809	0.002371
	.0601	1,151	20,152 FA	0.785	0.001466
2. To the left					
c. 0.9 V _L	.0100		15,699 BA		
	.1500		16,610 DA		
	.0900	1,723	17,705 FA	3.871	0.000445
d. V _L	.0067	128	18,170 BA	2.506	0.000051
	.1002	1,918	18,389 CA	2.108	0.000910
	.0601	1,151	19,957 EA	0.855	0.001346
3. Symmetrical					
b. 0.8 V _L	.0020		13,832 BA		
	.0301		15,656 DA		
	.0180	345	18,588 FA	1.826	0.000189
c. 0.9 V _L	.0033		16,180 AA		
	.0501	959	17,134 DA	8.125	0.000118
	.0301	575	18,733 EA	1.656	0.000347
d. V _L	.0007	13	19,054 AA	1.358	0.000096
	.0100	192	18,698 CA	1.695	0.000113
	.0060	115	18,377 EA	2.127	0.000054
S - Turns					
1. 0.8 V _H	.0040	77	18,385 AA	2.115	0.000036
	.0600		14,119 CA		
	.0360		16,282 FC		
2. V _H	.0051	98	19,855 AA	0.896	0.000109
	.0769	1,472	17,435 DA	5.290	0.000278
	.0462	883	19,397 FA	1.122	0.000787
Auto Landing	.0040		6,256 AA		
	.0600		11,681 DA		
	.0360	689	18,529 FA	1.903	0.000362
Total Damage in 100 Hr. (D) =					0.010713
Fatigue Life = 100/D = 9334 Hours					

TABLE XI. 209-010-403-1 SWASHPLATE OUTER RING
FATIGUE LIFE DETERMINATION

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
I. GROUND CONDITIONS					
A. NORMAL START	0.4000	7656	215 AA		0.0
B. SHUTDOWN W/COLL.	0.4000	7656	301 AA		0.0
II. ICE MANEUVERS					
A. TAKE-OFF					
1. NORMAL	0.0511	978	333 BA		0.0
	0.7668	14677	337 CA		0.0
	0.4601	8806	387 FA		0.0
2. JUMP	0.0057	109	399 AA		0.0
	0.0852	1631	449 CA		0.0
	0.0511	978	574 EA		0.0
B. HOVERING					
1. STEADY	0.0800	1531	271 BA		0.0
	1.2000	22968	319 DA		0.0
	0.7200	13781	362 FA		0.0
2. RIGHT TURN	0.0067	128	284 BA		0.0
	0.1002	1918	362 CA		0.0
	0.0601	1151	362 FA		0.0
3. LEFT TURN	0.0067	128	321 BA		0.0
	0.1002	1918	331 DA		0.0
	0.0601	1151	512 FA		0.0
4. CONTROL CORR.					
(A). LONGITUDINAL	0.0007	13	506 BA		0.0
	0.0100	192	602 DA		0.0
	0.0060	115	598 EA		0.0
(B). LATERAL	0.0007	13	358 BA		0.0
	0.0100	192	637 CA		0.0
	0.0060	115	674 FA		0.0
(C). RUDDER	0.0007	13	287 AA		0.0
	0.0100	192	399 CA		0.0
	0.0060	115	323 EA		0.0
C. SIDEWARD FLIGHT					
1. TO THE RIGHT	0.0096	184	512 AA		0.0
	0.1442	2761	399 CA		0.0
	0.0865	1656	562 FA		0.0
2. TO THE LEFT	0.0096	184	395 BA		0.0
	0.1442	2761	442 DA		0.0
	0.0865	1656	449 FA		0.0
D. REARWARD FLIGHT					
	0.0096	184	543 BA		0.0
	0.1442	2761	550 CA		0.0
	0.0865	1656	682 EA		0.0
E. ACCELERATION					
HOVER TO CLIMB A/S	0.0200	383	296 BA		0.0
	0.3000	5742	553 DA		0.0
	0.1800	3445	599 FA		0.0

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.		OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
F. DECELERATION					
1. NORMAL	0.0200	383	599 AA		0.0
	0.3000	5742	687 CA		0.0
	0.1800	3445	634 EA		0.0
2. QUICK STOP	0.0040	77	555 BA		0.0
	0.0600	1148	602 DA		0.0
	0.0360	689	766 EA		0.0
G. APPR. AND LANDING	0.2204	4218	605 BA		0.0
	3.3057	63271	749 CA		0.0
	1.9834	37963	742 EA		0.0
III. FORWARD LEVEL FLIGHT					
AIR SPEED	RPM				
A. 0.50 VH	314	0.0104	196	400 AB	0.0
		0.1563	2945	520 DA	0.0
		0.0938	1767	710 FC	0.0
	324	0.0938	1823	410 AB	0.0
		1.4070	27352	530 DA	0.0
		0.8442	16411	520 FA	0.0
B. 0.60 VH	314	0.0309	582	485 BA	0.0
		0.4634	8730	570 DA	0.0
		0.2780	5238	690 FC	0.0
	324	0.2780	5405	500 AC	0.0
		4.1705	81074	600 DA	0.0
		2.5023	48644	620 EA	0.0
C. 0.70 VH	314	0.0342	644	570 AA	0.0
		0.5131	9666	660 DA	0.0
		0.3078	5800	710 FC	0.0
	324	0.3079	5985	560 AC	0.0
		4.6178	89770	640 DA	0.0
		2.7707	53862	760 FC	0.0
D. 0.80 VH	314	0.0551	1038	660 AB	0.0
		0.8264	15569	765 DA	0.0
		0.4958	9341	740 FA	0.0
	324	0.4958	9639	695 AA	0.0
		7.4377	144588	705 DA	0.0
		4.4626	86753	800 FC	0.0
E. 0.90 VH	314	0.0160	301	830 AB	0.0
		0.2394	4510	890 DA	0.0
		0.1436	2706	830 FA	0.0
	324	0.1436	2792	808 AA	0.0
		2.1546	41885	805 DA	0.0
		1.2928	25131	810 FC	0.0
F. VH	314	0.0138	261	1180 AB	0.0
		0.2076	3911	1030 DA	0.0
		0.1246	2347	950 EA	0.0

