AD-757 267

ADVANCED ROTORCRAFT. VOLUME I

Advisory Group for Aerospace Research and Development
Paris, France

February 1973

DISTRIBUTED BY:

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

AGARD CONFERENCE PROCEEDINGS No. 121

on

Advanced Rotorcraft

Volume I

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY
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AGARD-CP-121

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.121

ADVANCED ROTORCRAFT

Volume I

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document may be better studied
on microfiche.

Papers presented at the 39th Meeting of the Flight Mechanics Panel of AGARD held at NASA Langley Research Center, Hampton, Virginia, USA, on 20–23 September, 1971.

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Published February 1973

629.735.45

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

PREFACE

In November 1954 the AGARD Flight Mechanics Panel held a Meeting on Rotary Wing Aircraft with emphasis on flight testing. Since that Meeting the Panel's technical program has stressed developments in non-rotary wing V/STOL aircraft. However, it was recognized that significant advances in the field of rotorcraft were occurring in the late 1950's and 1960's. In 1969 the Panel decided to organize a Symposium on the subject of advanced rotorcraft. The objectives of this Symposium were to:

- Review the experiences gained from existing helicopter operations
- Review the lessons obtained from flight tests of experimental helicopters
- Discuss the future of advanced rotorcraft
- Discuss improved ground test facilities for research and development of new rotorcraft.

The Symposium brought together research scientists, designers, flight test engineers and military personnel for an in-depth exchange of views on the past, present and future of rotorcraft. The papers presented at the Symposium are published in Volume I of the Proceedings. Volume II contains the lengthy open discussions which followed the presentations of the papers and the two round table discussions on the subjects of Advanced Rotorcraft Technology and Special Large Wind Tunnels for V/STOL and Helicopter Studies.

The Symposium was held at the NASA Langley Research Center, Hampton, Virginia, USA which gave the participants an opportunity to see many ground and flight test facilities used in helicopter research.

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AVANT PROPOS

En Novembre 1954, le Groupe de Travail de la Mécanique du Vol, de l'AGARD, organisa une réunion sur les Aéronefs à Voilure Tournante et plus particulièrement sur leurs essais en vol. Depuis cette réunion, le Groupe a mis l'accent, dans son programme technique, sur les développements dans le domaine des avions ADAC/ADAV à voilure non tournante. Il fut reconnu toutefois que des progrès importants avaient marqué les giravions vers la fin des années 50 et dans les années 60. Le Groupe décida donc, en 1969, d'organiser, sur le thème des giravions de conception avancée, un symposium où seraient poursuivis les objectifs suivants:

- Dresser le bilan de l'expérience acquise sur les hélicoptères en service
- Passer en revue les leçons tirées des essais en vol sur hélicoptères expérimentaux
- Etudier l'avenir des giravions de conception avancée
- Etudier les installations au sol améliorées pour la recherche et le développement de nouveaux giravions.

Ce Symposium réunit des chercheurs, des concepteurs, des ingénieurs d'essais et des militaires, et les fit participer à des échanges de vue approfondis sur le passé, le présent et l'avenir des giravions. Les communications présentées à l'occasion de ce Symposium font l'objet du Volume I du Compte-Rendu. Le Volume II est consacré aux importants débats publics qui suivirent la présentation des exposés, ainsi qu'aux deux "tables rondes" sur la Technologie des Giravions de Conception Avancée et sur les Souffleries Spéciales de Grandes Dimensions pour Etudes sur appareils ADAC/ADAV et sur hélicoptères.

Le Symposium eut lieu au Centre de Recherches Langley, de la NASA, à Hampton, Virginie, aux Etats-Unis, ce qui permit aux participants de rendre visite à de nombreuses installations d'essais en vol et au sol pour recherches sur hélicoptères.

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CONTENTS

LE VOL TACTIQUE DE L'HELICOPTERE ET LES REPERCUSSIONS SUR SA CONCEPTION

– Commandant M.BERTHOUX – Aviation Légère Armée de Terre –
– SERVICE TECHNIQUE DE L'AERONAUTIQUE –
4, avenue de la Porte d'Issy – PARIS 15ème

I°/ – AVANT PROPOS –

Le besoin pour les forces terrestres de réaliser une certaine aéromobilité constitue une des conséquences de l'apparition de l'arme atomique sur le champ de bataille.

Or, si les études relatives à l'emploi de l'arme atomique mettent en lumière ce caractère très mobile que doit revêtir désormais la manoeuvre des forces terrestres, elles soulignent également l'obstacle à cette mobilité que risque de constituer un terrain bouleversé par les frappes nucléaires.

C'est alors qu'apparait tout l'intérêt qu'offre l'emploi d'unités tactiques disposant en propre de véhicules aéromobiles leur permettant de mener un combat terrestre à partir d'un espace dit "aérien".

Aujourd'hui le véhicule aéromobile standard est "l'hélicoptère". Donne-t'il entièrement satisfaction à son utilisateur terrestre, pour mener sa bataille ? Sans faire preuve de trop d'intransigeance la réponse pourrait être négative, car à la limite le choix du combattant terrestre se porterait plutôt vers un véhicule à vocation terrestre capable de voler. Est-ce du domaine de l'utopie ? Peut-être !

Par contre, en s'en tenant à la technique éprouvée que constitue la voilure tournante, celle-ci pourrait donner plus grande satisfaction à son utilisateur si les personnes chargées de la conception de ce type de matériel percevaient mieux les contraintes d'emploi d'une telle plate-forme dans un combat terrestre.

Cet article a pour objet d'essayer d'exprimer certaines réflexions sur différents aspects de ce problème à la lumière de l'expérience acquise par l'Aviation Légère de l'Armée de Terre Française dans le domaine de l'aéromobilité.

Après avoir examiné l'environnement tactique nous essayerons de rechercher dans quels domaines la conception d'un hélicoptère pourrait en être influencée.

II°/ – ENVIRONNEMENT TACTIQUE –

2.1 – Eléments de base :

Nous retiendrons pour cadre, un conflit classique, sous menace nucléaire intéressant un théâtre d'opération européen.

Le fait nucléaire imposant
- un élargissement du champ de bataille,
- une dispersion et une concentration rapide des moyens,

la mobilité sera un des éléments fondamentaux des moyens de combat.

L'hélicoptère pouvant s'affranchir des obstacles naturels et de ceux créés par la frappe nucléaire constitue un des véhicules les mieux adaptés à ce genre de combat, dans la mesure où il pourra survivre dans ce milieu hostile afin que son équipage puisse mener à bien la mission reçue.

Que cette mission soit :
- .la lutte antichar
- .la lutte antihélicoptère
- .la reconnaissance
- .l'appui feu
- .le transport de combattants,

elle ne pourra être considérée comme réussie que si les chars et les hélicoptères sont détruits, l'ennemi repéré, les installations adverses neutralisées. Mais pour ce faire, il aura été capital que l'hélicoptère parvienne avec tout son "potentiel de combat" de son lieu de départ jusqu'au meilleur emplacement de tir pour utiliser ses armes, jusqu'à la zone de poser choisie pour y débarquer ses combattants dans les délais prévus, jusqu'à l'emplacement d'observation en x, y pour surveiller les mouvements de l'adversaire.

Cette phase de la mission représentant le "transit d'intervention" compte pour 50 % dans la réussite globale de la mission bien que se déroulant dans un milieu particulièrement hostile.

Nous arrivons ainsi à la notion de vulnérabilité, qui doit être prise dans son sens le plus large, comme un évènement défavorable interrompant ou interdisant la mission et plus particulièrement la phase de transit d'intervention.

2.2 – Vulnérabilité :

2.2.1. Trois types de vulnérabilité doivent être pris en considération :

- la vulnérabilité aux défaillances mécaniques,
- la vulnérabilité à l'environnement naturel (nuit – météorologie défavorable – neige – givrage – grêle),
- la vulnérabilité à l'environnement du combat.

Les deux premières concernent l'hélicoptère en général. Mais pour le cas particulier de l'hélicoptère militaire, il est fondamental de rechercher à diminuer sa vulnérabilité à l'environnement du combat, pour augmenter le pourcentage de réussite de ses missions.

2.2.2. Cet environnement du combat se concrétise par deux facteurs fondamentaux :

- facteur défavorable : l'ennemi caractérisé par son déploiement sur le terrain, ses moyens de détection et de repérage, le nombre et les performances de ses armes,

- <u>facteur favorable</u> : le terrain avec ses formes générales (vallées, collines), ses obstacles naturels (rideaux d'arbres, bosquets) et artificiels (villages, constructions isolées), qui permettent un défilement efficace aux vues et aux coups directs de l'ennemi.

Ces considérations nous font déboucher sur la notion de protection des hélicoptères qui représente un problème particulièrement complexe et qui ne se limite absolument pas à une utilisation plus ou moins rationnelle de plaques de blindage.

2.3 - <u>Protection des hélicoptères</u> :

2.3.1. <u>But</u> :

Diminuer la vulnérabilité de l'hélicoptère
- à la détection optique, électronique, acoustique initiatrice du feu ennemi,
- au feu des armes ennemies ;

en vue de
- lui permettre d'accomplir sa mission,
- d'augmenter ses possibilités de survie dans la zone des combats,
- permettre une remise rapide en service, après atteinte par le feu ennemi.

2.3.2. <u>Type de protection</u> :

Deux types de protection sont à considérer :
- protection active
- protection passive

Elles peuvent être définies ainsi :

a) - La protection active a pour objet de diminuer les possibilités d'atteinte de l'hélicoptère par le feu ennemi.

b) - La protection passive a pour objet de permettre à l'hélicoptère de <u>subir</u> le feu ennemi <u>sans</u> ou <u>avec</u> un certain endommagement.

2.3.3. <u>Utilisation des protections</u> - <u>Moyens envisageables</u> :

2.3.3.1— <u>Protection active</u> -

A- <u>Contre la détection</u>.

<u>Optique</u>..... : - peinture de camouflage (couleurs appropriées)
- suppression des reflets sur les surfaces vitrées ou métalliques
- formes
- dimensions (réduction)
- technique d'emploi (vol tactique)

<u>Electronique</u> : - peinture spéciale possédant un bon coefficient de reflectance aux émissions radar et aux infra-rouges
- détecteur d'émission radar
- technique d'emploi (vol tactique)

<u>Acoustique</u>.. : - diminution du bruit (radar - moteur)
- technique d'emploi (vol tactique)

B- <u>Contre le feu ennemi</u>.

- Connaissance de l'emplacement des armes ennemies et leurs caractéristiques (portée - efficacité)
- Technique d'emploi (vol tactique) en fonction des armes ennemies et de leur déploiement sur le terrain
- Utilisation d'armes ayant une portée supérieure à celles de l'ennemi
- Maniabilité de l'hélicoptère permettant de faire varier très rapidement la configuration de vol ou la route suivie

2.3.3.2— <u>Protection passive</u> -

La protection passive repose sur :
- l'organisation de l'hélicoptère
- l'emploi du blindage

Avant d'aborder l'organisation de l'hélicoptère, il sera nécessaire de déterminer les "points sensibles" dont l'atteinte par le feu ennemi pourra entraîner
*soit la perte de l'hélicoptère
*soit l'arrêt total de la mission
*soit la poursuite de la mission avec une efficacité réduite

Sous la dénomination "points sensibles" peuvent être rangés les organes et équipements nécessaires au vol et à la mission.

La détermination des "points sensibles " étant faite, l'organisation de l'hélicoptère pourra être envisagée suivant les cas :
- par concentration des "points sensibles" avec ou sans blindage,
- par dispersion des "points sensibles" avec ou sans blindage,
- par duplication des "points sensibles" avec ou sans blindage,

en fonction :
- du calibre, et des effets des projectiles susceptibles d'atteindre ces "points sensibles" au cours d'une mission type,
- des directions privilégiées d'atteinte,
- du pourcentage de perte ou d'arrêt de mission acceptable

Il apparait donc que la protection de l'hélicoptère doit être abordée comme un problème global. Il serait vain de couvrir un hélicoptère de plaques de blindage si celui-ci est parfaitement détectable par l'ennemi donc passible d'un tir précis avec une arme appropriée devant entraîner sa destruction.

La participation de l'utilisateur à la protection active se concrétise essentiellement par une technique d'emploi particulière : le vol tactique.

2.4 - Le vol tactique :

Il peut se définir ainsi :

"Le vol tactique est un déplacement qui s'accomplit en général à une hauteur très faible permettant d'utiliser au mieux, le terrain avec ses obstacles naturels et artificiels, les possibilités du véhicule, les équipements et armements de bord en vue : d'une part d'être en mesure d'assurer à chaque instant les actes élémentaires du combat terrestre et d'autre part d'échapper aux vues et aux tirs directs de l'ennemi".

Certes, le vol tactique n'est pas le seul mode de déplacement praticable sur le champ de bataille. Le vol à basse ou à très basse altitude est aussi utilisable. Mais, plus on se rapproche du contact plus la hauteur d'évolution diminue, pour se transformer en un déplacement, se rapprochant plus de celui pratiqué par un véhicule terrestre que par un aéronef.

"Caractéristiques générales du vol tactique -

- faible hauteur (2 à 3 mètres) ;
- entre et à proximité des obstacles, rarement au-dessus ;
- arrêts rapides nombreux ;
- stationnaires nombreux ;
- accélérations et décélérations sur trajectoire importantes ;
- évolutions très serrées ;
- utilisation de toute la gamme de vitesse. Pas de vitesse privilégiée ;
- adaptation de la vitesse au terrain et à la situation ;
- manoeuvres effectuées indépendamment du vent (direction et force).

Ces caractéristiques non exhaustives sont très contraignantes et peuvent se révèler déterminantes dans la conception de la plate-forme.

III°/ - INFLUENCES SUR LA CONCEPTION DE LA PLATE-FORME -

Nous examinerons l'influence du vol tactique dans les trois domaines suivants :
- les performances de la plate-forme
- la sécurité
- les possibilités tous temps

3.1 - Influences sur les performances :

Lors de la détermination des performances au stade de l'avant projet certains éléments devront toujours être présents à l'esprit des ingénieurs. En voici une simple énumération :

- Nécessité d'une bonne accélération et décélération sur trajectoire sans variation importante d'assiette pour permettre l'utilisation des armements et des systèmes de visée et d'observation.
- Facteur de charge de 2,5 rencontré couramment.
- Maniabilité très grande.
- Profil de vol très contraignant pour les durées de vie des ensembles mécaniques.
- Le vol s'effectuant couramment très près du sol, la panne moteur est fatale.
- Une grande partie de la mission s'effectue à des vitesses faibles 20 à 80 km/h.
- Nombreux arrêts rapides et stationnaires prolongés.
- Toutes les évolutions s'effectuent quelles que soient la direction et la force du vent.
- L'atteinte des limitations ne doit pas donner naissance à des phénomènes brutaux et dangereux.
- Facilité et rapidité de repliage des pales et de déplacement au sol soit d'une manière autonome, soit avec le concours exclusif de l'équipage, pour permettre un camouflage efficace.

3.2 - Influences sur la sécurité :

Le déplacement permanent à proximité immédiate et entre les obstacles nécessite :
- une visibilité parfaite pour le pilote,
- une diminution importante de la charge de travail de l'équipage

L'effort devra donc porter principalement sur
- l'organisation du poste de pilotage,
- le choix des instruments de pilotage,
- la détermination du nombre des membres de l'équipage.

3.2.1. Nombre de membres d'équipage :

En ce qui concerne la répartition des tâches au cours d'une mission en vol tactique nous pouvons dire que le pilote est principalement absorbé par la sécurité de sa machine au cours de ses évolutions parmi les obstacles. Son apport au reste de la mission peut être considéré comme négligeable. Il en découle qu'un ou deux autres membres d'équipage sont nécessaires pour assurer l'observation, la recherche des emplacements de tir, le tir, la transmission des renseignements et enfin une fonction primordiale : la navigation. Cette tâche particulièrement absorbante, sur laquelle repose pour une bonne part la réussite du "transit d'intervention", mobilise l'attention d'un membre d'équipage au détriment des autres fonctions, même en utilisant un système de navigation autonome.

