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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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AGARD CONFERENCE PROCEEDINGS No. 206

Specialists Meeting on Helicopter Design Mission Load Spectra

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY
ON BACK COVER

(14) AGARD-CP-206

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

(9)

(6) AGARD Conference Proceedings No.206

Specialists Meeting on

HELICOPTER DESIGN MISSION LOAD SPECTRA

(11) Aug 76

(12) 73p.

Papers presented at the 42nd Meeting of the Structures and Materials Panel
held in Ottawa, Canada on 8 April 1976.

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Published August 1976

ISBN 92-835-0172-1

Printed by Technical Editing and Reproduction Ltd
Harford House, 7–9 Charlotte St, London, W1P 1HD

PREFACE

As mission requirements cause the utilization of any helicopter fleet to vary significantly over the fleet's life time, the problem of predicting the structural life of fatigue-critical components becomes more complex. If component lives cannot be accurately predicted, catastrophic, premature failures could occur, and untimely replacements, maintenance and inspections will increase costs and decrease the operational availability of the helicopters.

A prime factor in the inability to accurately predict component lives is the lack of adequate mission load-spectra data, which is compounded by the increased aircraft performance that has resulted in more sophisticated mission profiles. The impact of the fatigue spectra on life has been well documented over the years, showing that, for the same load levels, merely changing loading sequence dramatically changes the number of cycles to failure. A more accurate representation of the effects of not only load levels but load sequence on helicopter components will provide more realistic fatigue analyses and life predictions.

In order to stimulate the collection of such data and to pool the approaches in the helicopter-producing NATO Nations, a Specialist Meeting was organized. The presentations, an overview and the subsequent discussions are published in this volume. Among the subjects covered are the development of load spectra for design, the adaptation of this for life predictions in the field, and mission load data gathering techniques.

The Conference Committee is indebted to the authors who, by their valuable presentations, contributed to the success of the meeting. Special thanks are extended to Mr R.B.Johnson, Jr., coordinator, and CDR E.R.Way, Executive of the Structures and Materials Panel, for their excellent work in the organization of the Conference.

Robert S.BERRISFORD
Chairman of the Conference
Committee for Helicopter
Design Mission Load Spectra

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SPECTRES DE MISSION POUR LE CALCUL DES DUREES DE VIE

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1 - INTRODUCTION :

Si le principe général de détermination des durées de vie des pièces d'hélicoptères est sensiblement le même dans les différents pays, on est frappé par la diversité des choix qui sont faits par le constructeur à chaque étape.

- la forme des courbes S/N pour un même matériau et un même type de pièce peut varier légèrement,
- parmi les choix possibles de variable statistique on trouve aussi bien le nombre de cycles, le logarithme du nombre de cycles, la contrainte ou le logarithme de la contrainte,
- la loi de probabilité choisie est, en général, la loi normale mais elle peut aussi être l'une des lois de Pierson ou encore la loi normale tronquée aux faibles valeurs de probabilités,
- Enfin le spectre de vol employé pour un type de mission déterminée peut varier assez notablement d'un constructeur à l'autre car il comporte une partie suggestive et chacun cherche plus ou moins à garder une certaine marge de sécurité.

En conséquence, les comparaisons entre méthodes sont délicates à moins que l'on ne les fasse globalement sur des exemples précis. Ajoutons que, fort heureusement, ces méthodes sont sanctionnées par l'expérience et donnent des degrés de sécurité comparables.

Il n'en reste pas moins, qu'une fois la méthode de calcul acceptée, la durée de vie dépend fortement du type de mission effectuée par chaque appareil.

On est donc généralement conduit à pénaliser les utilisations les moins sévères pour fixer une durée de vie unique par type de pièce ce qui n'est pas très économique mais simplifie la gestion du matériel et accroît la sécurité.

Dans de rares cas, cependant, le constructeur fixe une durée de vie particulière pour quelques types de missions bien définies. Le problème est alors d'assurer la sécurité tout en ne perturbant pas le circuit de fabrication et de révision. La discussion des solutions possibles à ce problème sera exposée plus loin.

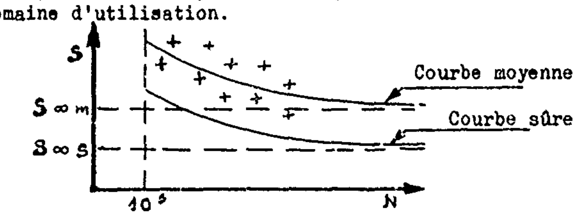
2 - NOTRE METHODE DE CALCUL DES DUREES DE VIE :

Etant donné que nous allons être amenés à comparer des durées de vie suivant les cas d'utilisation, il n'est pas inutile de rappeler brièvement les principales hypothèses de notre calcul bien que celles-ci aient déjà été exposées par ailleurs.

2.1 - Courbe moyenne et courbe sûre :

Ayant démontré que la résistance à la fatigue était un phénomène aléatoire, il existe des relations entre la contrainte S, le nombre de cycles N et la probabilité "p" dont on cherche une expression mathématique approchée dans le domaine d'utilisation.

Le problème est évidemment très complexe et sa solution exigerait un nombre d'essais impossible à réaliser dans la pratique. Néanmoins, les simplifications proposées, ci-après ont donné jusqu'alors de bons résultats :



- les courbes iso-probabilité de rupture $(S, N)p$ sont affines
- leur forme ne dépend que du matériau considéré et de la présence ou de l'absence de corrosion de frottement. On la détermine à l'aide de nombreux essais sur éprouvettes par méthode de régression sur l'équation:

$$S = S_{\infty} + \frac{A}{N^{\alpha}}$$

représentant la branche asymptotique pour $N > 10^5$ cycles.

- la variable statistique $v = \log S_\infty$ suit pratiquement la loi normale mais, pour tenir compte du fait que le contrôle élimine les pièces les plus mauvaises, on tronque cette loi aux faibles probabilités en annulant le risque de rupture à $\frac{S_\infty}{3}$

On peut ainsi, lorsque la moyenne et l'écart type sont connus, utiliser les règles du calcul statistique pour fixer la limite de fatigue sûre correspondant à un risque de l'ordre de 10^{-6}

La détermination de ces deux grandeurs doit être faite pour chaque pièce importante et l'analyse des nombreux résultats d'essais de contrôle statistique de nos pales nous a montré :

- que la moyenne de la famille était pratiquement définie à partir de 6 points d'essais (à ± 5 % près pour les dispersions usuelles)

- que, par contre, il fallait une centaine de points d'essais pour définir correctement l'écart type. Force était donc de le déterminer à l'aide de l'expérience antérieure et nous avons retenu les valeurs de l'écart type obtenues sur nos pales soit 0,06 pour les alliages légers et 0,045 pour les aciers.

Pour résumer le calcul de la courbe sûre, le facteur de probabilité $K_p = \frac{S_{\infty m}}{S_\infty}$ prend les valeurs usuelles suivantes :

- alliages légers........... : 2
- aciers.... .... : 1,77

2.2 - Calcul de la durée de vie :

La courbe de fatigue sûre étant ainsi définie, on mesure les efforts alternés correspondant aux différents cas de vol puis on détermine la durée de vie à l'aide de l'hypothèse des dommages cumulatifs de MINER et en utilisant le spectre de vol retenu pour la mission considérée.

A noter que quelques cas d'évolution donnent des contraintes présentant un certain caractère aléatoire. Il convient dans ce cas de faire plusieurs séries de mesures en vol pour tenir compte de ce fait.

Dans le passé, les contraintes étaient enregistrées sur papier et nous retenions en général la valeur de pointe supposée appliquée pendant toute la configuration de vol correspondante. Aujourd'hui, l'enregistrement se fait sur bande magnétique et une méthode de dépouillement automatique permet de mesurer les différents niveaux de contrainte alternée composant un cas de vol déterminé ainsi que les nombres de cycles correspondants. On peut ainsi déterminer avec précision l'endommagement par heure de vol pour les différentes configurations du spectre.

3 - SPECTRES DE VOL ET DUREES DE VIE :

3.1 - Présentation des spectres :

La figure 1 représente quelques uns des spectres couramment utilisés à l'Aérospatiale pour les hélicoptères de moyen tonnage et qui ont fait l'objet de discussions et d'accords avec les Services Officiels français.

Le spectre civil, du type transport, est surtout caractérisé par un faible temps en vol stationnaire et par des pourcentages élevés (50 % au total) de vol aux grandes vitesses de 0,85 et 0,9 VNE. Il comporte environ 9 % de virages et 10 % de vols dérapés. La durée du vol moyen est d'une demi-heure. Les calculs sont basés sur la masse maximale.

Le spectre A.S.M comprend au contraire 35 % de vol stationnaire H.E.S nécessités par l'écoute, des vitesses moins élevées, pas de dérapage et 6 % de virages. Les trempés se font toutes les 12 minutes, ce qui représente en moyenne 5 flares par heure de vol, et puisque les missions sont longues, on tient grossièrement compte de la consommation de carburant en considerant 2 masses de calcul.

Le spectre du travail à l'élingue, déterminé expérimentalement, comprend en moyenne 54 % de vol stationnaire à pleine charge, des vitesses faibles (0,5 à 0,7 VNE) et 8 % de virages à faible inclinaison. La durée d'une rotation est de 12 minutes (5 flares par heure de vol) et les retours à vide ne sont pas considérés parce que moins endommageants malgré leur vitesse plus élevée.

Un spectre un peu particulier est celui du remorquage ou du dragage représenté figure 2. Compte tenu du fait que le remorquage doit se faire dans une direction déterminée quel que soit le cap du vent, on peut avoir des dérapages importants dont l'influence sur le battement des pales du rotor arrière et la puissance passée par ce dernier est grande. On a donc été amené à considérer une répartition des vents relatifs, en grandeur et direction, basée sur des observations météorologiques diverses et sur des vitesses de remorquage déterminées expérimentalement. La direction des vents locaux et celle du remorquage ont été supposées aléatoires.

Les essais en vol ont permis de définir le meilleur angle de câble par rapport à l'appareil pour chaque direction du vent relatif.

La figure 3 donne la même information pour les appareils légers sous une forme légèrement différente. En effet plutôt que de fixer les créneaux de vitesse en fonction de la VNE atteinte en piqué seulement, on a choisi les vitesses d'utilisation basées sur les limites de pas : endurance maximale, croisière économique et croisière rapide.

Les spectres civils et militaires sont assez semblables à ceux des machines plus grosses sauf que les pourcentages de vol dérapés sont plus élevés et que, ayant à faire à des appareils plus maniables, les pourcentages de virages et de dérapages retenus sont plus grands (10 % de virages et 14 % de dérapages pour les civils, 15 % de virages et 14 % de dérapages pour les militaires)

Nous avons ajouté un spectre de pulvérisation agricole caractérisé par 95 % de temps de vol à vitesse réduite voisine de la vitesse de transition et par 28 % de virages nécessités par les rotations en bout de champ.

Pour donner une idée plus complète des spectres courants, il est bon de préciser les facteurs de charges considérés pour les virages étant donné que ces manoeuvres sont fortement endommageantes. Pour les hélicoptères de moyen tonnage, on considère des inclinaisons moyennes de virage de 30°, soit $n = 1,15$, avec quelques pointes (< 5 % du temps de virage) à 45°, soit $n = 1,4$, pour les spectres civil et A.S.M. Le spectre militaire moyen comporte des virages à 45° seulement.

En ce qui concerne les hélicoptères légers, la figure 4 donne la répartition des virages classés par facteur de charge aux différentes vitesses. On remarque que pour l'utilisation militaire, correspondant ici au vol tactique, l'ensemble du spectre est déplacé vers les facteurs de charges élevés. Il faut également signaler que ces facteurs de charge sont relativement élevés car ils correspondent aux machines de la génération la plus récente dont la surface de pale est dimensionnée par la vitesse et permet des Cz élevés.

3.2 - Sensibilité de la durée de vie au spectre de vol :

L'influence du type de mission et de la masse totale sur la durée de vie de quelques éléments d'un hélicoptère est représentée sur la figure 5 au sujet de laquelle il convient de faire tout d'abord quelques remarques générales.

- les différentes pièces qui ont servi à faire l'étude appartiennent souvent à des hélicoptères de type différent.

- les rapports de durée de vie représentés ne constituent pas une loi générale car ils peuvent être très fortement influencés par le degré de surabondance de la pièce considérée, par ses caractéristiques dynamiques ou encore par les limitations éventuellement précisées dans le Manuel de Vol.

  Il est évident que certaines pièces peuvent être dimensionnées pour avoir une durée de vie infinie pour toutes les utilisations de l'hélicoptère (c'est le cas de certaines pales plastiques) mais ce n'est pas toujours possible soit pour des raisons technologiques, soit tout simplement pour ne pas trop grever le devis de masse de l'appareil.

L'influence de la mission a été schématisée en prenant pour unité la durée de vie correspondant au spectre de transport. On voit alors que, pour d'autres types de missions, la durée de vie est divisée par trois ou par quatre ou multipliée par des facteurs du même ordre. Il est également curieux de constater que, pour une même mission, toujours comparée à la mission transport, certaines durées de vie augmentent et d'autres diminuent.

Ceci tient au fait que chaque élément de l'appareil n'est pas sensible aux mêmes paramètres de vol.

Par exemple, pour la mission militaire moyenne où les facteurs de charge sont plus élevés et plus nombreux et les vitesses plus faibles, la durée de vie des pales principales et du plateau cyclique diminue alors que celle du pylone rotor augmente, ce dernier élément étant surtout sensible à la vitesse et aux flares.

Pour les pales principales, nous trouvons deux diminutions sensibles de durée de vie : l'une en mission A.S.M qui comporte de nombreux flares endommageants, l'autre encore plus sensible en pulvérisation agricole mais qui se produit seulement lorsque la vitesse d'utilisation est voisine de la transition et que la réponse des pales est élevée. La figure 6 montre en effet que les contraintes dans la pale sont élevées en transition stabilisée alors qu'elle restent faibles tout le long d'une mise en vitesse progressive du stationnaire à la vitesse de croisière

Un autre cas intéressant est celui des pales arrière en remorquage ou en dragage pour lequel les vents relatifs latéraux agissant dans le sens du flux du rotor arrière augmentent la puissance à fournir par ce dernier et les contraintes au pied de pale. On a en fait simultanément approche du décrochage et action du sillage de la queue sur le rotor.

La durée de vie des pales peut alors être divisée par quatre.

La partie basse de la figure 5 donne l'influence d'une augmentation ou d'une diminution de masse maximale de l'appareil sur la durée de vie de différents éléments dans le cadre d'une mission militaire pour laquelle la manoeuvrabilité de l'appareil est importante. On voit que la durée de vie dimimue fortement pour la plupart des pièces lorsque la masse de l'appareil augmente, ceci étant dû essentiellement au fait que les virages à inclinaison donnée se font en décrochage de plus en plus prononcé.

Les pales arrière sont au contraire relativement peu sensibles à l'augmentation de masse. En effet, celles-ci sont essentiellement dimensionnées à la fatigue par les manoeuvres au palonnier et les dérapages plutôt que par une augmentation de la poussée nécessaire en vol stabilisé.

4 - DETERMINATION EXPERIMENTALE DU SPECTRE :

La première idée qui vient à l'esprit lorsqu'on veut connaître avec précision le spectre de contraintes ou d'efforts auxquels sont soumis les éléments de l'appareil est d'installer sur quelques machines représentatives de l'utilisation à évaluer un certain nombre de jauges extensométriques.

Ces jauges, placées sur chacun des éléments critiques au point de vue fatigue, peuvent être reliées à un mini ordinateur de bord chargé de faire du comptage de cycles par classes de contraintes alternées comprises entre des paires de niveau voisins. On peut également faire de l'enregistrement continu pendant un nombre d'heures de vol choisi puis faire le traitement à l'ordinateur des données ainsi recueillies.

Dans les deux cas on fait l'analyse statistique des résultats obtenus et on en déduit une répartition des contraintes alternées en pourcentage du temps de vol, établie pour chaque pièce importante, et qui servira à la justification à la fatigue de celle-ci.

Cette méthode à l'avantage d'être très directe mais, indépendamment de son coût, elle possède un certain nombre d'inconvénients qui nous l'ont fait écarter :

- Etant donné que la provenance des différents niveaux de contraintes alternées n'est pas connue, on ne pourra faire aucune extrapolation des résultats trouvés. Il faudra donc refaire une campagne de mesure chaque fois que l'on changera de manière appréciable les limitations de l'appareil (masse, vitesse, altitude...) et ceci même pour un type d'utilisation déterminé ;

- Si l'expérience montre que l'on a oublié un certain nombre de postes de mesures, il faudra également refaire une campagne complète ;

- Enfin, comme le mode de saisie des données ne permet pas de connaître l'ordre d'application des contraintes, il n'est pas possible de définir des essais à charges programmées à l'aide des résultats obtenus.

C'est la raison pour laquelle nous avons choisi d'enregistrer les paramètres essentiels du vol plutôt que les contraintes aux éléments. Un dépouillement des bandes d'enregistrement permet alors de déterminer le pourcentage du temps passé aux différentes configurations de vol et l'on peut relever séparément les contraintes correspondant à ces configurations au cours de vols d'essai.

Compte tenu de ce que nous avons vu au paragraphe précédent, il est aisé de fixer les paramètres de vol les plus influents sur la durée de vie. Très généralement il est souhaitable de pouvoir distinguer au dépouillement les configurations dommageantes suivantes :

- le vol horizontal aux différentes vitesses (y compris le vol stationnaire et la transition)

- le dérapage, surtout sensible pour le rotor arrière

- la montée, nécessitant des puissances élevées

- les manoeuvres (ressources, virages, lacet) avec le facteur de charge qui leur est lié.

- le flare, notamment pour en connaître le nombre par heure de vol

- l'altitude, importante pour son action sur le décrochage des pales

- le centrage longitudinal déterminant en grande partie le mât rotor principal

- et enfin, la masse qui influe très fortement sur les efforts pour presque toutes les pièces

Il est bien évident que certaines de ces données ne sont pas directement accessibles à la mesure (la masse et le centrage par exemple) et que d'autres s'obtiennent par comparaison d'enregistrement de différents paramètres (c'est le cas des manoeuvres, et de la montée ou de la descente).

Dans ces conditions, nous avons choisi une méthode mixte qui consiste : d'une part à enregistrer en continu tous les paramètres faciles à relever et d'autre part à demander aux équipages des renseignements complémentaires qui, pour la plupart, sont habituellement donnés par le pilote.

Ainsi, nous avons retenu l'enregistrement continu des paramètres suivants :

- vitesse indiquée
- altitude pression
- facteur de charge
- pas général
- pas rotor arrière

Par ailleurs, en même temps que les bandes d'enregistrement, on demande à l'utilisateur de fournir des bordereaux indiquant vol par vol :

- la masse au départ
- les conditions météorologiques (pression et température au sol)
- durée du vol entre décollage et atterrissage
- type de mission effectué

On voit que l'instrumentation peut se réduire à un simple enregistreur de bord à 5 pistes. Au début de l'étude, nous avions songé à enregistrer les données sous forme magnétique et à faire du dépouillement automatique par ordinateur après une conversion digitale de l'information analogique.

En fait, le programme correspondant s'est révélé difficile à écrire, d'autant plus qu'un certain nombre de données devaient être introduites aux endroits voulus à partir des comptes rendus de vols. Nous avons donc provisoirement renoncé à cette solution et opté pour un enregistrement papier dépouillé manuellement.

5 - APPLICATION A UN SPECTRE MILITAIRE :

La première applicationde cette méthode a été faite sur le Puma militaire il y a quatre ans. Cet hélicoptère, qui a maintenant une masse maximale de 7 t, était limité à l'époque à 6,4 t. L'évaluation a porté sur 5 machines dont l'une était affectée à une école de pilotage et les autres basées dans 4 formations éloignées les unes des autres qui avaient reçu la directive d'effectuer leur travail habituel avec les équipages les plus divers.

L'utilisation, assez variée, comportait des missions genre transport de troupe en terrain accidenté mais aussi du travail à l'élingue et des vols de liaison.

A l'origine, 50 heures de vol consécutives devaient être enregistrées sur les appareils en formation et 25 heures sur l'appareil école. En fait le dépouillement a porté sur 70 heures en moyenne pour le premier type d'utilisation, les 25 heures d'école étant conservées.

Le premier travail a consisté à déterminer la décomposition de chaque vol en configurations types et ceci par tranche d'altitude de 1000 m :

- stationnaire
- départ en translation
- montée
- descente
- vol latéral
- palier
- autorotation remise de pas
- approche et flare
- appareil au sol rotor tournant

Ces configurations types ont ensuite été décomposées de manière plus fine en déterminant : d'une part, la répartition des vitesses de palier en fonction de la VNE (compte tenu de la masse, de l'altitude pression et de la température au sol) et, d'autre part, les pourcentages de temps passés en virages classés par tranches de facteurs de charges. On a procédé de même pour les autres types de manoeuvres telles que les arrêts de virage en vol stationnaire.

On a ensuite évalué séparément pour chaque appareil du programme les temps ou pourcentages moyens par configuration. Cette opération a été faite en prenant la borne supérieure de l'intervalle de confiance à 90 % de la moyenne, ce qui a tendance à augmenter l'influence des configurations les plus dispersées qui sont aussi les plus endommageantes (manoeuvres surtout).

Ceci nous a permis d'établir un spectre par appareil comportant d'une part, les configurations types et d'autre part, une répartition des virages en fonction du facteur de charge et de la vitesse.

Pour éviter des analyses trop compliquées nous avons admis que le pourcentage de virages à une vitesse déterminée était proportionnel au pourcentage total d'utilisation de cette vitesse dans le spectre.

De là, nous sommes passés au spectre moyen de ce type d'utilisation.

Par ailleurs, nous avons déterminé séparément les pourcentages d'utilisation de l'appareil aux différentes vitesses et aux différentes altitudes.

Avant de montrer plus en détail les résultats obtenus il est bon de faire quelques remarques préalables.

Tout d'abord, il aurait peut être fallu éliminer les premières heures de vol pendant lesquelles la sévérité des manoeuvres a été très faible. (Il y avait sans doute la crainte du "mouchard"). Nous ne l'avons pas fait car la période d'adaptation des premiers pilotes a été très courte.

Enfin, à l'issue du travail, nous avons constaté que la dispersion des mesures entre les différents appareils était suffisamment faible pour qu'on puisse retenir un spectre moyen relatif au type d'utilisation considéré.

Comparons maintenant ce spectre moyen avec celui qui avait été imaginé après discussion avec les spécialistes : la figure 7 donne le spectre général où l'on voit que, par rapport aux prévisions, nous avons :

- un peu plus de vol stationnaire,
- très peu de vol en montée mais plus de vol latéral,
- une répartition beaucoup plus régulière des vitesses et des facteurs de charge
- des facteurs de charge jusqu'à la VNE comprise
- des approches plus longues
- 7 % de rotation au sol non endommageante

La figure 8 montre la répartition des facteurs de charge en fonction de la vitesse. On voit que l'on a pratiquement le même pourcentage total de virages, mais mieux répartis en fonction de la vitesse. La majorité des virages se fait à factour de charge de 1,15 (30° d'inclinaison) au lieu de 1,4 (45°) mais on atteint de temps à autre des facteurs de charge allant jusqu'à 2 (60°)

Il semble bien que, au moins pour certains éléments de l'appareil, le spectre mesuré soit plus sévère que le spectre estimé si l'on ne tient pas compte de la répartition des masses et des altitudes dont la figure 9 donne la comparaison avec les prévisions :

Pour les hélicoptères moyens, on voit que 70 % des vols se font à moins de 90 % de la masse maximale et seulement 5 % au voisinage de cette dernière (entre 95 et 100 %). A titre comparatif, nous avons indiqué dans le même tableau la répartition des masses au décollage obtenue sur notre hélicoptère léger Gazelle (5 places et 1900 kg de masse totale). On constate que, probablement par le fait qu'une machine légère est plus facile à remplir qu'une grosse, il y a un net déplacement de la répartition vers les masses élevées (28 % entre 95 et 100 %).

En ce qui concerne l'altitude, il y a évidemment beaucoup plus de dispersion dans les différents échantillons examinés, et, d'accord avec les utilisateurs, nous avons retenu une répartition voisine de celle de l'appareil qui faisait le plus de vol en montagne. Le domaine d'altitude retenu est malgré tout nettement plus favorable que le domaine estimé: 3 % au lieu de 10 à 3000 m et 89 % au sol au lieu de 70.

L'application du nouveau spectre au calcul des durées de vie du PUMA militaire a conduit à des gains très importants :

- durée de vie <u>double</u> pour les pales principales et arrière
- durée de vie <u>deux à trois fois plus grande</u> pour certains éléments des commandes de vol (plateau cyclique, boulon à oeil) et de la suspension du rotor principal, pour ne citer que les plus importants.

6 - DIFFERENCIATION DES DUREES DE VIE PAR TYPE D'UTILISATION :

Comme nous l'avons vu précédemment, les durées de vie sont très sensibles au spectre d'utilisation et, du point de vue économique, il y a tout à fait intérêt à cerner de plus près la réalité en affectant une durée de vie variable à un même élément suivant le type d'utilisation de l'appareil sur lequel il est monté.

Ceci pose des problèmes de suivi assez difficiles à résoudre en toute sécurité chez le constructeur et chez l'utilisateur et pour lesquels nous allons essayer de jeter quelques bases.

Tout d'abord, il est pratiquement exclu de donner aux utilisateurs une durée de vie "à la carte" qu'il faudrait suivre au prix de nombreux coefficients correcteurs et d'une comptabilité compliquée considérant à la fois le type de mission et la pièce elle-même.

Il en résulterait de nombreuses erreurs préjudiciables à la sécurité.

Dans ce cas, il est préférable de chercher à déterminer un spectre moyen comme nous l'avons déjà exposé.

Il se peut par contre qu'une machine soit, à temps partiel, utilisée pour une mission très endommageante pour l'un de ses éléments. C'est le cas par exemple pour la pulvérisation agricole en régime de transition (pales principales) ou le remorquage (pales arrière). On peut alors définir un coefficient correcteur par lequel l'utilisateur multipliera les heures de vol réelles de la mission pour obtenir les heures à comptabiliser pour l'élément considéré. Dans ce cas, l'élément en question sera retiré du service à un nombre d'heures constant précisé dans le Manuel d'Entretien.

Dans la mesure où la mission considérée est effectuée par une version donnée de l'appareil il faudrait, pour augmenter la sécurité, diminuer la durée de vie de l'élément pour la version considérée et signaler que les heures de vol effectuées hors de la mission dommageante peuvent être affectée d'un coefficient <u>minorateur</u>. C'est à la fois garantir qu'un oubli n'est pas dangereux et motiver l'utilisateur pour qui la comptabilité demandée représente un gain d'argent.

Il arrive enfin qu'une version d'un appareil donné conduise, pour une même pièce, à une durée de vie différente de celle des autres versions. Il n'y a dans ce cas aucun problème de documentation puisqu'on peut afficher une durée de vie différente par version dans le Manuel d'Entretien. Cependant, pour éviter des erreurs au moment des révisions, il faut pouvoir distinguer les pièces appartenant aux différentes versions de l'appareil, par exemple en leur donnant un numéro de pièce différent.

Nous avons résolu le problème dans des cas analogues en gardant le même numéro individuel pour toutes les pièces à l'état neuf puis en frappant, devant ce numéro et à la première révision de l'ensemble, une lettre caractéristique de la version. Ceci suppose évidemment que l'on porte sur la plaquette d'identification de l'ensemble le code de version au moment où celui-ci est affecté pour la première fois et que, au travers des révisions, cet ensemble soit toujours remonté sur des appareils de même version.

7 - CONCLUSION :

En matière de calcul des durées de vie, le spectre d'utilisation a une importance très grande comme nous venons de le montrer et il est tout à fait rentable de le déterminer expérimentalement.

Il me semble préférable de saisir les paramètres essentiels du vol (vitesse, facteur de charge, etc...) plutôt que des données plus élaborées (contraintes, efforts, etc...) dont il ne sera plus possible de tirer parti pour un appareil nouveau ou pour une extrapolation de l'appareil considéré. Ceci permet d'ailleurs des échanges faciles au niveau des constructeurs et des Services Officiels dans des groupes de travail comme celui auquel nous participons.

Le choix des hélicoptères à retenir pour faire l'évaluation a une grande importance : il ne s'agit pas de couvrir l'appareil qui a l'utilisation la plus sévère dans chaque cas mais d'avoir un échantillon statistique permettant d'estimer de manière valable le spectre moyen de la flotte considérée. En effet, puisque tout est ramené à un risque de rupture, on peut estimer que la probabilité d'avoir la pièce de résistance la plus faible montée sur l'appareil le plus malmené est extrêmement petite.

Dans toute la mesure du possible, on conservera une durée de vie unique pour une même pièce. Cependant, pour de rares utilisations fortement pénalisantes pour un très petit nombre d'éléments coûteux, on fixera un coefficient correcteur permettant de transformer les heures de vol réelles en durée de vie consommée.

Il est bien évident que l'on a intérêt à s'affranchir du problème des durées de vie, notamment en concevant des pièces dont la détérioration est progressive et visible comme c'est déjà le cas pour un certain nombre de réalisations en plastiques armés. Cependant, même dans ce cas, la connaissance du spectre est primordiale pour leur justification.