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
324	0.1246	2421	1090 AB	0.0
	1.8684	36322	920 DA	0.0
	1.1210	21793	930 EA	0.0
IV. NON-FIRING MANEUVERS				
A. FULL POWER CLIMB				
1. NORMAL				
	0.1000	1914	580 AC	0.0
	1.5000	28710	562 CA	0.0
	0.9000	17226	749 FC	0.0
2. HIGH-SPEED				
	0.0017	33	791 AC	0.0
	0.0256	489	1100 CA	0.0
	0.0153	294	848 FB	0.0
B. MAXIMUM RATE ACCEL CLIMB - CRUISE A/S				
	0.1870	3580	816 AA	0.0
	2.8056	53699	827 DA	0.0
	1.6834	32219	873 FC	0.0
C. NORMAL TURNS				
1. TO THE RIGHT				
(A) 0.5 VH				
	0.0668	1279	481 BA	0.0
	1.0020	19178	599 DA	0.0
	0.6012	11507	749 FC	0.0
(B) 0.7 VH				
	0.0668	1279	649 AA	0.0
	1.0020	19178	787 DA	0.0
	0.6012	11507	934 FB	0.0
(C) 0.9 VH				
	0.0043	83	889 AC	0.0
	0.0652	1247	988 DA	0.0
	0.0391	748	1113 EA	0.0
2. TO THE LEFT				
(A) 0.5 VH				
	0.0668	1279	469 AC	0.0
	1.0020	19178	575 DA	0.0
	0.6012	11507	637 FC	0.0
(B) 0.7 VH				
	0.0668	1279	629 BA	0.0
	1.0020	19178	750 DA	0.0
	0.6012	11507	873 FB	0.0
(C) 0.9 VH				
	0.0043	83	862 AA	0.0
	0.0652	1247	901 DA	0.0
	0.0391	748	955 EA	0.0
D. .9 VH CONTR. CORR				
1. LONGITUDINAL				
	0.0033	64	1299 AA	0.0
	0.0501	959	1037 DA	0.0
	0.0301	575	1094 FB	0.0
2. LATERAL				
	0.0033	64	1511 AA	44.086 0.000001
	0.0501	959	1136 DA	0.0
	0.0301	575	1106 FB	0.0
3. RUDDER				
	0.0006	11	926 AB	0.0
	0.0085	162	926 DA	0.0

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
E. SIDESLIP	0.0051 97 0.0080 153 0.1200 2297	877 FA 333 AC 493 DA		0.0 0.0 0.0
F. PART POWER DESCENT	0.0720 1378 0.0040 77 0.0600 1148 0.0360 689	664 FC 809 AB 787 CA 869 EA		0.0 0.0 0.0 0.0
V. GUNNERY MANEUVERS				
A. FIRING IN A HOVER	0.0050 96 0.0751 1438 0.0451 863	271 BA 319 DA 362 FA		0.0 0.0 0.0
B. STRAFING IN ACCEL. FROM A HOVER	0.0033 64 0.0501 959 0.0301 575	296 BA 553 DA 599 FA		0.0 0.0 0.0
C. GUNNERY RUNS				
1. PT. TARGET DIVES				
(A) TO 0.6 VL	0.0187 358 0.2806 5370 0.1683 3222	599 BA 679 DA 852 FC		0.0 0.0 0.0
(B) TO 0.8 VL	0.1040 1991 1.5602 29862 0.9361 17917	950 BA 1025 DA 919 EA		0.0 0.0 0.0
(C) TO 0.9 VL	0.3220 6164 4.8305 92455 2.8983 55473	1118 AA 1136 DA 1354 EA		0.0 0.0 0.0
(D) TO VL	0.0008 15 0.0120 230 0.0072 138	1419 AA 1387 CA 1727 EA	9.451	0.0 0.0 0.000015
2. SPRAY FIRE DIVES				
(A) TO 0.6 VL	0.0080 153 0.1202 2301 0.0721 1381	499 BA 703 DA 762 FC		0.0 0.0 0.0
(B) TO 0.8 VL	0.1079 2065 1.6184 30977 0.9711 18586	912 BA 1000 CA 1056 FA		0.0 0.0 0.0
(C) TO 0.9 VL	0.2291 4384 3.4358 65762 2.0615 39457	1042 AA 1150 CA 1354 FA		0.0 0.0 0.0
(D) TO VL	0.0040 77 0.0600 1148 0.0360 689	1306 AA 1337 CA 1615 FA	19.216	0.0 0.0 0.000036
D. GUNNERY RUN P/U				
1. TO THE RIGHT				
(A) 0.6 VL	0.0020 38	1012 BA		0.0

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
(B) 0.8 VL	0.0300 0.0180 0.0040 0.0600	574 345 77 1148	1309 DA 2099 EA 1625 BA 1649 CA	 1.833 17.919 15.239	0.0 0.000188 0.000004 0.000075
(C) 0.9 VL	0.0360 0.0100 0.1500 0.0900	689 191 2871 1723	1888 EA 1557 AC 1667 DA 1888 EA	4.202 29.756 13.563 4.202	0.000164 0.000006 0.000212 0.000410
(D) VL	0.0067 0.1002 0.0601	128 1918 1151	1662 BA 1825 CA 1963 EA	14.004 5.638 3.053	0.000009 0.000340 0.000377
2. TO THE LEFT					
(A) 0.6 VL	0.0020 0.0300 0.0180	38 574 345	1200 BA 1299 CA 1913 EA	 3.764	0.0 0.0 0.000092
(B) 0.8 VL	0.0040 0.0600 0.0360	77 1148 689	1537 BA 1864 DA 2101 FB	35.102 4.685 1.820	0.000002 0.000245 0.000379
(C) 0.9 VL	0.0100 0.1500 0.0900	191 2871 1723	2012 BA 1887 CA 1888 EA	2.517 4.220 4.202	0.000076 0.000680 0.000410
(D) VL	0.0067 0.1002 0.0601	128 1918 1151	1532 AA 1729 DA 1940 FB	36.631 9.344 3.357	0.000003 0.000205 0.000343
3. SYMMETRICAL					
(A) 0.6 VL	0.0002 0.0030 0.0018	4 57 34	937 BA 1148 DA 1416 FA	 29.997	0.0 0.0 0.0 0.000019
(B) 0.8 VL	0.0020 0.0301 0.0180	38 575 345	1224 BA 1556 DA 1876 FA	4.435 32.825 14.748	0.0 0.000078 0.000002 0.000065
(C) 0.9 VL	0.0033 0.0501 0.0301	64 959 575	1545 AC 1654 DA 2076 FB	1.987 17.195 14.004	0.000290 0.000001 0.000014
(D) VL	0.0007 0.0100 0.0060	13 192 115	1631 AC 1662 CA 1820 FC	5.778	0.000020
E. GUNNERY TURNS					
1. TO THE RIGHT					
(A) 0.5 VH	0.0250 0.3757 0.2254	479 7191 4315	840 AC 724 DA 799 FC	 57.421	0.0 0.0 0.0 0.000001
(B) 0.7 VH	0.0729 1.0972 0.6565	1396 20942 12565	1223 AC 1296 DA 1421 FA	 57.421	0.0 0.0 0.0 0.000001
(C) 0.9 VH	0.0016	31	1483 AC		