Il n'est donc pas raisonnable d'envisager de donner au pilote d'autres fonctions que celles concernant la conduite de sa machine en sécurité.

3.2.2. Organisation du poste de pilotage :

Le poste de pilotage devra donner au pilote une parfaite visibilité extérieure dans toutes les configurations de vol. Les angles morts, provoqués par les éléments de structure et par le tableau de bord sont à diminuer ou à éliminer d'une façon notable.

Les parties vitrées devront être conçues de manière à éliminer toutes déformations provoquées par leur courbure et tous reflets gênants, provenant en particulier de l'intérieur de l'habitacle, et posséder les dispositifs nécessaires pour maintenir une bonne vision malgré les intempéries (pluie - neige - givre).

3.2.3. Choix des instruments :

Evidemment un compromis est à trouver entre une visibilité parfaite vers l'extérieur, et une bonne vision du tableau de bord. Il est en effet difficile de concilier les impératifs du vol tactique et ceux du vol IFR, qui imposent une disposition particulière des instruments dans l'axe de vision du pilote.

De plus la duplication de certains instmuments entraîne une augmentation des dimensions de la planche de bord.

La solution de compromis semble résider dans un système permettant :
- la suppression de la planche de bord,
- la projection des informations,
- la sélection des informations indispensables selon le type de vol.

Afin de ne pas saturer l'attention visuelle du pilote, déjà mise à rude épreuve nous pensons que les systèmes d'alarmes devraient s'orienter vers les dispositifs auditifs prescrivant même au pilote les actions de sauvegarde à entreprendre.

3.2.4. Possibilité de vol tous temps :

Tous les avantages procurés par le vol tactique de jour se retrouvent de nuit et par mauvaise visibilité.

Mais le déplacement de nuit entre les obstacles ne semble pas avoir trouvé jusqu'à ce jour une solution technique valable, alors que l'utilisation de l'hélicoptère en conditions IMC sur le champ de bataille est en voie de résolution par des techniques classiques "aéronautiques" (infrastructure sol Radar G-C.A. - ILS tactique - Radio balise).

Nous pensons qu'un raisonnement et une technique plus "aéromobile" qu' "aéronautique" permettraient une approche plus efficace de la pratique du vol tactique de nuit et par mauvaises conditions météorologiques.

IV°/ - VOL TACTIQUE TOUS TEMPS -

4.1 - Deux idées peuvent servir de support à cette nouvelle approche :

*La première, technique : La vitesse de déplacement est fonction de la distance de perception des obstacles se trouvant sur le trajet.

*La seconde, tactique : Au cours d'un vol tactique on peut considérer qu'une vitesse raisonnablement supérieure à celle d'un véhicule terrestre évoluant dans le même environnement est tactiquement significative.

4.2 - Vitesse raisonnablement supérieure :

Nous pouvons supposer, que sur un champ de bataille un véhicule terrestre ami ou ennemi se déplaçant de nuit ou par mauvaise visibilité ne pourra avoir une vitesse supérieure à 10 km/h, malgré les aides à la vision dont il sera doté.

Choisissons par exemple 40 km/h comme vitesse de déplacement de l'hélicoptère, valeur qui en terme d'aéronautique est assimilable à une vitesse presque nulle. Elle représente quand même dans un combat terrestre une vitesse 4 fois supérieure à celle des véhicules terrestres utilisés dans les mêmes conditions.

Cette vitesse devient alors significative, car pour parcourir une distance de 10 km notre hélicoptère mettra 15 minutes, mais pendant ce temps l'ennemi n'aura parcouru que 2,8 km. Ce qui signifie que nous pouvons considérer que la situation tactique de l'ennemi n'aura pas évolué d'une manière significative.

Cet exemple n'est choisi que pour illustrer les avantages d'une vitesse qui reste faible et pour permettre ainsi de concevoir un système autonome "d'aide à la vue" pour hélicoptère effectuant du vol tactique de nuit et par mauvaises conditions météorologiques, relevant plus des moyens d'aide à la vue utilisés pour les véhicules terrestres que des systèmes typiquement aéronautiques.

Il est à noter que les vitesses faibles ne restent significatives que pour les distances réduites de 10 à 30 km. Ce sont des parcours qui intéressent la frange des contacts.

4.3 - Caractéristiques d'un "système autonome d'aide à la vue" :

Nous pouvons essayer de définir succinctement les caractéristiques d'un tel système.

4.3.1. But.

Donner au pilote d'un hélicoptère un moyen d'augmenter son acuité visuelle, pour lui permettre de se déplacer de nuit et par mauvaise visibilité entre ou à proximité des obstacles sans aide extérieure.

4.3.2. Environnement.

Milieu : .nuit de luminosités différentes,
.nuit et jour avec conditions météorologiques particulières :

brouillard
pluie
neige

Terrain : .obstacles naturels (lignes de crêtes - arbres),
.obstacles artificiels (poteaux - lignes électriques)

4.3.3. Elément vitesse.

Vitesse de déplacement déterminée en fonction des possibilités de perception des obstacles données par le système.

4.3.4. Elément hauteur.

La plus faible possible compatible avec les performances du système.

4.3.5. Elément relief.

Donner au pilote une sensation du relief la plus exacte possible.

4.3.6. Conséquences possibles.

- Nouvelle technique de pilotage
- Instrumentation de pilotage particulière
- Instrumentation double permettant de passer instantanément et indifféremment du vol tactique avec aide à la vue au vol en conditions IMC

V°/ - CONCLUSIONS -

Le vol tactique pratiqué couramment par l'Aviation Légère de l'Armée de Terre Française apporte une solution efficace à l'emploi de l'hélicoptère dans le milieu hostile du champ de bataille terrestre.

Cette technique particulière engendre des contraintes dans la conception du véhicule, de ses équipements et de ses armements. Les ingénieurs doivent en être conscients avant le premier tracé sur la table à dessin.

Les quelques idées exposées précédemment ne prétendent pas avoir fait le tour complet du problème. Elles ne représentent qu'un simple jalonnement tendant à orienter les réflexions des techniciens qui devraient se souvenir de ce que disait le général américain FULLER : "Les anti-conformistes sont les prophètes en fait de tactique car ils savent adapter les armes nouvelles".

THE OPERATION OF HELICOPTERS FROM SMALL SHIPS

by

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SUMMARY

For the past nine years the Royal Navy has been operating helicopters from its small ships, and during this time numerous flying trials have been carried out. This paper aims to set down the experience gained from these trials with particular reference to the landing and take-off manoeuvres. A brief description is given of the environment in which such operations take place, together with the types of helicopter, the classes of vessel, and the approach technique used. The problems created by the air turbulence around the ship, the restricted size of the landing platform, and the deck motion are discussed in relation to their effect on the performance and handling of the helicopter. Data is given on power demands, rates of descent at touchdown, and accuracy of landing position. Restrictions which affect the operational freedom of the ship are also mentioned.

1. INTRODUCTION

For the past nine years the Royal Navy has been using helicopters as an integral part of the weapon system on their anti-submarine destroyers and frigates, and for vertical replenishment tasks on the supply ships and tankers of the Royal Fleet Auxiliary Service (RFA). During this period numerous helicopter flying trials have been made to develop deck equipment and other landing aids and to determine the limits of safe operation. These trials have been necessary because of the relatively small deck landing areas which distinguish these vessels from the aviation ships or aircraft carriers. The aim of this paper is to report on the experienced gained, with particular emphasis on the landing and take-off manoeuvres. Most of the information given comes from the trials which were carried out in conjunction with the Royal Navy and with personnel from the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

A brief description is given of the environment in which such operations take place, together with the types of helicopter, the classes of vessel, and the approach technique used. All these factors, including those that are inherent to the design of the ship, influence the effectiveness with which helicopters can be operated. This paper is concerned mainly with those problems that relate to the helicopter and especially its handling, although restrictions imposed on the operational freedom of the ship are also mentioned.

As the object of the majority of the trials was to check that a particular helicopter could operate safely from a specific class of vessel, the time allocated was on many occasions restricted to a few days only and the test conditions to those prevailing at the time. Nevertheless, it is considered that the results taken over this period of about nine years, when collected together, provide sufficient data to show trends and to highlight the principal problem areas in the handling and performance of helicopters operating from small ships. Trials data has been given for the demand on engine power when landing-on, for rates of descent at touchdown, and for the accuracy of landing position. Inevitably, there are deficiencies in the information collected and gaps in our knowledge. Suggestions have been made on how these may be rectified.

2. OVERALL VIEW OF THE OPERATION

Within the UK most of our knowledge of operating helicopters from ships at sea is derived from experience with one basic design configuration. This is a single main rotor with a tail rotor in the vertical plane to provide torque compensation and yaw control. Furthermore, most of the main rotors have had three or more blades attached to a fully articulated hub. Initially, none of these helicopters was specifically designed for operation from the decks of ships. The Westland Wasp, developed from the Saunders Roe P 531 Scout, does have a special four-wheel undercarriage and an extra low minimum value of collective pitch to improve its deck-operating capability. But, in general, the pilots have had to take the helicopter as designed and develop their own techniques and skills to suit the task of operating on ships. But how does this task differ from other types of operation? The vessel and its superstructure will create a region of turbulent air which the helicopter must negotiate during the take-off and landing manoeuvres. However, turbulence can be experienced ashore, around buildings, or in mountainous localities. The flight deck of a ship is dimensionally very small. But again, these helicopters frequently operate from confined areas ashore, for example, in jungle clearings. The difference with the ship-borne situation is that the turbulence and constricted space almost always occur simultaneously. In addition, because calm conditions at sea are rare and because a relative wind can arise from the speed of the ship, the manoeuvres are normally made in the equivalent of high winds not often encountered on land. A third factor, and one that is unique to ship-borne operations, is deck motion. Ships have six degrees of freedom, of which the angular motions of roll and pitch and the vertical heave are the more important in the operation under discussion. These can present the pilot with a moving set of references, which could lead to disorientation when on the approach in poor visibility or at night, or when over the deck close to any superstructure. In addition, deck motion can increase the difficulty of judging the rate of descent and of the slope of the "ground". The technique adopted, if possible, is to hover or wait on the deck until a quiescent period of motion occurs before landing or taking-off.

This then is the ship-borne situation: the pilot has to manoeuvre his aircraft through moderate to severe turbulence on to or from a landing platform that is very restricted in size, and which, together with the visual references, can be rolling, pitching and heaving. Furthermore, this pattern of events may be repeated for every landing and take-off.

It was implied above that the flight characteristics of helicopters were not tailored to suit operation from ships. This is not to say that the importance of providing satisfactory handling qualities is not recognised - quite the contrary - but their achievement does present difficulties, as they cannot be treated independently. The main rotor has three tasks to perform. It is required to provide the lift force, to provide the propulsive force, and to provide control forces and moments for trim and during manoeuvres. Moreover, most helicopters are used in a multitude of differing roles in a wide variety of climatic conditions. The design task is to obtain a rotor that has adequate lifting capacity and performance together with structural integrity and the minimum of adverse dynamic characteristics under all conditions.

Additionally, manoeuvres inherently mean changes in the power demanded from the engine because the control forces and moments stem directly from the magnitudes and directions of the thrusts of the main and tail rotors. This is particularly true of the yaw control which is provided by the tail rotor that is often working in conditions near the regime of blade stall; quite modest increases in yawing moment applied by the pilot can thus create a demand for a large increase of power. In fact, the amount of control available to a pilot is often considered in terms of the helicopter power limitations. These are dictated either by what is available from the engine or by the strength of the transmission system. Certainly, when manoeuvring onto a ship's deck the engine torque indicator is an instrument that is constantly monitored by the pilot.

3. TYPES OF HELICOPTER IN USE

Most of the helicopter designs that have been operated in the UK have at sometime landed on the deck of a ship. Those in use by the Armed Services and which could be expected at the present time to operate from ships at sea are the Sioux, Scout, Wasp, Whirlwind, Wessex and Sea King (see Fig. 1). Within recent years by far the greatest amount of this type of flying has been done with the Wasp, the Wessex (variants 1, 3 and 5) and the Whirlwind (variants 3, 7 and 9). The Sea King has just entered front-line service with the Royal Navy and so the next few years should see a build up of experience with this larger helicopter.

With the exception of the Sioux, which has a two bladed see-saw rotor, all the helicopters have been similar in layout and the basic difference between these has been their size. As shown in the table below their mass has varied from 2360 kg. to 9300 kg. with respective main rotor diameters of 9.8 m and 18.9 m. The disc loading has ranged from about 170 N/m^2 up to 325 N/m^2.

Helicopter	Gross mass (kg)	Rotor diameter (metres)	Overall length, rotor turning (metres)
Sioux	1340	11.3	13.2
Scout	2360	9.8	12.3
Wasp	2500	9.8	12.3
Whirlwind 9	3500	16.2	18.9
Wessex	6150	17.1	20.1
Sea King	9300	18.9	22.1

The propulsion unit on a helicopter must be regarded as an integral part of the flying control system. These helicopters have gas turbine engines with a free turbine driving the main rotor. The speed of this rotor is normally controlled automatically, by varying engine fuel flow and hence power output, to match rotor power absorption as either the collective pitch or flight conditions vary. The flying controls are operated by hydraulic power, apart from the yaw control on the Whirlwind. On this aircraft and on the Wasp provision is made for manual reversion, but although full manual control of the main rotor on the Wessex is possible it is likely to be extremely difficult. The Wasp and Wessex have auto-stabilisation/auto-pilot systems fitted. The Sea King is similarly equipped.

4. TYPES OF SHIP USED

There are two main categories of ship that operate helicopters: aircraft-carriers and 'small' ships. The former are a distinct class of their own while the latter comprise a great variety of shapes and sizes. As can be seen from Fig. 2, operation from 'small' ships would be more accurately described as operation from ships with small flight decks. These smaller flight decks necessarily constrain and limit the operation of the helicopter much more than the large decks of the carriers. Moreover, as the carrier has been purpose-built to operate aircraft, it is likely to be sailed in the best way to suit the helicopter requirements. This is not always possible with the other classes of vessel, which may have a different primary role and only use the helicopter in an ancillary manner.

Almost without exception the 'small' ships have their flight decks placed at or near the stern of the vessel (Fig. 3a). This has the advantage of giving the pilot an approach path that is usually free from obstructions. However, the vertical displacement of the deck associated with the pitching motion will be considerable at this position. Apart from the Tribal class frigate, the forward end of these flight decks are obstructed by superstructure (Fig. 3b) often extending to the full width of the ship.

The majority of Royal Navy ships now regularly operating helicopters have stabilisers that limit the amplitude of their rolling motion, although at a ship's speed of less than about 10 knots this stabilisation becomes ineffective. As a general rule it can be said that for the same sea conditions the ship motion in terms of roll and pitch amplitudes will become greater as the size of the vessel decreases.

5. LANDING AND TAKE-OFF PROCEDURES FOR SMALL FLIGHT DECKS

In principle it is preferable to take-off and land the helicopter while it is facing into the relative wind, and it is also possible, with varying degrees of constraint according to the layout of the ships, to approach and land-on (or take-off) from any azimuth direction relative to the ship. Nevertheless, it is usual to follow a common pattern for all approaches, the helicopter coming in from the port astern sector. As the pilot usually sits in the right-hand seat of the cockpit this approach gives him the best view of the flight deck and is also one that can be used at night.

The operating routine can best be described if it is considered in three phases - the approach, the landing-on (or take-off), and the handling on deck.

For the first phase the helicopter is brought on to the approach at about 2 miles astern of the ship. At this point the airspeed is reduced to 60 knots and the descent commenced down a 3° glide slope at either 15° or 35° relative to the ship's course (Fig. 4) depending on the particular ship's equipment. Overall command of the operation is retained by the ship's commander, first of all through the Helicopter Controller if the ship is fitted with radar, and then by the Flight Deck Officer, who must be in visual contact with the helicopter. Transfer of control and visual contact takes place at a range of not less than ½ mile.

When in close proximity to the ship the pilot brings the helicopter along-side the port beam at about 15-18 m above sea-level. He can then adjust his position and speed relative to the vessel without fear of overshooting into the superstructure. Finally, he manoeuvres the helicopter sideways under the direction of the Flight Deck Officer to bring it to the required position over the flight deck.