Si j'avais enfin à me prononcer sur le point de l'étude des spectres qui serait le plus utile à régler dans l'immédiat par un groupe de travail comme le nôtre, je dirais que c'est la statistique des masses et de la durée des vols, pour différents types de missions et plusieurs tonnages d'hélicoptères. C'est à la fois le renseignement le plus simple à recueillir et le plus difficile à appréhender.

SPECTRE DES HELICOPTERES DE MOYEN TONNAGE

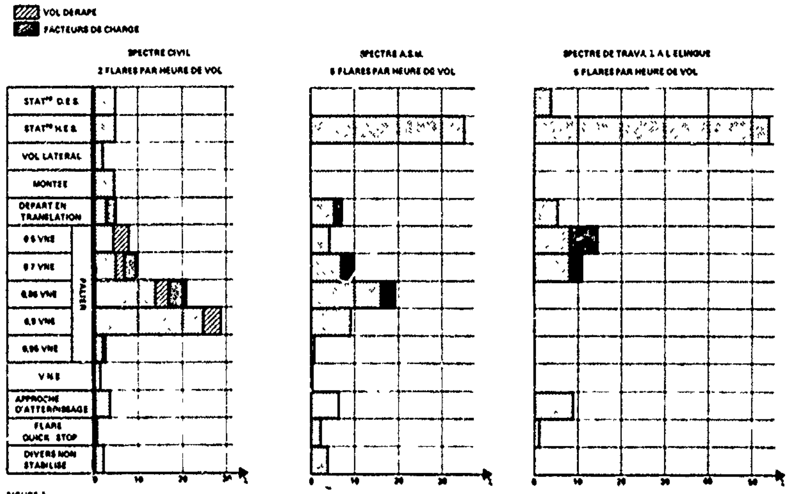


FIGURE 1

SPECTRE DE REMORQUAGE

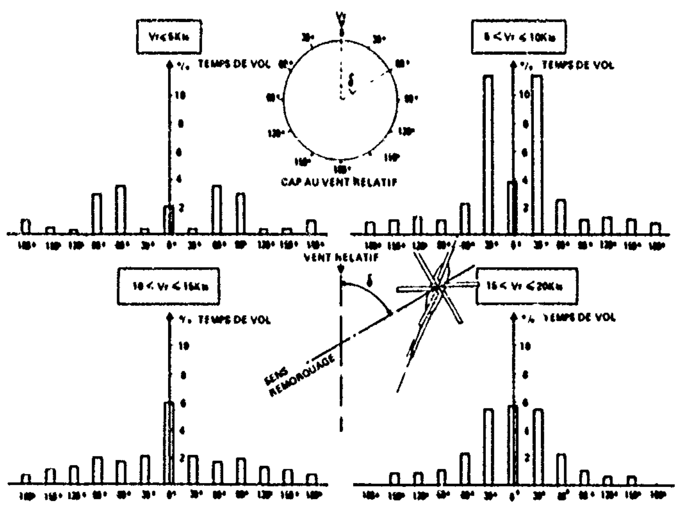


FIGURE 2

SPECTRE DES HELICOPTERES LEGERS

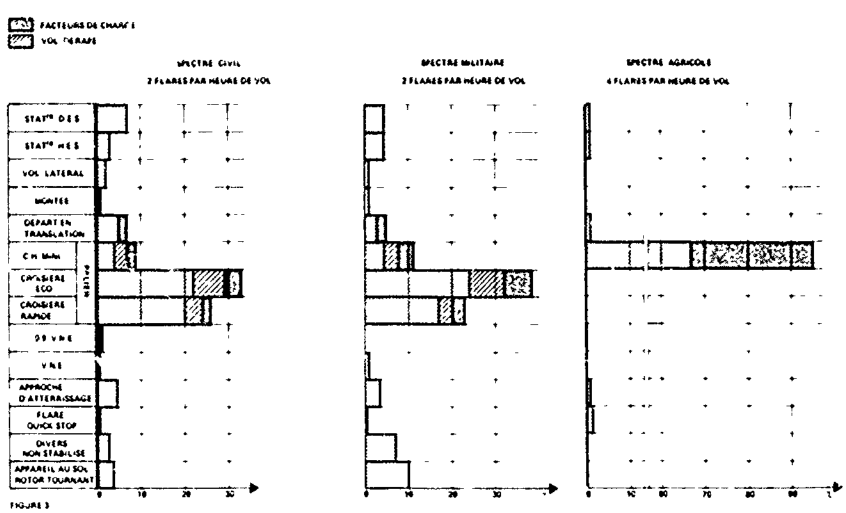


FIGURE 3

REPARTITION DES FACTEURS DE CHARGE EN PALIER

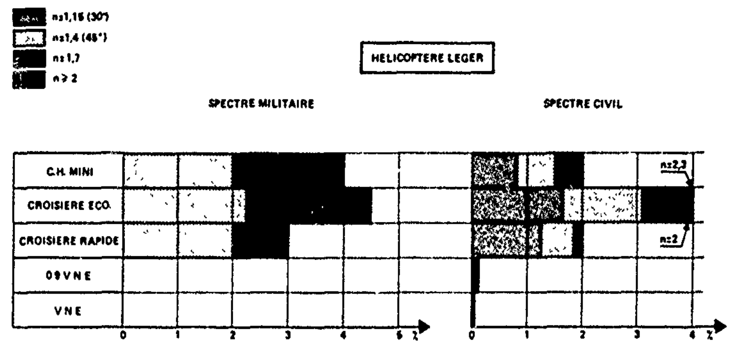


FIGURE 4

INFLUENCE DU SPECTRE SUR LA DUREE DE VIE

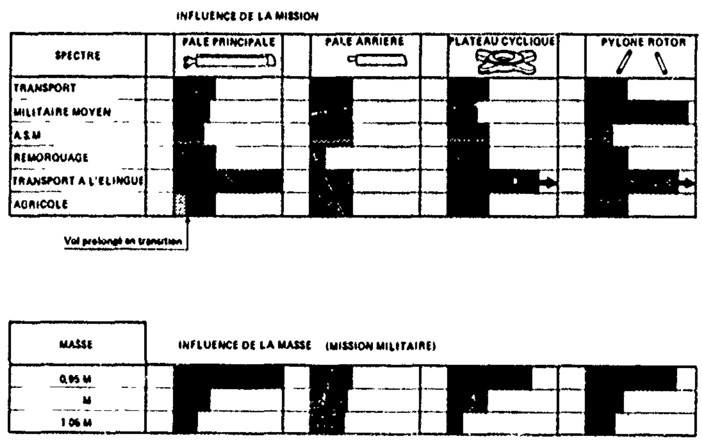


FIGURE 5

EVOLUTION DES EFFORTS EN PALIER

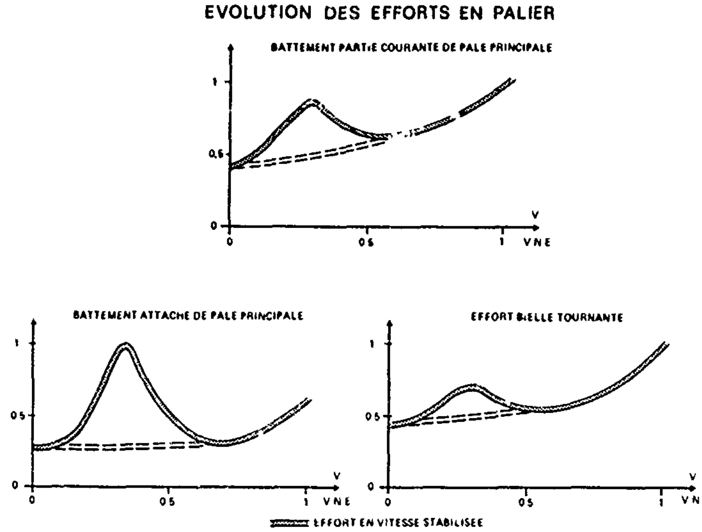


FIGURE 6

HELICOPTERE MOYEN – SPECTRE MILITAIRE

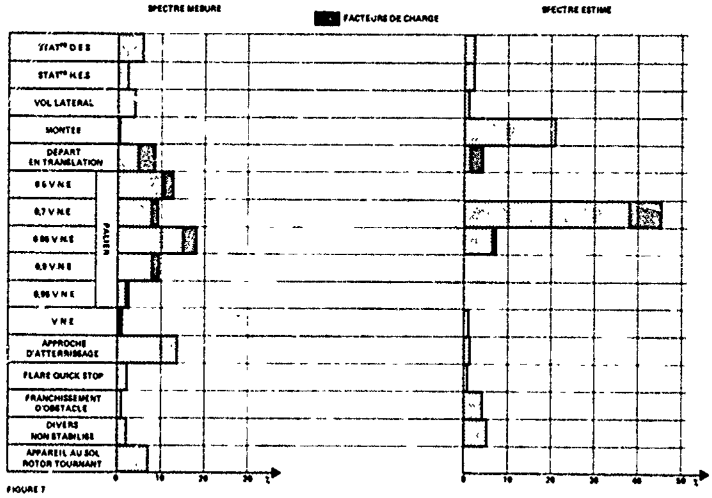


FIGURE 7

REPARTITION DES FACTEURS DE CHARGE EN PALIER

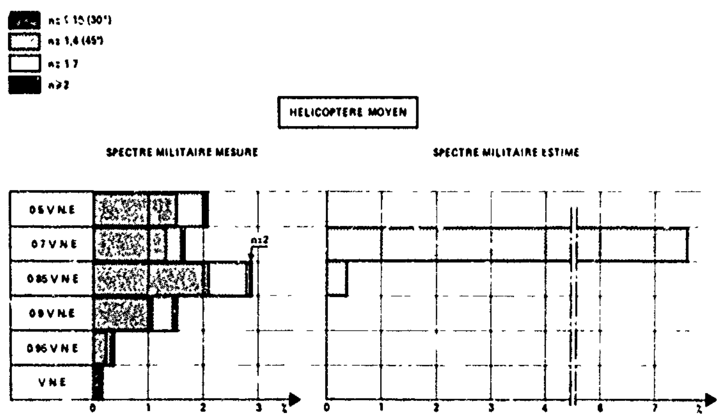


FIGURE 8

DOMAINES DE MASSE ET D'ALTITUDE ESTIMES ET MESURES

REPARTITION DES MASSES AU DECOLLAGE (% du temps de vol)

HELICOPTERE	REPARTITION	MASSE MAXIMALE 1		0.95		0.9		0.85		MASSE MINIMALE
MOYEN	ESTIMEE	100								
	MESUREE		5		25			70		
LEGER	ESTIMEE	100								
	MESUREE		28		53		12		7	

REPARTITION DES ALTITUDES DE VOL (% du temps de vol)

HELICOPTERE	REPARTITION	0	1500 m	3000 m
MOYEN	ESTIMEE	70	20	10
	MESUREE	89	8	3

FIGURE 9

THE IMPACT OF HELICOPTER MISSION SPECTRA ON FATIGUE

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SUMMARY

Helicopter components are requested to have long service lives, coming close to unlimited lifetimes. However, there are still real problems in the accurate prediction of the resulting component lives because this requires the availability of adequate mission load spectra.

The nature of the typical fatigue loading situation, in combination with the requested long lifetimes raises the question of the sensitivity to variations in mission requirements and load spectra. For a light helicopter with a hingeless rotor system using fiberglass rotor blades - in commercial as well as in military application - the influence of different mission requirements on load spectra and fatigue will be discussed. Normally, only a small portion of the complete mission is of importance for fatigue. Therefore, the impact of modified mission spectra on fatigue can be assessed, and no critical situation should arise if the overall design philisophy is adequate to prevent a high sensitivity.

1. INTRODUCTION

The design of the fatigue critical components of a helicopter, which are mainly the blades, the hub, and the drive system, requires the availability of adequate loads prediction methods and of a good representation of actual operational conditions to establish reliable mission load spectra. The rapidly expanding role of the helicopter in civilian as well as military operation has greatly changed the utilization of this aircraft and the resulting load spectrum has become more severe. The improving capability in the area of loads prediction will be of little value if it is not possible to establish realistic mission load spectra which are compatible with the operational requirements. Such data should be available during all steps of the development.

The mission profiles and the quantitative values for the flight conditions are different for the various helicopter concepts. It is obvious that a large transport helicopter will differ from a antitank helicopter, or a helicopter in civilian operation from a military helicopter which will do a high percentage of its operation in extreme nap-of-the-earth flying.

The Messerschmitt-Bölkow-Blohm Company has gained good experience with its light helicopter MBB - BO 105 (Figure 1, Ref. 1) which is engaged in civilian as well as in military operations. Up to now, about 250 ships have been delivered to customers, and some 150 000 hours of flight time have been accumulated. The first ships have about 3 600 hours of flight time. This experience is of special importance and interest because the BO 105 is the first production helicopter with a hingeless rotor and fiberglass rotorblades, which has been able to proof its broad ability in practical operation for a longer time. The broad spectrum of operation and experience includes utility, executive type, rescue, police, offshore, lighthouse supply mission in civilian operation, and LOH type, Scout and antitank missions in military operation.

Besides the problems resulting from the broad field of operation, which is typical for many light helicopters, the additional questions associated with the new technology like the different loading situation of the hingeless rotor and the behaviour of the fatigue loaded fiberglass blades had to be subject of special consideration.

2 LOADING AND FATIGUE SITUATION OF A HINGELESS ROTOR HELICOPTER

The historical story of the helicopter is marked by the struggle against dynamic forces at the rotor system and the dynamic characteristics of the whole helicopter. In the early days with the insufficient knowledge of the physical-technical correlations, the introduction of the blade attachment hinges was the only way to overcome the mechanical strength difficulities at the rotor blades and the control problems of the whole helicopter caused by the dissimmetry in the conditions of flow, and for a long period of time all successful helicopters were equipped with articulated rotors. Nowadays, rotors with rigid or so-called hingeless blade attachment are of great interest, because they offer mechanical simplification and improved handling qualities in comparison to the helicopter with an articulated rotor. The hingeless rotor is a new technology, which today is already a proven technology, offering a wide range for further improvements. The unterstanding and the theoretical background, as well as the materials to withstand the high vibratory loads are now available, (Ref. 2 ÷ 6).

For hingeless rotors without flapping hinges there is the possibility to transfer high moments from the blades to the hub and the fuselage, thereby the control of the helicopter becomes more powerful, faster and more direct, and nearly independent of rotor thrust. The hingeless rotor helicopter, therefore, is a step forward fulfilling the requirements for handling qualities of modern military helicopters with the ability of extreme nap-of-the-earth flying. The changed loading situation with high moment loads at the blade root area, at the hub, and at the rotor shaft have to be considered in the design. Figure 2 illustrates the different loading situation of a hingeless rotor and an articulated rotor with a small flapping hinge offset. If the aerodynamic lift at the blade is the same for both rotors, it produces the same force at the hinge or blade attachment respectively, and at the center of the hub; but for the moments there is a pronounced difference at the blade root and the hub. At a flapping hinge without restraint there will be no moment at all. For the hingeless rotor with blades of high elasticity there exists also an area at the blade with relatively low moments indicating an effective or equivalent hinge. Compared to a typical articulated rotor with a hinge offset of about 2 ÷ 3% of the radius, the equivalent hinge offset of a typical hingeless rotor is about 15%. Consequently, due to the different lever arm for the acting forces, the moments at the center of the hub will be quite different. The resulting moment at the hub can be reduced to a certain degree by coning the hub arms and thus producing an unloading moment out of centrifugal forces. Normally, the precone angle will be chosen for unloading with the design rotor thrust. Other thrust conditions and of course alternating thrust of the blades will result in corresponding moments at the hub. It should be pointed out that these moments resulting out of cyclic control inputs are intentioned; they are the basis of the improvements in handling qualities. The most stressed section of a hingeless rotor therefore is the blade attachment section, the hub and the rotor shaft. The blade main section is relatively low stressed with a proper design.

The possibility to transfer high moments from the rotor blades to the hub and the fuselage gives the changed situation for the control and trim behaviour. The control of helicopters with articulated rotors is mainly done by inclination of the thrust vector thus producing a moment around the center of gravity of the helicopter. For a helicopter with a hingeless rotor system an inclination of the thrust vector is combined with a strong hub moment, and the moment around the center of gravity is a combination of this hub moment and the moment due to the thrust inclination. The loading of the rotor shaft and the gearbox with its suspension is different in the two cases. For the articulated rotor the moment is built up linearly towards the center of gravity (cg); in the case of the hingeless rotor the hub and the shaft are subjected to a relatively high moment loading. Trim conditions, which need a rotor produced moment to overcome cg-travel or slope landing condition, for instance, require an alternating first harmonic moment in the rotating system for the hingeless rotor, whereas in the case of an articulated rotor, because of the equivalence of cyclic control and blade flapping, only an inclination of the thrust would be necessary. For the trim requirements in forward flight there are nearly no differences for both rotor systems, because the cyclic control is needed to overcome an aerodynamic nonuniformity, to which the dynamic systems of the rotors are of minor importance.

Higher harmonic blade loads resulting from the flow conditions of forward flight produce alternating forces at the hub for both rotor systems, and in addition for the hingeless rotor moments at the blade root and the hub. For a dynamically well tuned hingeless rotor these higher harmonic moments are relatively low compared to the first harmonic moments needed for trim or flight maneuvers. Figure 3 shows a typical case with a pronounced first harmonic part. The higher harmonic loads are exciting vibrations; and a low vibration level requires a good dynamic design for the rotor itself and in combination with the fuselage, for all rotor systems. However, for the component sizing for fatigue of a dynamically welltuned hingeless rotor only the first harmonic loads are determining.

To get optimal dynamic characteristics for a hingeless rotor system, the stiffness of the rotor system, which means for the MBB-system the stiffness of the elastic blades, should not be too high. The stiffness is defined by the flexural stiffness which is the product of the modulus of elasticity E and the moment of inertia J representing the cross-sectional area of the blade. The same flexural stiffness EJ can be realized for instance, with a high E of steel and a small J which means a thin cross-section or a low E like that of fiberglass reinforced plastic material and a larger J or crossection. In both cases the same moment loading must be tolerable for fatigue. The relation of the modulus of elasticity is about 5 for the two materials, but the relation of the fatigue stress allowables is only about 2. Therefore, the fiberglass material is much better than steel for a hingeless attached rotor blade. The stress allowables of fiberglass material result in blades of nearly unlimited life. Steel would be a very poor solution and also aluminum because of the low stress allowables. Figure 4 illustrates these conditions. The differences in the material properties are even more pronounced if the essential notch factors will be considered.

The main section of a hingeless attached blade outside the attachment area should normally not have fatigue problems. The loads are lower, and there will be enough sectional area, because no weight-saving construction is necessary normally. For other reasons, i.e., for flight dynamical reasons of autorotational behaviour a certain mass of the blades (or better a certain moment of inertia around the rotor axis) is desirable. This will result in favourable conditions for the stresses.

The typical fatigue problems of helicopters are high-cycle fatigue. The components of the dynamic system are loaded at frequencies which are multiples of the rotor speed with the highest loads in the first harmonic for the hingeless rotor, which means 7 cycles per second for the BO 105. This loading rate represents 25 million load cycles per thousand operating hours. Figure 5 shows a stress history of a typical helicopter flight on the blade root end. The helicopter components experience a relatively high frequency of cyclic loading. The high frequency cyclic loading is normally caused by bending moments. The low-cycle loads, which are the start-stop cycles, are resulting out of centrifugal forces. Their loading rate is with about 1 ÷ 5 000 cycles per 1 000 hours of flight low compared to the high-cycle loads. In most cases the start-stop cycles, therefore are of no influence for fatigue. But there can be some rotor components which are mainly loaded by the centrifugal forces - for instance the torsion-tension bars, which have the function to unload the blade feathering hinges and the hub from centrifugal forces. Such components have to be sized to withstand the low-cycle fatigue loading. Figure 6 shows the typical loads history of such a low-cycle fatigue component.

3. DETERMINATION OF FATIGUE LIFE

Most components of the dynamic system of helicopters are still today safe-life components. Only some of them are using failure-warning systems, others try to make use of some fail-safe features. Nevertheless, the basic fatigue problem of helicopters remains the problem of the prediction of safe lifetimes for the components which allow to remove the parts at a service-time with extremely remote probability of fatigue failure.

The prediction of fatigue life necessitates the knowledge of the load spectrum to which a part is exposed and of the fatigue strength of the part itself, see Figure 7. Load spectrum and fatigue strength are both statistical, therefore also fatigue life is a statistical quantity. The overall situation of the methods in prediction of fatigue life, especially for high-cycle loaded components, is not at all satisfactory. The more simple methods as for instance cumulative damage calculations with Miner's theory, which can easily be used in practice, are inaccurate and problematic; the more accurate methods using a direct load spectrum testing are only limited suited for practical use, because there is not enough information about loads in the early steps of development. In addition there will be difficulties if the spectrum has to be changed, and last but not least the expenditure for testing and of time is extremely high. Therefore, still today, the cumulative damage theories are mostly used in the helicopter industry. It is not intended here to discuss the differing methods - their advantages and disadvantages, only the situation from the viewpoint of the practical engineer will be illustrated.

The theories of cumulative damage allow the separation of the problem of fatigue strength of the components from the problem of load spectra. The connection is only done by a simple calculation. To determine the fatigue component strength no information is needed of the real load spectrum in service; and there is no problem in the calculation of lifetime if the loads are changing.

The determination of the fatigue strength is normally be done by S-N testing of full-scale components. Figures 8 ÷ 10 show typical results of such bench fatigue tests for the components blade, hub, and rotor shaft of the BO 105. The mean strength line (meaning 50% survival) and the line for 99.9% survival are shown. Since fatigue strength is of statistical nature a relatively large number of tests has to be used in establishing reliable S-N curves. Additional information, which is helpful in the statistical relations, can be taken from small specimen tests, i.e., information about the general shape of the S-N curve and about standard deviations (Ref. 7, 8).

In the figures the lines of 99.9% survival probability are compared with flight loads. All loads of normal level flight conditions are below fatigue endurance, only some loads of extreme maneuvers or extreme slope landing conditions will be somewhat higher. The situation is about the same for all three components. This means that blade, hub and rotorshaft are structurally well matched in design. Since the most frequent loads are the relatively low loads of the normal flight conditions, long fatigue lives can be expected.

In using Miner's theory of linear cumulative damage only such loads will have an effect on fatigue which are higher than the fatigue endurance loads. Considering cumulative damage theories to be sufficiently reliable, the whole fatigue problem is reduced to the problem of the prediction of the high loads and their frequency of extreme flight conditions, and that is the real problem. It can be stated that the uncertainty in this prediction is much higher than that of the methods of fatigue life prediction. The higher expenditure of improved fatigue life prediction methods, as for instance loads spectrum testing, would only make sense if load spectra of high reliability, especially in the range of the higher loads, could be established.

4. MISSION LOAD SPECTRA OF A LIGHT HELICOPTER

To establish mission load spectra, it requires the knowledge about all operational missions and flight profiles, and the determination of the loads for all flight conditions. The prediction of the loads maybe a problem for some extreme flight conditions, but the main loads, which are necessary for structural design, can be determined with adequate accuracy during all stages of the development of a helicopter. In the design phase, the loads will be calculated. The analytical models, both for stationary and instationary flight conditions, are sufficient to give enough information about the loads for nearly all conditions to be considered. The situation in loads prediction methods was discussed in an AGARD Specialists Meeting (Ref. 9), therefore it is not necessary to discuss the methods here. Also in the case of the hingeless rotor helicopter there are no problems. For the hingeless rotor the first harmonic rotor loads are normally most important, and these loads can be predicted with relatively high accuracy (Ref. 10). As already shown in Figure 3, there is good correlation between theoretical and measured values for level flight conditions. But also for extreme maneuvers, the correlation is sufficient as Figure 11 illustrates. During the flight testing phase, and for final qualification and certification of a helicopter, measured flight loads will be available. If there was adequate work in the previous stages of development, only minor corrections for the loads will be necessary. The determination of loads for specified conditions - analytically or experimentally - is in the line of rational work for engineers. The forces and moments are in confirmity with physical laws, and there is no necessity for speculation.

Quite different is the situation for the complete mission load spectrum which is the combination of the loads of all flight conditions to be considered. The first question is, which conditions of flight should be considered. The flight envelope which has to be established during the development generally gives the basic information. The demonstrated flight envelope, together with the regulations and restrictions of the flight manual, give the scope of all possible flight conditions determining both ultimate and fatigue loads. The next questions are, which conditions of flight will be probable, how much time will a helicopter fly in different conditions of flight, which is the distribution of the determining parameters, as for instance speed, altitude, and gross weight. For such questions there are no rational answers. The engineer has to leave the rational laws, and has to rely on speculation. The mission profile is usually established from considering all probable uses of a helicopter. Historical experience with helicopters is most helpful. Experience has resulted in standard mission spectra, which will be used for certification. Figure 12 shows the wellknown FAA mission spectrum (Ref. 11). The service-time is distributed to different flight conditions. For the design of most helicopters this or slightly modified spectra are used. There is no doubt, that different operational requirements will really result in different mission distributions. But as long as the certification of a helicopter allows to use it for all operations which can be flown within the qualified flight envelope, which is common practice for civilian certification, the same mission load spectrum has to be used independent of the effective future operational usage. Such a design and qualification philosophy is adequate for civilian, especially light helicopters, which will normally be used in utility operation with a large variety of mission, but the situation seems to be somewhat different for pure military application. Of course, there will be operational requirements as for instance for the L^H, liaison and training helicopters which will result in about the same mission load spectra as for civilian helicopters, but for special task helicopters, as for instance antitank or gunship helicopters, special mission spectra should also be established. It is generally possible to use for different versions of a military helicopter also different mission load spectra depending on the special operational requirements. But it is doubtful if such a procedure would be practical. Therefore it should be restricted to those cases only, which would result in an extreme reduction of service-life if the most severe operational loads would be considered for the standard helicopter. Different service-lives may be the result if a helicopter has to fulfill new mission requirements which were not within the primary requirements for the development; for instance, if a utility helicopter will be modified later on to a gunship-helicopter. In the future, such mission modifications should be less critical, because there is a pronounced tendency towards increased power and greater agility for new helicopters, especially for light and medium size helicopters which have to do most of their operation under extreme nap-of-the-earth flying. This requires consideration of the full flight and maneuver range for the structural and the fatige design.

The determination of the complete load spectrum of a helicopter has to be considered as a multidimensional statistical problem. The time distribution of the flight profile, as already mentioned, is found in a mixed procedure of operational considerations, resorts to tradition or experience, and probabilistic or speculative considerations. But this proportionate distribution of the different flights is only the first step. In a next step, the load distributions of the different flights have to be determined. The main parameters, which have to be considered are gross weight, cg-position, atmospheric conditions, and the factors defining the agility of maneuvers. The procedure again is the same. Operational consideration will give some support, but in this case probabilistic or speculative reasoning is also necessary, especially for the maneuvers.

For the helicopter BO 105, the FAA flight profile with the percentage of occurrence for the different flight conditions formed also the basis for the determination of the complete loads spectrum. Only minor modifications have been adopted. For instance, the high per cent occurrence of autorotation was reduced, because the BO 105 is twin-engined, and there will be no necessity for long autorotational flight. The per cent occurrence for some extreme maneuvers, for speeds higher than V_{NE} (never exceed speed), and for sideward and rearward flight was increased. In addition, hoist operations and slope landing conditions were introduced, because they may result in high loads for helicopters with a hingeless rotor system. The established flight profile is a mixed speed and maneuver spectrum. It contains no information about the spectra of grossweight and cg-position or atmospheric conditions. Therefore, the basic profile needs a further split. The result as it is used for the BO 105 is shown in Figure 13.

Now, the flight profile seems to be well enough defined to determine the loads spectrum; however an uncertainty about the severity of the maneuvers in real flight remains. It is the question of the load factors in pull-up maneuvers and turns, and also of the pilot's behaviour in maneuvers which may result in high loads without creating high load factors as for instance control reversal maneuvers. It should be noted here, that the spectrum of maneuver load factors seems not to be sufficient for establishing a complete maneuver spectrum. The maneuver loads spectrum will be highly influenced by the characteristics of the helicopter and the type of its rotorsystem. The hingeless rotor helicopter offers a high agility in maneuvers, and the structural design should be done in a way which allows to use its full potential. For the BO 105 the maneuver loads were determined in a special flight test program with several pilots, which were instructed to use the full maneuver potential for all segments of the complete flight program without any restrictions. Of course, these flight tests had been done fully instrumented. Their results, together with the results of the general loads flight program due to the established flight profile formed the complete loads spectrum. Figure 14 to 16 show typical results for rotorblade root moments (flapwise and inplane), and for the rotorshaft. Only measured loads were considered for these spectra. The frequencies of the different load grades were determined by use of special counting methods.

It is not intended to start a discussion about the shape of the distribution, e.g., whether it should be a normal distribution or not. Of course, it should be possible by modification of the flight profile to influence the distribution of the steps in the spectra to make it more smooth. But it seems to be worthwhile to mention that the distribution may slightly differ for the different components, and load conditions. The relatively high frequencies of the highest loads are the result of highly conservative assumptions for the extreme maneuvers and extreme slope landing conditions. The highest loads, for instance up to 11 900 Nm for the rotorshaft are beyond the range as it is defined in the flight manual. All BO 105 helicopters are equipped with a loads guide indicator showing the rotorshaft moment. The range of normal flight is up to 6 900 Nm, loads above 9 800 Nm (1 000 mkp) are not allowed. Experience has shown that such an indicator is helpful for some special maneuvers, as extreme slope landings; and in addition it provides long term data because all exceedings will be recorded. The admitted range is sufficient to do all maneuvers, and exceedings are extremely remote. Because of the correlation between the loads of the rotorshaft, the rotorhub, and the blade unrecorded overstressing of all components will be avoided. This maneuver guide indicator is no real flight restriction, even for very extreme maneuvers. The BO 105 demonstrated loops and rolls, and these maneuvers can be flown within the admitted range.