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION	
	0.0240	459	1457 DA	74.904	0.000006
	0.0144	276	1790 EA	6.719	0.000041
2. TO THE LEFT					
(A) 0.5 VH	0.0250	479	703 BA		0.0
	0.3757	7191	712 CA		0.0
	0.2254	4315	737 FB		0.0
(B) 0.7 VH	0.0729	1396	902 AC		0.0
	1.0942	20942	1099 DA		0.0
	0.6565	12565	1137 EA		0.0
(C) 0.9 VH	0.0016	31	1792 AC	6.651	0.000005
	0.0240	459	1432 DA	98.848	0.000005
	0.0144	276	1911 EA	3.797	0.000073
F. S-TURNS					
1. AT 0.8 VH	0.0040	77	1449 AA	81.652	0.000001
	0.0600	1148	1321 DA		0.0
	0.0360	689	1685 FC	12.120	0.000057
2. AT VH	0.0051	98	1774 AA	7.304	0.000013
	0.0769	1472	1519 DA	41.031	0.000036
	0.0462	883	1746 FC	8.495	0.000104
VI. POWER TRANSITIONS					
A. POWER TO AUTO					
1. 0.5 VH	0.0033	64	456 BA		0.0
	0.0501	959	543 DA		0.0
	0.0301	575	608 FA		0.0
2. 0.7 VH	0.0083	160	629 AB		0.0
	0.1252	2397	712 CA		0.0
	0.0751	1438	762 FB		0.0
3. 0.9 VH	0.0060	115	839 AB		0.0
	0.0900	1723	852 DA		0.0
	0.0540	1034	889 FB		0.0
B. AUTO TO POWER					
1. IN GROUND-EFFECT	0.0040	77	587 AA		0.0
	0.0600	1148	651 DA		0.0
	0.0360	689	799 FA		0.0
2. 0.4 VH	0.0067	128	592 BA		0.0
	0.1002	1918	716 DA		0.0
	0.0601	1151	639 FB		0.0
3. 0.6 VH	0.0324	620	926 AB		0.0
	0.4862	9307	937 CA		0.0
	0.2917	5584	1180 FB		0.0
4. MAX AUTO A/S	0.0012	23	1086 BA		0.0
	0.0180	345	1136 DA		0.0
	0.0108	207	1643 FB	15.857	0.000013
VII. AUTOROTATION					

TABLE XI (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY PITCH LINK AXIAL LOAD	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
A. STABILIZED FLIGHT					
1. 0.4 VH	0.0043	83	271 BA		0.0
	0.0652	1247	444 DA		0.0
	0.0391	748	298 FA		0.0
2. 0.6 VH	0.0809	1549	395 BA		0.0
	1.2142	23239	580 DA		0.0
	0.7285	13943	549 FC		0.0
3. MAX AUTO A/S	0.0040	77	662 AA		0.0
	0.0600	1148	703 DA		0.0
	0.0360	689	889 FC		0.0
B. AUTO TURNS					
1. TO THE RIGHT					
(A) 0.4 VH	0.0033	64	494 BA		0.0
	0.0501	959	568 DA		0.0
	0.0301	575	521 EA		0.0
(B) 0.6 VH	0.0541	1036	629 BA		0.0
	0.8119	15540	716 DA		0.0
	0.4872	9324	716 FC		0.0
(C) MAX AUTO A/S	0.0020	38	975 AB		0.0
	0.0300	574	1111 DA		0.0
	0.0180	345	1590 EA	23.040	0.000015
2. TO THE LEFT					
(A) 0.4 VH	0.0033	64	494 BA		0.0
	0.0501	959	592 DA		0.0
	0.0301	575	491 FB		0.0
(B) 0.6 VH	0.0541	1036	618 AC		0.0
	0.8119	15540	765 DA		0.0
	0.4872	9324	646 FA		0.0
(C) MAX AUTO A/S	0.0020	38	987 AB		0.0
	0.0300	574	1111 DA		0.0
	0.0180	345	1577 EA	25.423	0.000014
C. AUTO LANDING					
	0.0040	77	802 BA		0.0
	0.0600	1148	799 DA		0.0
	0.0360	689	1049 FA		0.0
ENDURANCE LIMIT = 1431.0 TOTAL DAMAGE (D) = 0.005143					
MATERIAL = ALUM					
FREQUENCY = 1 / REV OF M/R FATIGUE LIFE = 100/D = 19443 HOURS					

TABLE XII. 204-011-702-17 TAIL ROTOR BLADE
FATIGUE LIFE DETERMINATION

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TU FAILURE X 10**(-6)	DAMAGE FRACTION
I. GROUND CONDITIONS				
A. NORMAL START	0.4000	15312	1107 AA	0.0
B. SHUTDOWN W/COLL.	0.4000	15312	1825 AA	0.0
II. ICE MANEUVERS				
A. TAKE-OFF				
1. NORMAL	0.0511	1957	1631 AA	0.0
	0.7668	29353	2038 CA	0.0
	0.4601	17612	1988 FA	0.0
2. JUMP	0.0057	217	2141 BA	0.0
	0.0852	3261	2233 CA	0.0
	0.0511	1957	1148 FA	0.0
B. HOVERING				
1. STEADY	0.0800	3062	1684 AA	0.0
	1.2000	45936	1909 CA	0.0
	0.7200	27562	2240 FA	0.0
2. RIGHT TURN	0.0067	256	2259 BA	0.0
	0.1002	3836	2756 CA	0.0
	0.0601	2301	2734 EA	0.0
3. LEFT TURN	0.0067	256	3203 BA	0.0
	0.1002	3836	2485 CA	0.0
	0.0601	2301	2305 EA	0.0
4. CONTROL CORR.				
(A). LONGITUDINAL	0.0007	26	2235 BA	0.0
	0.0100	384	2027 CA	0.0
	0.0060	230	1964 FA	0.0
(B). LATERAL	0.0007	26	1979 AA	0.0
	0.0100	384	2537 CA	0.0
	0.0060	230	2246 FA	0.0
(C). RUDDER	0.0007	26	2431 BA	0.0
	0.0100	384	2139 CA	0.0
	0.0060	230	2275 EA	0.0
C. SIDEWARD FLIGHT				
1. TO THE RIGHT	0.0096	368	3217 BA	0.0
	0.1442	5522	2822 CA	0.0
	0.0865	3313	2888 FA	0.0
2. TO THE LEFT	0.0096	368	748 AA	0.0
	0.1442	5522	537 CA	0.0
	0.0865	3313	520 FA	0.0
D. REARWARD FLIGHT				
	0.0096	368	1508 AA	0.0
	0.1442	5522	1598 CA	0.0
	0.0865	3313	2383 FA	0.0
E. ACCELERATION				
HOVER TO CLIMB A/S	0.0200	766	1864 BA	0.0
	0.3000	11484	2862 CA	0.0
	0.1800	6890	2305 FA	0.0