In the second phase, over the deck, a hover is established with the wheels about 2 m above the deck. Control of the operation is still under the direction of the Deck Officer who is normally in radio communication with both the ship's command and the pilot, but who also has hand-held flags (or illuminated batons at night). When correctly positioned and at the appropriate moment with regard to ship motion the helicopter is landed firmly on to the deck.

At night and under conditions of poor visibility the procedure described above is strictly adhered to. Obviously there can be variations in the final flight path over the deck according to the strength and direction of the relative wind, size of flight deck, type of helicopter, etc. on the assumption that better performance and handling are ensured if the helicopter can be faced into the relative wind. The degree to which this is possible depends on the helicopter configuration and on certain features of the flight deck and of the ship. This is reflected in the limits that are laid down for each specific helicopter/ship combination (Fig. 5) to ensure that adequate margins of control and safety are maintained. These limits also include those to ship motion which is important in this phase of the operation.

Certain of the individual factors that determine these limits will be discussed in more detail below, but a brief reference here to the examples shown will help in setting the perspective. Thus, the Wasp with its special short, four-wheel, undercarriage can more readily be aligned to the relative wind than can the Wessex. For the latter, the distance between the main and tail wheels is larger in relation to the width of the flight deck and so it is constrained more nearly to the fore and aft direction. For the cases in which the helicopter has to land in a fore and aft direction relative to the ship then an overriding limit of 15 knots beam wind is imposed; this is to ensure an adequate manoeuvre margin.

The directional limitation is particularly applicable at night when the sighting of the required visual aids dictates that landings and take-offs must be made with the helicopter facing the bows of the ship. The more restrictive operating limits for nighttime reflect the greater difficulty in landing at night. Incidentally, the limitation to mainly forward facing relative winds for the example shown for the Wasp in daylight is a consequence of the fact that for the particular helicopter/ship combination in question, helicopters are not expected to take-off over, or land facing, the stern of the ship because there are no suitable visual references.

At take-off the helicopter becomes airborne in a comparatively very short time. Usually there are no problems provided that the pilot does not translate into forward flight until he is clear of the ship, hence avoiding impact between the main rotor and various aerials, masts, etc., on the superstructure. This of course assumes that the helicopter has an adequate power margin at the time. The limit in this respect may be reached on occasions when a combination of adverse effects arise, such as hot ambient conditions, ingestion of hot funnel gases devoid of oxygen, light winds, and maximum take-off weight.

In the third phase, the helicopter must be 'flown' by the pilot as long as the rotors are turning, for, even though the wheels are on the deck, the rotors will still retain their faculty for exerting lift forces and control moments. These are potent factors additional to the accelerations arising from oscillatory ship motion that tend to make the aircraft slide or topple over on deck. So once the touchdown has been firmly established the pilot will select the minimum collective pitch of the main rotor blades that is available to him. On the Wasp helicopter an extra low collective position is provided and future designs may be able to produce reverse thrust. This will have the effect of counteracting the increase in lift that can occur as the rolling motion of the ship causes the rotor disc to change its incidence.

It is now normal practice for helicopters to fly with autostabiliser equipment operative. It will be necessary, therefore, for the pilot to switch off this equipment as soon as possible after touchdown (conversely, it should be switched on just before take-off). If this action is not taken the autostabiliser will attempt to counteract the roll or pitch of the aircraft that is being impressed upon it by the deck motion. If large amplitudes develop this could lead to pounding of the blades on the flapping stops. Also, if not switched out, the yaw channel will react to changes in ship's course by moving or tending to move the tail of the helicopter across the deck.

Finally, the rotors have to be stopped, or started. In this operation, when the rotational speed is low, the blades will have lost their inherent centrifugal stiffening. Under this condition 'blade-sailing' may arise giving large deflections at the tips or even an unstable flapping motion. The situation is aggravated in high winds, by disc incidence changes as the ship rolls, and, by vertical gusts which are sometimes created at the side of the ship and which affect the local incidence at the tips of the blades.

6. THE NATURE AND EFFECT OF AIRFLOW AROUND THE SHIP

In relative winds from ahead the airflow around the ship and its superstructure, in the vicinity of the flight deck, is both variable and complex in character. In general, the wind speeds close to the deck are lower than the undisturbed, or free-stream, value, particularly at the forward end where reverse flow is likely to exist (Figs. 6b and c). In this region vertical downdraughts can occur which may entrain the funnel gases. The presence of the helicopter rotor may also emphasize this entrainment, leading to the discomfort of the aircrew and to the detriment of engine performance. The conglomerate of masts, funnels, boats, etc., give rise to eddy shedding, and vortices roll up along the junction between the deck and the side of the hull. These two effects combine to give a band of greater turbulence extending aft along the port (and/or starboard) area of the flight deck (Fig. 6b). The width of these bands is dictated by the dimensions of the forward bulkhead. A clearly defined edge to an area of more severe turbulence occurs if this structure is built right out to the side of the hull. A less abrupt band exists when a walkway is provided at the sides of the deck.

When the relative wind moves round to about 30° these bands of turbulence spread out over the whole flight deck (Fig. 6c). In addition a downdraught is created on the leeward side of the vessel. In a beam wind this 'curl over' becomes more pronounced, the airflow is curved over the deck, and is less turbulent. Over the deck the velocity may be slightly greater than the freestream value (Fig. 6d). The shape of the hull beneath the flight deck has most effect in the beam wind case. A 'solid' hull causes the greater disturbance, but some alleviation of the 'curl over' and some smoothing out of the flow will occur if the structure is cut away as on RFA Green Rover (see Figs. 3 and 6d).

On an aircraft carrier, beam winds create conditions downwind of the 'island' similar to those described above for ahead winds. But on a carrier these can usually be avoided by manoeuvring the ship or landing elsewhere on the deck.

The helicopter pilot is probably first made aware of the turbulence as he approaches alongside the stern of the vessel. In light winds there will be very little influence but above about 15-20 knots of wind over the deck it will become increasingly significant. The wake from a small superstructure forward of the flight deck will have little effect on a large helicopter though for a small helicopter downwind of a large structure the situation is very different. In addition to the size of the wake its direction must be considered. Ahead winds are possibly the least troublesome. When out to the port side the helicopter will be flying in the undisturbed free-stream, there will then be a short transient period as it translates through the band of turbulence at the deck-edge to reach the area of lower velocities over the deck. Probably the worst case occurs when the relative wind is approximately 30°-40° on the starboard bow. In this case all the hovering and translation on to the deck will be carried out with the aircraft completely immersed in the turbulent wake. If, at the same time, the deck size constrains the helicopter to land facing fore and aft then the pilot may need a considerable amount of rudder control to hold this heading (the majority of decks can accept a misalignment between helicopter and ship of at least ± 20° on this heading, but for night landings it is probably inadvisable to allow more than say ± 10°). Yet sufficient control margin must still remain to deal with wind fluctuations and for manoeuvring. Typically, in such a case, where a sideways manoeuvre is required to be made towards the deck in an upwind direction, there can be an additional power demand of some 15-20% (Figs. 7a and b). This could increase if the pilot allowed the helicopter to get too low, when he would also find himself in a region of downward moving air on the port side.

Thus there are obvious benefits to be gained by allowing the aircraft to approach facing into wind if this is possible, though there is an exception. This is when the wind is at an angle greater than, say, 20° on the port bow, in which case an approach from starboard is preferable to the standard routine used on the port side. Slight difficulties have also arisen on occasions when approaching directly into a strong beam wind. On reaching the far side of the deck, to ensure adequate tail wheel clearance with the other deck edge, pilots have reported a restraint on their forward progress. This may have been brought about by the rotor encountering a region of higher airspeed together with an upward curving flow (Fig. 6d). This would result in a backward and a lateral flapping of the rotor disc with corresponding tilts of the thrust vector.

7. THE PROBLEM OF DECK SIZE AND SUPERSTRUCTURE

As might be expected the smaller the deck relative to the dimensions of the helicopter, in particular those of the undercarriage and the overall length, the greater the difficulties involved. Allowances must be made to cover inaccuracies of landing and to give adequate clearance between the rotor tips and ships' structure. At the present time it is thought that an absolute minimum of about 1.5 m should be allowed for each of these two factors. The magnitude of both will be influenced by the handling characteristics of the helicopter, including the view and visual references available to the pilot. Good visual references must also be given to the Flight Deck Officer, who helps to relay the required information to the pilot, particularly at night. Estimation of distances in the longitudinal direction is not easy and as it is both difficult and undesirable for a pilot to turn his head and look out sideways from his cockpit for any length of time, a good design aim would be to provide a clear downward view to the right front where one undercarriage wheel could be seen relative to a transverse line marked over the full width of the deck.

Data on variations in landing positions are available from two sources. Firstly, 7 ships (mainly Leander class frigates that operate Wasp helicopters) regularly reported on every touchdown under operational conditions at sea over a period of three months in the winter, making a total of several hundred landings. The "errors" are shown in probability form in Figs. 8a and b. There were significant variations between individual ships but these could have been due to the occurrence of systematic errors in observations that were incidental to the main operational task. Further more accurate observations have been made during recent trials at sea. A typical result is shown in Fig. 9 but so far insufficient data of this latter type have been collected. Nevertheless, the results in Figs. 8 and 9 both show greater inaccuracies in the longitudinal landing position than in the lateral position. It must be pointed out that this information does not indicate the precision with which pilots can land on a given spot, but how they use the space available to them within the landing area marked out.

In fact, mainly because of a recent demand to widen the operational capabilities of the larger helicopters on some of the smaller flight decks, we have changed our policy on deck markings. No longer is the pilot shown the area in which he may safely place his undercarriage wheels (Fig. 3b), he is now given an aiming point on the deck. Because the pilot may wish to take-off or land facing into the wind, which could be in any azimuth direction relative to the ship, this aiming point is developed into a circular line as in Fig. 10.

If the width of the flight deck is small compared with the distance between the fore and aft wheels of the undercarriage then the helicopter will be constrained to land and take-off facing approximately in the fore and aft direction. This in turn imposes a limitation on the ship's movements. Either the helicopter must be launched or recovered only when the ship is steaming directly into wind, or the ship must restrict its operations to maintain a side wind component of less than 15 knots on the helicopter. Most of the helicopters are nominally cleared to fly sideways at 30 knots. The 15 knot limitation is indicative of the inadequacy of their rudder control and of the margins that must be maintained when operating from ships.

Further phenomena, but operationally less important ones, occur if the rotor-disc overlaps the deck edge. Firstly, with side wind on the hull, vertical airflows can impinge on the outer edges of the rotor disc and cause pulsating loads which may be fed back to the pilot's control stick. These vertical flows can also give changes in disc tilt, and hence control moments, contrary to those desired by the pilot. The interaction between these environmental conditions and the aerodynamics of the rotor have not yet been fully investigated, nor the subjective observations of the pilots fully explained. Secondly, the 'ground' beneath the rotor .. limited in area. The usual ('infinite') ground-effect is to reduce the power required to hover, or alternatively, to increase the lift from the rotor for the same power. Brief tests were carried out to check whether the fact that the deck dimensions were finite influenced the power required to hover. A reduction was measured, but it was small and unlikely to be operationally significant, especially since deck operations frequently take place in relative winds of between 15 and 25 knots, or can be made to do so by virtue of the ship's speed. Any advantageous ground effect would in any case be small at such speeds and the effect is soon outweighed - at about 15 knots - by the reduction of induced power that occurs with increase of speed irrespective of ground effect.

In the past, reports have been made of loss of main-rotor lift while hovering in ground-effect close to vertical walls. The effect has been to deflect the rotor downwash up the wall and through the rotor again, thus inducing vortex ring conditions. Some preliminary tests were conducted by R.A.E., Bedford, to check whether such a phenomenon was likely to be induced on board ship. Recirculation was established and the tests showed that the proximity of superstructure would increase the pilot's stick movements and power demands although no trim changes could be detected. During recent trials at sea with the Sea King a similar situation was observed when the aircraft approached close to the side of the ship, more or less at flight deck level in calm air. Sea water spray could be seen shooting up vertically at the side of the hull and recirculating through the outer region of the rotor disc.

8. THE PROBLEM OF SHIP MOTION

The chief characteristic of wave motion in the sea is its irregularity, and extensive simplifications are made when considering the motion of a ship in relation to the problem of helicopter operations. The velocities and accelerations of the deck are relevant as well as the amplitude of the motion which is assumed to be simple harmonic. Although in any given sea state the angular movements of the smaller vessels may be greater, the distance of the flight deck from the effective axes pitch and roll will be correspondingly less. Thus, since the periodic times are not widely different, it is found that the linear velocities and accelerations at the flight deck remain much the same value irrespective of ship size. Yawing and changes in ship's heading could be very significant and they are kept to a minimum when launching or recovering an aircraft. Ships usually roll at their own natural frequency whereas the pitch and heave motions depend among other things upon the period of encounter with the waves. For all but the very largest ships the roll period is taken to be between 8 and 10 seconds, and between 6 and 7 seconds for the pitch and heave.

Although in practice the wave motion is irregular and the amplitude of the deck motion is not constant, short intervals of time occur when the movement is relatively small. It is assumed that the helicopter can land-on or take-off at these times, even in rough seas, since the moment of touchdown or lift-off can be chosen by the pilot and the duration of the action is only a matter of a few seconds. For these manoeuvres the limits of roll and pitch have been set at $\pm 5^{\circ}$ and $\pm 2\frac{1}{2}^{\circ}$ respectively. For comparison it has been estimated that in sea state 5, which can be associated with a natural wind of about 30 knots, a small frigate would have a roll amplitude of $\pm 4^{\circ}$ and a pitch amplitude of $\pm 1^{\circ}$, with a probability of exceedance of 60%.

After land-on and just before take-off, the helicopter will be standing on deck with its rotors running for say one to two minutes. Thus there is a chance that the limits above may be exceeded during this time and it would be advisable for the aircraft to be lashed to the decks, since with rotors turning sizeable lift forces can still be generated. In calculating the forces and moments that tend to topple the helicopter or cause it to slide on deck under these conditions values of the order of $\pm 8^{\circ}$ and $\pm 2^{\circ}$ for roll and pitch are taken, which in the example, would have a probability of exceedance of about 13%. If there is a need to move the helicopter along the deck its rotors will not be turning but neither will it be firmly lashed down. Manoeuvring on deck in rough seas is not going to be accomplished quickly and the possibility must be considered of a deck motion of large amplitude. Values of the order of $\pm 14^{\circ}$ of roll and $\pm 4^{\circ}$ of pitch are used which on the small frigate would have a probability of exceedance of 0.1%. The corresponding values for heave at the above probabilities are $\pm$ 0.6 m, $\pm$ 1.2 m, and $\pm$ 2.1 m.

Unlike landing on sloping ground ashore the pilot cannot be certain beforehand in which direction the deck will be inclined at the instant of touchdown. He is unable, therefore, to anticipate the control movements that will be needed. His requirements will be for a control system that will give him the ability to move his helicopter rapidly and precisely without unduly large stick movements. All the same, the best technique is probably for the pilot to keep his 'wings level' and not attempt to follow the motion, otherwise there is a chance that he will become disorientated, particularly at night.

Evidence to support the presumption that ship motion increases the difficulty of landing was given in the reports on touchdown position referred to in para. 7. Figs. 11a and b show that the probability of landing further away from the optimum deck spot increased as the deck movement became greater. This was particularly so for the lateral position.

In contrast to landings ashore, the rate of descent of the helicopter will be a combination of the aircraft and the deck vertical velocities. It could be expected that pilots would thus have greater difficulty in judging this touchdown speed, leading to heavier loads in the undercarriage or possibly to overtorquing of the rotor transmission in an attempt to stop or reverse away from a rising deck. That such occurrences are rare is both a tribute to the pilots' skill and an indication that the lift control on present-day helicopters is on most occasions compatible with deck landing manoeuvres. The relative velocity between the helicopter and the deck has been measured on all the ship trials and it is from these that the curves given in Fig. 12 have been derived. The measurements were made with a high-speed cine camera and only daytime landings are included. The curves show that the Wasp helicopter is more likely to produce the higher values of touchdown velocity while the Sea King will achieve the lowest values. That is, the highest velocities appear to be associated with the smallest helicopter operating from the smaller decks, and conversely. On the other hand, the curve for the Sea King is based on the smallest number of results. It is known that the probability curves will move to the right as more results become available. A possible contribution to this trend is the familiarity and confidence that pilots acquire in their aircraft with time. Our experience from the trials is that piloting technique has a bigger influence on landing velocities and undercarriage loads than does ship motion. In fact, attempts to correlate the rates of descent with deck movement and with relative wind over the deck have yielded the result that high rates of descent are more likely to occur at the lower wind speeds and when the deck is level! This may merely reflect the greater caution exercised by pilots on the landing task as the sea state increases. More investigation is warranted such as the monitoring of the time spent in hovering over the deck before touchdown and of the workload in the cockpit.