In Figures 14 to 16 the fatigue endurance limits of the components are indicated. These limit lines are at relatively high load grades. All the loads of level flight conditions, and also most of the maneuvers necessary for normal flight are below fatigue endurance, meaning that they do not contribute to fatigue if cumulative damage theories are used. For the BO 105 only the extreme conditions of

right and left turns,

rolling pull-outs,

longitudinal reversal,

cyclic and collective pull-ups,

111 per cent V_{NE},

autorotational flight,

slope landings and starts

may result in loads higher than fatigue endurance. All other conditions, and their per cent of occurrence are not of importance for fatigue evaluation. Therefore, not to much effort should be applied for further improvements of the flight profile as long as the nondamaging portions are concerned.

As the BO 105 was primarily designed for civilian operation and certification, it has to be checked if military operational requirements for LOH-type and antitank missions will necessitate a remarkable modification. Longterm military evaluation, and special military flight programs simulating military tactics under extreme nap-of-the-earth flying have proven that the highly conservative mission loads spectrum of the civilian BO 105 is also sufficient for the military version Measurements indicate that in the tactical environment of nap-of-the-earth flying rapid and extreme maneuvers will be necessary with

maximum load factors	up to 2.5 g,
roll angles	up to 80 degrees,
rolling speeds	up to 50 degrees per second,
pitch angles	up to 40 degrees,
pitching speeds	up to 40 degrees per second.

The rapid rate of change of those parameters up to their extremes will, of course, result in high loads. But these maneuvers are all within the normal flight envelope, and the loads within the normal loads spectrum. All maneuvers can be performed within the conservatively admitted range of the loads or maneuver guide indicator.

5. IMPACT OF MODIFIED MISSION SPECTRA ON FATIGUE

Even if only a few flight conditions out of the complete mission profile will contribute to fatigue of the components, the sensitivity to changes in their percentage of occurrence has to be checked. A brief study which should give a better feeling for the overall situation was done (Ref. 12). This study was based on the assumption of linear damage accumulation following Miner's theory and on some probabilistic relations. The mission profile was modified in such a manner that all the different conditions of flight, also the related loads remained the same, only the percentage of their occurrence was changed. For this case it is very easy to reevaluate fatigue life. Often it will be sufficient to consider only one flight condition which exerts a prevailing influence neglecting the other conditions. It can be shown that a simple relation for the resulting fatigue life L can be derived as

$$L = \frac{c_1}{\alpha + c_2}$$

in which α is the relative occurrence of the modified flight condition; c_1 and c_2 are constants depending on the fatigue strength of the component to be considered, on the unchanged portion of the loads spectrum, and of course on the relation of the loads (see Appendix). The fatigue lives of the critical BO 105 components, which are blade, hub, and shaft, are, as already mentioned, influenced only by a few flight conditions. The data necessary for the calculation of the fatigue lives considering these damaging flight conditions with the obtained results are shown in the tables of Figures 17 to 19. With the per cent occurrence of the standard spectrum the calculated lifetimes are more than

5 000 hours for the rotor shaft,

11 000 hours for the rotor hub,

22 000 hours for the blades.

The tables in Figures 20 to 22 illustrate the influence of a changed percentage of occurrence of the different flight conditions. Most effective for the fatigue life of the rotor shaft are the maneuvers "Right and Left Turns". The certification of the BO 105 was done with 6 per cent occurrence for this flight condition. Modifying the occurrence to 3 per cent would increase whereas a modification to 12 per cent would reduce the fatigue life factor to 1.7 and 0.55 respectively. The modification effects of the other flight conditions are of less importance, as long as their percentage of occurrence is comparably small. For the rotor hub and the blades the situation is about the same. A relatively high sensitivity as the result of right and left turn maneuvers raises the question whether a further, more detailed consideration could be necessary; and a further study will be done. As for all maneuvers, also for the right and left turns very extreme conditions have been considered to determine the loads spectrum, therefore, in the next step only the loads spectrum of the right and left turns has to be reviewed. Most probably, the percentage of most extreme turns with maximum bank angles is to high in the turn maneuver spectrum. Perhaps, a more realistic distribution would result in a situation with a less pronounced sensitivity. This brief study shows that modified spectra can be of an immense effect on fatigue lives, but there are simple methods which allow the quick assessment of these effects. If there is enough information about the most damaging conditions of the flight profile - and this information should be available for all helicopters - expecially the detailed load spectra of these conditions should be further reviewed to avoid unnecessary high conservatism. If the operational requirements are changing or if some helicopters have to do special missions, resulting in widely changed loads spectra, the changed component lives can be calculated with the simple procedure as shown previously.

6. CONCLUDING REMARKS

Although the methods for the determination of loads spectra and for fatigue life evaluation have been all but satisfactory all the years, the overall situation is not as bad as it might seem at a glance. Helicopter technology is rapidly progressing, and the service-lives of all components are increasing. Requirements for new military helicopters ask for lifetimes of 5 000 or 10 000 operating hours of the dynamically loaded components, and these requirements can be materialized nowadays. The era of the older helicopter suffering under fatigue problems with low service-lives for the components has gone. Of course, this has become feasible by the enormous progress in the general technology with improved systems and materials, and with a better understanding, but also with the problematic methods of fatigue prediction as they are available for practical use. From the standpoint of airworthiness the helicopter is a safe vehicle; the still remaining fatigue problems are more or less due to unforeseen effects. The uncertainty in establishing flight profiles, and the problematic nature of the fatigue life prediction are compensated by very conservative assumptions.

The more and more pronounced tendency of the modern military helicopter towards higher power and improved performance, towards higher agility and qualification for extreme nap-of-the-flying, towards improved survivability and reliability, and towards reduced maintenance costs will further improve the situation. To fulfill the high requirements of maneuverability it is necessary to use the full flight potential of the helicopter, and the sizing of the components has to be adequate. The requirements for crashworthiness, for reduced vulnerability, and for damage tolerance ask for an adequate design philosophy. Fail safe devices and materials, which are, in addition to excellent strength properties, damage tolerant, insensitive to notches, and free of corrosion, as glassfiber reinforced plastic material, are getting more and more important.

The hingeless rotorsystem with fiberglass rotorblades offers a high potential although the dynamic loads are higher for some components compared to the articulated rotor. There is no problem in sizing its components; the theoretical determination of the loads is even easier, because the critical loads are mainly firstharmonic. The glassfiber rotorblades which are virtually failsafe are the only ones which achieve the requested fatigue lives of the future helicopter in today's operation.

The requested long lifetimes result in a design philosophy with the adequate sizing that all loads of level flight conditions and of all maneuvers with frequent occurrence should not contribute to fatigue damage. Mainly extreme maneuvers of infrequent occurrence, considering the full flight potential, will decide fatigue life. The uncertainty of the mission spectra will always remain, but only the spectra of extreme conditions will be of interest. These extreme conditions maybe considered with conservative assumptions without major penalties. The practical methods seem to be sufficiently safe, however, they should be continuously improved to avoid oversizing. More emphasis should be given to the spectra of extreme maneuvers, and also the methods to determine fatigue life under these conditions should be better established.

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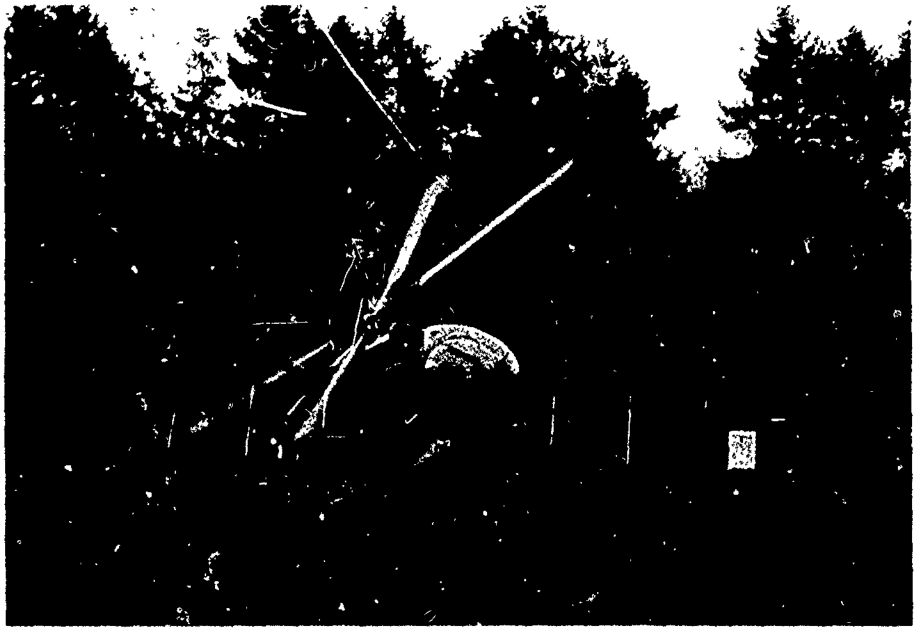


Fig. 1 Helicopter MBB - BO 105

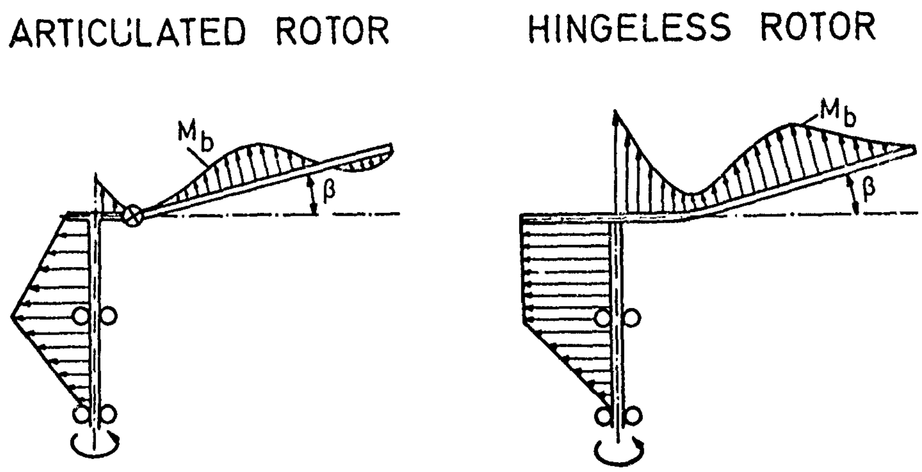


Fig. 2 Loading Situation of a Hingeless Rotor in Comparison to an Articulated Rotor

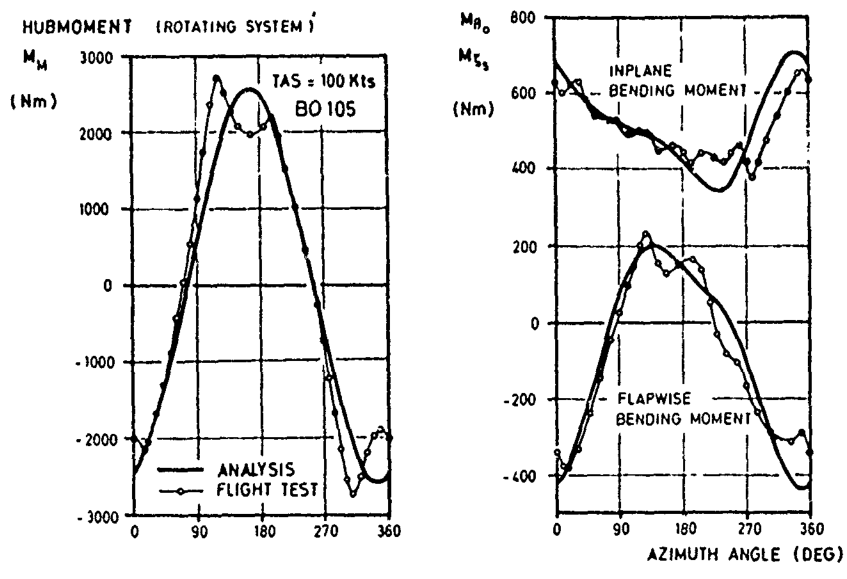


Fig. 3 Hub and Blade Root Moments in Forward Flight

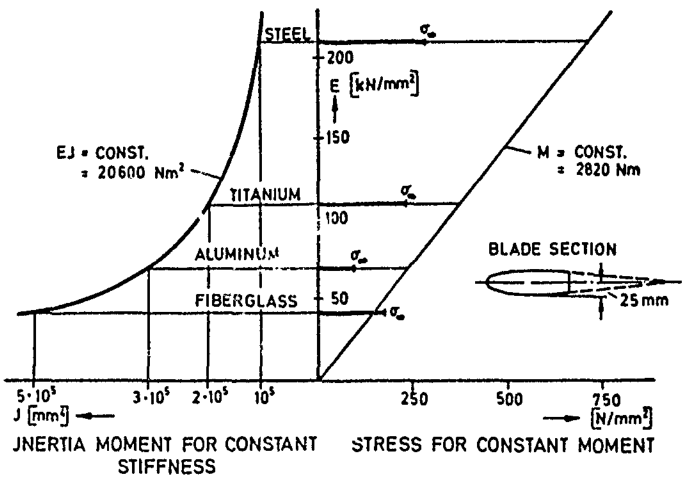


Fig. 4 Influence of Material Properties

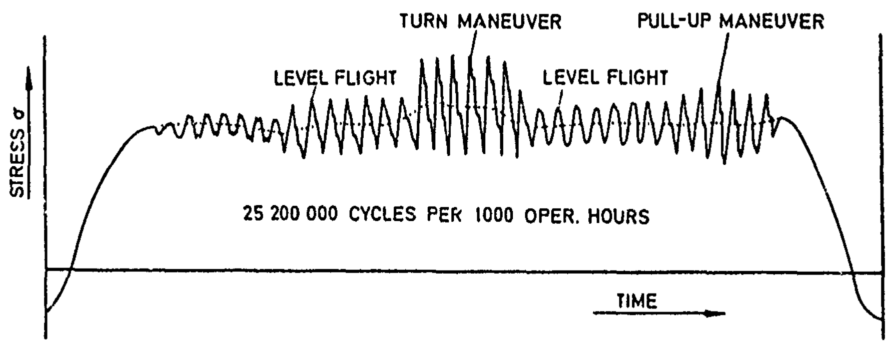


Fig. 5 Typical Stress History on a Rotorblade Root End

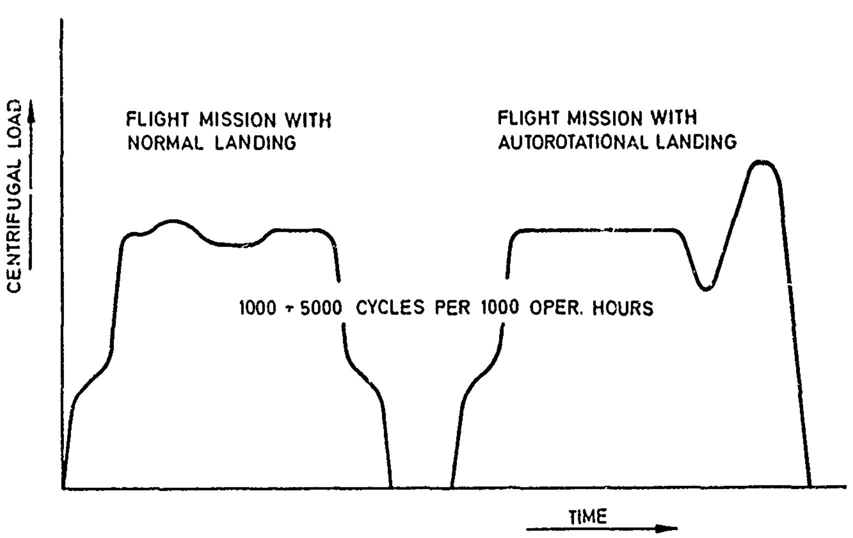


Fig. 6 Start-Stop Cycle History of Torsion-Tension Bar

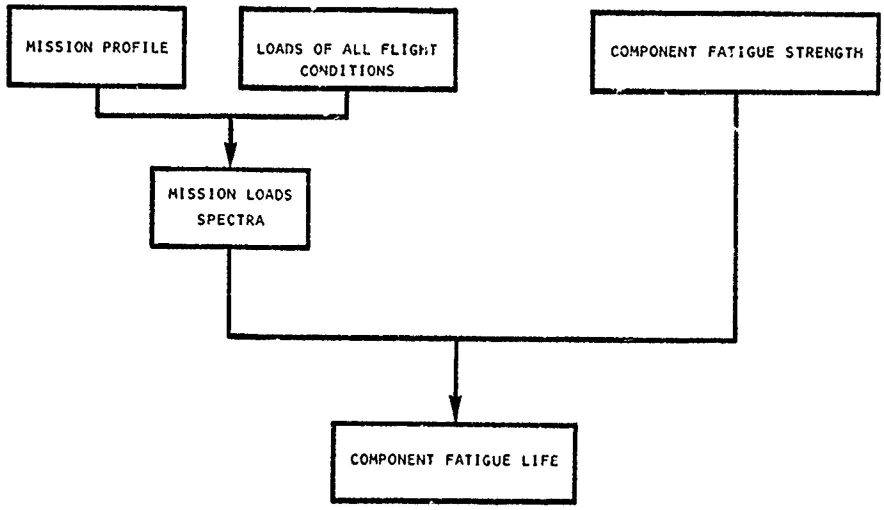


Fig. 7 Determination of Fatigue Life

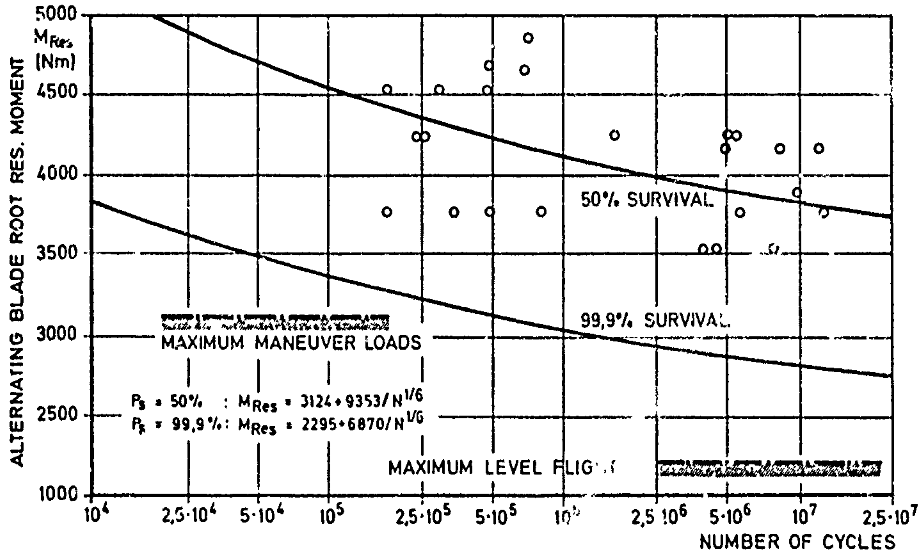


Fig. 8 Fatigue Strength of the Blade Spar

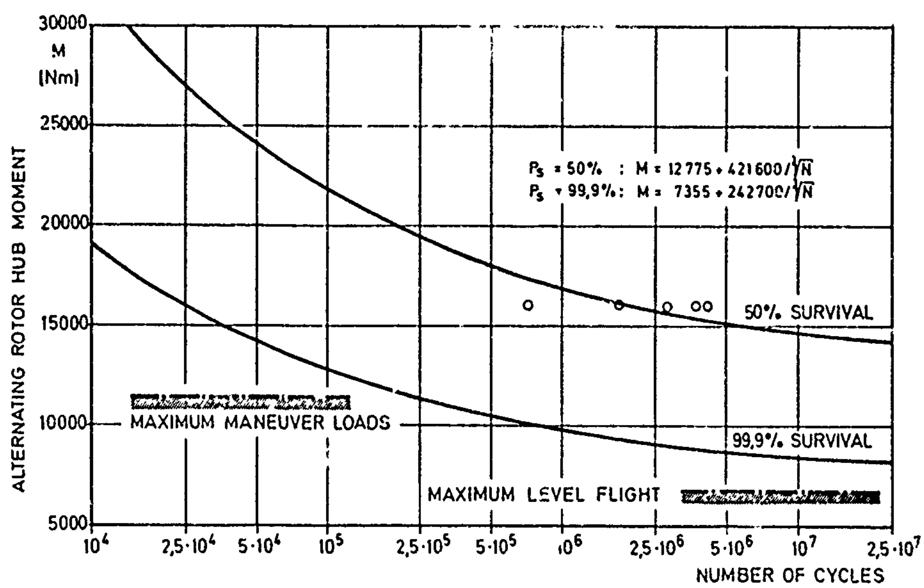


Fig. 9 Fatigue Strength of the Rotor Hub

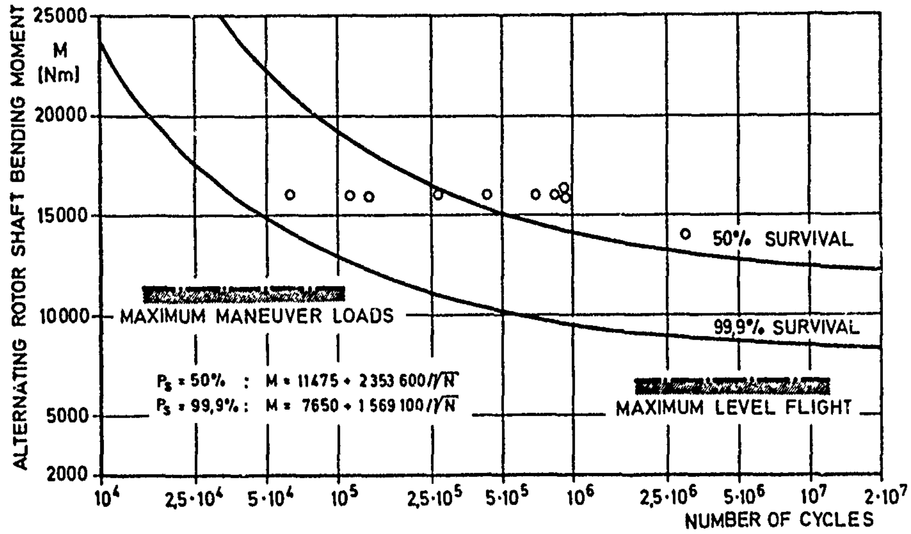


Fig. 10 Fatigue Strength of the Rotor Shaft

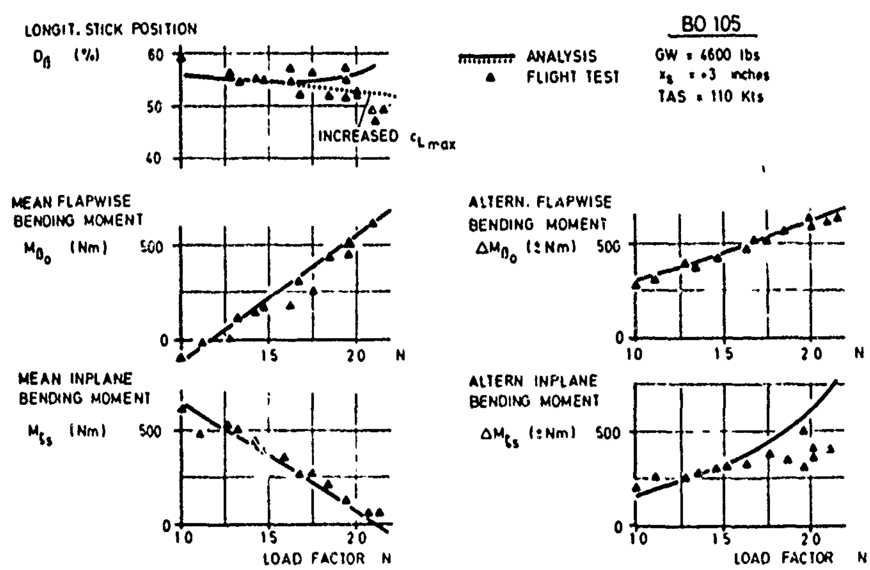


Fig. 11 Rotor Loads in Maneuver Flight

GROUND CONDITIONS:	
Rapid increase of rpm on ground to quickly engage clutch	0.5
Taxiing with full cyclic control	0.5
Jump takeoff	0.5
HOVERING:	
Steady hovering	0.5
Lateral reversal	1.0
Longitudinal reversal	1.5
Rudder reversal	1.0
FORWARD FLIGHT POWER ON:	
Level flight, 20 per cent V_{NE}	5.0
Level flight, 40 per cent V_{NE}	10.0
Level flight, 60 per cent V_{NE}	18.0
Level flight, 80 per cent V_{NE}	18.0
Maximum level flight (but not greater than V_{NE})	10.0
V_{NE}	3.0
111 per cent V_{NE}	0.5
Right turns	3.0
Left turns	3.0
Climb (max continuous power)	4.0
Cyclic and collective pull-ups from level flight	0.5
Change to autorotation from power-on flight	0.5
Partial power descent (including condition of zero flow through rotor)	2.0
Landing approach	3.0
Lateral reversals at V_H	0.5
Longitudinal reversals at V_H	0.5
Rudder reversals at V_H	0.5
Climb (takeoff power)	2.0
AUTOROTATION-POWER OFF:	
Steady forward flight	2.5
Right turns	1.0
Left turns	1.0
Lateral reversals	0.5
Longitudinal reversals	0.5
Rudder reversals	0.5
Cyclic and collective pull-ups	2.0
Landings (including flares)	2.5
	100.0

Fig. 12 Flight Maneuvers - Per Cent Occurrence

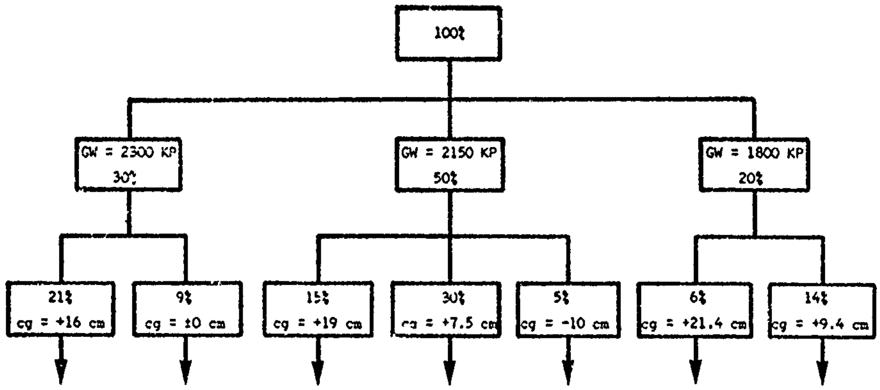


Fig. 13 Split of Flight Profile

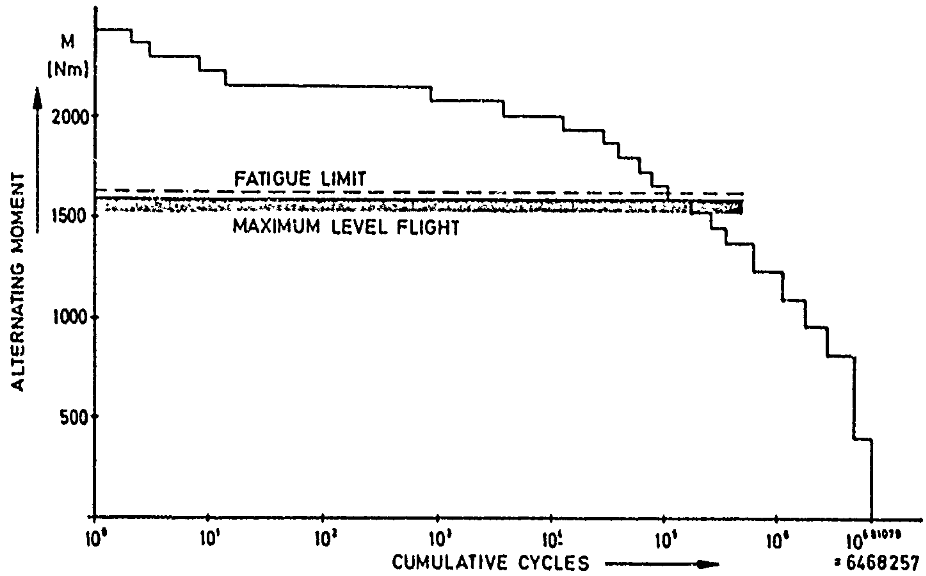


Fig. 14 Loads Spectrum for the Blade Flapping Moment

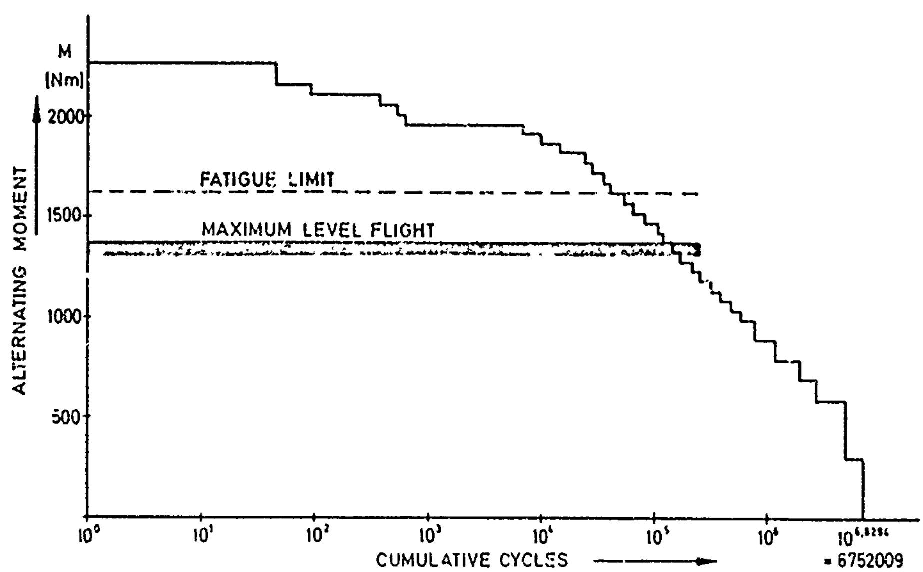


Fig. 15 Loads Spectrum for the Blade Inplane Moment

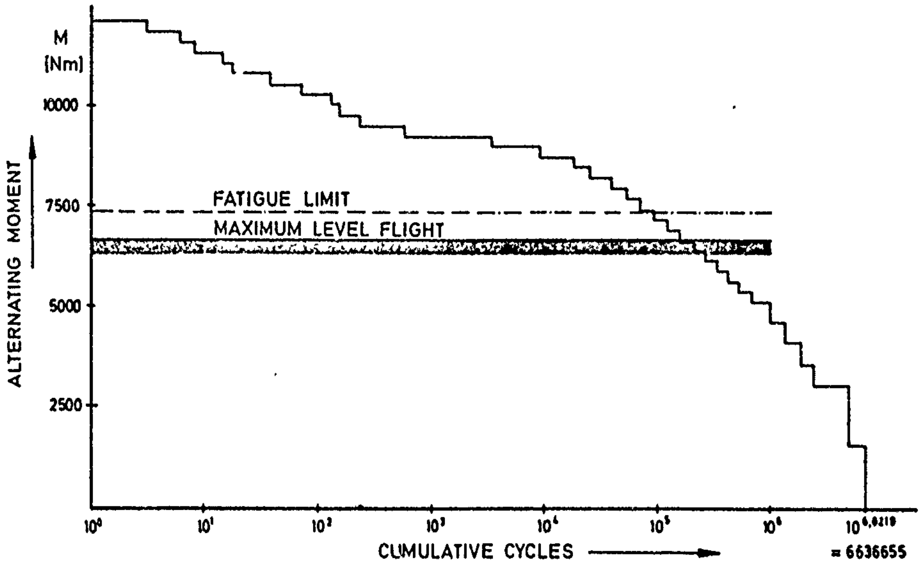


Fig 16 Loads Spectrum for the Rotor Hub & Shaft

FLIGHT CONDITION		RIGHT AND LEFT TURNS	SLOPE LANDINGS AND STARTS	ROLLING PULL-OUTS	LONGITUDINAL REVERSALS	111 PER CENT V_{NE}
NUMBER		1	2	3	4	7
MEDIAN	$\hat{M}_i$	216.7	196.5	23[illegible].1	198.2	199.4
PER CENT OCCURRENCE	a_i	6.00	.25	.25	.5	1.5
LIMIT LOAD RATIO	$\frac{M_\infty}{\hat{M}_i}$	1.080	1.191	1.013	1.181	1.174
DAMAGE INTEGRAL	A_i	.00002761	.0000027	.0000892	.0000034	.00000395