TABLE XII (Continued)

T/R BLADE					
FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
F. DECELERATION					
1. NORMAL	0.0200	766	1965 BA		0.0
	0.3000	11484	2686 CA		0.0
	0.1800	6890	2498 FA		0.0
2. QUICK STOP	0.0040	153	1901 BA		0.0
	0.0600	2297	2315 CA		0.0
	0.0360	1378	3010 FA		0.0
G. APPR. AND LANDING	0.2204	8436	2054 AA		0.0
	3.3057	126542	2099 CA		0.0
	1.9834	75925	2054 EA		0.0
III. FORWARD LEVEL FLIGHT					
AIKSPED	RPM				
A. 0.50 VH	314	0.0104	393	1010 AA	0.0
		0.1563	5889	1300 DA	0.0
		0.0938	3534	1187 FA	0.0
	324	0.0938	3647	910 AA	0.0
		1.4070	54704	981 DA	0.0
		0.8442	32822	1162 FB	0.0
B. 0.60 VH	314	0.0309	1164	1152 BA	0.0
		0.4634	17460	1246 DA	0.0
		0.2760	10476	1528 FA	0.0
	324	0.2780	10810	1295 AA	0.0
		4.1705	162.48	1126 DA	0.0
		2.5023	97289	1468 FB	0.0
C. 0.70 VH	314	0.0342	1280	1599 AA	0.0
		0.5131	19332	1373 DA	0.0
		0.3078	11599	1956 EA	0.0
	324	0.3079	11969	1606 AA	0.0
		4.6178	179539	1449 CA	0.0
		2.7707	107724	1803 FB	0.0
D. 0.80 VH	314	0.0551	2076	2168 BA	0.0
		0.8264	31138	1704 DA	0.0
		0.4958	18683	2250 EA	0.0
	324	0.4958	19278	2178 AA	0.0
		7.4377	289176	1927 DA	0.0
		4.4626	173506	2262 FB	0.0
E. 0.90 VH	314	0.0160	601	2220 AA	0.0
		0.2394	9021	2281 DA	0.0
		0.1436	5412	2967 EA	0.0
	324	0.1436	5585	2394 BA	0.0
		2.1546	83771	2583 DA	0.0
		1.2928	50262	2873 EA	0.0
F. VH	314	0.0138	521	2632 AA	0.0
		0.2076	7822	3414 CA	70.530
		0.1246	4693	3395 FB	76.879
					0.000111
					0.000061

TABLE XII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.		OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	324	0.1246	4843	3288 AC	0.0
		1.8684	72643	2952 CA	0.0
		1.1210	43586	3485 FB	52.060 0.000837
IV. NON-FIRING MANEUVERS					
A. FULL POWER CLIMB					
1. NORMAL					
		0.1000	3828	1755 AC	0.0
		1.5000	57420	3181 DA	0.0
		0.9000	34452	2084 EA	0.0
2. HIGH-SPEED					
		0.0017	65	2968 AC	0.0
		0.0256	978	3199 DA	0.0
		0.0153	587	3281 EA	0.0
B. MAXIMUM RATE ACCEL CLIMB - CRUISE A/S					
		0.1870	7160	2879 AC	0.0
		2.8056	107398	3231 DA	0.0
		1.6834	64439	2966 EA	0.0
C. NORMAL TURNS					
1. TO THE RIGHT					
(A) 0.5 VH					
		0.0668	2557	1247 AA	0.0
		1.0020	38356	1269 FB	0.0
		0.6012	23014	1331 FB	0.0
(B) 0.7 VH					
		0.0668	2557	1992 AA	0.0
		1.0020	38356	1831 CA	0.0
		0.6012	23014	2028 FB	0.0
(C) 0.9 VH					
		0.0043	166	2611 AC	0.0
		0.0652	2494	3281 EA	0.0
		0.0391	1497	3133 FB	0.0
2. TO THE LEFT					
(A) 0.5 VH					
		0.0668	2557	1286 AC	0.0
		1.0020	38356	1347 DA	0.0
		0.6012	23014	1451 FC	0.0
(B) 0.7 VH					
		0.0668	2557	2093 AB	0.0
		1.0020	38356	2140 DA	0.0
		0.6012	23014	2436 EA	0.0
(C) 0.9 VH					
		0.0043	166	3096 AB	0.0
		0.0652	2494	3860 DA	14.768 0.000169
		0.0391	1497	3387 EA	79.775 0.000019
D. .9 VH CONTR. CORR					
1. LONGITUDINAL					
		0.0033	128	2595 BA	0.0
		0.0501	1918	3191 DA	0.0
		0.0301	1151	2978 FB	0.0
2. LATERAL					
		0.0033	128	3269 AA	0.0
		0.0501	1918	3334 DA	0.0
		0.0301	1151	3381 EA	82.040 0.000014
3. RUDDER					
		0.0006	22	4584 AA	3.041 0.000007
		0.0085	324	3227 DA	0.0