Other more subjective effects of ship motion have been noticed. For example, at times pilots try to beat the motion by dropping the helicopter rapidly on to the deck. It is in situations such as this that the undercarriage needs to have good energy absorption properties without rebound. Very rapid reduction of the main-rotor collective pitch can be detrimental if there is no corresponding removal of rudder control to reduce tail-rotor thrust, because high side loads will then be induced in the tail wheel and rear fuselage on impact with the deck.

There are occasions when deck motion can be disconcerting to the pilot at take-off. Should conditions which require full power at take-off, such as maximum weight, high air temperature, and zero wind, coincide with a sea swell that causes large vertical movements of the landing platform then the pilot may find that he does not lift off the deck straight away. This will be particularly so if he initiates the take-off at the wrong moment in the deck pitching cycle, for then the deck could be rising at a faster rate than the helicopter.

9. CONCLUDING REMARKS

The task of flying a helicopter on to or from the deck of a ship differs from similar activities ashore because the environment involves a greater degree of turbulence, a restricted space, and deck motion. The flying control systems of the helicopters currently used have not been specifically altered to deal with this environment, but there seems to be a general opinion among Royal Navy pilots that the Wasp is the helicopter most suited for the task. A number of factors must influence this opinion and it is possible that the pilot's view and undercarriage performance are taken into account, as well as the control characteristics. Freedom from ground resonance and aircraft size may be significant also. More studies and comparisons are required to determine the importance of the various parameters and to help to formulate criteria for the required handling qualities.

Ideally, a ship should be capable of operating a helicopter from its deck without imposing restrictions on its freedom to sail in any direction, by day or by night, whatever the weather. That this is not always achieved is shown in Fig. 13. In this diagram 100% freedom of manoeuvre would represent the ideal, 0% would indicate that the ship could not operate the helicopter under any conditions. This representation is the result of empirical integrations of limits such as those shown in Fig. 5, including the limits for stopping and starting the rotors, for different values of ship speed and direction and natural wind. The restrictions can be expressed in this way because the limiting factor for present-day operations is the ability of the helicopter to manoeuvre within the turbulent environment of a ship. To be more specific, there are boundaries of relative wind speed and direction beyond which the helicopter has insufficient power (or transmission torque allowance) to manoeuvre. Yaw control is especially demanding on the power available, and when holding the helicopter against a side wind the limit is soon reached. This situation is again reflected in Fig. 13 by the curves for the Wessex (day and night) and the Wasp (night) which represent conditions when the helicopters are restricted to landings approximately in line with the ship's heading. Allowing the helicopter to face directly into wind gives greater operational freedom as indicated by the curve for the Wasp (daytime). Deck motion has not been found to be so limiting as was once thought, particularly in roll. Tribute for this must go in part to the pilots for the skills and techniques that they have developed. At the same time, the introduction of ships stabilised in roll must have been a contributory factor.

The task now is to extend the current operational limits towards the ideal. At present there are palliatives which provide some extension. For example, if the helicopter can be placed in the lee of the ship's superstructure then its rotors can be stopped or started in relative winds from ahead greater than those indicated in the diagrams. Also, the limits can be increased if the helicopter is flown at weights less than the maximum, which is frequently the case. This is a useful procedure when high ambient temperatures and light winds are prevailing. The trials have shown areas of potential improvement. A considerable gain would be brought about by allowing the helicopter always to face into the relative wind. For the larger helicopters this would require many of the decks to be made wider - not an easy thing to achieve. Similar gains would be made in the present restrictive night limits, not only by improving the visual landing aids but by making them omni-directional as well. But to make provision for approaches from any azimuth direction at night would be very difficult. Alternatively, the advantages to be gained by facing directly into wind could be met partially by improvements in the design of the helicopter. The greatest need is for more powerful yaw control but an all-round increase in power-margins would also be beneficial. Another area that has not yet received sufficient attention is the airflow around the ship. It should be possible to reduce the eddy shedding and hence the small scale turbulence by redesigning or repositioning masts, funnels, etc. Investigations should be made to determine the effects of superstructure width and shape, and to ascertain whether clear spaces beneath the flight deck (Fig. 10) are beneficial.

Quantitatively the trials have not yielded all the data that we would like. For instance, very few results have been obtained under rough sea conditions. The requirement now is to know how the helicopters are used by the service pilots during their routine flying. To achieve this several programmes have been initiated:

(i) A Wessex helicopter is to be instrumented so that the pitch angles of blades of the tail rotor can be monitored.

(ii) Investigations are underway on the application of a doppler radar device to measure rates of descent. This instrument is small enough to be attached to each individual undercarriage leg and will give information on night landings for the first time.

(iii) A small electrical spirit-level is on trial as a means of measuring deck motion more precisely. Direct readings will be available to the Ship's Command which will enable them to work more accurately within the limits set down. It is hoped that the device can be used also for recording deck motion.

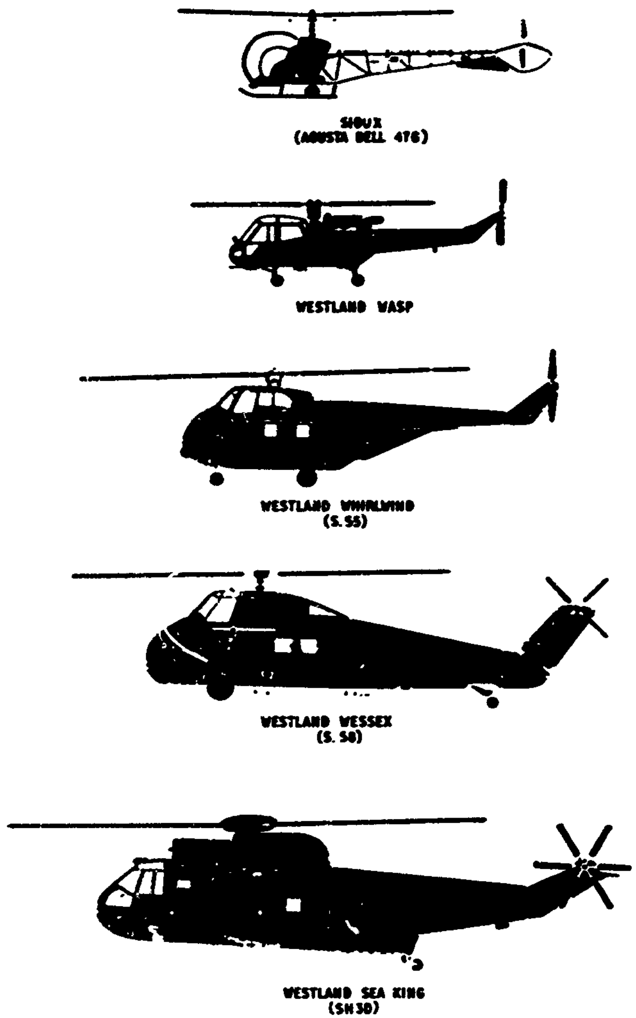


FIG.1 HELICOPTERS USED ON SHIP TRIALS

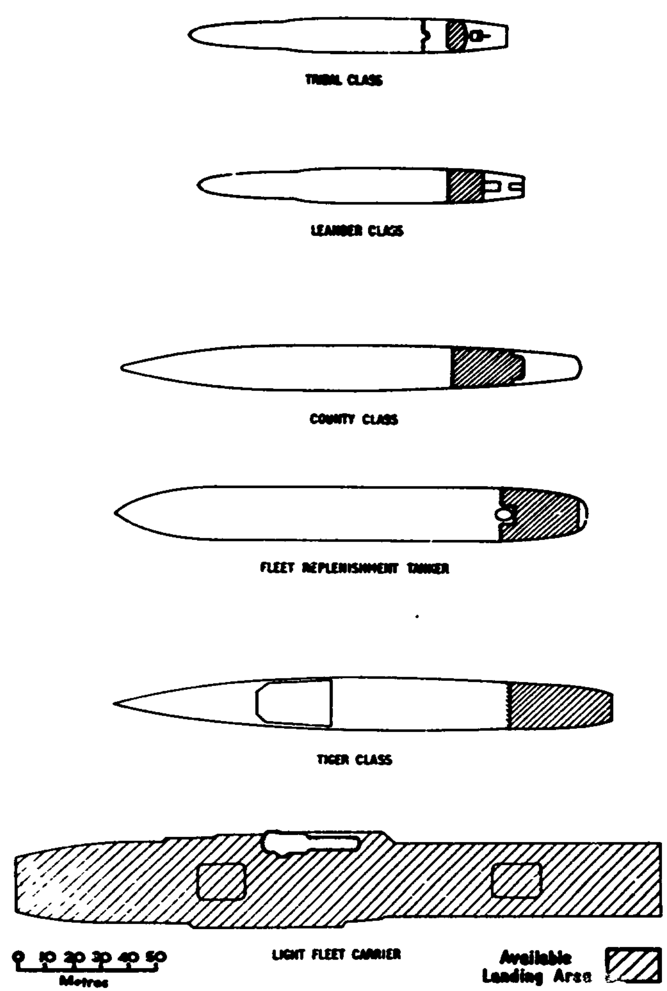
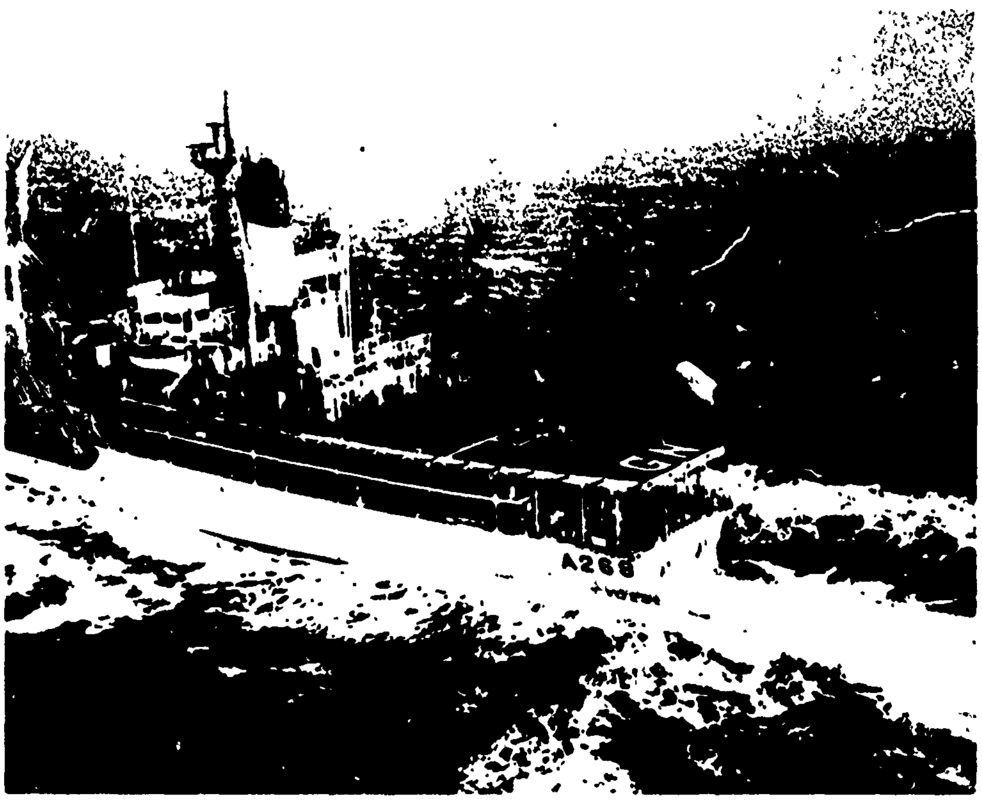