FATIGUE LIFE IN HOURS $L = \dfrac{46939}{28.590a_1 + 1.5543a_2 + 135.883a_3 + 2.0611a_4 + 2.4828a_7} = 22282$ HOURS

ALLOWABLE FATIGUE DATA: (FOR 99.9% SURVIVAL) M_∞ = 234 MKP C = 701 MKP
X = 1/6 (FOR GLASS FIBER REINFORCED EPOXY)

Fig. 17 Fatigue Life of the Blade Spar

FLIGHT CONDITION		RIGHT AND LEFT TURNS	SLOPE LANDINGS AND STARTS	ROLLING PULL-OUTS	LONGITUDINAL REVERSALS	CYCLIC AND COLLECTIVE PULL-UPS	AUTOROTATION FLIGHT
NUMBER		1	2	3	4	5	6
MEDIAN	$\hat{M}_i$	725.2	768.2	713.0	668.5	653.5	612.0
PER CENT OCCURRENCE	a_i	6.00	.25	.25	.5	.2	1.4
LIMIT LOAD RATIO	$\frac{M_\infty}{\hat{M}_i}$	1.034	.976	1.052	1.122	1 148	1.225
DAMAGE INTEGRAL	A_i	.00195	.00375	.00155	.00060	.00041	.000115

FATIGUE LIFE IN HOURS $L = \dfrac{601622900}{743717a_1 + 1700019a_2 + 561824a_3 + 179248a_4 + 114425a_5 + 26360a_6} = 11621$ HOURS

ALLOWABLE FATIGUE DATA (FOR 99.9% SURVIVAL) M_∞ = 750 MKP C = 24750 MKP
X = 1/3 (FOR TITANIUM ALLOY)

Fig. 18 Fatigue Life of the Rotor Hub

FLIGHT CONDITION		RIGHT AND LEFT TURNS	SLOPE LANDINGS AND STARTS	ROLLING PULL-OUTS	LONGITUDINAL REVERSALS	CYCLIC AND COLLECTIVE PULL-UPS	AUTOROTATION FLIGHT
NUMBER		1	2	3	4	5	6
MEDIAN	$\tilde{M}_i$	725.2	768.2	713.0	668.5	653.5	612.0
PER CENT OCCURRENCE	α_i	6.0	.25	.25	.5	.2	1.4
LIMIT LOAD RATIO	$\frac{M_\infty}{\tilde{M}_i}$	1.075	1.014	1.094	1.167	1.194	1.275
DAMAGE INTEGRAL	A_i	.0051	.0101	.00405	.0016	.00107	.00031

FATIGUE LIFE IN HOURS

$$L = \frac{1015871}{2682\alpha_1 + 5960\alpha_2 + 2059\alpha_3 + 715\alpha_4 + 457\alpha_5 + 116\alpha_6} = 5430 \text{ HOURS}$$

ALLOWABLE FATIGUE DATA: M_∞ = 780 MKP C = 160 000 MKP
(FOR 99.9% SURVIVAL) X = 1/2 (FOR STEEL)

Fig. 19 Fatigue Life of the Rotor Shaft

RIGHT AND LEFT TURNS:	α_1	.03	.06	.12
$L = \frac{1641.77}{\alpha_1 + .0136808}$	L	37586	22282	12281
	$L/L_{.06}$	1.687	1.000	.551
SLOPE LANDINGS AND STARTS:	α_2	0	.0025	.005
$L = \frac{30199}{\alpha_2 + 1.352798}$	L	22323	22282	22241
	$L/L_{.0025}$	1.002	1.000	.998
ROLLING PULL-OUTS:	α_3	0	.0025	.005
$L = \frac{345.44}{\alpha_3 + .013003}$	L	26566	22282	19188
	$L/L_{.0025}$	1.192	1.000	.861
LONGITUDINAL REVERSALS:	α_4	0	.005	.010
$L = \frac{22773.5}{\alpha_4 + 1.01705}$	L	22392	22282	22174
	$L/L_{.005}$	1.005	1.000	.995
111 PER CENT V_{NE}:	α_7	0	.015	.030
$L = \frac{18905}{\alpha_7 + .83345}$	L	22683	22282	21907
	$L/L_{.015}$	1.018	1.000	.983

Fig. 20 Influence of Modified Spectra for the Blade Spar

RIGHT AND LEFT TURNS:	α_1	.03	.06	.12
$L = \frac{808.94}{\alpha_1 + .00961218}$	L	20421	11621	6241
	$L/L_{.06}$	1.757	1.000	.537
SLOPE LANDINGS AND STARTS:	α_2	0	.0025	.005
$L = \frac{353.89}{\alpha_2 + .0279536}$	L	12660	1162[illegible]	10739
	$L/L_{.0025}$	1.089	1.000	.924
ROLLING PULL-OUTS:	α_3	0	.0025	.005
$L = \frac{1070.84}{\alpha_3 + .089650}$	L	11945	11621	11314
	$L/L_{.0025}$	1.028	1.000	.974
LONGITUDINAL REVERSALS.	α_4	0	.005	.010
$L = \frac{3356.36}{\alpha_4 + .283827}$	L	11825	11621	11232
	$L/L_{.005}$	1.018	1.000	.983

Fig. 21 Influence of Modified Spectra for the Rotor Hub

RIGHT AND LEFT TURNS:	α_1	.03	.06	.12
$L = \dfrac{378.75}{\alpha_1 + .0097545}$	L	9527	5430	2919
	$L/L_{.06}$	1.755	1.000	.538
SLOPE LANDINGS AND STARTS:	α_2	0	.0025	.005
$L = \dfrac{170.44}{\alpha_2 + .0288898}$	L	5900	5430	5029
	$L/L_{.0025}$	1.087	1.000	0.926
ROLLING PULL-OUTS:	α_3	0	.0025	.005
$L = \dfrac{493.42}{\alpha_3 + .0883735}$	L	5593	5430	5284
	$L/L_{.0025}$	1.028	1.000	.973
LONGITUDINAL REVERSAL:	α_4	0	.005	.010
$L = \dfrac{1420.67}{\alpha_4 + .2566446}$	L	5536	5430	5328
	$L/L_{.005}$	1.020	1.000	.981

Fig. 22 Influence of Modified Spectra for the Rotor Shaft

APPENDIX

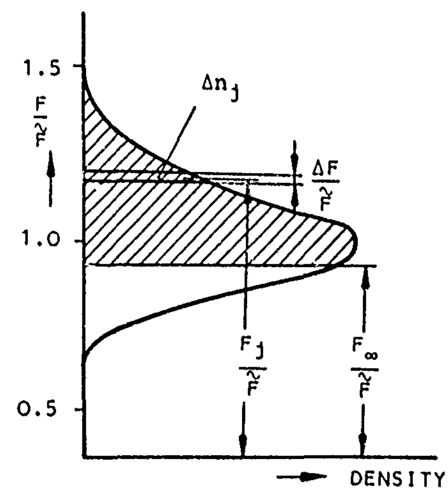
EVALUATION OF LOAD SPECTRA AND FATIGUE LIVES BY STATISTICAL METHODS

Approaching the frequency curve of each flight condition by a logarithmic normal distribution and using the Miner hypothesis of damage accumulation a practical method to evaluate load spectra and fatigue lives is obtained. To get the most possible simplicity a special representation for scatter and size of loads is necessary (Ref. 12).

There are two items by which a normal distribution is completely defined: The median $\tilde{F}$ and the standard deviation s or scatter factor ε of the loads. Statistical analysis of flight measurement data of the helicopter BO 105 showed, that within certain limits the same standard deviation for all interesting flight conditions may be used:

$$s = \log \varepsilon = .06000; \qquad \varepsilon = 1.14815 .$$

Another simplification is obtained by introducing a dimensionless parameter dividing loads by their median. One single frequency curve then is valid for all flight conditions (see Figure). The damage of a multilevel loading is obtained by the Miner Rule as the sum of cycle ratios. The load range is subdivided into a number of intervals, each of the same width ΔF; the number of cycles Δn belonging to each level is given by the enclosed area of the frequency curve. For interval or level number j of the flight condition number i the following expression can be deduced from the logarithmic normal distribution:



$$(1) \qquad \Delta n_{ij} = \frac{a}{\left(\frac{F_{ij}}{\tilde{F}_i}\right)^{1 + b \log \frac{F_{ij}}{F_i}}}$$

$$\text{where} \qquad a = \frac{\Delta F}{\tilde{F}_i} \Big/ \sqrt{2\pi}\, s \ln 10$$

$$\text{and} \qquad b = 1/2\, s^2 \ln 10 .$$

The sum of all Δn_{ij} of the flight condition corresponding to the whole area under the distribution curve is one event. As only loads in excess of the part's fatigue limit produce damage, the lower part of the curve is without influence. Moreover loads exceeding a certain value given by the load ratio $F_{max}/\tilde{F} = \varepsilon^{3.090} = 1.5325$ have a probability of occurrence so low (1/1000) that they practically do not exist. Only loads between these two limits (shaded area in above Figure) are affecting fatigue life.

The denominator of the cycle ratio by which damage is evaluated is the number of cycles-to-failure of the load level. Using again dimensionless parameters the following equation deduced from the konwn Stromeyer formula for single level loading may be used:

$$(2) \qquad \frac{1}{N} = \left(\frac{F/\tilde{F} - F_\infty/\tilde{F}}{C/\tilde{F}}\right)^{1/x} .$$

The fatigue limit F_∞ and the material constant C are reduced by a tolerance factor in order to obtain allowables for a 99.9 per cent survival probability.

The fractions of damage produced by the load level number j of the flight condition number i is given by the ratio:

$$(3) \qquad \frac{\Delta n_{ij}}{N_j} = \frac{a\left(\frac{F_{ij}}{\tilde{F}_i} - \frac{F_\infty}{\tilde{F}_i}\right)^{1/x}}{\left(\frac{F_i}{\tilde{F}_i}\right)^{1 + b \log \frac{F_{ij}}{\tilde{F}_i}}} \Big/ \left(\frac{C}{\tilde{F}_i}\right)^{1/x}$$

The sum of all fractions taken between the limits $F_\infty/\tilde{F}_i$ and 1.5325 is the total damage of the flight condition number i. The $(C/\tilde{F}_i)^{1/x}$-fold value of this integral is only a function of the dimensionless fatigue limit $F_\infty/\tilde{F}_i$:

$$(4) \qquad A_i = \left(\frac{C}{F_i}\right)^{1/x} \Sigma \frac{\Delta n_{ij}}{N_j} = \Sigma \frac{a \left(\frac{F_{ij}}{\tilde{F}_i} - \frac{F_\infty}{\tilde{F}_i}\right)^{1/x}}{\left(\frac{F_{ij}}{\tilde{F}_i}\right)^{1 + b \log \frac{F_{ij}}{\tilde{F}_i}}}$$

where $a = .05775277$ and $b = 60.31866$ } for $s = .06000$ see equation (1).

The exponent x in equation (4) is a material constant, appropriate values are for steel 1/2, for titanium and aluminum alloys 1/3, and for GFE (glass fibre reinforced epoxy) 1/6. The integral A_i is the $(C/\tilde{F}_i)^{1/x}$-fold damage generated by one cycle of the flight condition number i. To obtain the damage accumulated during 1 hour of operation one has to multiply it by the number of cycles per hour. The main rotor frequency being 7 Hz a number of 25 200 (= 7 Hz • 3 600 sec) cycles is obtained. The damage for 1 hour operation is then:

$$(5) \qquad D_i = 25\,200 \frac{A_i}{(C/\tilde{F})^{1/x}}$$

The total damage of the mission profile consisting of a number of flight conditions, each having α_i per cent occurrence, is obtained by the following equation:

$$(6) \qquad D = \Sigma \alpha_i D_i = \frac{25\,200}{C^{1/x}} \Sigma \alpha_i A_i \tilde{F}_i^{1/x}$$

The life in hours is the reciprocal value of this total damage:

$$(7) \qquad L = \frac{C^{1/x} / 25\,200}{\Sigma \alpha_i A_i \tilde{F}_i^{1/x}}$$

By modification of the mission profile of a given aircraft whose flight performance and part sizes are maintained unchanged only the per cent occurrences are concerned. It is then easy to reevaluate fatigue life by the aid of equation (7).

Often it is sufficient to treat only the one flight condition which exerts a prevailing influence and neglecting the others. In this case equation (7) takes the simple form

$$(8) \qquad L = \frac{c_i}{\alpha + c_1}$$

in which α = per cent occurrence of modified flight condition

c_1, c_2 = constants obtained by application of equation (7).

HELICOPTER DESIGN MISSION LOAD SPECTRA

by

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INTRODUCTION

It is virtually axiomatic that fatigue is a criterion in the design of rotating machinery. It is equally axiomatic that fatigue is inherent in the structural design of any modern aircraft. When the helicopter was invented, however, it became clear that the fatigue problems associated with rotating machinery would be combined with the fatigue problems of aircraft structures and would dominate design, often to the exclusion of the usual criteria hitherto considered by the Engineer and Designer.

Why should this be so? In my view there are at least three main reasons. Firstly, there are many components in a helicopter where we have yet to achieve a 'Fail Safe' design concept, and therefore the 'Safe Life' approach with all its implications is essential to ensure a satisfactory and economic life. Secondly, in many of these components fatigue damage is accumulated at rotor order frequency as well as at frequencies corresponding to atmospheric turbulence, manoeuvre and flight by flight conditions leading to a most complex interaction of stresses over the full range of the S/N diagram. Last, but not least, the magnitude and occurrence of this fatigue loading is dependent upon the in service usage, which in the case of a helicopter has often been inadequately defined at the design stage and may change with the helicopters changed capability and usage. In this situation we have to use what we like to call 'Engineering Judgement' but the validity of this may be limited by our appreciation of changes in operational use.

I think that it would be of interest if I described briefly the various stages of design and substantiation of the fatigue loaded parts of a helicopter.

At the outset of a project our first requirement is to establish the spectrum of loading that the various components will be expected to withstand in their future service life. Now a load spectrum can be considered from two aspects, magnitude and occurrence. The latter may be expressed as a percentage of total flying time or as a number of occurrences per hour of flight. The magnitude of loading can be derived in the design stage from calculation or parametric read out from previous similar aircraft types. Occurrence however can only come from the best assessment of the in-service role usage that we can make using, perhaps, such standard manoeuvre spectra that we can find in, say, the British Civil Airworthiness Requirements (which were derived from the old CAM6 spectra) or from documents such as the U.S. Naval requirements of AR56.

The components are designed for strength on this basis and it soon becomes clear that in the case of rotating parts such as rotor hubs and blades and primary control components the fatigue considerations often swamp the normal everyday static stressing cases. Development fatigue testing is then carried out, often to a spectrum of loads based upon the design load spectra, and modified, when necessary, to lead towards the production definition. During this development phase, prototype aircraft are built and flown and measurements of stress are made in all the critical components throughout the flight envelope and in all manoeuvre conditions appropriate to the eventual service use of the aircraft. The measured loads are analysed and, in association with the assumed spectrum of manoeuvres, a production load spectrum is formed for each component.

We can immediately see that the critical link running through this procedure is the assumed manoeuvre spectrum. This manoeuvre spectrum is used at the design stage, throughout the development and production fatigue tests, and is still used in the final fatigue substantiation. It should be emphasised however that the fatigue substantiation procedures used over many years at Westlands have been well justified by the virtually negligible occurrence of catastrophic fatigue failure on aircraft in service. It is evident, however, that an increased knowledge of in service load spectra at the design stage can only lead to increased life or reduced weight either of which must be greatly beneficial to the operating costs of the aircraft.

Now we measure the loads in flight, we measure the fatigue strength by means of fatigue tests, then why don't we measure the occurrence of loads? Well, of course, we do, but it has to be on a synthetic basis as it would be impractical to wait for extensive tests on a new aircraft in an operational environment.

For the remainder of this paper, therefore, I would like to put forward a few thoughts upon the measurements that should be made and how we should make them. In addition I would like to discuss some of the trials that we have carried out from Westlands with their results, and finally where we are trying to go in the near future.

LOAD SPECTRUM MEASUREMENTS

If we wish to carry out a fatigue substantiation it is necessary to have a spectrum of load levels and their associated percentage times. What could be more easy? We only have to attach strain-gauges to all the appropriate places and tell the service user to go away and fly it! What then are the problems?

Let us look at the nature of the loads that we are trying to measure. Firstly, they are loads or stresses of a vibratory nature and moreover their frequency may be of the order of the rotational speed of the Main or Tail rotor. The vibratory magnitude of the load does not stay constant but varies with the flight conditions of the helicopter and the state of atmospheric turbulence. In addition, of course, 'steady' loads which may, or may not, be present and which are superimposed upon the vibratory load pattern, are themselves continually varying. For example, the centrifugal stress upon a main rotor blade is removed every time the rotor stops. In another case the mean drive torque in a tail rotor drive shaft fluctuates with the power demand from the tail rotor and is superimposed upon torsional resonancies in the transmission system.

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What is more, it is clear that a small sample obtained from a few minutes or even hours of flight does not give us the full statistical information from which we can derive a spectrum of occurrence in service. It is in fact necessary to take measurements over a very long period of time, perhaps hundreds of flying hours, to obtain a satisfactory statistical sample. Different aircraft used in widely differing roles by the various service users would need to partake in the programme.

The measurements themselves need to take place in a service environment, even in the front line if possible. Furthermore, any load measurements which may be made satisfactorily in the short term contractors trials using straingauges and sliprings or their equivalents pose a problem in the long term from a mechanical and electrical reliability point of view.

Finally, assuming that the measurements could be made, there is then a colossal task of recording, analysing and reducing the data from tape or trace to form usable spectra.

Now supposing all the problems surrounding the accumulation of this load data were solved, what have we achieved? We would certainly have a very accurate knowledge of the actual loads or stresses on certain components of the helicopter directly measured under service conditions and this no doubt would allow us to allocate fatigue lives with a greater degree of certainty than hitherto. It is perhaps probable that extensions of life could be made, as the original spectrum assumptions must inevitably be very conservative, but the cost and timescale could be quite excessive. It may be more economic to stay with our conservative life calculations and throw away the components prematurely. The timescale may be such that the aircraft is becoming obsolescent before the results could be used in a practical manner. Furthermore, it is likely that the results obtained from a specific aircraft could not be directly read across to new designs and the information gained is therefore only applicable to the type upon which it was measured.

However, we at Westlands have made some in-service measurements of this nature which I shall describe later on in the paper. These exercises however are limited to the investigation of a single component upon which a substantiation problem has occurred and where the assumptions made for the loading spectrum are critical.

If we wish to establish load spectra on a worthwhile scale we will have to reduce the task to manageable proportions. What can be done? It seems that a first possibility would be to relinquish all thoughts of measuring the actual vibratory loads but to define the aircraft manoeuvre conditions from flight parameters that are in themselves relatively easy to measure. Now providing that we can allocate actual loads to these conditions (and this can be done by means of the Contractors flight trials for example) then it seems possible that we can see a solution to the problem.

Initial probing of this problem in conjunction with the Royal Aircraft Establishment at Farnborough has suggested that a limited series of flight parameters can be used to define the manoeuvre condition of the aircraft at any moment in time. These parameters can be measured without the use of straingauges and in many cases the aircraft instrumentation or other standard instrumentation can be used.

We consider that the immediate future of spectrum measurement lies in the establishment of flight manoeuvre spectra rather than load spectra. Certainly we consider that the possibility of reading across to future types when operating in the same role will be quite feasible.

DATA COLLECTION AT WESTLANDS

This section will consist of a brief review of the various attempts that we have made at Westlands to establish load or manoeuvre spectra in service.

(a) An Early Manual Data Recording Exercise

One of our early attempts to gather data first took place in 1963/64. In this exercise we simply circulated copies of flight manoeuvre spectra that we were using at that time to the aircrew of Wessex and Whirlwind squadrons in the Navy and Air Force and asked them to make their own estimates of the percentage occurrences for the various manoeuvre conditions of the spectra.

The spectra that we were using at that time were of two types, the B.C.A.R. or C.A.M.6 type where percentage times were allocated to various flight conditions, and, because of the inadequacies of this presentation, we also used a supplementary spectrum of the number of occurrences per hour of certain conditions such as control inputs, take offs and landings etc.

It has to be admitted that this exercise was not a great success despite the excellent co-operation we had from the aircrews. One of the problems was due to the differences of opinion regarding the measuring of the various manoeuvre descriptions. For example, we had considered that a 'control reversal' meant a deliberate sequence of control inputs to perform a given manoeuvre. The pilots, however, interpreted this to mean the corrective actions required to maintain steady flight conditions. We also considered that some of the percentage times estimates must have been subject to bias because the stated spectrum was in front of them as they filled out their questionnaires. We were not able to make extensive use of the results.

(b) Whirlwind Tail Rotor Blades

From the time of that early attempt to gather service flight data and up to the present we have confined our investigations to the collection of actual load data by instrumentation to try to solve particular problems of life substantiation. These problems may have arisen from the difficulty of obtaining an adequate life with predicted load data, and we believed that a worthwhile result might be obtained if we had actual load measurements. In some cases, however, the problem may have arisen from unpredicted premature in-service failures where we believed that real knowledge of the load spectrum would lead to necessary flight restrictions for safety reasons in the first instance and secondly would lead to the appropriate modification action.

An example of the latter situation arose on Whirlwind Mk. 7 aircraft some years ago. We had a series of tail rotor blade failures (none of which actually caused catastrophe) and although our original flight trials had not uncovered any excessive stresses we believed that there was a possibility of a near resonant situation under certain conditions of flight. We carried out in-service recording with five aircraft in 1966/67 with straingauges and slip rings fitted to the tail rotors. An average of 100 hours recording per aircraft was obtained, the results being on paper trace. The results showed that on one of the aircraft occasional bursts of stresses much higher than expected had occurred although this did not entirely account for the low life failures. We used the results however to obtain a spectrum of stresses which was used in conjunction with service failure times initially to establish safe lives on existing blades and then to establish the amount of modification needed to obtain satisfactory lives. No problems have occurred since the final modification was incorporated.

(c) Wessex Tail Drive Shaft Gears

Another problem of a different type arose during the fatigue substantiation of the Wessex tail and intermediate gearbox gears. We have always had difficulty in establishing a spectrum of power for the tail rotor of a helicopter and our conservative estimates for this substantiation had led to very low estimates of service life. Although we had not had any in-service problem we considered that it would be prudent to modify our original designs to increase their strength and life.However we believed that it was essential to obtain spectrum information for tail transmission and a programme was instigated to obtain in-service data.

In order to avoid the problems of slip rings and straingauges we felt that we had to make measurements of a parameter that would provide a measure of tail drive shaft power without these drawbacks. We had obtained a large amount of information from our own flight trials which enabled us to make a global plot of pitch angle against power for the tail rotor. Although this relationship is, of course, speed dependent, we found that such a plot indicated that a conservative upper boundary curve could be established. We decided therefore to try to obtain a spectrum of pitch angle believing that the slight conservatism referred to above would be more than compensated for by the improved realism of the occurrences obtained.

An instrument was attached to the tail rotor pitch control rod that enabled us to establish the time spent in a number of ranges of pitch angle. A recorder in the aircraft had a series of digital displays and was read at convenient intervals, usually weekly, and the results entered upon a standard form and returned to Westlands.

An initial exercise was started some years ago in 1969 and collected over 300 hours on two aircraft. This was not very successful, unfortunately, due to both instrumentation problems and also significantly, a lack of continuity of personnel at the operational stations which tended to cause gaps and inconsistencies in the programme. However, enough was learnt by this exercise to lead the Ministry and ourselves to restart the recording in view of the expected future use of the Wessex. We now have another two aircraft in service which have so far gathered a further three hundred hours without the above problems. This exercise is still continuing.

(d) Wessex Main Gear Box Drive Gears

Recent re-investigation of the lives of the input gear train for the Wessex helicopter had led us to consider that our assumptions for the main drive power spectrum are critical. We considered that it would be worthwhile to investigate the possibility of instrumenting in-service aircraft to improve this knowledge.

An exercise was mounted some years ago on six Wessex Mk. 3 aircraft used in an anti-submarine role and a total of some 1400 hours of recording was obtained. We have recently considered that we should extend the range of power covered by the original exercise and we have started a new programme involving three Wessex Mk. 3's on anti-submarine duties and three Wessex Mk. 5's in a Commando carrying role. The output is recorded digitally from signals taken from pressure switches from the Mk. 3 pressure torquemeter and electrical switches from the electronic torquemeter of the Mk. 5 aircraft. We have collected to date about 800 hours of data from the six aircraft in service. So far the recordings have tended to confirm that our original assumptions of spectra were remarkably accurate. They even reflected our assumption that a 'hump' would exist in the region of the red line marking on the torquemeter which the pilot would often wish to use in take off or maximum climb conditions. This exercise is also continuing.

The Manual Data Recording Exercise

With the exception of the early exercise involving an aircrew questionnaire, all the data collection undertaken so far has been in the form of direct load measurement for the solution of specific problems.

Believing, as we do, that the immediate future lies in the collection of manoeuvre data, we consider that effort should be directed to the establishment of broad flight manoeuvre spectra that is essential to a real understanding of the use of helicopters in service situations. This will enable future fatigue substantiation to be based on solid foundations. Not of course that direct load data collection would be superceded when taking this approach, because we shall always have a need to solve these specific problems while the present safe life methods of fatigue substantiation are with us.

In early 1974 the Ministry and ourselves instigated a new manual recording exercise on a very large scale. We felt that there was much useful data that could be obtained without waiting for a fully instrumented survey. We believed that it was possible to ask the aircrew to fill in a relatively simple form at the end of every flight. These forms were headed with the aircraft type and serial number and location. The information requested was as follows:-

Data
Sortie Airborne Time
Number of Rotor Starts
Maximum All up Weight during Sortie
Number of landings
Number of decelerations, flares or approaches
(auto pilot or manual)
Time at torque above 90%
Time in hover
Time above 0.95 V_{max}
Sortie Code
Ground outside air temperature
Operating base, ship or shore

Codes were provided for over forty different sortie types in military use. The forms were provided to virtually all users in the Navy, Army and Air Force and the data was logged for every flight over a period of a year. Data has been obtained on nearly 75,000 flying hours on five hundred aircraft of ten different types. The aircraft types include the various marks of Sea King, Gazelle, Scout, Wasp, Wessex, Whirlwind and Puma.