TABLE XII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.		OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
E. SIDESLIP	0.0051	194	3146 FB		0.0
	0.0080	306	904 AC		0.0
	0.1200	4594	1165 DA		0.0
	0.0720	2756	1600 FC		0.0
F. PART POWER DESCENT	0.0040	153	2383 BA		0.0
	0.0600	2297	2421 CA		0.0
	0.0360	1378	2141 FA		0.0
V. GUNNERY MANEUVERS					
A. FIRING IN A HOVER	0.0050	192	1684 AA		0.0
	0.0751	2876	1909 CA		0.0
	0.0451	1725	2240 FA		0.0
B. STRAFING IN ACCEL. FROM A HOVER	0.0033	128	1864 BA		0.0
	0.0501	1918	2862 CA		0.0
	0.0301	1151	2305 FA		0.0
C. GUNNERY RUNS					
1. PT. TARGET DIVES					
(A) TO 0.6 VL	0.0187	716	1554 AB		0.0
	0.2806	10740	1729 CA		0.0
	0.1683	6444	2135 FA		0.0
(B) TO 0.8 VL	0.1040	3982	2154 BA		0.0
	1.5602	59724	2632 DA		0.0
	0.9361	35834	2588 EA		0.0
(C) TO 0.9 VL	0.3220	12327	3002 AA		0.0
	4.8305	184911	2960 CA		0.0
	2.8983	110946	3038 FA		0.0
(D) TO VL	0.0008	31	3291 AA		0.0
	0.0120	459	3719 DA	22.468	0.000020
	0.0072	276	4057 EA	8.872	0.000031
2. SPRAY FIRE DIVES					
(A) TO 0.6 VL	0.0080	307	1666 BA		0.0
	0.1202	4603	2203 DA		0.0
	0.0721	2762	2217 FA		0.0
(B) TO 0.8 VL	0.1079	4130	2200 BA		0.0
	1.6184	61954	2837 CA		0.0
	0.9711	37172	2500 EA		0.0
(C) TO 0.9 VL	0.2291	8768	2782 BA		0.0
	3.4358	131524	3424 CA	67.461	0.001950
	2.0615	78914	3104 FA		0.0
(D) TO VL	0.0040	153	3228 AB		0.0
	0.0600	2297	3381 CA	82.040	0.000028
	0.0360	1378	4293 FC	5.262	0.000262
D. GUNNERY RUN P/U					
1. TO THE RIGHT					
(A) 0.6 VL	0.0020	77	1718 AB		0.0

TABLE XII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	0.0300	1148	1940 DA	0.0
	0.0180	689	2616 FA	0.0
(B) 0.8 VL	0.0040	153	3089 AA	0.0
	0.0600	2297	3399 CA	75.483 0.000030
	0.0360	1378	4193 EA	6.501 0.000212
(C) 0.9 VL	0.0100	383	4261 BA	5.622 0.000069
	0.1500	5742	3611 CA	32.257 0.000178
	0.0900	3445	4629 EA	2.815 0.001224
(D) VL	0.0067	256	3586 BA	35.282 0.000007
	0.1002	3836	4641 CA	2.759 0.001390
	0.0601	2301	4959 EA	1.678 0.001371
2. TO THE LEFT				
(A) 0.6 VL	0.0020	77	2142 BA	0.0
	0.0300	1148	2779 CA	0.0
	0.0180	689	2466 FB	0.0
(B) 0.8 VL	0.0040	153	3563 BA	38.399 0.000004
	0.0600	2297	3218 FA	0.0
	0.0360	1378	3836 FA	15.804 0.000087
(C) 0.9 VL	0.0100	383	5182 BA	1.232 0.000311
	0.1500	5742	4420 CA	4.096 0.001402
	0.0900	3445	4475 EA	3.696 0.000932
(D) VL	0.0067	256	4812 BA	2.092 0.000122
	0.1002	3836	4842 CA	1.998 0.001920
	0.0601	2301	5325 EA	1.025 0.002245
3. SYMMETRICAL				
(A) 0.6 VL	0.0002	8	1728 AA	0.0
	0.0030	115	2042 DA	0.0
	0.0018	69	2462 FA	0.0
(B) 0.8 VL	0.0020	77	3603 BA	33.187 0.000002
	0.0301	1151	3639 CA	29.257 0.000039
	0.0180	690	3975 EA	10.866 0.000064
(C) 0.9 VL	0.0033	128	3821 BA	16.499 0.000008
	0.0501	1918	3838 CA	15.714 0.000122
	0.0301	1151	4759 EA	2.274 0.000506
(D) VL	0.0007	26	4461 BA	3.792 0.000007
	0.0100	384	4273 CA	5.483 0.000070
	0.0060	230	4604 FB	2.938 0.000078
E. GUNNERY TURNS				
1. TO THE RIGHT				
(A) 0.5 VH	0.0250	959	1432 AB	0.0
	0.3757	14383	1436 CA	0.0
	0.2254	8630	1641 FB	0.0
(B) 0.7 VH	0.0729	2792	2307 AB	0.0
	1.0942	41884	2085 DA	0.0
	0.6565	25131	2418 FC	0.0
(C) 0.9 VH	0.0016	61	2957 AC	0.0

TABLE XII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	0.0240	919	3085 DA		0.0
	0.0144	551	2967 FB		0.0
2. TO THE LEFT					
(A) 0.5 VH	0.0250	959	2010 AB		0.0
	0.3757	14383	1565 CA		0.0
	0.2254	8630	1900 FA		0.0
(B) 0.7 VH	0.0729	2792	2373 BA		0.0
	1.0942	41884	2387 DA		0.0
	0.6565	25131	2778 FA		0.0
(C) 0.9 VH	0.0016	61	3953 AB	11.498	0.000005
	0.0240	919	3340 DA	100.000	0.000009
	0.0144	551	4226 EA	6.053	0.000091
F. S-TURNS					
1. AT 0.8 VH	0.0040	153	3428 AA	66.282	0.000002
	0.0600	2297	3880 CA	13.971	0.000164
	0.0360	1378	3518 FA	45.614	0.000030
2. AT VH	0.0051	196	4754 BA	2.292	0.000086
	0.0769	2944	4825 DA	2.051	0.001436
	0.0462	1767	5789 FB	0.600	0.002943
VI. POWER TRANSITIONS					
A. POWER TO AUTO					
1. 0.5 VH	0.0033	128	1601 AA		0.0
	0.0501	1918	1184 DA		0.0
	0.0301	1151	1331 FA		0.0
2. 0.7 VH	0.0083	320	2171 AA		0.0
	0.1252	4793	1646 DA		0.0
	0.0751	2876	2129 EA		0.0
3. 0.9 VH	0.0060	230	2861 BA		0.0
	0.0900	3445	2155 DA		0.0
	0.0540	2067	3260 FB		0.0
B. AUTO TO POWER					
1. IN GROUND-EFFECT	0.0040	153	1829 AA		0.0
	0.0600	2297	2175 CA		0.0
	0.0360	1378	2090 EA		0.0
2. 0.4 VH	0.0067	256	1630 AB		0.0
	0.1002	3836	1848 DA		0.0
	0.0601	2301	1892 FB		0.0
3. 0.6 VH	0.0324	1241	2131 AB		0.0
	0.4862	18613	2339 DA		0.0
	0.2917	11168	2474 FB		0.0
4. MAX AUTO A/S	0.0012	46	3332 AB		0.0
	0.0180	689	2641 DA		0.0
	0.0108	413	3440 FB	62.903	0.000007
VII. AUTOROTATION					