FIG.2 PLAN VIEW OF FLIGHT DECKS

(a) GENERAL VIEW



(b) CLOSE UP OF FLIGHT DECK

FIG.3 R.F.A. GREEN ROVER

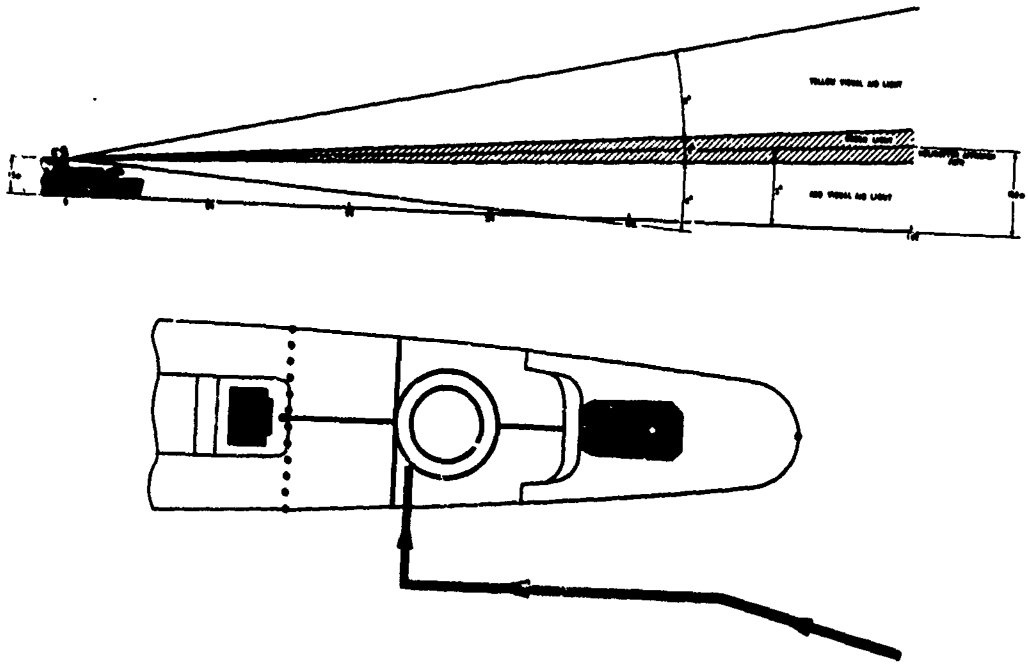


FIG. 4 STANDARD APPROACH PATH

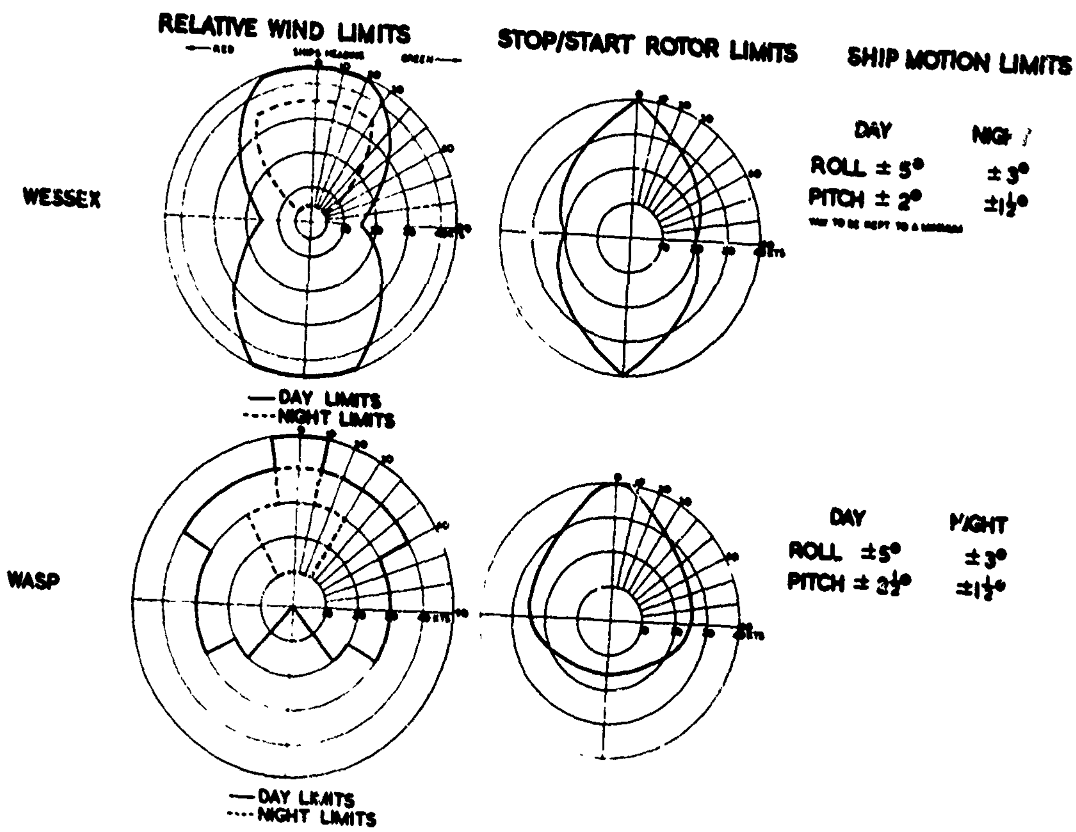


FIG. 5 RELATIVE WIND AND SHIP MOTION LIMITS

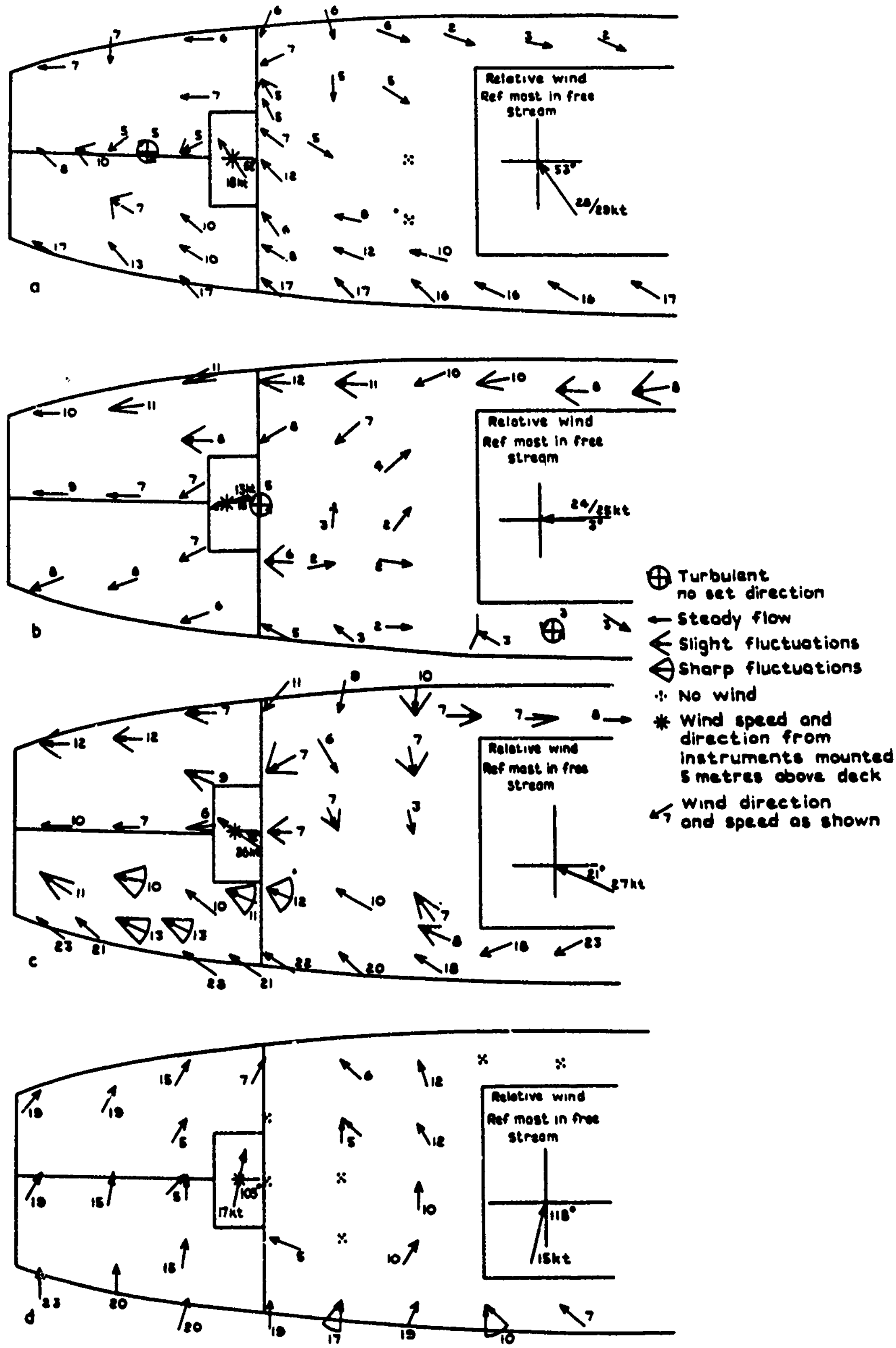
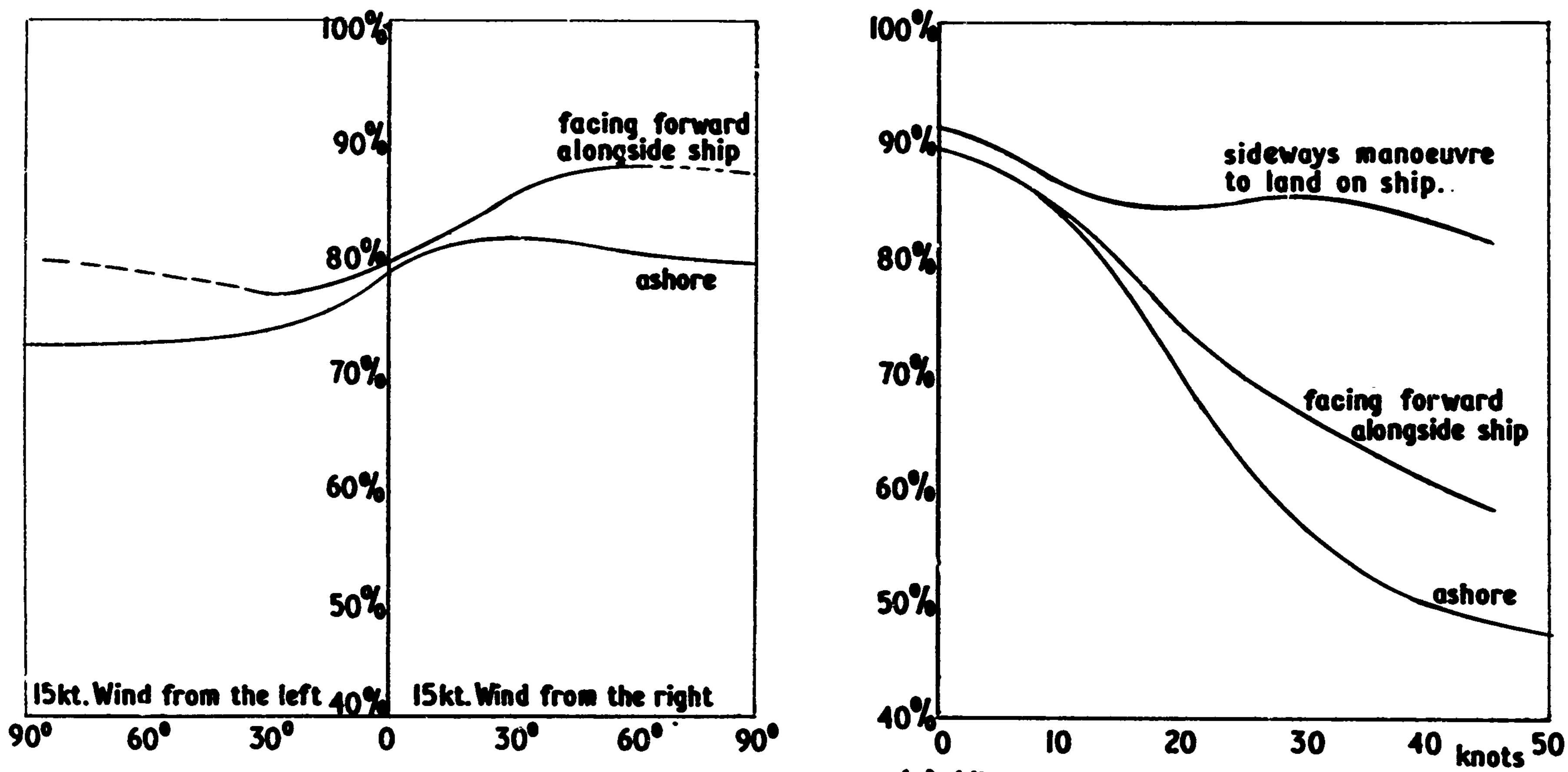


FIG. 6 WINDFLOW ON RFA GREEN ROVER 2 METRES ABOVE DECK LEVEL



(a) Wind direction relative to the helicopter

(b) Wind speed relative to the helicopter (helicopter facing into an ahead wind)