The forms were returned to Westlands where we transcribed and stored the information on our computer. This exercise is now complete.

We had had some criticism from aircrew regarding the accuracy achievable for the parameters of torque, hover and high speed flight where we had asked for a time estimate. The timing was not provided by stop watch but had to be supplied from memory after the flight. I agree that the reliability of such data may be debatable although the results have been shown to have some consistency with our pre-conceived ideas of the spectra and I think that, statistically, they contain much of interest.

In my view the major value of this exercise has been in the improvement in our knowledge of how the aircraft have been used i.e. sortie usage, flight duration, rotor stop-starts etc. This can provide us with positive data which can be used for the new generation of aircraft to come.

A few brief points from the exercise are of interest.

(a) The number of landings per hour was frequently of the order of 4, 5 or 6 in certain roles This was in excess of our early spectrum estimates, where we had believed that 2 per hour might be reasonable. This does line up however with the assumptions made when we established 5 per hour for the Lynx usage.

(b) The number of manual flares per hour - which is critical for main rotor blades and control systems - compares well with the assumptions from AR56 (also used for the Lynx) of 5 per hour

(c) Time spent in hover seems to be much higher than our assumed values, up to 45% in anti-submarine roles.

(d) Our original assumptions of time spent in high speed flight have generally been conservative.

There were, of course, a number of occasions when an individual aircraft produced maxima greatly in excess of the average results for the fleet. These can sometimes be traced back and we would like to investigate some of these occurrences if possible.

We have found that, with the awareness in the Services etc., of the availability of this data, which can be produced in virtually any desired form from the computer, that we are starting to have requests for specific data from certain operations. For example, request for rotor start-stop statistics for the Gazelle fleet has been requested for engine data.

This has proved to be a most interesting exercise, some data of appreciable value has come from it. Obviously the results are only broad outlines and in some cases we would have to take care in the interpretation of results. Only a full instrumentation exercise will of course provide us with a spectrum of manoeuvres and detailed aircraft conditions for the future. This does not detract from the intrinsic value of this exercise, however, which has provided us with basic data from the whole sphere of military helicopter activity.

<u>Data collection from instrumented aircraft</u>

I have said earlier in this paper that we consider that the future lay in the establishment of manoeuvre spectra for helicopters in their various roles. This we believe would have more value for the design background for future generations of helicopters than the collection of load data on any one type.

Now, as I mentioned earlier in this paper, we also believe that it should be possible to recognise the manoeuvre condition of a helicopter from a number of relatively easily measured aircraft parameters. An initial exercise had already been carried out in 1970 when a number of flight parameters were continuously measured on a Wasp aircraft over a short period of flying. From this exercise we were able to obtain a feel for the measurements to be made and methods of analysis.

R.A.E. and ourselves consider that the basic parameters to be measured should be as follows:-

Pressure Altitude
Indicated Airspeed
Normal acceleration at the C.G.
Pitch Angle
Roll Angle
Heading
Collective pitch control position
Lateral control position
Pitch control position
Rudder control position
Rotor speed
Rotor torque
Tail rotor torque

We consider that it should be possible to recognise the following flight groups from these parameters:-

Longitudinal Flight Conditions:

Lift off and touch down
Transitions to forward flight
Level flight
Climb
Descent - power on
Descent - minimum pitch
Deceleration
Pull up from descent
Collective Flare
Longitudinal reversals

Lateral Flight Conditions:

Left or right turns - sensibly co-ordinated
Left or right sideways (or quatering) flight
Sideslip to port or starboard
Spot turns, left or right

On Ground Conditions:

Rotor run up
On ground rotors turning
Rotor run down

The next step planned in this exercise will take place next year when it is planned to instrument a Scout aircraft and fly it to a service manoeuvre pattern and try to show that our belief in the methods of flight recognition are indeed practical.

Following this exercise it is planned to institute an in-service digital recording trial which will collect large scale data from several service aircraft. The final specification for the instrumentation will be dependent upon the outcome of the Scout trial. It has been decided that the first in service aircraft to be fitted with the digital equipment will be the Naval Sea King. It is intended that the parameters will be recorded in flight on a suitable tape recorder. The results will be digitised on the ground and computer programmes will be developed to provide the required manoeuvre recognition.

This, we hope, will be the first steps in a rational programme to provide us with the data that will one day form the basis of fatigue re-substantiation of existing aircraft and the design spectra of the future.

U.S. AIR FORCE HELICOPTER OPERATIONAL FLIGHT SPECTRA SURVEY PROGRAM - PAST AND PRESENT

by

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SUMMARY

This paper summarizes the activities of Technology Incorporated over the last five years in developing and refining various techniques to process helicopter operational usage data for the U.S. Air Force. In particular, this paper presents the development and application of the Flight Condition Recognition (FCR) technique for the processing of such data.

The FCR technique identifies (1) aircraft operations and transitions (such as touchdown and entry to autorotation) which are called "flight conditions," and (2) flight phases (such as ascent and level flight) which are called "mission segments." Each identification is based on the characteristic behavior of the in-flight parameters (such as airspeed and altitude). With such identifications, the data processing consists basically of determining the occurrences and durations of the flight conditions and mission segments, of measuring the in-flight parameters, and accordingly of presenting these data so that the flight condition and mission segment time and frequency distributions are quantitatively defined by selected ranges of the flight parameters.

This application of the FCR technique permits the fatigue analyst to comprehend and apply more effectively the operational usage spectrum to the calculation of the fatigue life of critical helicopter components.

Aircraft Structural Integrity Requirement

As prescribed by U.S. Air Force Regulation MIL-STD-1530 (see Reference 1), a formal Aircraft Structural Integrity Program (ASIP) is required for each active fixed-wing and rotary-wing aircraft within the Air Force inventory. The intent of ASIP is to systematically substantiate or re-evaluate the fatigue life of critical aircraft components and structure. The flow chart in Figure 1 basically describes the current ASIP methodology. For fixed-wing aircraft, the Air Force is redefining this methodology because of the application of damage tolerant concepts during the design and substantiation of the aircraft's structure (see Reference 2). However, for rotary-wing aircraft, the Air Force still applies the existing ASIP methodology to its operational helicopters.

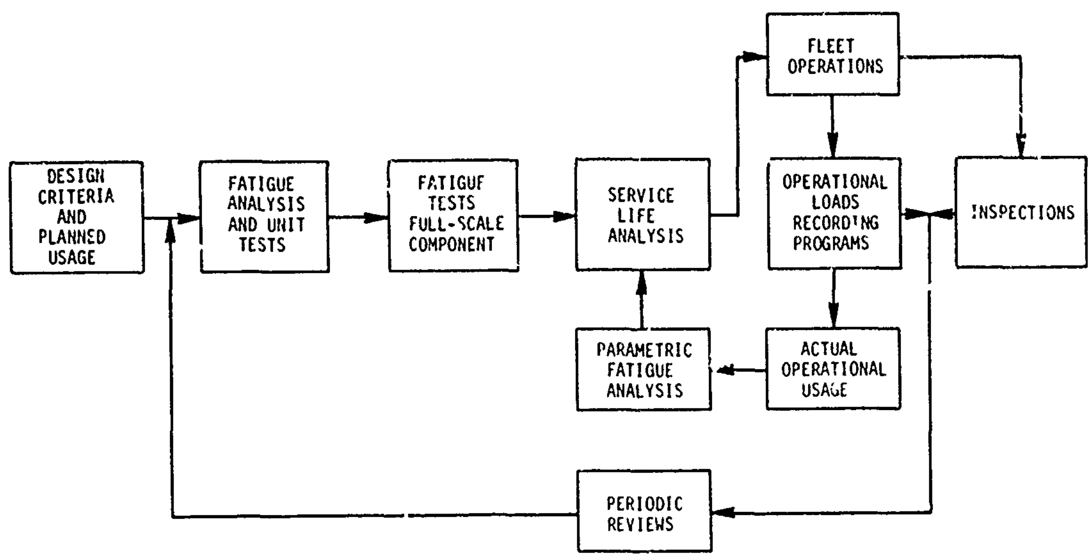

Figure 1. Interrelationship of ASIP Phases

Both the existing ASIP and the new program evolving from the damage tolerant concept require surveying the actual operational usage of the aircraft through recording programs. Extensive surveys on virtually every type of fixed-wing aircraft within NATO have quantified fleet operations in terms of airspeed, altitude, and gust and maneuver vertical accelerations, as well as other critical parameters. These surveys have been well documented in literature by AGARD and individual governmental organizations. However, for rotary-wing aircraft, fewer surveys have been conducted, often for purposes other than the verification or revision of the fatigue predictions of critical components.

Early Helicopter Operational Usage Surveys

Early helicopter operational usage surveys for the Air Force were conducted on the YH-40 helicopter in 1959 (Reference 3); the H-13H, H-21C, and H-34A in 1960 (Reference 4); and the HU-1A in 1962 (Reference 5). As described in Reference 5, the objective of these programs was "to depict the frequency distribution of certain in-flight parameters within arbitrarily selected ranges of magnitude. The information was to be used to verify or modify estimates, assumptions, and calculations of the distribution of the variables, for the fatigue life computations." Typically, the parameters monitored during these surveys included airspeed, pressure altitude, c.g. vertical acceleration, and various stationary or rotating component strains. The processed data were grouped without consideration for the operational mode. For example, data concerning vertical acceleration were presented as a total frequency distribution without any breakdown by operations such as takeoff, landing, level flight, or turns.

From September 1964 through March 1972, Technology Incorporated, under contract to the Eustis Directorate of the U.S. Army Air Mobility Research and Development Laboratory, collected operational usage data on seven different types of helicopters, as detailed in Table 1. These helicopters ranged from small observation to large crane models. The objective of these programs was to present the operational usage data so that the helicopter designer and fatigue analyst would have a better understanding of the operational flight spectra and their application in defining reliable design criteria for helicopters.

TABLE 1. OPERATIONAL USAGE SURVEYS FOR ARMY HELICOPTERS

Aircraft	Location	Mission	Time Frame	Processed Data (hrs)	Reference
UH-1B	Continental U.S.	Training	9/64-5/65	219	6
CH-47A	Southeast Asia	Cargo	1/66-5/67	235	7
CH-47A	Southeast Asia	Assault	7/66-5/67	207	8
AH-1G	Southeast Asia	Attack	7/58-11/69	408	9
CH-54	Southeast Asia	Crane	7/68-2/70	410	10
OH-6A	Southeast Asia	Observation	3/70-9/70	216	11
UH-1H	Southeast Asia	Utility	9/71-3/72	203	12

Table 2 lists the parameters recorded during these surveys. As apparent in the table, airspeed, altitude, vertical acceleration, engine torque, outside air temperature, and rotor speed were consistently monitored. During the early programs, longitudinal cyclic and collective stick positions were recorded to derive the instantaneous center of gravity; however, since this derivation proved to be impractical, the center of gravity was subsequently derived from information logged each flight on supplemental data sheets. During later programs, lateral and longitudinal accelerations were recorded in addition to vertical acceleration.

TABLE 2. PARAMETERS RECORDED DURING ARMY HELICOPTER OPERATIONAL USAGE SURVEYS

Parameter	UH-1B	CH-47 Cargo	CH-47 Armed	AH-1G	CH-54	OH-6A	UH-1H
Airspeed	o	o	o	o	o	o	o
Pressure Altitude	o	o	o	o	o	o	o
OAT	o	o	o	o	o	o	o
Rotor Speed	o	o	o	o	o	o	o
Vertical Acceleration	o	o	o	o	o	o	o
Longitudinal Acceleration			o	o	o	o	o
Lateral Acceleration			o	o	o	o	o
Engine Torque	o	o		o	o	o	o
Engine Exhaust Temperature		o					
Engine Gas Generator Speed		o					
Longitudinal Cyclic Position	o	o	o	o	o	o	
Collective Position	o	o	o	o	o	o	
Control Tube Strains							o

During the data processing, the recorded data were separated into four mission segments: Ascent, Descent, Steady State, and Maneuver. Ascent included the takeoff and climb to steady flight or maximum altitude and all unsteady ascents. Descent included the unsteady portion of descent, the flare, and the landing. Steady State was defined as the portion of the flight during which the stick traces had a relatively steady value and the airspeed and altitude traces were steady or changing smoothly. Steady State included cruising, hovering, steady climb, and steady descent. Maneuver consisted of any transient phases during the flight which did not fall into one of the other mission segments. The vertical acceleration trace in Maneuver was usually very active. During the data editing, all vertical acceleration peaks were identified as being either maneuver- or gust-induced. The maneuver peaks were defined as those vertical acceleration peaks that were preceded by the movement of either the longitudinal cyclic control or the collective control or both. After the edited data were processed, the various parameter data were presented in terms of exceedance curves, histograms, and maneuver and gust spectra. (Exceedance curves depict the time to reach or exceed a particular n_z level.)

The resultant operational usage spectra for the surveys (as documented in References 6 through 12) were of general interest to the helicopter manufacturers and provided the Government with information for defining the design criteria for future helicopters. However, these data could not be used directly by the fatigue analyst in determining the effect of various operational environments on fatigue-critical components, because the frequency and severity of damaging flight conditions were not identified. For example, all data for the maneuver mission segment were grouped into parameter ranges such as those for airspeed, altitude, and engine torque without any breakdown by the type of maneuver. Since the data were not distinguished according to maneuver type, such as turns and pull-ups, it could not be used for detail fatigue calculations.

<u>Pilot Program Study</u>

During 1971, Technology Incorporated, under contract to the U.S. Air Force, conducted a study (Reference 13) to determine how operational flight spectra data could be acquired, processed, and presented so that it would be compatible with typical fatigue analysis requirements, more understandable to fatigue analysts, and representative of flight operations, such as left or right turns. Specifically, the study objectives were to define the following:

(1) The optimum set of parameters required to define the UH-1F service usage.

(2) The data processing technique and graphic and tabular data presentations that would provide the most useful information.

(3) The minimum data sample that would provide a consistent and significant operational usage spectrum.

(4) The most cost-effective recording system for the recommended set of parameters, data processing-presentation procedure, and sample size.

As a result of this study, several recommendations were made for the development of the UH-1F flight loads program. Although these recommendations were based on the operating characteristics of the UH-1F models, they were generally applicable to other helicopters with the same or similar operating characteristics, since most helicopters have similar requirements for service life analysis and a minimum-cost, minimum-duration program would be desirable for all helicopters.

To achieve the most cost-effective program for the UH-1F aircraft, it was concluded that the parameters listed in Table 3 should be recorded by airborne recorders, logged on supplemental data forms, and derived from the recorded and logged data.

TABLE 3. RECORDED, SUPPLEMENTAL, AND COMPUTED DATA FOR ARMY HELICOPTER OPERATIONAL USAGE SURVEYS

RECORDED DATA	SUPPLEMENTAL DATA	COMPUTED DATA
Collective Control Position Longitudinal Control Position Lateral Control Position Rudder Pedal Position Vertical Acceleration Engine Torque Rotor Speed Indicated Airspeed Pressure Altitude Terrain Clearance Altitude Outside Air Temperature	Flight Identification Mission Identification Indicated Airspeed at Check Points Rotor Speed at Check Points Abrupt Discontinuities Transducer Serial Numbers Takeoff and Landing Data: Gross Weight, CG Position, Base Elevation, Barometric Pressure, Outside Air Temperature	Gross Weight* CG Position* Density Altitude Rate of Climb* Velocity in Percent Redline Velocity in Percent Design Airspeed Acceleration * Instantaneous

It was further concluded that this information should be processed by a flight condition identification procedure which uses the time histories of the control stick positions, engine torque, vertical load factor, rotor speed, airspeed, and altitude to detect and classify each flight condition. The processed data should then be presented in flight time tabulations with breakdowns by flight condition, mission segment, and various related parameter ranges.

The sample size investigation indicated that a 200-hour sample should be used in the operational usage survey for each helicopter model, mission type, and base of operations.

On the basis of the data acquisition, reduction, and presentation requirements and of a minimum-cost program, the oscillograph recorder should be initially employed for this type of flight loads program. Depending on the development of reliable data processing software and on the availability of a reliable and durable digital tape recorder for use in the helicopter environment, future programs should employ a digital magnetic tape recorder system for greater efficiency.

FCR Technique Development

During the winter of 1972-1973, data similar to that described in Table 3 were collected on several UH-1H Army helicopters operating at Fort Greeley, Alaska. These data were intended to describe Arctic environment operations so that the fatigue analyst would have a better understanding of the operational flight spectrum of Army helicopters and its effects in defining reliable design criteria for helicopters (Reference 14).

With the Army's consent, Technology Incorporated used two methods to reduce the data. First, the data were reduced by the Four-Mission Segment technique previously used on other U.S. Army surveys (References 6 through 12). The data were also independently processed by the technique recommended in an Air Force study (Reference 13); this technique was called the Flight Condition Recognition (FCR) technique. In processing the data by this technique, the data are divided in mission segments such as Ground Operations, Hover, Ascent, Level Flight, Descent, and Autorotation. Within each of these mission segments, the data are further categorized by flight conditions such as steady state, left or right turns, pull-ups, pushovers, and flares. In the Arctic data, 24 flight conditions were identified within the various mission segments. The data were then presented as occurrences of flight conditions and percentage of time within these flight conditions. Consequently, the data were displayed similarly as in the manufacturer's design spectrum, Table 4. Thus, the fatigue analyst could evaluate his derived fatigue spectrum and make appropriate modifications where milder or more severe conditions warrant them.

TABLE 4. DESIGN USAGE SPECTRUM FOR THE UH-1F/P HELICOPTER

Flight Condition	Percent Time
I. Ground Conditions	1.000
II. Power-On Flight	
A. Vertical Takeoff	0.400
B. Hovering IGE	
1. Steady	4.290
2. Right Turn	0.100
3. Left Turn	0.100
4. Control Rev.	
a. Longitudinal	0.010
b. Lateral	0.010
c. Rudder	0.010
C. Normal Accel.	1.000
D. Normal Decel.	1.000
E. Max. Rate Accel.	0.250
F. Max. Rate Decel.	0.250
G. Sideward Flt.	
1. To the Right	0.250
2. To the Left	0.250
H. Rearward Flt.	0.250
I. Full Power Climb	4.000
J. Fwd. Level Flt.	
1. 0.8 Vne	7.000
2. 0.9 Vne	30.000
3. Vne	40.000
K. Right Turns	
1. 0.3 V_H	0.500
2. 0.6 V_H	1.000
3. 0.9 V_H	0.500
L. Left Turns	
1. 0.3 V_H	0.500
2. 0.6 V_H	1.000
3. 0.9 V_H	0.500
M. Cyclic Pull-ups	
1. 0.6 V_H	0.200
2. 0.9 V_H	0.050
N. Collective Pull-ups	
1. 0.6 V_H	0.200
2. 0.9 V_H	0.050
P. 0.9 V_H Control Rev.	
1. Longitudinal	0.050
2. Lateral	0.050
3. Rudder	0.050
Q. Normal Landing	1.000
III. Transitions	
A. Power to Auto	
1. 0.3 V_H	0.100
2. 0.6 V_H	0.200
3. 0.9 V_H	0.050
B. Auto to Power	
1. 0.4 V_H	0.100
2. 0.6 V_H	0.200
3. 0.8 V_H	0.050
IV. Autorotation	
A. Steady Fwd. Flt.	
1. 0.4 V_H	0.800
2. 0.6 V_H	1.000
3. 0.8 V_H	0.200
B. 60 Kt. Cont. Rev.	
1. Longitudinal	0.010
2. Lateral	0.010
3. Rudder	0.010
C. Right Turns	
1. 0.4 V_H	0.200
2. 0.6 V_H	0.250
3. 0.8 V_H	0.050
D. Left Turns	
1. 0.4 V_H	0.200
2. 0.6 V_H	0.250
3. 0.8 V_H	0.050
E. Auto Landing Appr. W/Pwr. Recv. IGE	
1. 0.4 V_H	0.080
2. 0.6 V_H	0.100
3. 0.8 V_H	0.020
F. Full Auto Land.	0.250

FCR Application

As a result of the Reference 13 study and the Arctic survey of the UH-1H helicopter, the Air Force, through the Warner Robins Aeronautical Engineering Office of Warner Robins Air Logistics Center, contracted with Technology Incorporated to acquire, reduce, and present 300 hours of data on the UH-1F utility and TH-1F training missions and 900 hours of data on other Air Force helicopters. The 900 hours of data were recorded as follows: 100 hours on the HH-1H rescue training mission, 200 hours on the CH/HH-53B/C training mission, 200 hours on the HH-53C rescue training mission, 200 hours on the HH-3E training mission, and 200 hours on the CH-3E cargo mission. Under separate contracts with each helicopter manufacturer, the Air Force intends to update each model's fatigue spectrum and revise the fatigue lives of the critical dynamic components. The rest of this paper will deal with the techniques used in acquiring, reducing, and presenting the data gathered on the UH-1F and TH-1F helicopters. Separate publications will present the data collected on the H-53 and H-3 helicopters.

Data Acquisition

The contract with Warner Robins specified that two recording sites would be used to acquire the data on the UH-1F and TH-1F aircraft. At the first site, four aircraft were instrumented until a total of 200 hours of data were collected on the utility mission of the UH-1F helicopters; and, at the second site, two aircraft were instrumented until a total of 100 hours of data were collected on the training mission of the TH-1F helicopters.

Technology Incorporated provided field support services for the data acquisition on a non-interference basis; that is, the data collection did not affect the operations of the using command. The field services included the following: the installation of the recording systems, the servicing of the systems during the data collection, the collection of recorded and supplemental data, the removal of the recording systems, and the restoration of the aircraft at the completion of the recording effort. Although these services minimized the using command's support of the program, its maintenance group had to provide access to the instrumented aircraft and make available necessary ground support equipment (such as ground power units), and its operations group had to coordinate the flight schedules with the company field technician and provide some supplemental data. These support tasks, however, were within the scope of normal operations. To further minimize interference with the using command, the recording systems were not classified as mission critical to prevent aborting regularly scheduled missions because of recording system malfunction.

Special oscillograph recording systems were designed and fabricated for the data acquisition. As shown schematically in Figure 2 the major components include parameter sensors, a signal conditioning unit, and an oscillograph recorder. The parameter sensors are shown grouped around the signal conditioning unit, which prepares the sensor output for the oscillograph recorder. In addition to the basic recording system, mounting brackets, wiring harnesses, power interconnections, and assemblies for tapping the pitot-static and transmission oil systems were fabricated.

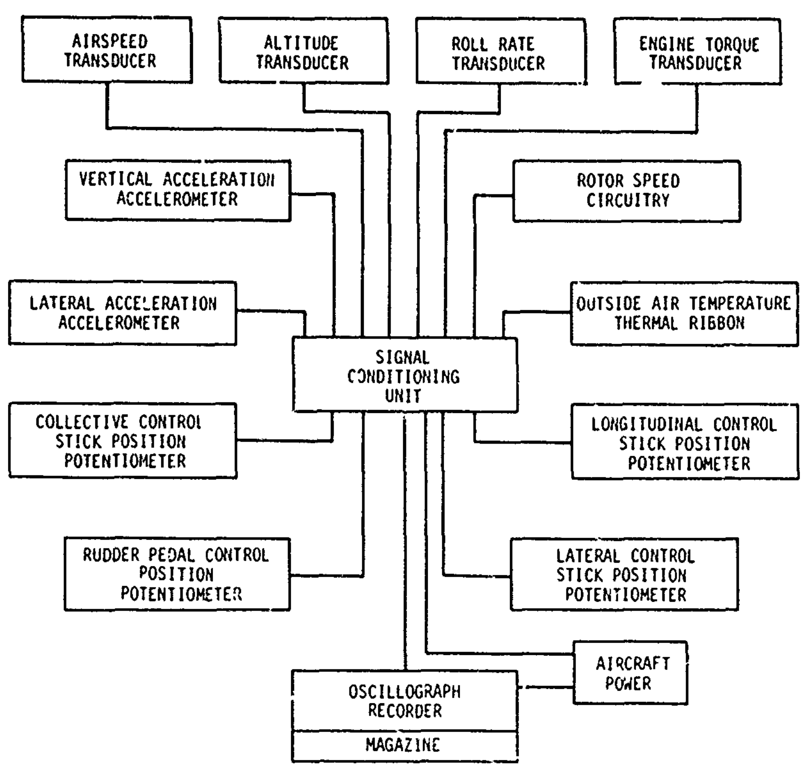

Figure 2. Functional Relationship of Major Components in Oscillograph Recording System for H-1 Operational Usage Survey

In passing, it should be noted that the parameters shown in Figure 2 differ slightly from those of Table 3; the latter are the recorded parameters recommended by the Reference 13 study. The differences were minor and did not affect the ability to reduce the data by the FCR method. As the figure illustrates, terrain clearance altitude was deleted from the data collection, and lateral acceleration and roll rate were added. Terrain clearance altitude was deleted because a radar altimeter was not available on the instrumented aircraft and its installation in the recording system would not be cost-effective. The loss of terrain-clearance altitude did not seriously affect the data reduction, even though it was to be used in identifying the in-ground-effect (IGE) segments of flight. The IGE segments of flight could still be identified by monitoring the control stick position because the turbulence caused by the ground effect is apparent in such data. Lateral acceleration and, later, roll rate were added to provide an indication of the severity of turns. Lateral acceleration, which was originally included for this purpose, was retained for correlation purposes after the Bell Helicopter Company requested that the aircraft roll rate be monitored. After the recording systems were modified, so little roll rate data were recorded in the rest of the T/UH-1F operations that they will not be discussed here.

The in-flight parameters shown in Figure 2 were recorded on standard oscillograph strip chart (photosensitive paper 3-7/8 inches wide). The oscillographs supplied by the Government were selected because of their capability of recording at paper speed slow enough to monitor the duration of most flights. With 150 feet of chart loaded in the oscillogram magazine, the oscillograph could record approximately 4.6 hours of data.

The narrow chart size posed the problem of displaying 12 dynamic parameters, a measurement-reference line, and a constant cycling pattern for time correlation so that the traces would be distinct, legible, and cover the prescribed ranges with sufficient deflection to permit reading the traces to the desired accuracy, for example, within ±300 feet for a pressure altitude reading at 10,000 feet. As illustrated in Figure 3, the trace positions selected for the program recording were based on the optimum trace separation during level flight. As apparent, the distinct traces are easily read. When traces crossed and overlapped during maneuvers, they were still visually distinguishable since maneuvers usually produce large deflections for short periods.

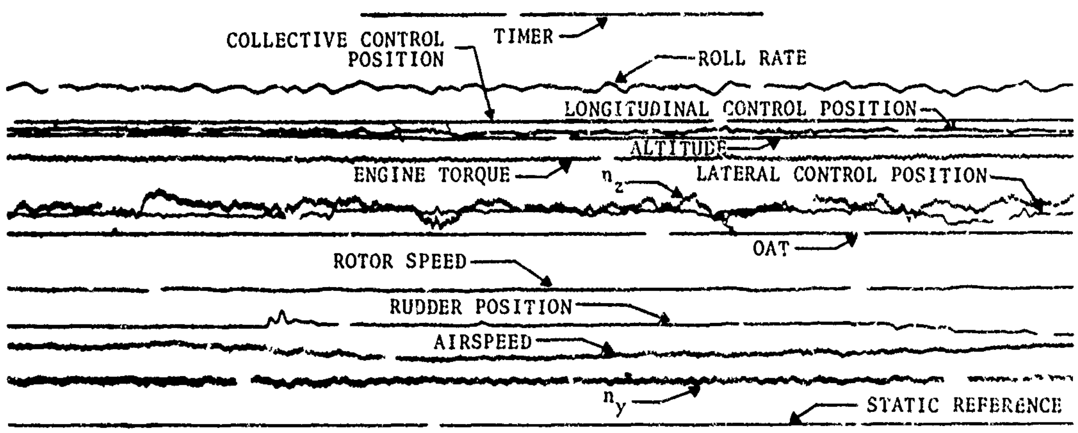

Figure 3. Oscillogram Trace Positions During Typical Level Flight

To complement the recorded data, field technicians logged the supplemental data listed in Table 3. Except for the airspeed and rotor speed check points, all items in the supplemental data listing were prepared for each flight. Since the check point data were intended primarily to verify the recording system functioning by comparing the system outputs with the aircraft instrument readings, the checks were made only during the initial installation checkout flights.

DATA REDUCTION

The recorded and supplemental data were reduced to the required format by the FCR method. As explained earlier, the FCR method is currently a manual technique which identifies specific flight conditions and mission segments in oscillogram data. In general, flight conditions, such as turn , pull-ups, and pushovers, were identified by the characteristic patterns in the control stick position and vertical acceleration traces while mission segments, such as ascent, descent, and IGE hover, were identified by characteristic patterns in the airspeed, altitude, and engine torque traces. After the flight conditions and mission segments were identified, their durations were timed and then compiled to form a usage spectrum for the particular mission type. In addition, the flight conditions were described in terms of the percentage of time spent in gross weight, center-of-gravity position, altitude, airspeed, rotor speed, engine torque, vertical acceleration, lateral acceleration, and rate-of-climb ranges. This parameter-range data provided the means of determining the fatigue-damage severity of the flight conditions by comparing such data with similar data recorded in the original flight loads survey. With the usage spectrum and parameter-range data, the manufacturer can evaluate the component lives for the recorded mission.