TABLE XII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5	CYC. TO FAILURE X 10 ⁴ (-6)	DAMAGE FRACTION
A. STABILIZED FLIGHT				
1. 0.4 VH	0.0043	166	1446 AA	0.0
	0.0652	2494	771 DA	0.0
	0.0391	1497	994 FA	0.0
2. 0.6 VH	0.0809	3099	1400 AA	0.0
	1.2142	46478	1084 DA	0.0
	0.7285	27887	1135 FA	0.0
3. MAX AUTO A/S	0.0040	153	2836 AA	0.0
	0.0600	2297	1914 DA	0.0
	0.0360	1378	2518 FA	0.0
B. AUTO TURNS				
1. TO THE RIGHT				
(A) 0.4 VH	0.0033	128	2025 AA	0.0
	0.0501	1918	1143 DA	0.0
	0.0301	1151	1074 EA	0.0
(B) 0.6 VH	0.0541	2072	1772 AB	0.0
	0.8119	31080	1131 DA	0.0
	0.4872	18648	1385 FA	0.0
(C) MAX AUTO A/S	0.0020	77	2653 AC	0.0
	0.0300	1148	2661 CA	0.0
	0.0180	689	3396 EA	76.527 0.000009
2. TO THE LEFT				
(A) 0.4 VH	0.0033	128	1228 BA	0.0
	0.0501	1918	1067 CA	0.0
	0.0301	1151	1328 EA	0.0
(B) 0.6 VH	0.0541	2072	1481 AC	0.0
	0.8119	31080	1436 CA	0.0
	0.4872	18648	1415 FA	0.0
(C) MAX AUTO A/S	0.0020	77	2992 AA	0.0
	0.0300	1148	2483 CA	0.0
	0.0180	689	2789 EA	0.0
C. AUTO LANDING				
	0.0040	153	1529 BA	0.0
	0.0600	2297	3194 CA	0.0
	0.0360	1378	3428 EA	66.282 0.000021
ENDURANCE LIMIT = 3340.0 TOTAL DAMAGE (D) = 0.020713				
MATERIAL = ALUM				
FREQUENCY = 2 / REV OF M/R FATIGUE LIFE = 100/D = 4827 HOURS				

TABLE XIII. 204-011-728-5 TAIL ROTOR GRIP
FATIGUE LIFE DETERMINATION

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. TIME	CYCLES IN 100 HRS.	OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
I. GROUND CONDITIONS					
A. NORMAL START	0.4000	15373	1090 CA		0.0
B. SHUTDOWN W/COLL.	0.4000	15373	1291 CA		0.0
II. ICE MANEUVERS					
A. TAKE-OFF					
1. NORMAL	0.0511	1965	813 AA		0.0
	0.7668	29471	1024 CA		0.0
	0.4601	17682	979 FA		0.0
2. JUMP	0.0057	218	841 AA		0.0
	0.0852	3274	1520 CA		0.0
	0.0511	1965	1884 FA		0.0
B. HOVERING					
1. STEADY	0.0800	3075	631 AA		0.0
	1.2000	46120	799 CA		0.0
	0.7200	27672	887 FA		0.0
2. RIGHT TURN	0.0067	257	1739 AA		0.0
	0.1002	3851	1612 CA		0.0
	0.0601	2311	1249 FA		0.0
3. LEFT TURN	0.0067	257	772 AA		0.0
	0.1002	3851	1137 DA		0.0
	0.0601	2311	882 FA		0.0
4. CONTROL CORR.					
(A). LONGITUDINAL	0.0007	26	946 AA		0.0
	0.0100	385	979 CA		0.0
	0.0060	231	1038 FA		0.0
(B). LATERAL	0.0007	26	869 AA		0.0
	0.0100	385	1074 DA		0.0
	0.0060	231	940 FA		0.0
(C). RUDDER	0.0007	26	948 AA		0.0
	0.0100	385	1160 CA		0.0
	0.0060	231	921 FA		0.0
C. SIDEWARD FLIGHT					
1. TO THE RIGHT	0.0096	370	1589 AA		0.0
	0.1442	5544	1708 DA		0.0
	0.0865	3326	1371 FA		0.0
2. TO THE LEFT	0.0096	370	1255 AA		0.0
	0.1442	5544	1881 CA		0.0
	0.0865	3326	2556 FA		0.0
D. REARWARD FLIGHT					
	0.0096	370	1411 AA		0.0
	0.1442	5544	5082 CA	0.664	0.008350
	0.0865	3326	1920 FA		0.0
E. ACCELERATION					
HOVER TO CLIMB A/S	0.0200	769	1270 AA		0.0
	0.3000	11530	1193 CA		0.0
	0.1800	6918	1603 FA		0.0

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE		OSCILLATORY RESULTANT	CYC. TO FAILURE	DAMAGE FRACTION
	PCT. TIME	CYCLES IN 100 HRS.	MOMENT @ STA 2.65	X 10**(-6)	
F. DECELERATION					
1. NORMAL	0.0200	769	1466 AA		0.0
	0.3000	11530	1256 DA		0.0
	0.1800	6918	1413 FA		0.0
2. QUICK STOP	0.0040	154	1460 AA		0.0
	0.0600	2306	1298 CA		0.0
	0.0360	1384	1958 FA		0.0
G. APPR. AND LANDING	0.2204	8470	1226 AA		0.0
	3.3057	127048	1315 CA		0.0
	1.9834	76229	1892 FA		0.0
III. FORWARD LEVEL FLIGHT					
AIRSPEED	RPM				
A. 0.50 VH	314	0.0104	394	946 AA	0.0
		0.1563	5913	962 BB	0.0
		0.0938	3548	1244 FA	0.0
	324	0.0938	3662	1027 AA	0.0
		1.4070	54923	1031 BB	0.0
B. 0.60 VH	314	0.8442	32954	1182 FA	0.0
		0.0309	1169	1242 AA	0.0
		0.4634	17530	1218 BB	0.0
	324	0.2780	10518	1284 FA	0.0
		0.2780	10853	1107 AA	0.0
C. 0.70 VH	314	4.1705	162797	1195 BB	0.0
		2.5023	97678	1294 FA	0.0
		0.0342	1294	1309 AA	0.0
	324	0.5131	19409	1501 BB	0.0
		0.3078	11646	1410 FA	0.0
D. 0.80 VH	314	0.3079	12017	1400 AA	0.0
		4.6178	180257	1379 BB	0.0
		2.7707	108154	1463 FA	0.0
	324	0.0551	2084	1451 AA	0.0
		0.8264	31263	1791 BB	0.0
E. 0.90 VH	314	0.4958	18758	1594 FA	0.0
		0.4958	19356	1642 AA	0.0
		7.4377	290333	1601 BB	0.0
	324	4.4626	174200	1702 FA	0.0
		0.0160	604	1725 AA	0.0
F. VH	314	0.2394	9057	2108 BB	0.0
		0.1436	5434	1810 FA	0.0
		0.1436	5607	1922 AA	0.0
	324	2.1546	84106	1815 BB	0.0
		1.2928	50463	1962 FA	0.0
	314	0.0138	524	2181 AA	0.0
		0.2076	7854	2356 BB	0.0
		0.1246	4712	2066	0.0