FIG. 7 TYPICAL RESULTS OF % MEAN POWER REQUIRED FOR HOVERING

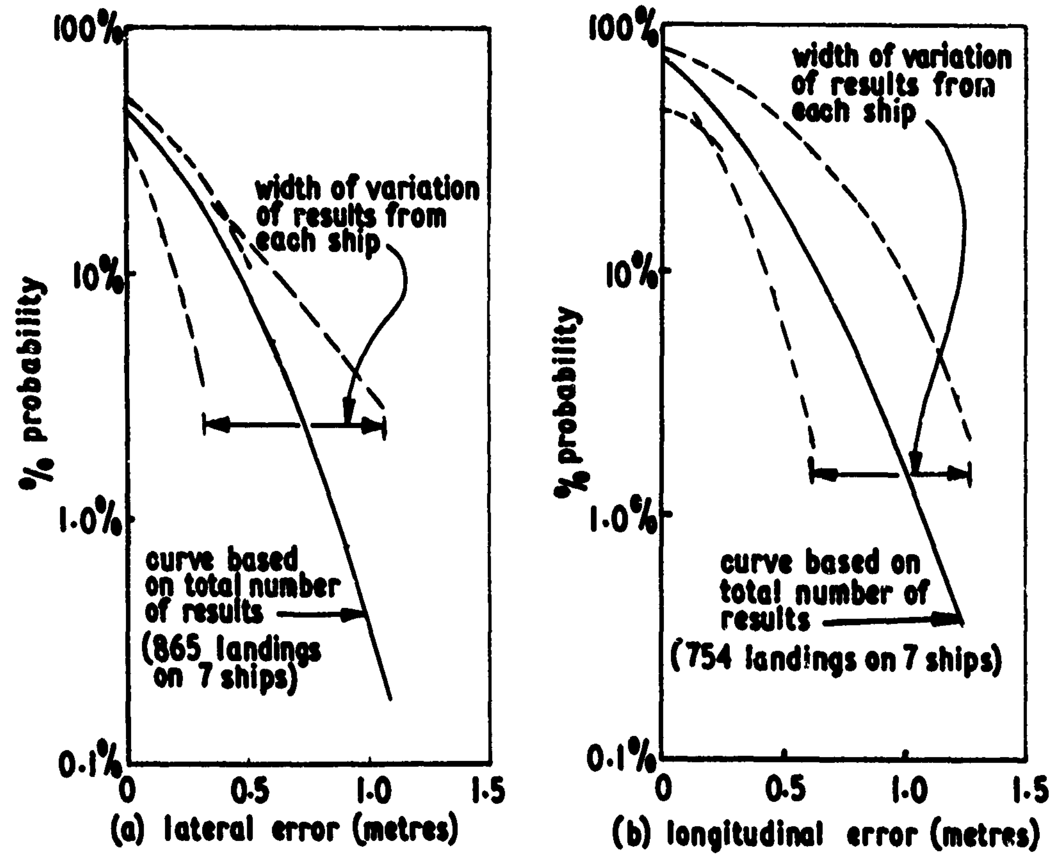


FIG. 8 PROBABILITY OF EXCEEDING POSITION ERROR FROM CORRECT LANDING SPOT

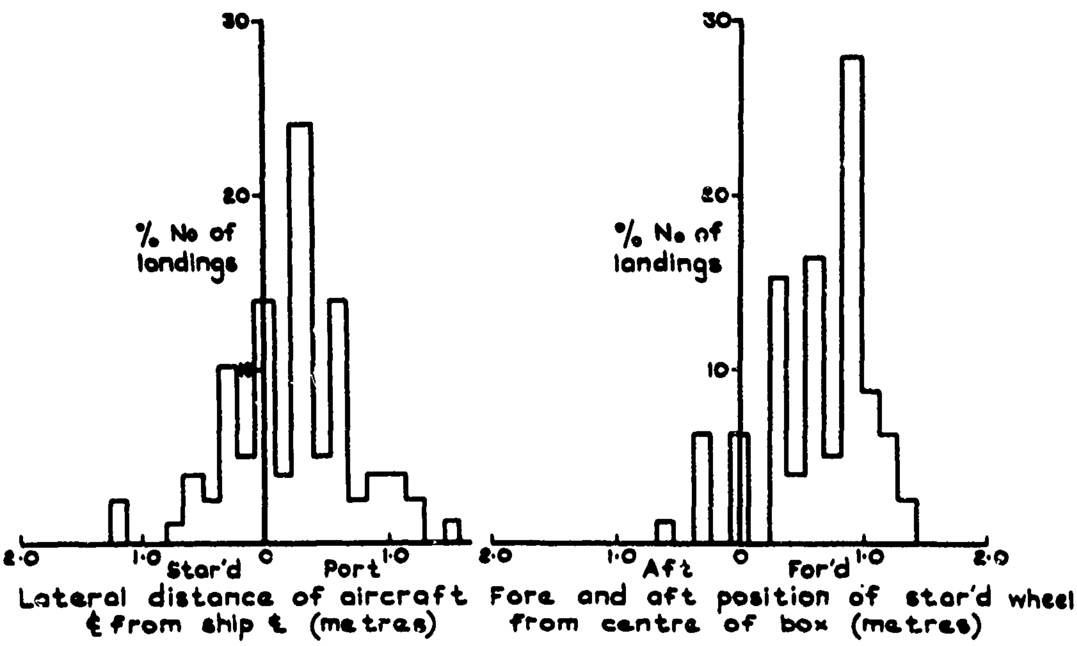


FIG.9 WHEEL POSITIONS RECORDED DURING SHIP TRIAL

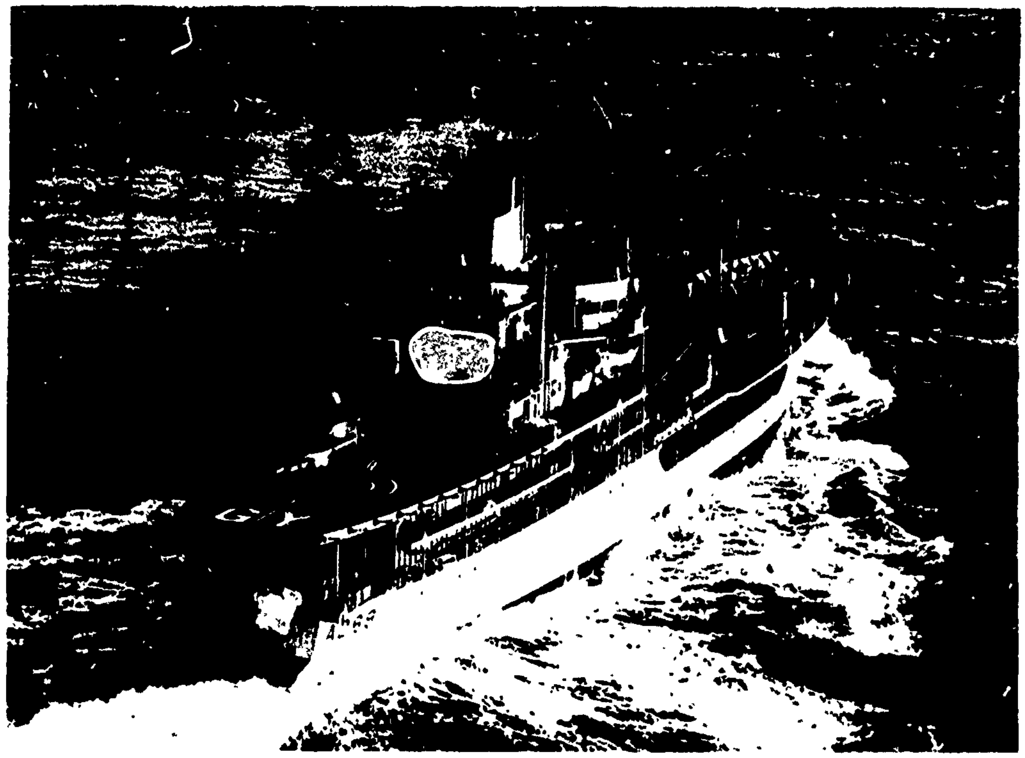


FIG 10 NEW PATTERN OF DECK MARKING

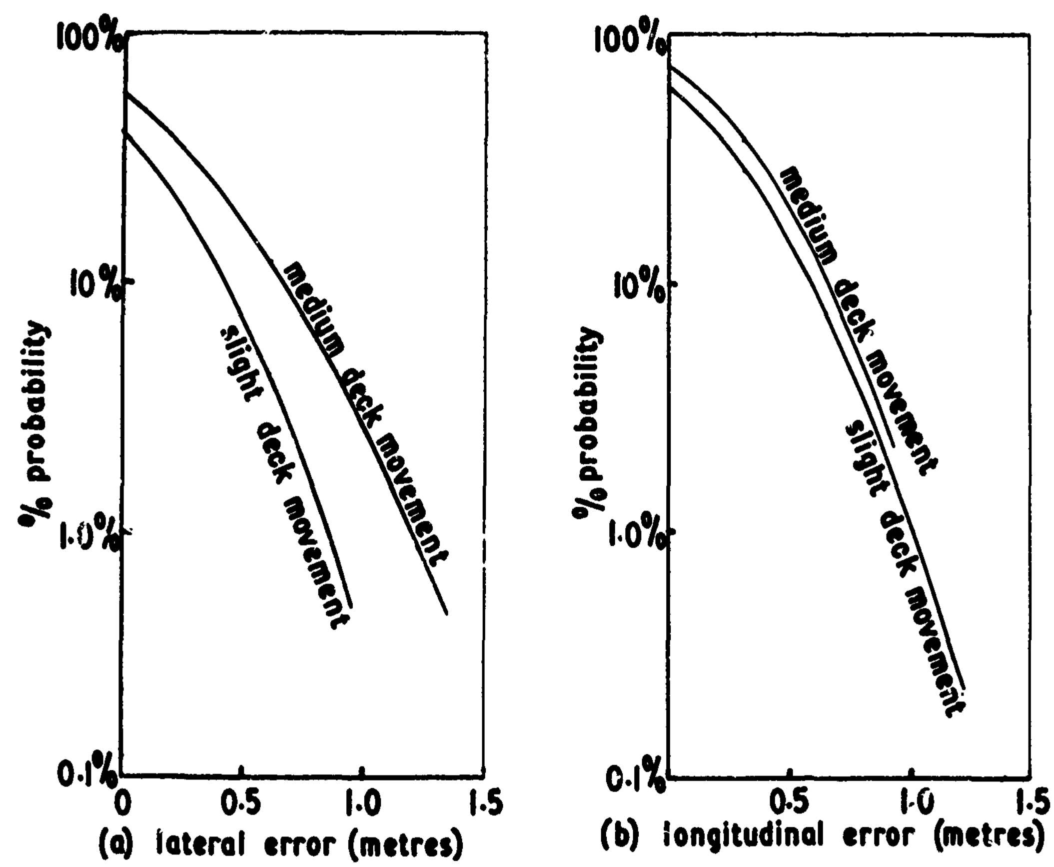


FIG. 11 EFFECT OF SEA STATE ON PROBABILITY OF LANDING POSITION ERROR

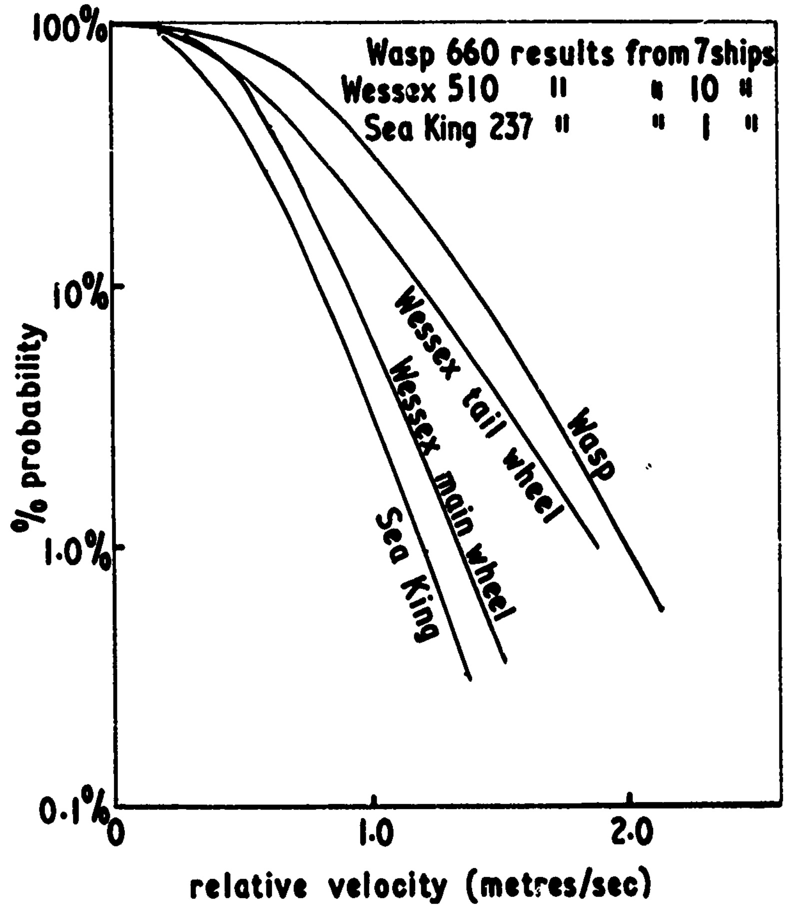


FIG. 12 PROBABILITY OF EXCEEDING RELATIVE RATE OF DESCENT

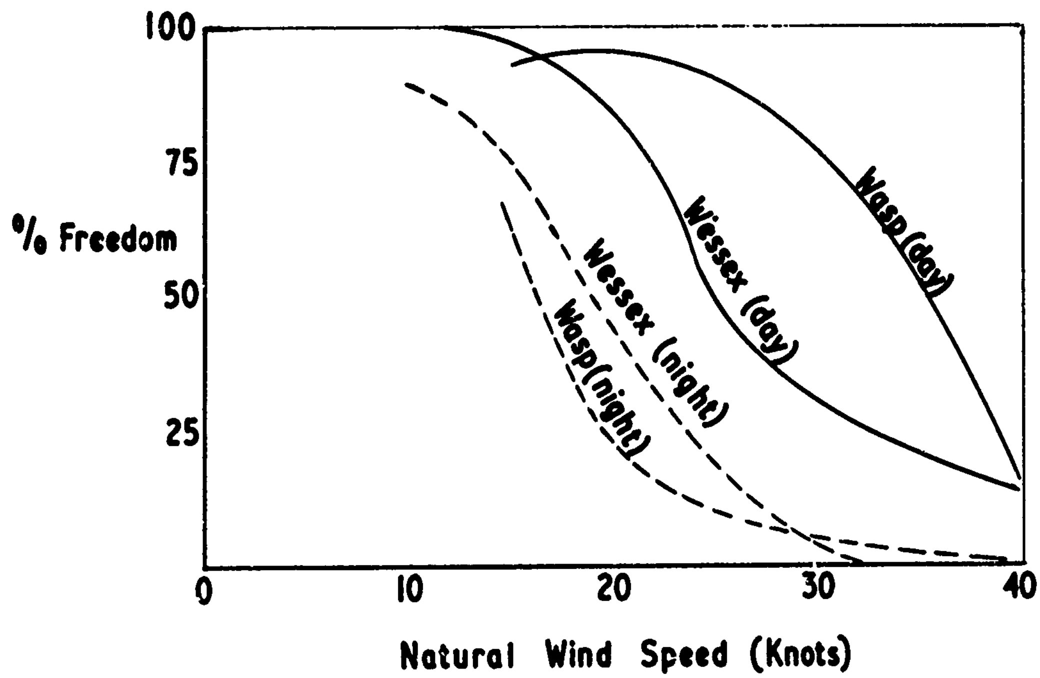


FIG. 13 EFFECT OF HELICOPTER OPERATION ON A SHIP'S FREEDOM TO MANOEUVRE

DIX ANS D'EXPERIENCE
AVEC LES HELICOPTERES EN OPERATION DANS LES ARMEES FRANCAISES

par

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Il n'est pas rare que dans les assemblées où constructeurs et utilisateurs sont réunis pour échanger leurs idées, certaines voix s'élèvent pour mettre en évidence un fait qui semble être universel, à savoir l'incompréhension qui préside aux rapports entre ceux qui fabriquent des appareils et ceux qui les utilisent. Il a donc semblé intéressant d'examiner à la lueur d'exemples fournis par quinze années d'études et de production d'hélicoptères à l'Aérospatiale, si cet état de fait est celui qui existe en France, dans ce cas, où il trouve son origine, ce qui a été réalisé pour le faire disparaître et ce qui reste à fair pour atteindre ce but.

Il paraît cependant nécessaire de faire une remarque préliminaire importante: les rapports entre constructeur et utilisateur diffèrent évidemment sensiblement selon qu'un hélicoptère est construit pour répondre à une demande précise d'un utilisateur ou qu'un programme est lancé de la propre initiative de l'industriel. Ceci ne veut cependant pas dire qu'il ne faut tenir compte des remarques des utilisateurs potentiels que lorsqu'ils s'intéressent, de manière précise, à un appareil.

C'est ainsi que la famille des Alouette n'a pas trouvé son origine dans une demand formelle d'un utilisateur. Mais, petit à petit, le nombre des clients s'étant accru, des idées se sont faites jour qui ont donné et donnent encore lieu à des modifications des appareils.

C'est plutôt dans le programme suivant de la production d'Aérospatiale que l'on trouve trace des échanges entre utilisateur et constructeur. Ce programme est celui du Super Frelon. Cet hélicoptère devait correspondre à l'idée suivante de la Marine F^se^: puisque le monde semble s'acheminer vers une période où le temps de guerre déclarée sera remplacé par des alternances de crise et de détente, il faut disposer, en matière de lutte contre les sous-marins, d'un système d'armes qui permette de connaître la présence de ces bâtiments, de les suivre aussi longtemps qu'il est nécessaire et, éventuellement, de les détruire. Aérospatiale fut chargé par le Gouvernement français de coordonner la réalisation de ce système d'armes, qui devait être efficace contre des sous-marins marchant au moins à 25 noeuds en plongée et capables de demeurer immergés pendant de très longues périodes.

Contre un tel type de bâtiment, outre l'écoute passive, l'utilisation des ultra-sons était, aux environs de 1960, le seul moyen de détection envisageable. Un sonar de la famille AQS-13 fut donc retenu. Mais, compte-tenu des portées de détection possibles, il était évident que le sous-marin ayant une grande marge de vitesse risquait de pouvoir s'échapper, s'il n'était pas possible de diriger, sur de nouveaux points de recherche, d'autres hélicoptères, lorsque l'objectif arriverait en limite de portée du sonar. Cette manoeuvre devrait être accomplie avec précision, avec une certaine souplesse et avec un échange suffisant d'informations. Précision pour retrouver le sous-marin, souplesse pour changer le lieu de mise en station d'un hélicoptère pendant son transit, en cas de tentative d'évasion du sous-marin et échange d'informations pour donner, à l'autre hélicoptère, à son arrivée en station, la possibilité de trouver son but aussi rapidement que possible.

Pour accomplir cette tâche, if fut fait appel à un radar fabriqué par la Société Française C.F.T.H. appelé SYLPHE permettant de localiser, à partir d'un appareil porteur, trois autres hélicoptères et de leur transmettre des informations codées. La synthétisation de la situation tactique étant de première importance, celle-ci fut présentée sur une table dite table tactique, dérivée de celle construite par la Sté Crouzet pour le patrouilleur Atlantic. Sur cette table apparaît un quadrillage de coordonnées grille par rapport auquel la position de l'hélicoptère porteur est constamment entretenue en utilisant les informations fournies par un radar doppler. Cette position est matérialisée par un petit cercle gradué sur lequel est repéré le cap de l'hélicoptère. Les trois autres appareils sont représentés par des croix de différentes couleurs dont les positions sont automatiquement entretenues par l'intermédiaire du radar SYLPHE.

Les objectifs, détectés par le sonar de l'hélicoptère porteur, apparaissent également sur la table tactique, de telle manière qu'il est possible, connaissant la position de l'objectif et celle des quatre hélicoptères, de déplacer un de ceux-ci pour pister le sous-marin. Pour faire exécuter cette manoeuvre, le commandant du dispositif envoie, également par l'intermédiaire du radar SYLPHE, à l'appareil qu'il a choisi, une route à suivre et une distance à parcourir. Ces deux éléments sont entretenus pendant le transit de l'hélicoptère en déplacement, et peuvent être modifiés à tout moment, si le sous-marin change soit sa route, soit sa vitesse.

Ce qui intéresse surtout les utilisateurs, c'est la précision d'ensembe d'un tel système d'armes. L'exploitation des nombreux essais en vol effectués avec des sous-marins, ont montré que le Super Frelon peut larguer sa torpille MK.44 dans un cercle, autour de la position du sous-marin, tel que, compte-tenu des performances de la tête chercheuse de cette arme, il existe de très grandes chances de faire but sur l'objectif.

On peut donc conclure que le système d'armes du Super Frelon est à la hauteur des espoirs que la Marine française avait fondé sur lui. Comment est-on arrivé à ce résultat?

La complexité des équipements mis en œuvre était telle qu'il fut décidé de donner plus de souplesse et d'efficacité aux procédures administratives habituelles. Pour ce faire, Aérospatiale reçut la tâche de coordonner les activités de tous les fabricants d'équipements et de prendre la responsabilité de l'intégration de tous les équipements à l'intérieur de ce système d'armes. En particulier, cette Société devait faire tous les essais au sol et en vol permettant d'aboutir à une mise au point satisfaisante. En cas de difficultés risquant de mettre en cause soit les délais, soit le budget de l'opération, elle devait proposer toute modification jugée souhaitable ou tout compromis permettant d'aboutir au résultat recherché, au moindre coût, par exemple accepter des performances moindres sur un équipement, si celles-ci n'étaient pas nécessaires à l'obtention de la précision visée pour l'ensemble du système d'armes.

Pour l'aider dans sa tâche et faire prendre à temps, par les Services Officiels, les décisions nécessaires, un Groupe de travail fut créé comprenant, outre Aérospatiale et les Services Techniques Officiels, un représentant de la Marine. Toutes les fois qu'il était nécessaire, les fabricants d'équipements ou les ingénieurs des centres d'essais officiels étaient convoqués aux réunions de ce groupe. Celles-ci se tenaient systématiquement tous les mois et tous les problèmes non résolus étaient passés en revue. De plus, sous l'impulsion de ce groupe, il fut décidé que, dans toute la mesure du possible, le opérations de contrôle et d'essais en vol seraient menées simultanément par le constructeur, les Services Officiels et la Marine. L'adoption de telles procédures a permis de réaliser, en environ 15 mois, la mise au point d'un système d'armes très complexe et répondant à la demande des utilisateurs.