Figure 4 presents the basic steps and the general data flow in the data reduction process. Except for the manual identification of the flight conditions and mission segments and some data checks, all steps were performed with automatic or semiautomatic equipment.

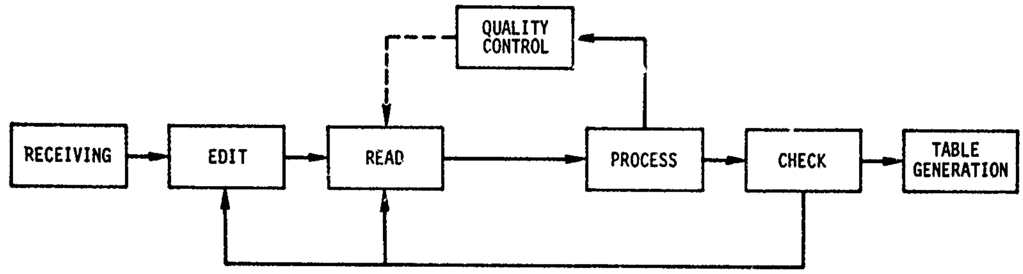

Figure 4. Sequence of FCR Data Reduction Phases

Upon receipt of each set of recorded and supplemental data, the data analysts identified each oscillogram and its accompanying supplemental data sheets by a unique number, examined the oscillogram for trace aberrations indicative of equipment malfunctions, checked the supplemental data sheet for completeness and accuracy, interpreted the preflight calibration data, added the calibration data to the supplemental data, and logged all pertinent information for weekly reports summarizing both the weekly and the cumulative number of usable and unusable hours of recorded data. Only the oscillograms with usable data were forwarded to data editing, the next step in the data reduction process.

In addition to identifying the flight conditions and mission segments listed in Table 5 and demarcating their durations such that the entire oscillogram was sectioned into mission segments which in turn were separated into flight conditions, the data editors marked for reading those parameter points whose measurements would adequately delineate their time histories and the peak occurrences of such parameters as airspeed, engine torque, and vertical acceleration. In addition, they noted whether a vertical acceleration peak was maneuver- or gust-induced according to established criteria. Except for the transient flight condition, all flight condition and mission segment titles are self-explanatory. The transient flight condition represents a period in which the rapid variations in engine torque and rotor speed are not typical of any other flight condition.

TABLE 5. FLIGHT CONDITIONS AND MISSION SEGMENTS IDENTIFIED BY THE FCR TECHNIQUE

	Rotor Start	Steady State	Transient	Take-off	Collective Pushover	Collective Pull-up	Flare	Touchdown	Rotor Stop	Left Turn	Right Turn	Cyclic Pushover	Cyclic Pull-up	Longitudinal Reversal	Lateral Reversal	Rudder Reversal	Left Sideward	Right Sideward	Initiation of Ascent	Entry to Autorotation	Recovery from Autorotation	Dive	Dive Pull-out	Cargo Pick-up	Cargo Drop
Ground Conditions	●	●	●						●	●	●														
IGE Hover		●	●	●	●	●		●		●	●	●	●	●	●	●	●	●	●					●	●
Ascent		●		●	●	●				●	●	●	●	●	●	●							●		
Level Flight		●		●	●	●	●	●		●	●	●	●	●	●	●									
Descent		●			●	●	●	●		●	●	●	●	●	●	●						●			
Autorotation		●	●		●	●	●	●		●	●	●	●	●	●	●				●	●				
OGE Hover		●			●	●				●	●	●	●	●	●	●			●						
Nap of Earth		●			●	●	●	●		●	●	●	●	●	●	●									

After the data editing, the oscillogram trace deflections at the marked points were measured on semiautomatic readers in terms of the number of counts per inch, and the counts were automatically digitized on keypunch cards. The result of this operation was a deck of keypunch cards representing in sequential time the parameters for the flight conditions and mission segments on each oscillogram. Then the supplemental and calibration data were keypunched on other cards and added to the oscillogram data on a flight-by-flight basis.

Next a series of computer programs performed the following operations: (1) printout of the oscillogram data for continuity and format checks, (2) plotting of the oscillogram data for review, (3) conversion of the oscillogram data to engineering units by applying the calibration data, (4) checking for and listing unusual or extreme values, (5) calculating the derived parameters (such as density altitude), and (6) grouping the data for each parameter in prescribed ranges (see Table 6) to permit the comparison of the current data with previous data. Except for rate of climb, all derived parameters were computed as instantaneous values.

TABLE 6. RANGES FOR THE RECORDED AND COMPUTED PARAMETERS

Vert. Acc. (n_z)	Lat. Acc. (n_y)	Rotor Speed (rpm)	OAT (°F)	Rate of Climb (ft/min)	Gross Weight (lb)
<0.2	<-0.40	<296	<-60	<-2100	<6000
0.2 to 0.4	-0.40 to -0.35	296 to 300	-60 to -40	-2100 to -1800	6000 to 7000
0.4 to 0.5	-0.35 to -0.30	300 to 304	-40 to -20	-1800 to -1500	7000 to 8000
0.5 to 0.6	-0.30 to -0.25	304 to 308	-20 to 0	-1500 to -1200	8000 to 9000
0.6 to 0.7	-0.25 to -0.20	308 to 312	0 to 20	-1200 to -900	9000 to 10000
0.7 to 0.8	-0.20 to -0.15	312 to 316	20 to 40	-900 to -600	≥10000
0.8 to 1.2*	-0.15 to -0.10	316 to 320	40 to 60	-600 to -300	
1.2 to 1.3	-0.10 to 0.10	320 to 324	60 to 80	-300 to 300	
1.3 to 1.4	0.10 to 0.15	324 to 328	80 to 100	300 to 600	
1.4 to 1.5	0.15 to 0.20	328 to 332	100 to 120	600 to 900	
1.5 to 1.6	0.20 to 0.25	332 to 336	≥120	900 to 1200	
1.6 to 1.7	0.25 to 0.30	336 to 340		1200 to 1500	
1.7 to 1.8	0.30 to 0.35	≥340		1500 to 1800	
1.8 to 2.0	0.35 to 0.40			1800 to 2100	
2.0 to 2.2	≥0.40			≥2100	
2.2 to 2.4					
≥2.4					

C.G. Position	Airspeed (%Vne)	Density Altitude (ft)	Roll Rate (deg/sec)	Torque (psi)	A/S Acc. (knots/sec)
FWD 125 to 131.5	<40	<-2000	<10	<10	<-2
AFT 131.5 to 138	40 to 50	-2000 to 0	10 to 15	10 to 20	-2 to 2
	50 to 60	0 to 2000	15 to 20	20 to 30	≥2
	60 to 70	2000 to 4000	20 to 25	30 to 40	
	70 to 80	4000 to 6000	≥25	40 to 50	
	80 to 90	6000 to 9000		50 to 60	
	90 to 100	9000 to 12000		60 to 70	
	100 to 110	12000 to 15000		≥70	
	≥110	≥15000			

* For turns add:

0.8 to 0.9
0.9 to 1.1
1.1 to 1.2

After the foregoing computer operations, the following manual operations were performed to check the data: (1) the inspection of the data format to check that the data were digitized to the correct order of magnitude, (2) the inspection for data continuity to check that successive parameter values are within acceptable limits, (3) the comparison of the data with normal values to detect measurements of the wrong trace because of trace crossing and overlapping, (4) the investigation of unusual or extreme values, and (5) the verification of computer-generated error messages. After all detected errors were corrected, the data was subjected to a quality control check.

In the quality control check, randomly selected oscillogram points were manually measured and the measurements were compared with the digitized data for the same points. Depending on the accuracy standards and the agreement of the digitized data with the manually measured data, the digitized data was accepted or rejected on a flight-by-flight basis. Then the oscillograms corresponding to the rejected data were reprocessed through all the foregoing steps in the data reduction process.

In the final steps of the data reduction process, all data found valid in the quality control check were merged onto a master tape which was maintained until 200 hours of utility mission data or 100 hours of training mission data were acquired. Then after the data on the master tape were sorted according to the required table formats, the data tables were printed.

DATA PRESENTATION

In accord with the manufacturer's requirements and approval, the mission data were presented primarily in terms of the time distribution in flight conditions and mission segments. In addition, the data represents the percentage of time recorded in parameter ranges during the respective flight condition-mission segment combinations. As presented and discussed in the final reports for each mission type and the composite missions, seven types of tabular printouts were prepared: (1) the usage spectrum, (2) the usage spectrum normalized to 100 hours, (3) the time distributions for the recorded and derived parameter ranges, (4) the time distributions for coincident parameter ranges, (5) the concurrent lateral and vertical acceleration peaks, (6) the maneuver-induced vertical acceleration peaks in coincident parameter ranges, and (7) the airspeed accelerations in level flight. Because of the volume of data in each report, only samples of each table for the composite H-1F missions will be described. Copies of the final reports may be requested by writing to the Helicopter System Manager, Robins AFB, Georgia 31098.

For the H-1F helicopter, Table 7 presents the composite (utility and training mission) usage spectrum normalized to a 100-hour basis. (The composite usage spectrum represents a 300-hour data sample, 200 hours of utility data and 100 hours of training data.) This table presents for each combination of mission segment and flight condition the number of occurrences, the total time in minutes, and the percentage of 100 hours.

TABLE 7. OPERATIONAL USAGE SPECTRUM FOR THE H-1 HELICOPTERS

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

	TIME (HRS)	PERCENT TIME	OCCUR.	PERCENT OCCUR.
MISSION SEGMENTS				
GROUND CONDITIONS	50.90	16.6	1025	14.4
IGE HOVER	25.50	8.3	1276	17.9
ASCENT	40.98	13.4	1506	21.1
LEVEL FLIGHT	140.32	45.9	1624	22.8
DESCENT	44.55	14.6	1351	18.9
AUTOROTATION	3.54	1.2	351	4.9
TOTALS	305.80	100.0	7133	100.0
FLIGHT CONDITIONS				
ROTOR START	0.00	0.0	205	.8
STEADY STATE	269.61	88.2	10376	42.9
TRANSIENT	2.87	.9	855	3.5
TAKE-OFF	2.23	.7	883	3.7
COLLECTIVE PUSHOVER	2.84	.9	1691	7.0
COLLECTIVE PULL-UP	1.52	.5	951	3.9
FLARE	2.79	.9	553	2.3
TOUCHDOWN	0.00	0.0	876	3.6
ROTOR STOP	0.00	0.0	197	.8
MISSION SEGMENT VARIATION	0.00	0.0	2399	9.9
LEFT TURN	9.95	3.3	1407	5.8
RIGHT TURN	8.55	2.8	1260	5.0
CYCLIC PUSHOVER	.28	.1	173	.7
CYCLIC PULL-UP	1.73	.6	702	2.9
LONGITUDINAL REVERSAL	.05	.0	67	.3
LATERAL REVERSAL	.11	.0	159	.7
RUDDER REVERSAL	.13	.0	169	.7
INITIATION OF ASCENT	1.70	.6	61[illegible]	2.5
ENTRY AUTOROTATION	.57	.2	352	1.5
RECOVERY AUTOROTATION	.85	.3	352	1.5
DIVE	.01	.0	2	.0
DIVE PULL-OUT	.00	.0	2	.0
ENDS IN FLIGHT	0.00	0.0	8	.0
TOTALS	305.80	100.0	24190	100.0

Table 8 presents the time and percent of time in gross weight ranges for each mission segment and for each flight condition in each mission segment. In this and similar tables, each range is identified by its lower bound; for example, 6000 pounds represents 6000 to 7000 pounds. Table 9 presents the same data for the airspeed as a percentage of the never-to-exceed velocity ($\%V_{ne}$). Similar tables were prepared for rotor speed, outside air temperature, maneuver-induced vertical accelerations, gust-induced vertical accelerations, and engine torque. Such tabular formats permit comparing the current data with previous usage survey data.

TABLE 8. TIME AND PERCENT OF TIME IN GROSS WEIGHT RANGES FOR EACH H-1 MISSION SEGMENT AND FLIGHT CONDITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

	PERCENT OF TIME IN RANGES						TIME
MISSION SEGMENTS	BEL	6000	7000	8000	9000	10000	(HRS)
GROUND CONDITIONS	7.9	76.7	15.4				50.90
IGE HOVER	19.2	73.7	7.2				25.50
ASCENT	16.1	77.7	6.2				40.98
LEVEL FLIGHT	16.6	74.2	9.2				140.32
DESCENT	20.0	72.6	7.4				44.55
AUTOROTATION	18.2	80.3	1.5				3.54
TOTAL	15.8	74.9	9.3	0.0	0.0	0.0	305.80
FLIGHT CONDITIONS							
STEADY STATE	15.6	74.6	9.8				269.61
TRANSIENT	10.8	73.9	15.3				2.87
TAKE-OFF	13.0	77.7	9.3				2.23
COLLECTIVE PUSHOVER	19.6	75.3	5.1				2.84
COLLECTIVE PULL-UP	22.2	72.9	4.9				1.52
FLARE	17.9	74.8	7.3				2.79
LEFT TURN	18.2	79.0	2.8				9.95
RIGHT TURN	17.5	77.7	4.8				8.55
CYCLIC PUSHOVER	13.6	80.8	5.6				.28
CYCLIC PULL-UP	16.3	79.3	4.4				1.73
LONGITUDINAL REVERSAL	11.8	88.2					.05
LATERAL REVERSAL	23.0	73.8	3.1				.11
RUDDER REVERSAL	16.8	78.0	5.2				.13
INITIATION OF ASCENT	15.9	75.9	8.1				1.70
ENTRY AUTOROTATION	18.3	80.6	1.1				.57
RECOVERY AUTOROTATION	18.8	79.5	1.8				.85
DIVE		100.0					.01
DIVE PULL-OUT		100.0					.00
TOTAL	15.8	74.9	9.3	0.0	0.0	0.0	305.80

TABLE 9. TIME AND PERCENT OF TIME IN AIRSPEED RANGES FOR EACH H-1 MISSION SEGMENT AND FLIGHT CONDITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

MISSION SEGMENTS	PERCENT OF TIME IN RANGES REL	40	50	60	70	80	90	100	110	TIME (HRS)
GROUND CONDITIONS	100.0									50.90
IGE HOVER	99.2	.6	.1	.0	.0					25.50
ASCENT	6.7	4.3	10.3	23.3	32.7	18.4	3.6	.7		40.98
LEVEL FLIGHT	.4	1.2	4.4	11.8	25.8	37.1	16.0	3.0	.1	140.32
DESCENT	10.0	7.1	10.3	15.7	22.9	22.8	9.1	2.0	.0	44.55
AUTOROTATION	21.2	8.3	14.5	26.8	23.4	5.4	.4	.1		3.54
TOTAL	27.7	2.3	5.1	11.2	19.8	22.9	9.2	1.8	.0	305.80
FLIGHT CONDITIONS										
STEADY STATE	26.5	2.2	4.8	10.6	19.7	24.2	10.0	2.0	.1	269.61
TRANSIENT	99.8			.0	.2					2.87
TAKE-OFF	99.4	.4	.1	.0						2.23
COLLECTIVE PUSHOVER	5.1	6.4	16.4	19.7	26.3	18.8	5.6	1.7	.1	2.84
COLLECTIVE PULL-UP	19.2	12.4	16.6	20.1	20.1	9.2	2.1	.3	.0	1.52
FLARE	96.2	3.6	.2							2.79
LEFT TURN	17.7	1.8	6.4	10.0	31.2	19.5	3.1	.2	.0	9.95
RIGHT TURN	10.1	2.9	9.4	23.6	30.3	17.6	5.8	.3	.0	8.55
CYCLIC PUSHOVER	23.2	24.3	12.8	13.6	19.7	3.9	2.1	.4		.28
CYCLIC PULL-UP	2.6	5.3	10.6	27.1	29.8	20.5	3.8	.2	.2	.73
LONGITUDINAL REVERSAL	72.6	13.8	3.3	3.9	4.5	1.9				.05
LATERAL REVERSAL	81.9	.7	3.5	1.4	5.1	6.3	1.0			.11
RUDDER REVERSAL	99.1						.9			.13
INITIATION OF ASCENT	89.9	7.6	1.8	.6	.1					1.70
ENTRY AUTOROTATION		.2	5.0	23.8	51.3	17.7	1.6	.3		.57
RECOVERY AUTOROTATION	80.7	5.9	4.6	5.6	2.6	.5				.85
DIVE				34.8	26.1	39.1				.01
DIVE PULL-OUT				22.7	40.9	27.3	9.1			.00
TOTAL	27.7	2.3	5.1	11.2	19.8	22.9	9.2	1.8	.0	305.80

Table 10 presents for each combination of flight condition, mission segment, gross weight, and center-of-gravity position, the time in coincident ranges of density altitude, airspeed, rotor speed, engine torque, and rate of climb. Since this table indicates the relative severity of the flight conditions, its data can be directly correlated with the data in the manufacturer's H-1F strain survey.

TABLE 10. TIME IN COINCIDENT PARAMETER RANGES BY MISSION SEGMENT, FLIGHT CONDITION, GROSS WEIGHT, AND C.G. POSITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

STEADY STATE IGE HOVER
GROSS WEIGHT: 6000 CG POSITION: FWD

DENSITY ALT	PERCENT VNF	ROTOR SPEED	TORQUE	TIME (MIN)	
2000	<40	296	30	.13	
2000	<40	300	30	.30	
2000	<40	304	20	.09	
2000	<40	304	30	.05	
2000	<40	308	20	2.38	
2000	<40	308	30	.21	
2000	<40	312	20	.15	
2000	<40				3.32
4000	<40	300	20	.09	
4000	<40	300	30	.08	
4000	<40	304	20	.63	
4000	<40	304	30	.47	
4000	<40	308	20	.97	
4000	<40	308	30	.78	
4000	<40	312	20	1.68	
4000	<40	312	30	.19	
4000	<40	316	20	2.07	
4000	<40	316	30	.79	
4000	<40	320	20	.38	
4000	<40				8.12
6000	<40	296	20	.04	
6000	<40	296	30	.02	
6000	<40	300	20	.59	
6000	<40	304	20	.88	
6000	<40	304	30	.10	
6000	<40	308	20	1.43	
6000	<40	312	20	.87	
6000	<40	312	30	.38	
6000	<40	316	30	.05	
6000	<40				4.38
TOTAL					15.82

Table 11 presents the occurrences of concurrent vertical and lateral accelerations in coincident density altitude and airspeed ranges for each combination of flight condition, mission segment, gross weight, and center-of-gravity position. The lateral accelerations to the left were given positive values. In the computer-generated tables, the vertical accelerations are separated into those inside and outside threshold; only the latter are presented here.

TABLE 11. COINCIDENT LATERAL AND VERTICAL ACCELERATION PEAK OCCURRENCES IN COINCIDENT PARAMETER RANGES FOR FLIGHT CONDITION AND MISSION SEGMENT COMBINATIONS

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

FLIGHT CONDITION	MISSION SEGMENT	GROSS WGT	CG POSITION	DENSITY ALT	PERCENT VNE	CORR NZ	NY	TOTAL OCCUR
CYCLIC PULL-UP	DESCENT	<6000	AFT	4000	70	1.3	0.10	1
CYCLIC PULL-UP	DESCENT	6000	AFT	6000	70	1.7	-0.15	1
CYCLIC PULL-UP	DESCENT	6000	AFT	6000	110	1.4	0.10	1
CYCLIC PULL-UP	AUTOROTATION	<6000	AFT	6000	60	1.5	0.10	1
CYCLIC PULL-UP	AUTOROTATION	<6000	AFT	2000	70	1.2	-0.15	1
CYCLIC PULL-UP	AUTOROTATION	6000	AFT	2000	50	1.3	-0.15	1
CYCLIC PULL-UP	AUTOROTATION	6000	AFT	4000	60	1.2	-0.15	1
CYCLIC PULL-UP	AUTOROTATION	6000	AFT	4000	60	1.4	-0.15	1
LONGITUDINAL REVERSAL	ASCENT	6000	AFT	2000	60	1.4	-0.15	1
ENTRY AUTOROTATION	AUTOROTATION	<6000	AFT	6000	80	0.7	-0.15	1
ENTRY AUTOROTATION	AUTOROTATION	6000	AFT	4000	70	0.2	-0.15	1
ENTRY AUTOROTATION	AUTOROTATION	6000	AFT	6000	70	0.2	0.10	1
ENTRY AUTOROTATION	AUTOROTATION	6000	AFT	4000	70	0.5	-0.15	1
ENTRY AUTOROTATION	AUTOROTATION	6000	AFT	6000	80	0.7	0.15	1

Table 12 presents the maneuver-induced vertical acceleration peaks. This table lists the following data: (1) the number of maximum n_z peaks (one for each flight condition), (2) the duration of these peaks outside threshold, (3) the time of the flight conditions containing the peaks, (4) the total number of n_z peaks (including any secondary peaks) outside threshold, and (5) the duration of all the n_z peaks outside threshold for each combination of flight condition and mission segment. (The range labeled NONE represents the threshold from 0.9g to 1.1g.)

TABLE 12. MANEUVER-INDUCED VERTICAL ACCELERATION PEAK OCCURRENCES AND DURATIONS BY MISSION SEGMENT AND FLIGHT CONDITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

RIGHT TURN NO. FLT. CONDITIONS: 431
LEVEL FLIGHT FLT. TIME (MIN): 207.71

RANGE	MAX. NZ PER FLT. COND.			TOTAL NZ PEAKS	
	OCCUR.	DURATION	FLT. TIME	OCCUR.	DURATION
BELOW			0.00		
0.2			0.00		
0.4			0.00		
0.5			0.00		
0.6			0.00		
0.7			0.00		
0.8			0.00		
NONE	24	.	7.93		
1.1	148	9.57	67.12	330	15.31
1.2	161	15.84	78.52	244	21.51
1.3	65	9.06	35.07	79	10.83
1.4	22	5.26	12.10	26	5.82
1.5	10	2.35	6.51	11	2.57
1.6	1	.26	.47	1	.26
1.7			0.00		
1.8			0.00		
2.0			0.00		
2.2			0.00		
2.4			0.00		
SUM	431	42.33	207.71	691	56.28

Table 13 presents the maneuver-induced vertical accelerations in concurrent density altitude and airspeed ranges for the combinations of flight condition, mission segment, gross weight, and center-of-gravity position.

For each flight condition in the level flight mission segment, Table 14 lists the number of occurrences and the durations of both accelerations and decelerations outside the 2-knot-per-second threshold.

The computer-generated tables were supplemented by figures or summary tables to present the data more vividly and concisely. For example, the data in Table 8 for both the mission segments and the flight conditions were illustrated by percentage-of-time histograms and frequency distribution curves, and the vertical acceleration data in Table 12 were depicted by exceedance (hours to reach or exceed a given vertical acceleration level) curves. In general, the summary tables list the number of occurrences, total time, and average duration for each flight condition in each mission segment.

TABLE 13. MANEUVER-INDUCED VERTICAL ACCELERATION PEAKS IN COINCIDENT PARAMETER RANGES BY MISSION SEGMENT, FLIGHT CONDITION, GROSS WEIGHT, AND C.G. POSITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

CYCLIC PULL-UP GROSS WEIGHT: 6000
AUTOROTATION CG POSITION: AFT

DENSITY ALTITUDE	PERCENT VNE	0.7	0.8	0.9	1.1	1.2	1.3	1.4	1.5	1.6	1.7	1.8	2.0	TIME (HRS)
		OCCURRENCES IN RANGES												
0	50									1				.00
0	60					1		1	2	1		1		.01
2000	40					1								.02
2000	50					8	2	3		1				.03
2000	60					8	7	2	2	3				.03
2000	70					2		1						.00
4000	40							1						.03
4000	50					6	5	5	2	1				.05
4000	60					15	12	13	4	8	1			.10
4000	70					8	7	5	4	3	1	2		.05
4000	80					1	1		1					.00
6000	50					4								.01
6000	60					4	2	1			1	1		.02
6000	70					3	5	6	3	1	3			.05
6000	80					5	3	2		2				.02
TOTAL		0	0	0	0	66	44	40	18	21	6	4	0	

TABLE 14. AIRSPEED ACCELERATION OCCURRENCES AND DURATIONS DURING LEVEL FLIGHT FOR EACH FLIGHT CONDITION

AIRSPEED ACCELERATION OCCURRENCES AND FLIGHT TIME (MIN) FOR LEVEL FLIGHT BY FLIGHT CONDITION

MISSION: COMPOSITE AIRCRAFT: UH/TH-1F

			BELOW -2.0 KT/SEC			ABOVE 2.0 KT/SEC		
FLIGHT CONDITION	TOTAL OCCUR	TOTAL TIME	RANGE OCCUR	FLT. COND. DURATION	RANGE DURATION	RANGE OCCUR	FLT. COND. DURATION	RANGE DURATION
STEADY STATE	2460	7889.59	135	654.71	16.43	231	895.32	33.19
TAKE-OFF	10	.93	3	.31	.22	2	.31	.20
COLLECTIVE PUSHOVER	521	51.17	159	14.19	11.55	10	1.32	.53
COLLECTIVE PULL-UP	168	14.91	17	1.84	.95	50	4.19	4.00
FLARE	8	2.43	2	.41	.41	0	0.00	0.00
LEFT TURN	445	221.39	44	23.12	4.67	49	25.97	5.26
RIGHT TURN	431	207.71	34	15.62	3.84	37	19.03	4.35
CYCLIC PUSHOVER	25	3.03	2	.24	.13	8	1.10	.96
CYCLIC PULL-UP	174	27.54	63	9.87	6.21	17	2.70	1.48
LONGITUDINAL REVERSAL	2	.10	0	0.00	0.00	1	.06	.06
LATERAL REVERSAL	8	.52	1	.02	.02	1	.02	.02
RUDDER REVERSAL	1	.07	0	0.00	0.00	0	0.00	0.00

Survey Results

During the current H-1F program, 300 hours of data were acquired, processed, and presented in an operational usage spectrum where time is distributed in flight conditions and mission segments and in parameter ranges for each flight condition and mission segment. The parameters include instantaneous gross weight, center-of-gravity position, density altitude, airspeed, engine torque, rotor speed, vertical acceleration, and lateral acceleration. Since the 300-hour sample consists of 200 hours of utility mission data and 100 hours of training mission data, the composite mission spectrum is the weighted average of the utility and training mission spectra. In the following summary, first for the composite mission spectrum and then for the utility and training mission spectra, the time distributions in the parameter ranges for each flight condition and misison segment as well as those in the flight conditions and mission segments themselves must be considered in evaluating the current operational usage.

Figure 5 presents the operational usage spectrum for the composite mission data. In the comparison of this table with that for the design usage spectrm (Table 4), the more significant differences were as follows. For the mission segments, the operational spectrum had more time in ground operations (16 percent vs 1 percent) and IGE hover (7 percent vs 4.5 percent) but less time in autorotation (0.5 percent vs 3.5 percent) than the design spectrum. For the flight conditions, the operational spectrum had less time in forward level flight (43 percent vs 77 percent) than the design spectrum. Moreover, in the operational spectrum, the rotor speeds during autorotation were occasionally below the normal continuous operating limit, and the airspeed during level flight were lower than those in the design spectrum.