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
324	0.1246	4862	2247 AA	0.0
	1.8684	72934	2036 BB	0.0
	1.1210	43760	2189 FA	0.0
IV. NON-FIRING MANEUVERS				
A. FULL POWER CLIMB				
1. NORMAL	0.1000	3843	1209 AA	0.0
	1.5000	57650	1576 BB	0.0
	0.9000	34590	1338 FA	0.0
2. HIGH-SPEED	0.0017	65	1786 AA	0.0
	0.0256	982	1945 CA	0.0
	0.0153	589	1796 FA	0.0
B. MAXIMUM RATE ACCEL CLIMB - CRUISE A/S				
	0.1870	7189	2066 AA	0.0
	2.8056	107828	1949 CA	0.0
	1.6834	64697	1755 FA	0.0
C. NORMAL TURNS				
1. TO THE RIGHT				
(A) 0.5 VH	0.0668	2567	1050 AA	0.0
	1.0020	38510	1266 BB	0.0
	0.6012	23106	1325 FA	0.0
(B) 0.7 VH	0.0668	2567	1644 AA	0.0
	1.0020	38510	1929 BB	0.0
	0.6012	23106	1614 FA	0.0
(C) 0.9 VH	0.0043	167	1909 AA	0.0
	0.0652	2504	2051 BB	0.0
	0.0391	1503	2047 FA	0.0
2. TO THE LEFT				
(A) 0.5 VH	0.0668	2567	1142 AA	0.0
	1.0020	38510	1350 BB	0.0
	0.6012	23106	1393 FA	0.0
(B) 0.7 VH	0.0668	2567	1515 AA	0.0
	1.0020	38510	1881 BB	0.0
	0.6012	23106	1771 FA	0.0
(C) 0.9 VH	0.0043	167	1933 AA	0.0
	0.0652	2504	2291 BB	0.0
	0.0391	1503	2211 FA	0.0
D. .9 VH CONTR. CORR				
1. LONGITUDINAL				
	0.0033	128	1950 AA	0.0
	0.0501	1925	2385 BB	0.0
	0.0301	1155	1874 FA	0.0
2. LATERAL				
	0.0033	128	1825 AA	0.0
	0.0501	1925	2002 DA	0.0
	0.0301	1155	1758 FA	0.0
3. RUDDER				
	0.0006	22	2068 AA	0.0
	0.0085	325	1784 BB	0.0

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
E. SIDESLIP	0.0051 195 0.0080 307 0.1200 4612	1904 FA 1071 AA 1115 BB		0.0 0.0 0.0
F. PART POWER DESCENT	0.0720 2767 0.0040 154 0.0600 2306 0.0360 1384	1335 FA 1923 AA 1625 CA 1451 FA		0.0 0.0 0.0 0.0
V. GUNNERY MANEUVERS				
A. FIRING IN A HOVER	0.0050 192 0.0751 2887 0.0451 1732	830 AA 1062 CA 999 FA		0.0 0.0 0.0
B. STRAFING IN ACCEL. FROM A HOVER	0.0033 128 0.0501 1925 0.0301 1155	1270 AA 1194 CA 1604 FA		0.0 0.0 0.0
C. GUNNERY RUNS				
1. PT. TARGET DIVES				
(A) TO 0.6 VL	0.0187 719 0.2806 10783 0.1683 6470	1129 AA 1329 BB 1557 FA		0.0 0.0 0.0
(B) TO 0.8 VL	0.1040 3998 1.5602 59963 0.9361 35978	2003 AA 1999 CA 2138 FA		0.0 0.0 0.0
(C) TO 0.9 VL	0.3220 12377 4.8305 185650 2.8983 111390	2498 AA 2347 CA 2641 FA		0.0 0.0 0.0
(D) TO VL	0.0008 31 0.0120 461 0.0072 277	2991 AA 2464 CA 2947 FA	94.104	0.000000 0.0 0.0
2. SPRAY FIRE DIVES				
(A) TO 0.6 VL	0.0080 308 0.1202 4621 0.0721 2773	1244 AA 1426 DA 1587 FA		0.0 0.0 0.0
(B) TO 0.8 VL	0.1079 4147 1.6184 62202 0.9711 37321	1968 AA 2211 DA 2137 FA		0.0 0.0 0.0
(C) TO 0.9 VL	0.2291 8803 3.4358 132050 2.0615 79230	2710 AA 2361 DA 2595 FA		0.0 0.0 0.0
(D) TO VL	0.0040 154 0.0600 2306 0.0360 1384	3291 AA 3147 CA 3310 FA	24.781 44.008 23.132	0.000006 0.00002 0.000060
D. GUNNERY RUN P/U				
1. TO THE RIGHT				
(A) 0.6 VL	0.0020 77	2075 AA		0.0