Chronologiquement, le SA 330 PUMA suit de très près le Super Frelon dans la gamme des productions de l'Aérospatiale. Cet appareil a été conçu pour répondre à une fiche-programme publiée par l'Armée de Terre française en 1963 et précisée en 1964. Il devait pouvoir assurer le transport de 12 hommes équipés ou le ravitaillement de troupes au sol ou évacuer 6 blessés.

Ces missions peuvent ne pas sembler particulièrement difficiles à accomplir et, de fait, le Puma fait beaucoup mieux que ce qui était exigé à l'origine, puisqu'il peut emmener vingt passagers à une distance franchissable maximale d'environ 625 km et est capable d'une vitesse de croisière supérieure à 250 km/h. Néanmoins, il faut bien avouer que, sur certains points, l'Armée de Terre n'a pas obtenu tout ce qu'elle demandait.

Examinons quels sont les plus significatifs de ces points. Le plus évident est le dégivrage de la voilure de l'appareil. Les connaissances techniques de l'époque ne permettaient pas de réaliser un tel dispositif, de manière économique, en 1965. Plus originale était la demande d'un train d'atterrissage automoteur qui aurait été capable de faire déplacer l'appareil, au sol, à une vitesse d'environ 10 km/h et de lui faire franchir des fossés à bords francs de 0,50 m de largeur ou des pentes de 12°. Ceci correspondait au souci légitime de l'Armée de pouvoir mettre à l'abri et camoufler ses appareils. Malheureusement, la réalisation d'un tel dispositif aurait été trop onéreuse et fut finalement abandonné.

De même, un système de navigation autonome, donnant une précision de 0,5% de la distance parcourue sur 200 km était demandé. Il aurait donné ses indications à la fois par lecture directe, sur une carte, par exemple, et par entretien de la route et de la distance à parcourir pour rallier cinq points affichables en vol par l'équipage. Cet ensemble de navigation ne put être réalisé pour le Puma pour des raisons budgétaires; par contre, Aérospatiale a fait étudier, de sa propre initiative, un système moins ambitieux et donc plus économique, disponible à ce jour.

C'est également pour des raisons financières que l'Aviation de l'Armée de Terre française a dû renoncer à munir le SA 330 d'un détecteur d'obstacles, pouvant signaler la présence de câbles même non métalliques, et d'un système permettant le vol de groupe par mauvaises conditions météorologiques.

Inutile de préciser que l'abandon de ces différents équipements a été difficile à faire accepter par l'Armée française. Il eut sans doute été infiniment préférable qu'un dialogue entre constructeur et utilisateur mit bien en évidence, avant la rédaction de la fiche programme de l'appareil, les problèmes de finance et de délais auxquels il fallait s'attendre pour intégrer de tels systèmes à un hélicoptère qui par ailleurs devait être robuste et devait pouvoir être entretenu par des unités peu équipées.

De ce dernier point de vue, un gros effort a été fait pour donner satisfaction aux utilisateurs. Un groupe de travail comprenant des représentants de l'Armée, des Service Officiels et d'Aérospatiale a étudié dès le stade initial de sa conception l'entretien de l'appareil. Il s'est efforcé avec succès de simplifier les travaux à exécuter pour obtenir le maintien en disponibilité de cet hélicoptère et d'utiliser, au maximum, des ensembles intégrés à la structure pour faciliter le travail des équipes au sol.

Autres motifs de satisfaction pour l'utilisateur, mis à part les performances, dont il a été déjà parlé, la grande maniabilité de l'appareil due à sa réserve de puissance, qui est de l'ordre de 45%, et, évidemment, la sécurité qui résulte de cette motorisation, surabondante pour les cas normal de vol. Celle-ci permet également le transport de la charge utile maximale jusqu'à 1500 mètres d'altitude et 35° de température ambiante.

Enfin, s'il a pu apparaître que l'Armée de Terre française avait eu des vues quelque peu prophétiques au moment de la conception du Puma, cela est encore plus clair si l'on sait qu'elle avait également, pour cet appareil, demandé un dispositif permettant de supprimer le rotor anticouple.

Comme chacun sait, ce dispositif a été mis au point, sous le nom de Fenestron, pour le dernier né de la famille hélicoptères de l'Aérospatiale: le SA341 Gazelle. Une conférence particulière traitera de ce sujet intéressant. Aussi est-il plus opportun de signaler les points qui, sur cet appareil, apporteront, espérons-nous, des motifs de satisfaction aux utilisateurs.

Ceux-ci seront sans doute constitués par l'amélioration des performances, amélioration considérable par rapport à celle des Alouette II, auxquelles cet appareil va succéder. Citons d'abord la vitesse qui passe de 180 km/h à 260 en croisière, la vitesse maximale de Gazelle pouvant atteindre près de 300 km/h puisque cet appareil a battu le record de sa catégorie avec plus de 310 km/h. Non moins intéressante est l'amélioration de la charge utile: dans le cas de l'Aouette II, le rapport de la charge utile à la masse maximale de l'appareil était de 45% ; il devient 50% pour Gazelle, dès le début de la production, avec des espoirs très précis d'améliorer encore ce chiffre par la suite.

Mais les préoccupations d'Aérospatiale ont porté surtour sur les manières de faciliter l'entretien de cet appareil et de rendre sa mise en œuvre plus économique. C'est ainsi que ses pales sont en stratifié fibre de verrerésine, ce qui permet de conférer, à ces ensembles, une durée de vie infinie et a autorisé, également, l'emploi d'un moyeu rotor principal simplifié. Si celui-ci conserve l'articulation classique de battement, la nécessité d'utiliser une articulation de traînée compliquée, comme sur les hélicoptères précédents, est éliminée entraînant une simplification des opérations d'entretien de la tête rotor. De même, il a été possible de ne pas utiliser d'amortisseurs pour le train d'atterrissage généralement nécessaires pour éviter la résonance sol.

Comme pour le Puma, un groupe de travail s'occupe de trouver, pour Gazelle, des solutions simples aux problèmes d'entretien courant et il est à peu près certain que le rapport utilisé habituellement, caractérisant la fréquence et la complexité de ces opérations, sera inférieur à un homme heure de travail par heure de vol.

A travers ces exemples, il est possible de dégager quelques conclusions concernant les rapports constructeur-utilisateur. Tout d'abord, une fiche-programme est faite dans la majorité des cas pour un appareil qui sera en formation, cinq à six ans après; elle est, donc, basée sur les connaissances techniques disponibles au moment de sa rédaction et doit tenir compte des évolutions envisageables dans le courant des années suivantes. Enoncer cela, c'est évidemment montrer la nécessité des échanges qui doivent exister entre les Etats-Majors, les Ingénieurs des Services Officiels et ceux du Constructeur, au moment de la conception d'un programme.

Ensuite, toute fiche-programme doit être soumise à des remises à jour, car ou bien elle aura fait un pari trop audacieux sur les évolutions techniques à venir ou bien certaines améliorations se seront développées plus rapidement que prévu; dans un cas comme dans l'autre, il ne faut pas hésiter à modifier les clauses techniques de l'appareil et, là encore, les échanges entre les différents participants du programme sont nécessaires.

Enfin, par dessus tout, il ne faut pas perdre de vue le but poursuivi, c'est-à-dire la mission que doit accomplir l'appareil. Pour y atteindre, certaines exigences initiales peuvent se révéler superflues, par contre d'autres demandes peuvent avoir été trop timides. Il est, alors, nécessaire de modifier l'une et l'autre, en fonction du budget imparti au programme; pour ce faire, il peut être utile de créer un groupe de travail comprenant des représentants de l'Etat-Major, des Services Techniques Officiels et du constructeur, pour faire prendre rapidement les décisions nécessaires.

Reste un problème très important: l'amélioration de l'exécution et la réduction du nombre des opérations d'entretien des appareils. Il semble que ce problème soit traité, à peu près partout dans le monde, par une équipe, groupant une fois encore, des personnes des organismes déjà cités. Inutile de dire que ce problème est abordé avec tout le soin nécessaire par les constructeurs, ne serait-ce que parce que les hélicoptères ont de plus en plus de débouchés, dans le domaine civil où les considérations économiques sont primordiales.

Au travers de ces idées générales, il apparait que la conception d'un hélicoptère moderne et sa mise au point nécessitent un travail d'équipe soutenu auquel doivent participer: Etats-Majors, Services Techniques et constructeurs. Cela n'est pas là le travail le moins passionnant.

– FIABILITE ET SECURITE EN OPERATION DES PIECES MECANIQUES – – POUR HELICOPTERES –

Ingénieur de l'Armement S.BERNER
– SERVICE TECHNIQUE AERONAUTIQUE –
– 4, avenue de la Porte d'Issy – PARIS XVe –

Les réflexions qui suivent ne prétendent pas faire l'inventaire de toutes les questions que pose l'emploi des hélicoptères du point de vue de la sécurité et de la fiabilité. Elles s'efforcent plutôt de faire le point à l'échelle française des progrès réalisés sur ce plan dans la conception des pièces mécaniques pour hélicoptères.

On pourra au premier abord s'étonner de l'accent que nous mettons, en lui consacrant cet exposé, sur la défaillance des pièces mécaniques. En effet les collisions en vol, le feu de l'ennemi, les pertes de contrôle de l'appareil, les fautes caractérisées du personnel navigant ou au sol, précèdent, et parfois de loin, les défaillances des pièces mécaniques dans l'analyse par importance des causes d'accidents ou d'incidents d'hélicoptères (tableau n° 1). Il en découle que l'on devrait s'attacher à instruire le personnel, mettre au point des dispositifs anti-collision, diminuer la vulnérabilité de l'hélicoptère à l'action adverse, avant de demander aux constructeurs des ensembles mécaniques plus fiables et plus sûrs.

Force nous est de constater que la situation dans la réalité est un peu différente, et que si on ne peut nier les efforts réels faits dans ces divers domaines, ils ne porteront probablement leurs fruits qu'à plus long terme. Par contre les organes mécaniques ont été, ces dernières années, l'objet de très nombreuses études visant à augmenter leur durée de vie ou leur potentiel d'utilisation.

Après avoir souligné à l'aide d'un ou deux exemples les aléas rencontrés dans la détermination des durées de vie des pièces mécaniques pour hélicoptères, nous essayerons dans une première partie de présenter, en nous appuyant sur une réalisation pratique, ce qui nous semble être la solution d'avenir : la conception à caractère "fail-safe". Dans une deuxième partie, nous parlerons de la qualification des boîtes de transmission mécaniques et de l'attribution de leur potentiel initial d'utilisation. Ces sujets sont actuellement au coeur d'un certain nombre de discussions entre constructeur et Services Officiels français et nous tâcherons de préciser au mieux notre optique dans ce débat, en la confrontant avec celles, parfois assez différentes, adoptées dans d'autres pays.

– DETERMINATION DES DUREES DE VIE –

La fixation de la durée de vie en service des pièces vitales d'hélicoptères travaillant à la fatigue est un problème important et difficile.

Toutes les méthodes qui peuvent être employées par les constructeurs pour justifier aux yeux des Autorités une durée de vie en service tendent à prendre en considération le caractère aléatoire de la résistance en fatigue et en conséquence à estimer la marge à prendre sur la résistance moyenne pour maintenir le risque de rupture à un niveau acceptable.

Notons déjà qu'à ce stade, la fixation du risque acceptable peut faire l'objet d'appréciations très différentes. Appliquer par exemple à une pièce la "règle Américaine des 3 σ", consistant à prendre pour la courbe de Wohler sûre (Working curve) S, N une marge égale à trois fois la dispersion par rapport à la courbe moyenne, conduit à multiplier 5 à 10 fois la durée de vie calculée par la Méthode Française (fig. n° 1). Il faut indiquer tout de suite que ceci ne pénalise pas de façon grave les constructeurs français. En effet, la marge de résistance entre une bonne pièce (2 000 heures par exemple) et une mauvaise (100 heures) tient à peu de chose. Environ 20 % de résistance en fatigue supplémentaire, qui ne dépendent souvent que d'un dessin soigné des pièces, effort toujours possible au stade prototype.

Certains prétendront que l'expérience en service justifie à postériori la validité de telle ou telle méthode. Le raisonnement est inexact. La survie effective de 2 000 pales jusqu'à la durée de vie fixée permet seulement d'affirmer qu'il y a peu de chances que le risque initial ait été supérieur à 1/ 2 000. Vouloir prendre argument de cette réussite pour essayer de prolonger la durée de vie au delà, comme cela pourrait se faire pour des pièces non vitales, constitue le meilleur moyen de réduire la vie des passagers.

Après avoir seulement mentionné ces divergences de méthodes, qui peuvent déjà modifier à elles seules l'ordre de grandeur des durées de vie accordées, venons en à ce qui leur est commun. Dans la détermination rationnelle de la durée de vie d'une pièce entre la connaissance de trois facteurs de base :

- Les contraintes associées à chaque cas de vol prévu dans le spectre d'utilisation de l'appareil.
- La fréquence de chacun de ces cas de vol.
- La résistance en fatigue de la pièce.

Or la connaissance de chacun de ces trois facteurs est associée à des aléas que l'on s'efforce de couvrir mais rarement de façon satisfaisante.

Citons-en une illustration qui nous semble assez significative. Elle concerne la dégradation en service des conditions de travail des pièces mécaniques.

Il s'agit de la rupture en vol d'un boulon à oeil de commande de pas sur pale principale survenu au début de l'année 1969, sur un appareil Super-Frelon de notre Marine Nationale (Fig. n° 3).

Le pilote avait écourté sa mission à la suite d'un niveau vibratoire anormal de l'hélicoptère et seule la visite après vol révéla la rupture totale de la cage de rotule du boulon à oeil.

Les premières conclusions de l'expertise montrèrent que la rupture n'était pas due à un défaut métallurgique mais à des surcontraintes excessives.

Après un examen systématique des éléments du moyeu de l'appareil, sans résultats, un premier vol de contrôle montra que les efforts dans les biellettes étaient très supérieures à la normale. L'état d'érosion du bord d'attaque des pales conduisit alors à penser que l'augmentation des efforts était due à une anomalie aérodynamique ou à une variation du centrage des pales due à l'érosion.

Les essais ultérieurs devaient montrer que l'augmentation brutale des efforts pouvait résulter d'une érosion d'aspect tout à fait normal et que la seule disparition de la protection anodique dure du bord d'attaque suffisait à endommager considérablement toute la chaîne de commande.

La figure n° 2 montre qu'en croisière à 230 km/h, on volait au-dessus de la VNE avec des pales érodées, ce qui explique la rupture constatée.

L'établissement des durées de vie reposait sur des contraintes mesurées sur un appareil avec des pales en bon état et ne tenait pas compte de la dégradation due à l'érosion.

C'est cet incident qui a motivé la mise au point d'un indicateur de contraintes en vol, instrument aujourd'hui utilisé par de nombreux constructeurs. Il permet de signaler les dégradations éventuelles et indique la VNE effective. Toute une série de travaux ont par ailleurs porté sur la protection des bords d'attaque des pales, travaux qui se poursuivent encore à l'heure actuelle. La principale difficulté est d'offrir une protection efficace à la fois à la pluie et au sable et aucune solution ne semble donner jusqu'à présent entière satisfaction.

Un deuxième exemple destiné à mettre en évidence les aléas que comportent les méthodes actuelles de détermination des durées de vie peut être trouvé dans l'utilisation de matériaux relativement nouveaux, comme le titane dans la fabrication de pièces de très grande dimension travaillant à la fatigue.

L'établissement de "facteurs d'échelle" dans les matériaux classiques tels que l'acier ou les alliages légers a fait l'objet de travaux extrêmement étendus et constitue une des bases de la longue expérience que la plupart des constructeurs ont aujourd'hui acquise et gardent d'ailleurs très jalousement pour eux. Aussi ne s'étonnera-t-on pas de ne voir citer ici aucun chiffre. La dispersion des caractéristiques de tenue en fatigue des grosses pièces matricées en titane ne leur a pas moins réservé de très désagréables surprises. Elle peut en effet atteindre des valeurs que les dissections et essais préalables étaient loin d'indiquer. On est conduit, en attendant que les essais de fatigue sur grandes pièces en titane s'accumulent, ou bien à dimensionner trop largement des éléments qui déjà n'ont pas un poids négligeable, ou bien à adopter des facteurs d'échelle "raisonnables", reposant sur un nombre d'essais très limité. Personne ne peut affirmer aujourd'hui que le risque qu'on accepte ainsi de courir est homogène avec le niveau de sécurité offert par des techniques plus éprouvées.