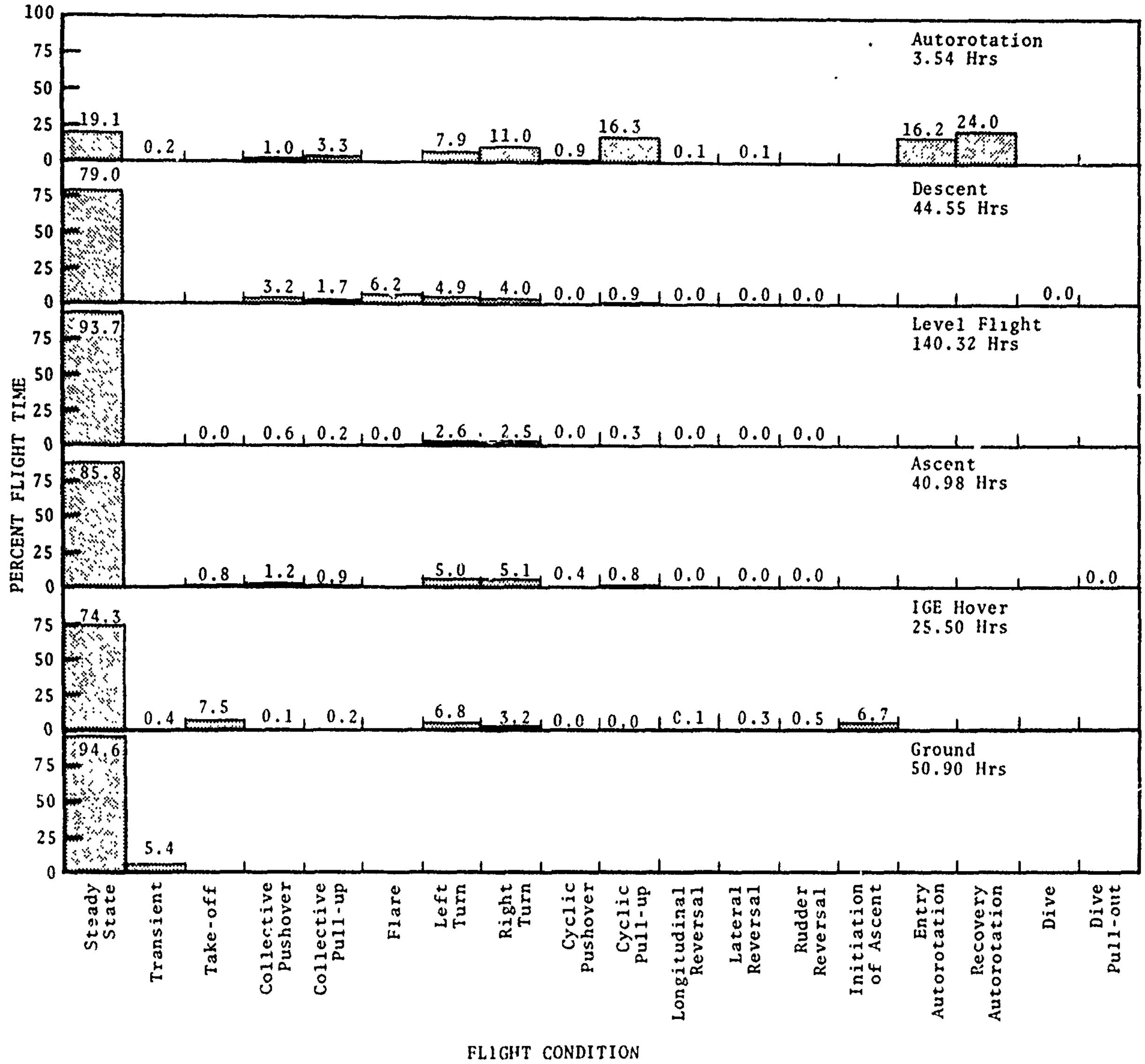

Figure 5. Percent of Time in Each Flight Condition by Mission Segment for Composite Mission Data

Figures 6 and 7 present the operational usage spectra for the utility and training mission data, respectively. The following are the more significant differences between these two spectra. For the mission segments, the utility mission spectrum had more time in level flight (52 percent vs 35 percent) but less time in ascent (12 percent vs 16 percent) and descent (13 percent vs 17 percent) than the training mission spectrum. Consequently, the training mission spectrum had shorter ground-air-ground cycles (15 minutes vs 24 minutes) than the utility mission spectrum. For the flight conditions, the training mission spectrum had more time in turns (4 percent vs 2 percent) than the utility mission spectrum. In both spectra, turns were occasionally accompanied with pull-ups which produced vertical acceleration peaks larger than those sustained during normal turns. Moreover, the training mission spectrum more closely adhered to the airspeed limit since it had less time beyond the never-to-exceed velocity (0.7 percent vs 2.4 percent) than the utility mission spectrum.

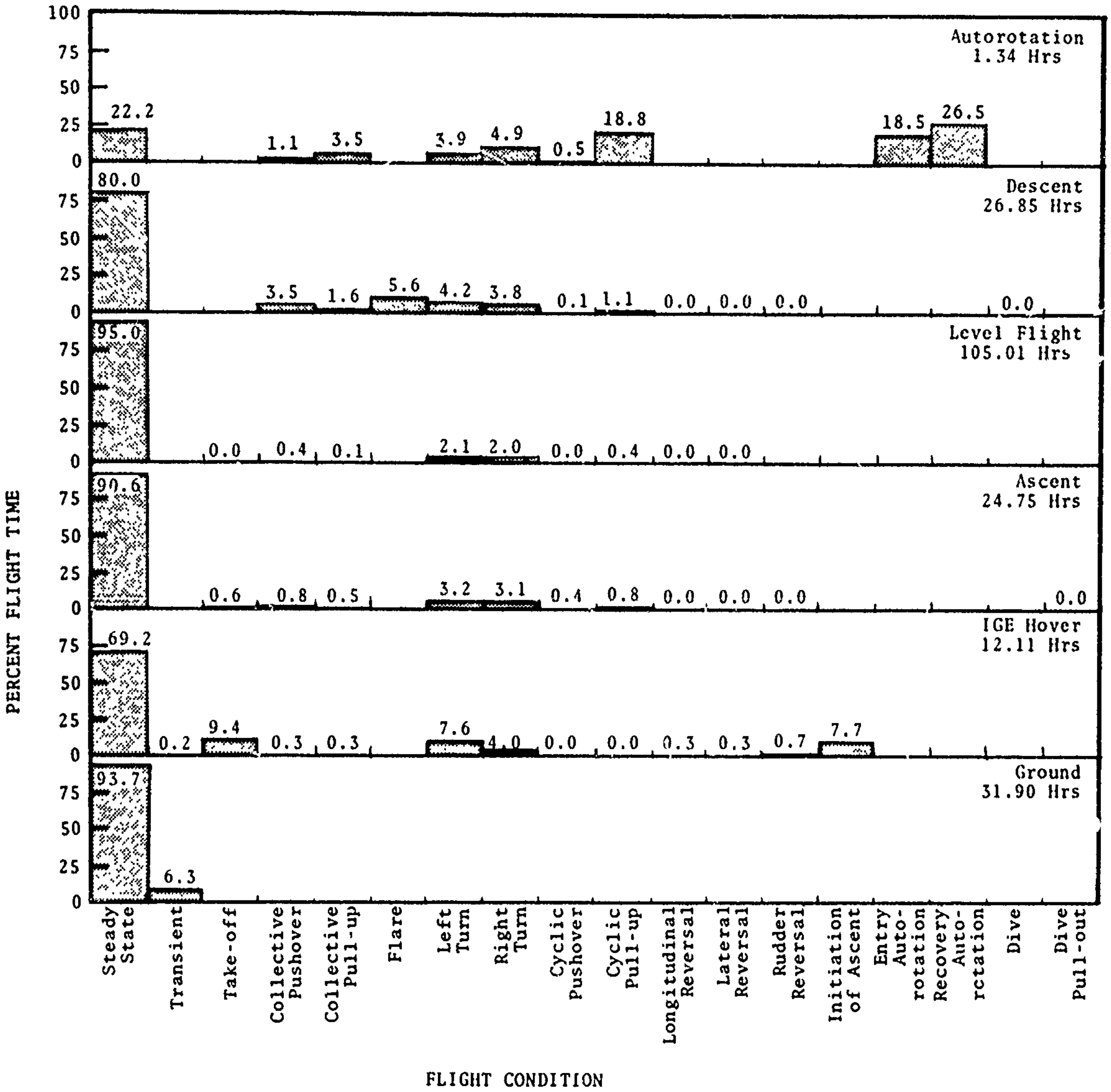

Figure 6. Percent of Time in Each Flight Condition by Mission Segment for Utility Mission Data

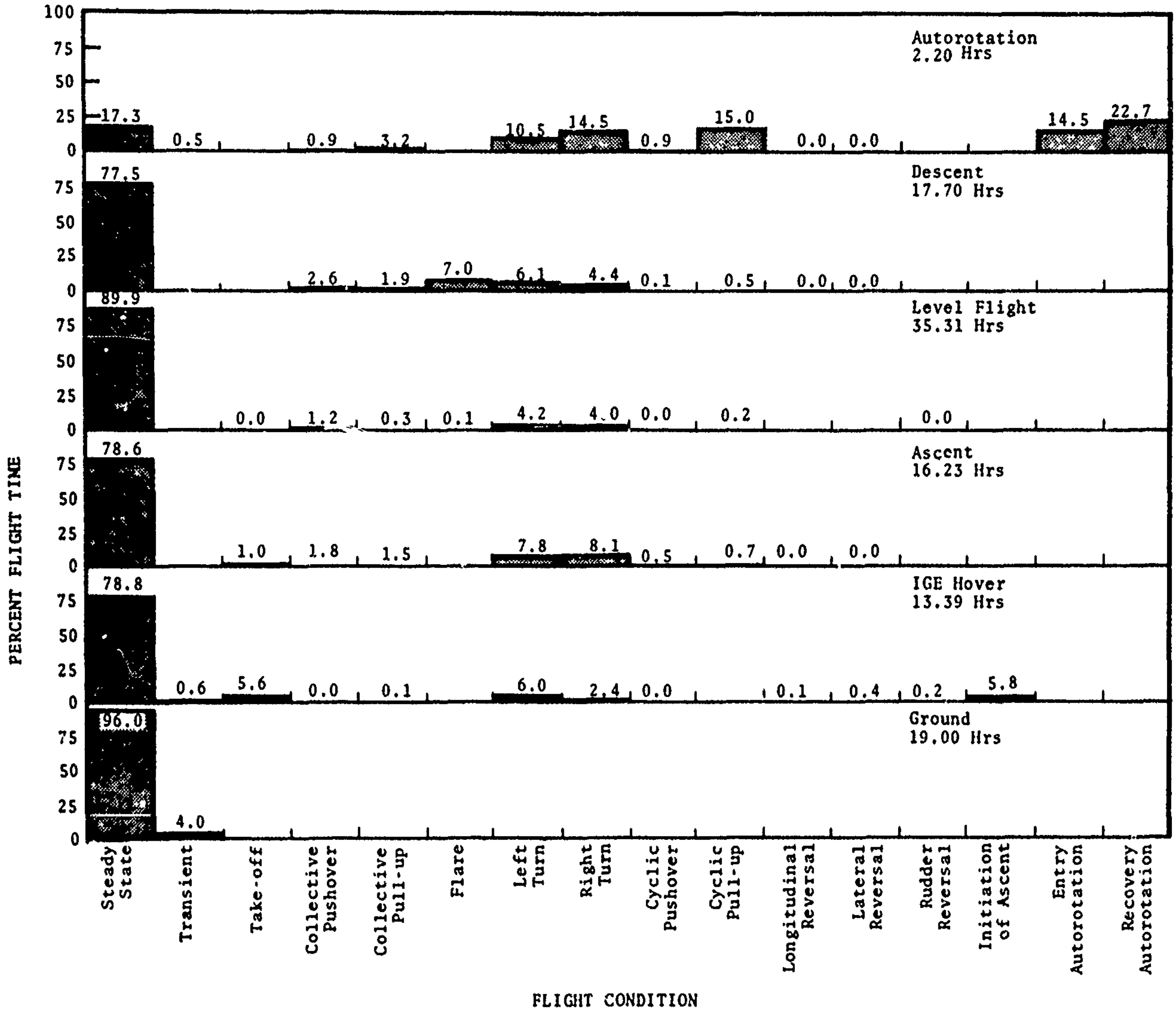

Figure 7. Percent of Time in Each Flight Condition by Mission Segment for Training Mission Data

SUMMARY AND CONCLUSIONS

The change in the requirements for aircraft operational usage recording programs from design criteria to component life data (in keeping with ASIP requirements) has fostered the development of the flight condition recognition (FCR) technique for processing and presenting the operational data. Developed in a study program for the Army, the FCR technique was later applied to 1200 hours of operational usage data recorded on Air Force H-1, H-53, and H-3 helicopters. Not only did the FCR technique prove both technically and economically feasible, but it significantly improved the operational usage spectrum by detailing the flight conditions and mission segments and the parameter ranges within them so that the fatigue analyst can apply the data directly to the computation of the dynamic component lives.

Future applications of the FCR technique will distinguish the operations in asymmetrical maneuvers. In addition, projected developments include (1) recording cargo pickups and drops by monitoring the external sling load, (2) airborne magnetic tape recorders such that steady state segments can be processed directly from the tape data with maneuvering segments displayed for manual editing as in the current application, and (3) ultimately computer programs for maneuvering segment editing to completely automate all data processing operations.

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U. S. NAVY HELICOPTER OPERATIONAL FLIGHT SPECTRUM SURVEY PROGRAM - PAST AND PRESENT

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SUMMARY

U. S. Navy operational requirements for rotary wing aircraft cover an extreme range of missions, from search and rescue over hostile terrain and sea, to mine countermeasures, anti-submarine warfare, attack, cargo replenishment at sea, utility, training, et. al. These vastly different operations necessitated the establishment of a broad program, to measure the structural loading environment during such missions.

In the early 1950's the first in-flight operational survey was performed on the HUP-1 helicopter. Since that time in-flight load spectra surveys have been performed on the SH-3A, CH-53A, CH-46D, UH-1E, TH-1L and HH-2D helicopters. Data from these surveys have been used to establish a more rational basis for static and fatigue structural design criteria as well as to more realistically establish the service lives of existing critically loaded structural components.

In addition to these in-flight load surveys, data has been obtained during operational landing surveys performed on the HUP-1, HTL-3, HTL-4, HTL-5, HO3S, and HRS-1 helicopters. These surveys were performed during the early 1950's. In the late 1960's, data was obtained in confined area landing operations with the CH-53A and CH-46F helicopters. Recently, surveys were made during landing operations with the HH-2D and SH-2F aircraft on the landing platforms of small non-aviation type ships (destroyers and frigates) at sea. These data have been used to up-date the Navy's design and test criteria for structural strength for landing.

Results and details of these surveys as well as the Navy's current plans for future surveys such as flight surveys in mine countermeasures operations with the RH-53D helicopter and at-sea hauldown landing operations with the SH-2F helicopter are discussed.

1.0 INTRODUCTION

The U. S. Navy accepted its first helicopter into the fleet in 1943. This aircraft, designated the HNS-1, was a U. S. Army R-4B Sikorsky helicopter. Usage of this aircraft by the Navy however, was somewhat different from that of the Army, partly because of Navy at-sea mission requirements. The Navy throughout the last thirty years has typically taken helicopters such as the HNS-1 and used them for operations that the aircraft contractor has little or no knowledge of, and thus had not completely accounted for in their structural static or fatigue strength criteria.

Thus during the early 1950's the Navy decided to perform operational surveys to quantitatively define their helicopters' mission profiles. From these surveys would evolve criteria to be used for current and future aircraft structural design. In particular, these surveys provided the quantitative data necessary for realistic static and fatigue strength design.

The first in-flight operational survey with a Navy helicopter was performed in the early 1950's by the National Advisory Committee for Aeronautics (NACA), reference (1), using their airspeed (V), normal acceleration (G), and altitude (H) recorder. Also in the early 1950's the Aeronautical Structures Laboratory (ASL) of the Naval Air Engineering Center performed operational surveys on a number of Navy helicopters to determine operational landing sink speeds. In the mid 1960's, ASL under the direction of the Airframe Division, Naval Air Systems Command (NAVAIR) initiated a program to perform operational flight spectra surveys on different classes of U. S. Navy and Marine helicopters. Also included in this program, were landing surveys performed to measure the landing contact conditions of land and ship based helicopters. During the early 1970's the Naval Air Test Center (NATC) performed landing surveys on ship based helicopters involved in at-sea landing operations aboard small non-aviation type ships (destroyers and frigates).

This paper presents an overview of all the surveys performed to date on U. S. Navy and Marine helicopters. Aircraft descriptions, parameters measured, instrumentation characteristics, results and conclusions obtained from the surveys are presented. Current plans for an in-flight operational survey to be performed on an RH class helicopter, and a landing survey to be performed on the SH-2F RAST program helicopter are discussed. These operational surveys will investigate Airborne Mine Counter Measure operations and at-sea hauldown landing operations, respectively.

2.0 HELICOPTER OPERATIONS

For static and fatigue strength U. S. Navy and Marine Helicopters have in the past been designed to military specification MIL-S-8698, reference (2). However, as a result of in-flight and landing surveys performed on operational aircraft and studies made, for example reference (3), and many years of research and investigation conducted by the Structures Branch, Airframe Division, Naval Air Systems Command, a new structural design specification, AR-56 reference (4), was written for U. S. Navy and Marine helicopters. AR-56 presents recommended mission profiles for various classes of helicopters that should be used for the design fatigue loadings. The data needed to generate the design mission profiles and the landing contact conditions reflected in AR-56 were obtained from surveys involving many Navy and Marine helicopters. The surveys covered in this report are concerned with the classes and type of helicopters given in Table (1).

Landing surveys were performed on the HTL-3, HTL-4, HTL-5, HUP-1, HO3S, HRS-1, CH-46F, CH-53A, SH-2F and HH-2D aircraft. In-flight surveys were performed on the HUP-1, UH-1E, TH-1L, CH-46D, CH-53A, SH-3A and HH-2D. The landing surveys for the HUP-1 and CH-53A aircraft were performed separately from the in-flight surveys; however, the HH-2D's landing and in-flight surveys were performed concurrently. Present plans are to perform an in-flight survey on the RH-53D AMCM helicopter and also an at-sea landing survey on the SH-2F helicopter as used during Recovery Assist Secure and Traverse (RAST) operations as a function of increased ship motion/sea state.

The gross weight range and maximum speed for the above aircraft are given in Table (2). The missions of the helicopters were diverse and ranged from an ASW (anti-submarine warfare) type operation which involves landing on non-aviation ships, to an AMCM operation involving the towing of mine countermeasure sleds which imposes large dynamic tow cable loads on the helicopter.

This unique usage of rotary wing aircraft by the Navy, has been the driving force behind the Navy's need to gather operational usage data.

3.0 PARAMETERS MEASURED

Depending upon the nature of the survey and the data desired, the parameters measured were as wide and varied as the mission requirements. The parameters measured on the various surveys are listed in Table (3). The number and type of parameters measured were chosen to provide the amount of information necessary to define the structural loading environment during normal operational conditions. State-of-the-art instrumentation also played a key role in the selection, as well as maintenance of the equipment and equipment capacity.

All the landing surveys were photographic, however, the HH-2D and SH-2F aircraft were also equipped with vertical velocity doppler radar units to measure landing sink speeds directly. The in-flight surveys did not measure data in the rotating system, with the exception of the HH-2D survey which made use of the aircrafts' existing rotor slip ring assembly to measure rotor blade flap and lead-lag motion. The CH-46D, UH-1E and TH-1L surveys measured vibratory loads in the non-rotating control system. Control tube loads were measured by strain gage bridges installed on the control links which were then calibrated against load in a testing machine.

Present plans are to strain gage the tail rotor pylon for the RH-53D survey in order to look closely at an area of the aircraft that was suspect during a recent accident involving that particular helicopter.

The SH-2F RAST operational landing survey will collect data from landing gear that have been adequately strain gaged and calibrated; doppler radar units attached to the airframe/landing gear, as well as photographic equipment will be employed. Ship motion, (e. g. roll, pitch, heave, etc.) will be obtained simultaneously, so as to collect data sufficient to determine the helicopters' landing contact conditions relative to the ship for various high sea state conditions.

4.0 DESCRIPTION OF INSTRUMENTATION

The in-flight survey performed on the HUP-1, made use of a VGH type recorder. This instrument weighed 15 pounds and had a maximum recording time of 20 hours. The in-flight surveys for the SH-3A, CH-46D, CH-53A, HH-2D, TH-1L and UH-1E aircraft made use of the Century 409 oscillograph, which was capable of recording nine channels of information. Weight of the oscillograph was 20 lbs. Because of the large number of parameters that were monitored on the UH-1E and TH-1L surveys, it was necessary to use two oscillograph recorders. One oscillograph recorded vibratory data, i. e. the control tube loads and two pilot's seat accelerations. The other parameters, i. e. steady state data, were recorded on the second oscillograph. Present plans for the RH-53D survey are to make use of a magnetic tape digital data acquisition system. This system will give the capability to record more parameters, and is expected to greatly reduce data reduction time.

Additional qualitative and quantitative flight information was obtained from pilot report sheets. A typical report sheet is shown in figure (1). Navy personnel gathered the pilot report sheets and changed the flight record magazines immediately after each flight. The number of aircraft instrumented for each of the in-flight surveys is listed in Table (4).

Instrumentation was also needed for the at-sea landing survey to record ship motion data. Ships motion parameters for the SH-2F landing survey on the U.S.S. Biddle, reference (5), were measured with Naval Air Test Center (NATC) FM tape instrumentation. This motion was simultaneously recorded with the aircraft telemetered Pulse Code Modulated (PCM) data. Present plans for SH-2F RAST operational landing survey are to make use of the same type of instrumentation recording system that was used during the U.S.S. Biddle survey.

The landing surveys performed on the HUP-1, HTL-3, HTL-4/5, HRS-1 and HO3S, references 6 through 10 respectively, made use of a 16mm camera acquisition system. The survey performed on the CH-46F and CH-53A aircraft employed the Naval Air Development Center 70mm photo-electronic data acquisition system. Details concerning the set up and location of the cameras are given in references (11) and (12). Description of the photographic system can be found in reference (13).

5.0 DATA REDUCTION

Information collected during in-flight surveys was recorded on oscillograph records which were then processed by the Naval Air Development Center (NADC). A sample oscillograph trace is given in figure (2). Records were edited to determine aircraft gross weight, and flight conditions. The record reading was performed on semi-automatic data-reduction equipment. Typically two methods were used to read the flight data. In the first method, all parameters on the flight record were read simultaneously at intervals of one minute as indicated by the timing trace on the record. In the second method, all parameters were read simultaneously whenever any one of three selected parameters exceeded a specified threshold value, e. g. the following three parameters and their threshold values were used in reference (14).

A. Normal Load Factor (N_Z) - read whenever it was greater than 1.2 or less than +0.8.
B. Engine Torque - read whenever it was greater than 100%.
C. Cruise Guide - read whenever it was greater than 30%.

In addition however, two other criteria must have been satisfied before the above readings were made:

A. The controlling parameter (either N_Z, engine torque, or cruise guide) must increase an amount equal to or greater than one-half of the amount by which it previously decreased.
B. The controlling parameter must decrease an amount equal to or greater than one-half that by which it previously increased.

All parameters on the flight record were read whenever one of the three selected parameters satisfied the threshold values and criteria.

The resulting data punched on computer cards were then transferred to magnetic tape for processing on the IBM 360 computer.

Photographs obtained from the landing surveys were analyzed to determine aircraft landing gear sink speeds, horizontal velocity, pitch and roll angles, as well as pitch and roll rates. A detailed description of the analytical techniques used is given in reference (14).

Typically for the at-sea landing surveys, parameters such as ship roll and pitch motion as a function of sea state were correlated with aircraft motion and the resulting aircraft touchdown attitude relative to the ship was determined, reference (5).

6.0 RESULTS

Typical results are presented for the following groups of surveys:

Group (I)	Land Based Landing Surveys -	HUP-1, HTL-3,4,5, HRS-1, CH-46F, CH-53A
Group (II)	At-Sea Landing Surveys -	HO3S, SH-2F, HH-2D
Group (III)	In-Flight Surveys -	HUP-1, UH-1E, TH-1L, HH-2D, SH-3A, CH-46D, CH-53A

All data obtained during the Group (I) surveys was based on photographic information. Figures (3) and (4) give typical sinking speed distributions obtained from survey data. The landing survey performed on the CH-53A helicopter resulted in an increase in the design loading conditions for landing gear strength. MIL-S-8698 limit sinking speed of 8 fps at the basic design gross weight was revised to a design sink speed of 12 fps in AR-56.

Group II results are given in figures (5) and (6), and Table (5). Data obtained from the U.S.S. Sims landing survey, reference (15), as well as operational data obtained from ship motion surveys, demonstrated a need for an increased strength landing gear for the HH-2D helicopter. The increased strength gave the capability to make 12 fps vertical landings. Operational surveys to date, references (16) and (17), have substantiated this increase to a 12 fps sink speed.

Group III survey results were presented in many histograms and exceedance plots. A summary of the types of these are given in Table (6). Total usable flight hours of data obtained during the in-flight surveys are given in Table (4). Typical data obtained from the group III surveys are given in figures (7) through (10). Data obtained from the UH-1E survey, reference (18), indicated that the aircraft in-flight operates on occasion at a rotor rpm below the minimum limit. Data from this survey is now under study to determine the nature of operating below the minimum rpm. (Note: There have been recent accidents involving this type of helicopter, i.e. the H-1 series, where it was suspected that operation below minimum rotor rpm may have existed and caused the rotor to contact the fuselage.) The survey performed on the HH-2D showed that the aircraft spent a large portion of its flight time in the taxi mode. By operating in this mode, with its high main rotor hub moments, resulted in a lower fatigue life for the hub than was originally calculated. The original mission profile used for the fatigue design loading did not reflect this type of usage for the aircraft. The dynamically loaded components that were monitored on the CH-46D survey reference (19), however, showed that the design loading was sufficient, figure (8) illustrating the margin between the measured and design loads.

7.0 CONCLUSIONS

The operational surveys performed on U. S. Navy and Marine helicopters have been beneficial in establishing quantitative mission profile data. Operational usage data has been used also to update design and test criteria for structural static and fatigue strength. Future surveys are needed however, to constantly assess operational usage.

Present plans call for an in-flight survey on the RH-53D Mine Counter Measures Helicopter and for an at-sea survey of the SH-2F aircraft equipped with recovery assist equipment (RAST) for landing aboard non-aviation ships in heavy seas.

The amount of time and effort spent by personnel at NADC in reducing data collected from oscillograph recorders will decrease sharply when the magnetic tape system is used. Initial studies have indicated that large amounts of data collected with the magnetic tape system can be handled more accurately and processed more quickly.

The data obtained from the in-flight surveys have been used to define the life of fatigue critical components. The parameters measured on the surveys were determined with this in mind and the data was presented in a manner such that the fatigue analyst could compute the service component fatigue lives. It is well known that each helicopter company has its own method of determining component lives and must be given data from an operational survey in a form suitable for its own fatigue analysts to use. With this in mind it is planned to present the data obtained from the RH-53D survey in a manner consistant with that outlined in refernce (20), modified appropriately in response to contractor recommendations.

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9.0 ACKNOWLEDGEMENT

The author wishes to acknowledge the cooperation of personnel of the U. S. Naval Air Development Center, U. S. Naval Air Test Center and the Airframe and Technology Administrators Divisions of the Naval Air Systems Command, (NAVAIR), who contributed their efforts and knowledge in the preparation of this paper. In particular the author wishes to express his appreciation to Mr. Dean Lindquist of NAVAIR for his contributions in the area of operational landing surveys.