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE		OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
	PCT. TIME	CYCLES IN 100 HRS.			
(B) 0.8 VL	0.0300	1153	1988 DA		0.0
	0.0180	692	1878 FA		0.0
	0.0040	154	2735 AA		0.0
	0.0600	2306	2934 BB		0.0
(C) 0.9 VL	0.0360	1384	2726 FA		0.0
	0.0100	384	2537 AA		0.0
	0.1500	5765	3390 CA	17.566	0.000328
	0.0900	3459	3137 FA	45.980	0.000075
(D) VL	0.0067	257	3522 AA	11.664	0.000022
	0.1002	3851	3821 CA	5.375	0.000716
	0.0601	2311	3828 FA	5.289	0.000437
2. TO THE LEFT					
(A) 0.6 VL	0.0020	77	1717 AA		0.0
	0.0300	1153	1983 CA		0.0
	0.0180	692	2006 FA		0.0
	0.0040	154	2567 AA		0.0
(B) 0.8 VL	0.0600	2306	2871 BB		0.0
	0.0360	1384	3226 FA	31.724	0.000044
	0.0100	384	3562 AA	10.400	0.000037
	0.1500	5765	3731 CA	6.663	0.000865
(C) 0.9 VL	0.0900	3459	3147 FA	44.008	0.000079
	0.0067	257	3976 AA	3.826	0.000067
	0.1002	3851	4083 CA	3.082	0.001249
	0.0601	2311	3986 FA	3.747	0.000617
3. SYMMETRICAL					
(A) 0.6 VL	0.0002	8	1950 AA		0.0
	0.0030	115	2116 CA		0.0
	0.0018	69	2021 FA		0.0
	0.0020	77	2759 AA		0.0
(B) 0.8 VL	0.0301	1155	2824 CA		0.0
	0.0180	693	2850 FA		0.0
	0.0033	128	3654 AA	8.102	0.000016
	0.0501	1925	2915 DA		0.0
(C) 0.9 VL	0.0301	1155	3139 FA	45.577	0.000025
	0.0007	26	3209 AA	33.948	0.000001
	0.0100	385	3633 CA	8.563	0.000045
	0.0060	231	4141 FA	2.757	0.000084
E. GUNNERY TURNS					
1. TO THE RIGHT					
(A) 0.5 VH	0.0250	963	1599 AA		0.0
	0.3757	14440	2170 BB		0.0
	0.2254	8664	1580 FA		0.0
	0.0729	2803	1971 AA		0.0
(B) 0.7 VH	1.0942	42052	2745 BB		0.0
	0.6565	25231	2821 FA		0.0
	0.0016	61	2855 AA		0.0

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.		OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FAILURE X 10**(-6)	DAMAGE FRACTION
2. TO THE LEFT	0.0240	922	2955 DA		0.0
	0.0144	553	2782 FA		0.0
(A) 0.5 VH	0.0250	963	1410 AA		0.0
	0.3757	14440	1957 BB		0.0
	0.2254	8664	1613 FA		0.0
	0.0729	2803	1882 AA		0.0
(B) 0.7 VH	1.0942	42052	2371 BB		0.0
	0.6565	25231	2575 FA		0.0
	0.0016	61	2688 AA		0.0
	0.0240	922	2978 BB		0.0
(C) 0.9 VH	0.0144	553	2866 FA		0.0
F. S-TURNS					
1. AT 0.8 VH	0.0040	154	2862 AA		0.0
	0.0600	2306	2577 DA		0.0
	0.0360	1384	2899 FA		0.0
	0.0051	197	3118 AA	50.055	0.000004
2. AT VH	0.0769	2956	2739 CA		0.0
	0.0462	1774	2386 FA		0.0
VI. POWER TRANSITIONS					
A. POWER TO AUTO					
1. 0.5 VH	0.0033	128	1257 AA		0.0
	0.0501	1925	1095 BB		0.0
	0.0301	1155	1003 FA		0.0
	0.0083	321	1451 AA		0.0
2. 0.7 VH	0.1252	4813	1398 CA		0.0
	0.0751	2888	1626 FA		0.0
	0.0060	231	2029 AA		0.0
	0.0900	3459	1837 CA		0.0
3. 0.9 VH	0.0540	2075	1642 FA		0.0
B. AUTO TO POWER					
1. IN GROUND-EFFECT	0.0040	154	1487 AA		0.0
	0.0600	2306	1469 CA		0.0
	0.0360	1384	1812 FA		0.0
	0.0067	257	1531 AA		0.0
2. 0.4 VH	0.1002	3851	1465 DA		0.0
	0.0601	2311	1219 FA		0.0
	0.0324	1246	1901 AA		0.0
	0.4862	18688	1564 DA		0.0
3. 0.6 VH	0.2917	11213	1994 FA		0.0
	0.0012	46	2133 AA		0.0
	0.0180	692	2270 CA		0.0
	0.0108	415	1836 FA		0.0
VII. AUTOROTATION					

TABLE XIII (Continued)

FLIGHT CONDITION	FREQUENCY OF OCCURRENCE PCT. CYCLES IN TIME 100 HRS.	OSCILLATORY RESULTANT MOMENT @ STA 2.65	CYC. TO FA'URE X 10 ⁻⁴ (-6)	DAMAGE FRACTION
A. STABILIZED FLIGHT				
1. 0.4 VH	0.0043	167	1181 AA	0.0
	0.0652	2504	921 BB	0.0
2. 0.6 VH	0.0391	1503	956 FA	0.0
	0.0809	3111	1184 AA	0.0
	1.2142	46664	1061 CA	0.0
	0.7285	27998	1518 FA	0.0
3. MAX AUTO A/S	0.0040	154	2014 AA	0.0
	0.0600	2306	1988 DA	0.0
	0.0360	1384	2126 FA	0.0
B. AUTO TURNS				
1. TO THE RIGHT				
(A) 0.4 VH	0.0033	128	1428 AA	0.0
	0.0501	1925	1076 DA	0.0
	0.0301	1155	1027 FA	0.0
(B) 0.6 VH	0.0541	2080	1750 AA	0.0
	0.8119	31205	1771 DA	0.0
	0.4872	18723	1350 FA	0.0
(C) MAX AUTO A/S	0.0020	77	2317 AA	0.0
	0.0300	1153	1927 CA	0.0
	0.0180	692	1892 FA	0.0
2. TO THE LEFT				
(A) 0.4 VH	0.0033	128	1378 AA	0.0
	0.0501	1925	1050 DA	0.0
	0.0301	1155	1006 FA	0.0
(B) 0.6 VH	0.0541	2080	1817 AA	0.0
	0.8119	31205	1522 CA	0.0
	0.4872	18723	1501 FA	0.0
(C) MAX AUTO A/S	0.0020	77	2167 AA	0.0
	0.0300	1153	2035 DA	0.0
	0.0180	692	1991 FA	0.0
C. AUTO LANDING				
	0.0040	154	1333 AA	0.0
	0.0600	2306	1347 DA	0.0
	0.0360	1384	1494 FA	0.0
ENDURANCE LIMIT = 2980.0			TOTAL DAMAGE (D) = 0.013179	
MATERIAL = ALUM				
FREQUENCY = 2 / REV OF M/R			FATIGUE LIFE = 100/D = 7587 HOURS	