TRAINING:	HTL-3,4,5 TH-1L
UTILITY:	HUP-1 UH-1E
OBSERVATION:	HO3S
RESCUE:	HRS-1
CARGO:	CH-46D CH-46F CH-53A
SEARCH:	SH-2F SH-3A
MINE COUNTER MEASURES:	RH-53D
PLANE GUARD: (LAMPS)	HH-2D

TABLE 1. VARIOUS CLASSES OF U. S. NAVY HELICOPTERS INVOLVED IN OPERATIONAL SURVEYS

AIRCRAFT	GROSS WEIGHT RANGE (LBS)	MAXIMUM SPEED (KNOTS) @S.L.
HTL-3	1800-2350	80
HTL-4	1800-2350	80
HTL-5	1800-2350	80
HUP-1	5000-6000	104
HO3S	5393-8070	100
HRS-1	5000-8070	90
TH-1L	7000-9500	120
UH-1E	7000-8500	120
SH-2F	9000-12800	145
HH-2D	9000-12800	140
SH-3A	13000-19000	140
CH-46D/F	14000-23000	144
CH-53A	25000-42000	170
RH-53D	33000-50000	170

TABLE 2. AIRCRAFT GROSS WEIGHT RANGES AND MAXIMUM SPEED

Parameter	Aircraft															
	HTL-3	HTL-4,5	HRS-1	HO3S	HUP-1	HUP-1	UH-1E	TH-1L	HH-2D	SH-3A	CH-46D	CH-46F	CH-53A	CH-53A	SH-2F	RH-53D
Airspeed						✓	✓	✓	✓	✓	✓		✓			✓
C. G. Acceleration (N_Z)						✓	✓	✓	✓	✓	✓		✓			✓
Altitude (Pressure)						✓	✓	✓	✓	✓	✓		✓			✓
Rotor RPM							✓	✓	✓	✓	✓		✓			✓
OAT							✓		✓	✓	✓		✓			✓
Engine RPM							✓									
Cruise Guide Indicator													✓			
Landing Event							✓		✓				✓			
A/C Acceleration Aft XMSSN											✓					
A/C Acceleration (Ramp Door)											✓					
For. Rotor Control Link Load											✓					
Aft Rotor Control Link Load											✓					
C. G. Acceleration (N_Y)									✓							
Rotor Blade Flapping Angle									✓							
Rotor Blade Lead-Lag Angle									✓							
Longitudinal Stick Position								✓								
Lateral Stick Position								✓								✓
Collective Stick Position								✓								✓
Longitudinal Cyclic Control Tube Load							✓	✓								
Lateral Cyclic Control Tube Load							✓	✓								
Collective Control Tube Load							✓	✓								
Sink Speed	✓	✓	✓	✓	✓				✓			✓		✓	✓	
Tail Boom Acceleration								✓								
Pilot Acceleration (N_Z)							✓	✓								
Pilot Acceleration (N_Y)							✓	✓								
Rudder Position								✓								
#1 Engine Torque							✓	✓					✓		✓	✓
#2 Engine Torque																✓
Tail Pylon Strain																✓
Cable Tension																✓
Cable Skew Angle																✓
Altitude (Radar)																✓

TABLE 3. PARAMETERS RECORDED ON U. S. NAVY HELICOPTERS DURING OPERATIONAL SURVEYS

	NUMBER OF A/C IN SURVEY	USABLE FLIGHT HOURS OF DATA OBTAINED
HUP-1	1	37
SH-3A	3	105
CH-46D	4	89
CH-53A	4	133
HH-2D	2	137
UH-1E	3	181
TH-1L	4	300
RH-53D	5	200

TABLE 4. NUMBER OF AIRCRAFT MONITORED DURING IN-FLIGHT OPERATIONAL SURVEYS

LANDING CASE ID	A/C ATTITUDES (DEGREES) ROLL	PITCH	YAW	RATES OF ATT. (DEG/SEC) ROLL	PITCH	YAW	SINK RATE RADARS (FT/SEC) LH	RH	TAIL	CG	A/C G.WT. (KIPS)	ROTOR LIFT (KIPS)	DECK ATTITUDES (DEGREES) ROLL	PITCH	YAW
3.004	-0.7	2.1	-12.0	0.8	1.8	0	6.77	7.40	6.14	7.01	12.20	9.80	-0.8	-0.6	21.0
3.005	-2.8	0.7	62.4	7.9	-3.5	0.6	5.04	5.93	4.41	5.82	12.11	10.42	-3.7	0.2	4.5
3.006	-2.8	4.3	58.1	-6.5	3.5	-3.0	5.20	4.41	4.57	4.81	12.08	12.10	-1.0	-0.4	21.7
3.007	-2.8	0.7	49.6	-0.2	-1.8	-0.2	5.67	5.35	6.30	5.57	12.05	10.60	0	-0.6	12.8
3.008	1.4	5.0	28.3	-10.4	6.9	0	4.41	3.15	5.67	3.92	11.96	10.55	-2.3	-0.2	15.6
3.009	-2.8	5.0	11.3	0	5.3	0	5.51	7.87	6.61	6.61	11.90	10.56	-1.4	0	1.0
3.010	-1.4	-0.7	9.2	2.0	2.4	-3.9	5.83	6.93	5.67	6.32	11.87	10.45	-1.2	-0.2	17.5
3.011	0	1.4	-2.1	-0.2	2.0	0	5.04	5.04	5.67	5.08	11.83	9.82	-0.4	-1.0	10.8
3.012	-1.4	0	-7.1	0	0.6	0	7.09	8.98	7.72	7.06	11.82	9.72	-1.4	-0.4	6.2
3.013	-1.4	2.8	-12.0	-0.2	-0.2	0	3.15	8.03	2.20	5.21	11.78	11.28	-3.9	0	3.4
3.014	-0.7	1.4	-14.2	3.9	4.7	-5.5	5.20	6.77	6.30	5.98	11.69	9.00	-0.8	-0.2	21.2
3.015	-0.7	0.7	-12.0	0	4.5	-0.2	4.72	6.30	6.93	5.58	11.65	11.43	-0.8	-0.4	10.0
3.016	-1.4	-0.7	-12.8	-0.4	1.4	0	4.72	5.35	6.46	5.09	11.61	9.75	-2.3	-0.2	19.8
3.017	-2.1	-2.1	-16.3	0.4	-3.5	4.9	5.67	4.88	3.94	5.26	11.56	5.00	-4.5	0	20.0
3.018	0	-0.7	-16.3	-2.0	2.6	-5.1	7.56	6.93	8.66	7.33	11.51	7.45	-1.4	-0.4	21.0
3.019	-2.8	1.4	-18.4	0	0.6	-0.2	6.30	6.61	8.03	6.54	11.46	8.22	-2.3	-0.4	0.2
3.020	-1.4	0.7	-13.5	-0.8	4.7	0.8	8.66	7.87	9.45	8.36	11.41	8.02	-0.6	-0.4	3.3
3.021	- -	0	-7.8	-4.9	2.0	0	7.09	5.83	7.56	6.55	11.33	9.12	-2.3	-0.2	6.7
3.022	- -	-2.1	-7.1	-0.2	-2.2	12.4	8.50	8.82	8.19	8.64	11.27	6.32	-1.6	-0.2	9.4

TABLE 5. PARAMETERS RECORDED ON SH-2F
AT-SEA LANDING SURVEY

TH-1L STUDY

PERCENT TOTAL FLIGHT TIME vs AIRSPEED
PERCENT TOTAL FLIGHT TIME vs ALTITUDE
PERCENT TOTAL FLIGHT TIME vs AIR TEMPERATURE
PERCENT TOTAL FLIGHT TIME vs ROTOR RPM
PERCENT TOTAL FLIGHT TIME vs GROSS WEIGHT
PERCENT TOTAL FLIGHT TIME vs FLIGHT REGIMES
PERCENT TOTAL FLIGHT TIME vs FLIGHT DURATION
PERCENT TOTAL FLIGHT TIME vs TORQUE
PERCENT TOTAL FLIGHT TIME vs HORSE POWER
CUM. FREQ. PER 1000 FLT. HRS. vs POSITIVE A/C C.G. ACCEL.
CUM. FREQ. PER 1000 FLT. HRS. vs NEGATIVE A/C C.G. ACCEL.
CUM. FREQ. PER 1000 FLT. HRS. vs N_y ON TAIL BOOM
FREQUENCY OF EXCEEDING A GIVEN CONTROL TUBE LOAD
FREQUENCY OF EXCEEDING A GIVEN TOUCHDOWN VERT. VELOCITY
FREQUENCY OF EXCEEDING A GIVEN PILOT SEAT ACCELERATION
A/C C.G. ACCEL. (VERTICAL) vs GROSS WEIGHT
C_t vs FLIGHT TIME
μ vs FLIGHT TIME

TABLE 6. TYPICAL HISTOGRAMS AND EXCEEDANCE
CURVES OBTAINED FROM IN-FLIGHT SURVEYS

TO THE PILOT: This helicopter is equipped with a flight recorder which automatically produces a record of airspeed, normal acceleration, altitude and other flight environment conditions. This information will contribute to improvement of structural design requirements and fatigue life determination. The following information is required for proper evaluation of the recorded data. Your assistance is greatly appreciated.

DATE OF TAKEOFF ______________ FLIGHT DURATION ______________

AIRCRAFT MODEL ______________ SQUADRON NUMBER ______________

AIRCRAFT SERIAL NUMBER ______________ FLIGHT PURPOSE CODE ______________

MISSION DESCRIPTION ______________

LOCATION OF OPERATIONS	Takeoff	Landing
TIME (LOCAL)	Takeoff	Landing
FUEL WEIGHT	Takeoff	Landing
CARGO WEIGHT	Takeoff	Landing

ALTIMETER SETTING ______________

SEA STATE AT LANDING ______________

ANGLE OF APPROACH TO SEA STATE ______________

LANDING IMPACT: MILD AVERAGE SEVERE

REMARKS: (UNUSUAL MANEUVERS, WEATHER CONDITIONS, ETC.)

FIGURE 1. TYPICAL PILOT REPORT FORM

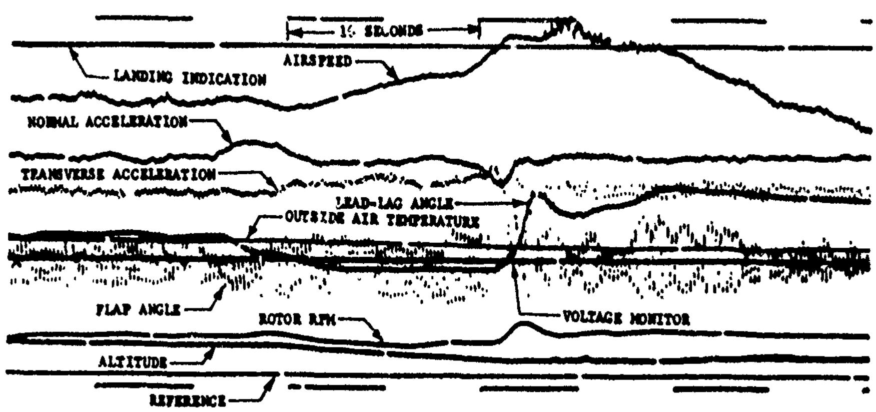


FIGURE 2. SAMPLE OSCILLOGRAPH TRACE

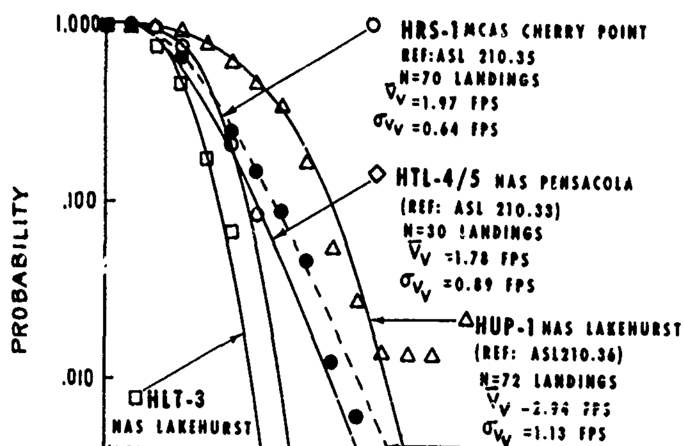


FIGURE 3. SUMMARY OF SINK SPEED DISTRIBUTION

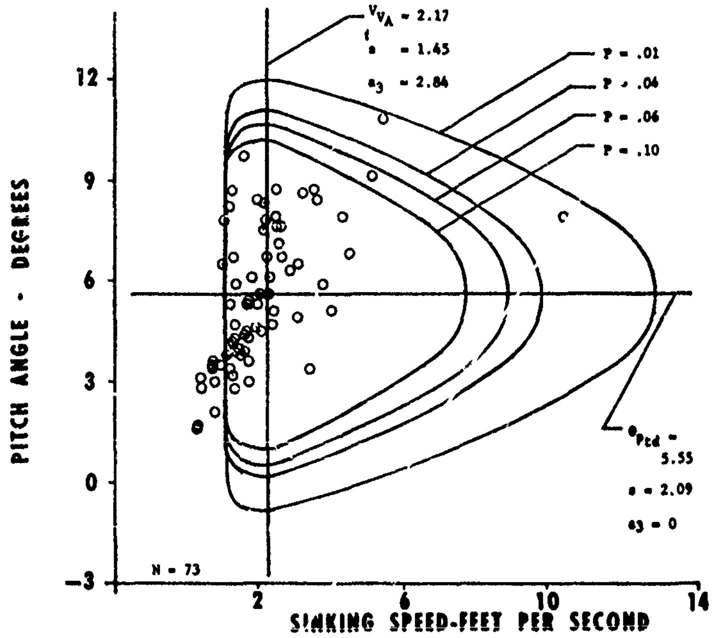


FIGURE 4. CH-53A LANDING SURVEY - PROBABILITY THAT COMBINED VALUES WILL LIE OUTSIDE A GIVEN ENVELOPE

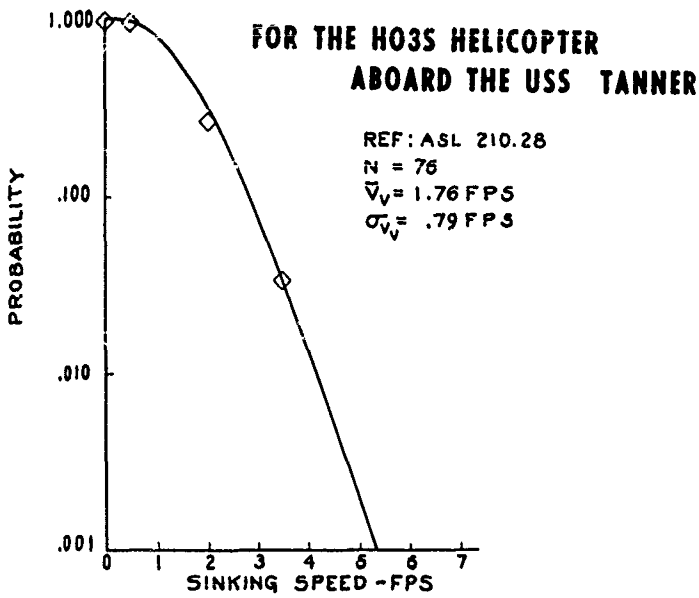


FIGURE 5. HO3S AT-SEA LANDING SURVEY - SUMMARY OF SINK SPEED DISTRIBUTION

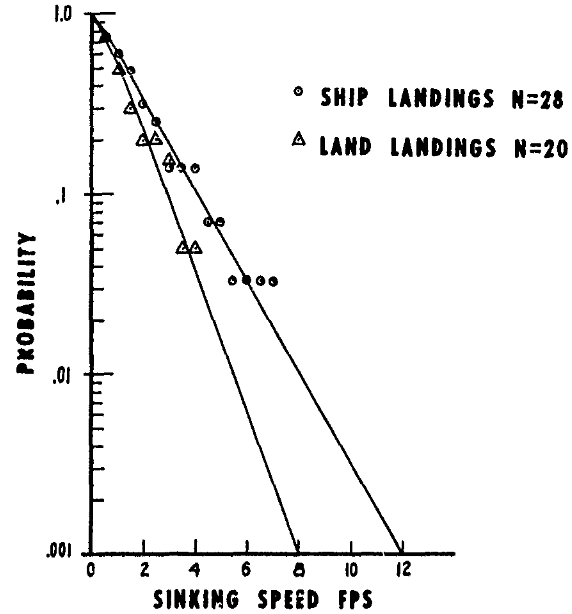


FIGURE 6. HH-2D SINKING SPEED DISTRIBUTION
DV-98 FLIGHT LOADS STUDY

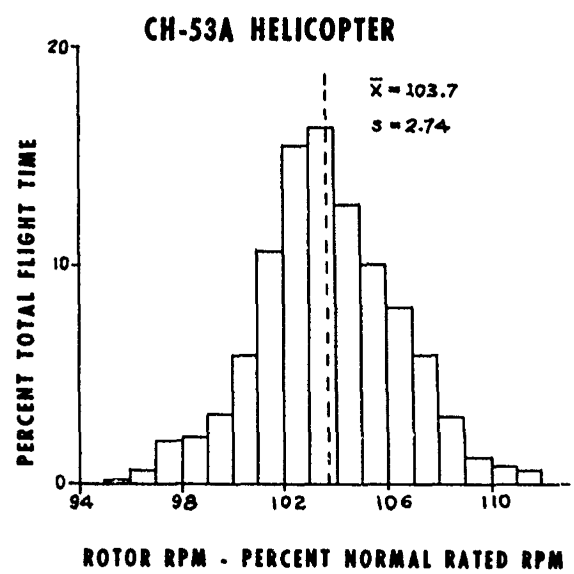


FIGURE 7. CH-53A IN-FLIGHT SURVEY - PERCENT
TOTAL FLIGHT TIME VERSUS ROTOR RPM

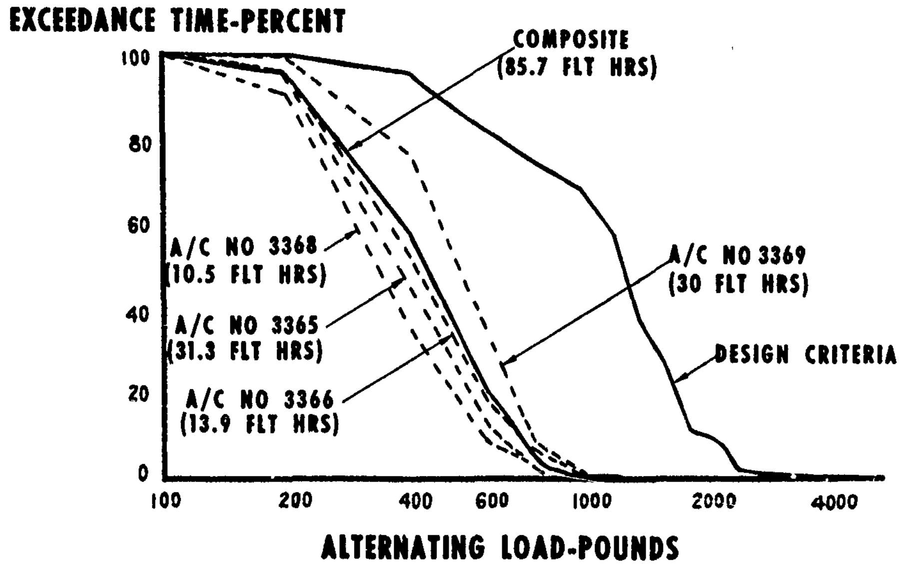


FIGURE 8. CH-46D IN-FLIGHT SURVEY - PERCENT EXCEEDANCE TIME VERSUS AFT LONGITUDINAL LINK LOADS

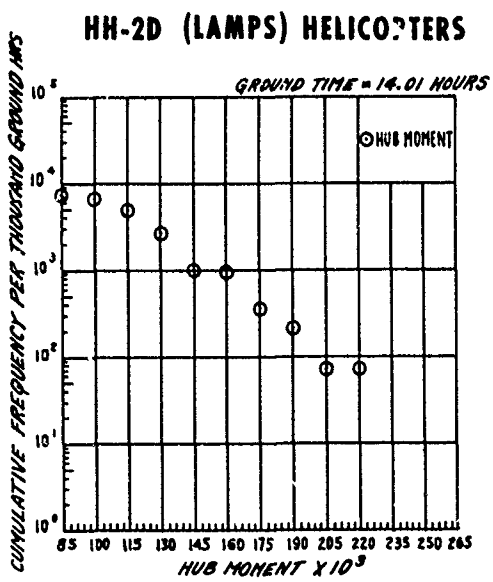


FIGURE 9. HH-2D IN-FLIGHT SURVEY - FREQUENCY OF EXCEEDING A GIVEN HUB MOMENT

UH-1E HELICOPTER

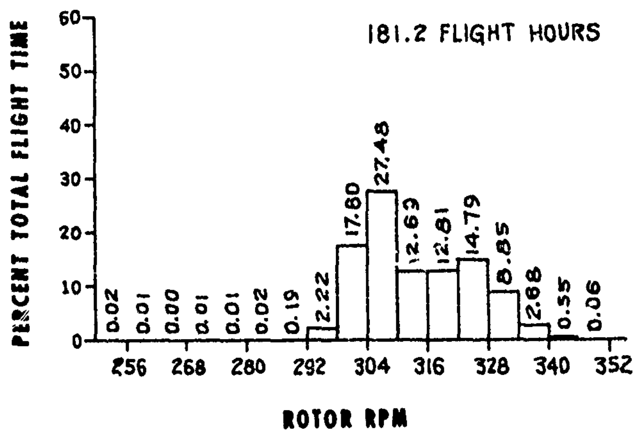


FIGURE 10. UH-1E IN-FLIGHT SURVEY - PERCENT TOTAL FLIGHT TIME VERSUS ROTOR RPM

CRITIQUE AND SUMMARY

of the Specialists Meeting on

HELICOPTER DESIGN MISSION LOAD SPECTRA

Frederick H. Immen
Chief of the Advanced Systems Research Office
of the U.S. Army Air Mobility R&D Laboratory
Ames Research Center
Moffett Field, California

INTRODUCTION

I feel very honored to have the opportunity of critiquing these excellent papers on mission spectra definition. My experience in this area stems from the frustrations of defining fatigue loads criteria for the Chinook CH-47 models A, B, and C helicopters, computing "safe" replacement lives for critical dynamic components and guaranteeing that these lives defined a high probability of successful operation for a large fleet of aircraft. The CH-47 found its way into the thick of battle in Viet Nam. As calendar time and flight time accumulated on these aircraft and service reports, accident reports and incident reports started to return to us, it became obvious that we didn't have adequate knowledge of how the aircraft were being used in actual operational conditions. One element of the failure investigation, which all too often became necessary, consisted of conjecture about what uses the aircraft was being put to. I have observed during these presentations that the quest for accurate knowledge of aircraft usage that could critically affect fatigue life continues and the tools used in this quest are becoming more sophisticated.

Before commenting on the efficacy of these tools as described, let me describe another tool which is being developed in an exploratory way by the US Army Air Mobility R&D Laboratory.

As part of our structural integrity program for helicopters, we are currently developing a low cost structural integrity monitoring system which will permit the determination of in-service fatigue damage of critical structural components. The approach includes designing a parametric data recording system for the Cobra AH-1G helicopter. Ten flight recorders will be fabricated, qualified, and flight tested. Two ground-based data retrieval units will be designed and fabricated. Supporting computer software will be prepared.

The flight recorder has four basic elements; it is solid state, has a memory device, and has a microprocessor for segregating the data. It has an integrated circuit memory and an auxiliary battery pack in case of main power failure and for holding the data during ramp time. This recorder operates on a slightly different principle than the usual flight recorder. Since the memory is solid state, it does not suffer the drawback of tape recorders which require moving elements, loading tape, and complex mechanical mechanisms which may easily develop faults. Information is abstracted from the flight recorder by use of a ground data processing unit which performs initial processing of the data such as conducting parity checks and other types of checks to verify the data. The data is then recorded on a standard tape cassette. The cassette is then forwarded or transmitted electronically to a central agency which will conduct the fatigue damage assessment.

Parameters which are recorded are as follows:

Indicated airspeed.

Static pressure.

Outside air temperature.

Main rotor RPM.

Roll attitude.

Vertical acceleration.

Landing gear touchdown.

Engine torque.

Flight loads are determined by measuring a combination of the above parameters which have been found from flight strain survey data to be representative of one or several critical damaging flight conditions. As an example, a damaging pull-up might be defined by the following combination of parameter measurements:

Vertical acceleration above threshold, Nz>1.3g

Airspeed below threshold, $A/S \leq .7\ V_L$

Roll attitude below threshold, $\beta<10°$

Density altitude above threshold

A damaging turn might be defined as:

Vertical acceleration between threshold, $1.3 \leq Nz < 1.5$

Airspeed between threshold, $.65\ V_H \leq A/S < .8\ V_H$

Roll attitude above threshold, $\beta \geq 10°$

In this way a large number of flight conditions can be recorded. The length of time in some flight conditions is recorded and the number of times other flight conditions occur are recorded. The flight loads are calculated by using this information and an algorithm developed from Bell Helicopter flight strain surveys. Although this method is not as accurate as measuring the loads directly, the flight condition technique allows a greater economy in the number of channels required to define flight loads, and they are all in the fixed system. A potential improvement of this system is to add some channels for actual strain on critical components. This would serve two purposes: the fatigue history of those components would be more accurately known and the actual strain could be used to improve the algorithm for calculating loads on other components.

A problem area which has not been resolved in the development of a completely integrated structural integrity monitoring system is the technique of recording aircraft gross weight and including this important parameter in the fatigue damage computation. Several avenues are open for exploratory development considering trade-offs of cost and accuracy. In descending order of accuracy (but also descending order of cost), the following approaches are being considered:

1. Measure gross weight directly.
   a. Calibrated transmission mounting measurements.
   b. Landing gear oleo pressure.
2. Combine measurable parameters to compute G.W.
3. Utilize pilot logs.
4. Use historical gross weight statistics.

SUMMARIZATION

All the techniques and systems for defining mission spectra that have been described by my august colleagues and by myself will do the job and will do it well enough to provide safe operation of the aircraft for which they are devised when combined rationally with the other two members of the triumvriate; namely, measured loads (strain) and fatigue strengths.

Mr. Liard - Argued convincingly for parameter measurement rather than direct loads measurement in mission spectrum development. He spoke of Aerospatiale's program of developing critical component replacement times for aircraft of the same model but which fly different missions by application of correction coefficients to maintenance data.

Mr. Hall - Spoke of Westland's attempts at collecting statistical samples of data for definition of mission spectra by use of pilot reports in order to acquire large samples of data over long periods of time. He concurred with Mr. Liard that mission spectra should be defined by parameter measurement and described a comprehensive program planned for a survey on the Sea King helicopter in the near future.

Mr. Martin - Spoke of the new US Air Force mission spectra definition program called Flight Condition Recognition (FCR) with which the operator can define in much greater detail the actual flight conditions that the aircraft is experiencing.

Mr. Reichert - Emphasized the unique characteristics of the Boklow hingeless rotor and the special considerations which must be observed in developing a mission loads spectrum tailored to these unique characteristics. The modern hingeless rotor design implies that a conservative mission spectrum can be accommodated without undue weight penalty.

Mr. Malatino - Described several mission spectra surveys conducted by the US Navy mainly to investigate specific flight condition problems peculiar to Navy operations.

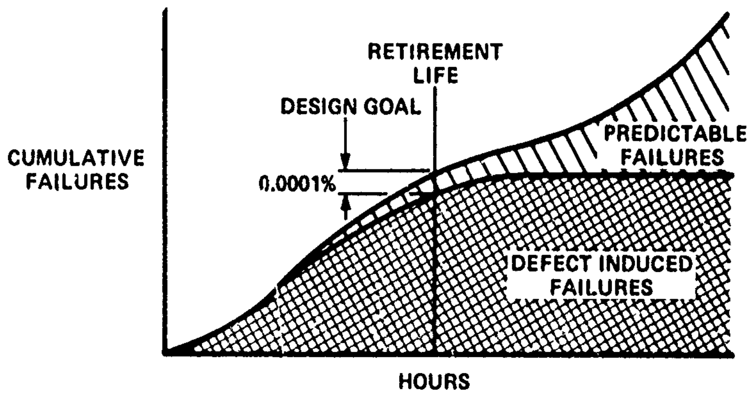
And I described the US Army's exploratory program to monitor structural integrity of a fleet of helicopters by use of an on-board parameter measuring system and automated component life computation system.

CRITIQUE

In the case of safe life design where safety of flight relies on definition of a very high probability of nonfailure for critical components, it is ironic that statistical manipulations are performed on measured loads and fatigue strengths to define life but the loads profile, which is equally powerful in defining life, is defined by empericism, or at best by a small sampling of flying hours and measured parameters. Admittedly, things are getting better than they were 25 years ago when no attempts at probabilistic analysis were made and 15 years ago when only strengths were treated statistically. Perhaps with the techniques described today, even more probabilistically stringent life computation methods can be devised.

However, we shouldn't think that this added precision in measuring mission spectra is going to solve all our problems. Looking back over operational helicopters' dynamic system failures (see figure), it turns out that there are very few failures that one could say were caused because statistics predicted them. Also, of the defect-induced failures, the defect is seldom attributable to a defect in the mission spectrum.

CUMULATIVE FAILURES vs. FLIGHT HOURS



Measuring mission loads spectra in actual service is a necessity but it is addressing the problem of structural integrity (structural reliability) from a position of desperation (Thoreau said that man lives his life in quiet desperation. That certainly is true for the helicopter stress man.) because the designer has not been completely successful in designing components which can be replaced on condition; i.e., replaced when the condition of the part has obviously degraded in a soft failure mode and is replaced when it needs to be. How much better it would be to concern ourselves with mission spectra to define economical operation of fail-safe components rather than defining lives which represent the catastrophic thresholds of use. Several of the speakers stated that some "safe-life" components are inevitable in helicopter design. I don't agree that a fully fail-safe goal can't be reached. Perhaps we are not trying hard enough.

Another approach to fatigue design is to use a highly truncated fatigue loads spectrum during initial design which assumes the aircraft operates in the worst loading regime within G load, power, vibration, performance, or stability limits 100% of the time. Since fatigue strength is logarithmically related to life and helicopters usually operate at the high cycle end of the S-N curve, it doesn't cost much weight to accomplish an acceptable safe life using this conservatism. Also, usually only a small portion of a component must be reinforced to achieve a large life improvement. Problems then arise when component strengths don't come up to expectations and the designer starts to manipulate his loads spectra to eliminate "conservatism." This might take the form of assuming gross weight splits or assuming a large portion of time is spent in some docile flight regime. In the long run, this may turn out to be more expensive from a life cycle cost basis than redesigning the component immediately. The confidence that the designer gains to manipulate his mission spectra, of course, comes from his mission loads spectra measurement system. It seems chancy to me to use measured spectra to increase the safe life of components above what had been conservatively defined in the initial design of the component, even using the most sophisticated spectrum measurement system described today. Conversely, if the measurement system warns of a critical safe life problem, take heed. It could be the harbinger of serious operational fatigue failures.

REPORT DOCUMENTATION PAGE			
1. Recipient's Reference	2. Originator's Reference AGARD-CP-206	3. Further Reference ISBN 92-835-0172-1	4. Security Classification of Document UNCLASSIFIED
5. Originator	Advisory Group for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France		
6. Title	HELICOPTER DESIGN MISSION LOAD SPECTRA		
7. Presented at	the 42nd Meeting of the Structures and Materials Panel, held in Ottawa, Canada on 8 April 1976		
8. Author(s)	Various		9. Date August 1976
10. Author's Address	Various		11. Pages 72
12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.		
13. Keywords/Descriptors	Heliccpters Life (durability) Components	Design Fatigue (materials) Loads (forces)	

14. Abstract

As mission requirements cause the utilization of any helicopter fleet to vary significantly over the fleet's life time, the problem of predicting the structural life of fatigue-critical components becomes more complex.

A prime factor in the inability to accurately predict component lives is the lack of adequate mission load-spectra data, which is compounded by the increased aircraft performance that has resulted in more sophisticated mission profiles. The impact of the fatigue spectra on life has been well documented showing that, for the same load levels, merely changing loading sequence dramatically changes the number of cycles to failure. A more accurate representation of the effects of not only load levels but load sequence will provide more realistic fatigue analyses and life predictions.

In order to stimulate the collection of such data and to pool the approaches of the helicopter producing NATO Nations, a Specialist Meeting was organized. The presentations, an overview and the subsequent discussions are published in this volume. Among the subjects covered are the development of load spectra for design, the adaptation of this for life predictions in the field, and mission load data gathering techniques.

AGARD Conference Proceedings No.206
Advisory Group for Aerospace Research and Development, NATO
HELICOPTER DESIGN MISSION LOAD SPECTRA
Published August 1976
72 pages

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ISBN 92-835-0172-1

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Printed by Technical Editing and Reproduction Ltd
Harford House, 7 9 Charlotte St, London W1P 1HD

ISBN 92-835-0172-1