Citons enfin ce que tout le monde se plait à nommer les cas d'espèce : coups d'outil malheureux ou défaut métallurgique échappant à plusieurs contrôles successifs. Chacun a à l'esprit au moins un cas de rupture en vol de cette nature, que les méthodes de calcul ne sont pas sensées couvrir.

Nous pensons que dans ces conditions, partout où elle est réalisable, la notion de caractère fail-safe doit compléter celle de résistance à la fatigue.

Cette idée a déjà donné lieu, dans le domaine de la fabrication des pales, à quelques applications. Citons l'utilisation du BIM ou de tout autre système reposant sur la détection d'une fuite de gaz, le dessin de certaines attaches de pales, comportant une disposition judicieuse de boulons, et tant d'autres pièces dont le caractère fail-safe ne s'est révélé que forfuitement, à la suite d'un incident.

Mais le progrès le plus remarquable nous parait résider dans l'utilisation des fibres à haut module d'élasticité (verre, carbone, bore). Nous sommes, pour notre part convaincus que les avantages du point de vue de la sécurité sont tels qu'un effort doit être fait pour généraliser cette technologie à un grand nombre de pièces mécaniques. Notre conviction est fondée sur un certain nombre d'études et de réalisations portant sur les pales, les moyeux, les transmissions.

Sur les pales nous dirons peu, si ce n'est que les travaux réalisés dans le cadre du développement de l'appareil SA 341 ont démontré que les problèmes posés par le comportement dynamique et le réglage de ce type de pales pouvaient être résolus de façon satisfaisante. La seule question qui restait en suspens était la variation dans le temps des caractéristiques des fibres. Avec le recul de ces trois dernières années, il semble acquis que cette variation relative ne dépasse en aucun cas 3 à 4 % et ne devrait pas donner lieu à inquiétude.

Mais le caractère fail-safe de ces pales ne fait que reporter le problème de la rupture éventuelle plus en amont, sur leur attache au moyeu, sur le moyeu lui-même, sur la chaîne de commande, autant d'éléments à caractère aussi vital que les pales.

C'est à ce stade qu'il n'est peut être pas inutile de revenir sur les principes de la conception fail-safe pour voir comment elle peut s'appliquer aux pièces pour hélicoptères.

D'abord on peut noter que fail-safe n'est pas forcement synonyme de redondance, encore moins d'une redondance externe telle que la duplication. Tout pousse à penser d'ailleurs que mettre en parallèle 2 éléments identiques n'offre jamais la sécurité donnée par le simple calcul de fiabilité. Il n'est pas rare en effet de constater la défaillance simultanée de 2 éléments indépendants et identiques en parallèle. La panne dépend plus de la nature des conditions de fonctionnement critiques (vibrations, chaleur, chute de tension etc...) que du nombre de systèmes mis en parallèle. La solution semble être du côté de la superposition en parallèle d'éléments de conceptions différentes, avec des conditions de fonctionnement critiques différentes :

Le caractère fail-safe peut être défini de la façon suivante :

Une pièce présente un caractère fail-safe si la proximité d'une rupture est toujours détectable un certain temps avant l'issue catastrophique. Ce délai d'anticipation doit permettre la poursuite de la mission, et éventuellement la possibilité d'en entreprendre d'autres avant réparation.

L'alarme peut se réduire à un aspect extérieur anormal, repérable aisément lors des visites avant vol, à un niveau vibratoire de la machine inhabituel mais ne mettant pas en danger l'équipage, ou prendre la forme d'une alarme visuelle ou auditive.

Les pièces en fibres mises au point, réalisées et essayées par le constructeur français d'hélicoptères l'Aérospatiale ont présenté en fatigue le comportement suivant :

- Résistance en fatigue conduisant à une durée de vie théorique infinie.

- Une délamination en surface des fibres et une coloration inhabituelle de la zone détériorée sont les premiers symptômes de l'affaiblissement de la pièce. Leur apparition précède de l'équivalent de plusieurs dizaines d'heures la rupture véritable de la pièce.

- Variation des caractéristiques dynamiques de la pièce : fréquences propres et amortissement structural.

Ces caractères ont été jugés comme pleinement suffisants pour constituer le caractère fail-safe recherché.

L'exemple d'application le plus intéressant, et sur lequel nous allons donner quelques détails, nous semble être le moyeu d'hélicoptères semi-rigide, réalisé en fibres de verre, et destiné à supprimer les articulations de battement et de traînée. La tenue en fatigue et les grandes déflections permises par ce matériau le prédestinent, nous semble-t-il, à un tel emploi.

En effet, sans rentrer dans le détail des problèmes liés à un rotor semi-rigide, on peut schématiquement dire que dans le cas où une certaine souplesse du premier mode est souhaitée ou inévitable, il y a intérêt à la concentrer le plus possible dans le moyeu, en amont de l'articulation d'incidence. Ceci pour éviter tous les couplages par les déformées statiques de battement ou de traînée.

La figure n° 4 représente un tel moyeu, réalisé en fibres de verre. Sa simplicité rejoint celle d'un moyeu comme celui du LYNX.

Ce moyeu comporte essentiellement :

- Un structure centrale en étoile (le corps de moyeu) réalisée en matériau composite fibres de verre résine et dont les trois branches constituent les éléments souples en battement et traînée, dimensionnés de façon convenable pour assurer la souplesse et la résistance requises.

- Des articulations de pas fixées à l'extrémité des bras souples, constituées par des paliers et des faisceaux torsibles de retenue d'effort centrifuge.

- Des amortisseurs de traînée à faible course. Leur présence qui peut paraître étonnante nous semble justifiée tant que des amortisseurs structuraux suffisants n'auront pas été obtenus sur ce type de réalisations. C'est probablement un point sur lequel tout progrès futur sera très payant.

L'utilisation de fibres à haut module d'élasticité dans des pièces telles que les transmissions arrières, les moyeux de rotors arrière donne lieu également à de très importants travaux, les avantages recherchés étant, à côté de la tenue excellente en fatigue et du caractère fail-safe, la diminution de poids, la simplicité, l'absence de maintenance.

Mais ces efforts consentis pour augmenter les caractéristiques de tenue en fatigue des pièces d'hélicoptères et compléter la notion de durée de vie sûre par celle de caractère fail-safe ne sont qu'une partie d'une préoccupation plus générale qui est l'augmentation des durées d'utilisation des ensembles mécaniques. Ceux-ci comportent un certain nombre d'ensembles de transmission, de renvois et de réducteurs auxquels il convient d'accorder une attention toute spéciale.

- QUALIFICATION DES ENSEMBLES DE TRANSMISSION -
- ATTRIBUTION DE LEUR POTENTIEL INITIAL D'UTILISATION -

C'est qu'un certain nombre de contraintes nouvelles sont venues sensiblement bousculer les habitudes prises dans ce domaine. Ce sont principalement :

- Les spectres de vol très sévères caractéristiques du vol tactique à proximité du sol, et dont les répercussions sur les mécaniques arrières sont considérables. Les déplacements tactiques comprennent aujourd'hui un nombre de manoeuvres en lacet élevé, qui peut s'avérer comme l'élément déterminant dans le dimensionnement des éléments arrières.

- La nécessité de démontrer des durées de vie infinies pour les pignons de la boîte de transmission principale, et une durée de vie supérieure, à disons 10 000 heures pour les pignons de la transmission arrière. Ces derniers sont en effet amenés à travailler sous des efforts qui endommageront forcément ne serait-ce que de façon temporaire, et garantir une durée de vie théoriquement infinie reviendrait à en faire des monstres.

- La tendance à offrir aux utilisateurs, dans une machine qui se veut réussie, des potentiels initiaux entre révisions les plus grands possibles. 1 000 heures étant considérées comme souhaitables, et 2 000 heures satisfaisantes.

- Enfin le cadre de coopération internationale dans lequel la plupart de nos machines sont réalisées et le seront probablement à l'avenir. Il faut envisager qu'une boîte de transmission soit dessinée par un bureau d'étude français, mise en gamme par un bureau spécialisé italien et fabriquée en Grande-Bretagne. On comprendra qu'en dehors des difficultés de langage, des problèmes puissent surgir de cet éclatement. Il est essentiel de vérifier la qualité d'une fabrication sur une base acceptée par tous les coopérants. Sinon, on ne pourra éviter que l'un mette sur le compte des caractéristiques du matériau ce que le deuxième impute à la largesse des tolérances et le troisième au dimensionnement même des pièces.

Devant ces nouvelles exigences, force nous est de constater que les procédés actuels de qualification et de justification des potentiels initiaux entre révisions s'avèrent caduques.

Ceux-ci reposent sur 3 points :

- Justification en fatigue par le calcul.

- Essais d'endurance au sol et en vol.

- Essais dits de "surpuissance".

- La tenue en fatigue des pignons est justifiée par le calcul. Une entorse importante est cependant faite à la règle sacro-sainte du tiers (1/3)*. On se contente en effet de démontrer que les contraintes calculées sont inférieures à un niveau considéré comme admissible par les règles de l'art, niveau qui se situe plutôt à la moitié (1/2) de la limite de fatigue moyenne du pignon qu'à son tiers. Nous avons jusqu'à présent, en temps que Services Officiels, toléré cette justification. L'argument essentiel est que l'expérience accumulée par le constructeur l'Aérospatiale dans la fabrication des engrenages est considérable et n'est jamais venue infirmer cette méthode, du moins, et cette remarque est importante, tant que ce constructeur était seul en cause. Cet argument devient beaucoup plus discutable dès que l'on envisage un cadre de travail international.

- Le comportement en service des roulements, joints, flectors, paliers etc... est jugé par des essais dits d'endurance, effectués pour une part en vol et pour une autre part au banc sous les charges usuelles rencontrées en service. La règle est simple. Pour justifier par exemple 500 heures de potentiel initial entre révisions, il faut avoir réalisé sans détérioration des pièces :

Banc :		1 ensemble	1 000 heures
Vol :	(	1 ensemble	500 heures
	(+	1 ensemble	500 heures
		TOTAL........	2 000 heures

- Cette règle nous semble mériter la critique suivante : elle conduit à des essais longs et coûteux, dont la valeur statistique est à priori discutable.

- Enfin un troisième essai dit de "surpuissance", ou de "qualification" au cours duquel on applique à l'ensemble pendant environ 100 heures des charges plus sévères que celles rencontrées en vol sans excéder 115 % de la puissance maximale au décollage. Les détériorations de roulements sont tolérées. Les ruptures de pignons ne le sont pas. Lorsqu'on s'interroge sur la signification de cet essai, on constate qu'il en a une certaine : déceler une erreur grossière de conception ou de fabrication, faisant chuter la tenue en fatigue d'un ensemble d'au moins 30 à 40 %. Lorsque les roulements subissent cet essai sans dommage, ce qui n'est pas exigé, il constitue de plus une bonne présomption sur la réussite de la boîte. Sans pouvoir en conclure plus.

- Devant les critiques dont cette façon de procéder peut faire l'objet, et devant les exigences nouvelles que nous avons soulignées plus haut, nous aurions tendance à défendre aujourd'hui la proposition suivante, qui nous semble plus raisonnable :

- <u>Justification de la tenue en fatigue</u> :

On revient à la règle traditionnelle qui autorise la justification par le calcul quand les contraintes calculées sont inférieures au tiers de la contrainte limite moyenne. Dans les autres cas, la justification en fatigue du couple de pignons incriminé doit faire l'objet d'un essai sur banc de fatigue dit "universel" (fig. n° 5). Car il ne reprend de l'ensemble complet que le couple en question, avec ses roulements originaux ou des roulements renforcés. Le but de l'essai est de provoquer la rupture en un nombre de cycles voisin de 10^6 en appliquant les charges correspondantes. Les résultats de l'essai permettent ensuite, par les méthodes de calcul habituelles, de remonter aux charges admissibles avec un risque acceptable (10^{-6}).

Décrivons à titre d'exemple l'essai d'un jeu de 3 pignons :

Deux jeux de pignons sont en fait essayés simultanément dans deux boîtes A et B. Les 2 pignons 3 qui engrènent avec les pignons libres 2 sont solidaires d'un même arbre tandis que le pignon 1/A est monté sur un moyeu pouvant être immobilisé en rotation.

Le même pignon 1/B est solidaire d'un arbre libre en rotation traversant de part en part le pignon 1/A.

Il est alors possible d'introduire dans la chaîne ainsi constituée un couple statique de torsion en faisant pivoter le pignon et en maintenant fixe l'autre extrémité de la chaîne soit le moyeu portant le pignon 1/A.

Les mises en charge sont effectuées au moyen de 2 bras de leviers, l'un équipé d'une contrefiche fixée rigidement sur le bâti, l'autre d'un vérin permettant la rotation du pignon 1/B.

Le couple introduit dans la chaîne est maintenu par une cale bloquée entre les deux bras de levier. La contrefiche et le vérin peuvent ensuite être désolidarisés.

La chaîne est alors libre en rotation.

L'un des deux bras de leviers est accouplé à un excentrique réglé de telle façon que l'engrènement sur les pignons 2 s'effectue sur un minimum de 5 dents ce qui correspond au dégagement total au moins de 2 dents équipées de postes de jauges extensométriques et à une rotation du pignon de $\pm$ 12°.

- <u>Essais d'endurance</u> :

Dans la mesure où les ensembles mécaniques dépendent, du point de vue de leur endurance, de la tenue des éléments les plus fragiles : les roulements, il est possible d'établir une relation entre la durée de vie à fiabilité donnée de ceux-ci et le niveau des charges appliqué pendant l'essai. La forme de cette relation peut être discutée mais la plupart des spécialistes s'accordent pour proportionner la durée de vie au cube de la charge dynamique appliquée.

On peut dans ces conditions raccourcir sensiblement la durée des essais d'endurance en appliquant un taux de surcharge convenable. Ceci devrait conduire à des temps de réaction plus courts dans le cas où une modification s'avère nécessaire, mais surtout permettre l'essai d'un plus grand nombre d'ensembles et rétablir la valeur statistique de la justification.

* Une contrainte inférieure au tiers de la limite de fatigue moyenne de la pièce est jugée comme apportant un endommagement nul.

L'accélération imprimée à l'essai dépend du taux de surcharge jugé comme acceptable et qui se situe aux environs de 115 % de la puissance maximale au décollage.

Au delà de ce seuil, la conformité des déformations ne peut plus être assurée, ce qui enlèverait une partie de sa valeur à l'essai.

Les ensembles mécaniques qui équipent les hélicoptères actuels peuvent prétendre, s'ils sont réussis, mériter une attribution initiale de potentiel entre révisions supérieure ou égale à 1 000 heures. Ces progrès sont dus non seulement au soin porté sur la fiabilité globale dès le stade de la conception mais aussi à l'emploi de plus en plus généralisé de moyens de contrôle tels que : bouchon magnétique muni d'une alarme cabine permanente, analyse spectrométrique des huiles par prélèvements, analyse des bruits d'engrenages, etc... Ces moyens de contrôle assurent le caractère "fail-safe" des boîtes de transmission, au sens où nous l'entendons. Encore convient-il de le vérifier au cours d'un essai. L'essai d'endurance accéléré en surcharge nous semble tout indiqué pour assurer ce rôle.

- Quand à l'essai dit de "qualification" nous serions tout disposés dans cette nouvelle optique... à le supprimer. Certains penseront à l'appliquer pour justifier l'équivalence de qualité entre plusieurs fournisseurs. Mais nous avons assez dit qu'il est seulement capable de faire apparaître lorsqu'elles existent les divergences de qualité de grosse amplitude. Le reste passera inaperçu.

- COMPARAISON AVEC LES DEMARCHES AMERICAINES ET BRITANNIQUES -

Justifications américaines.

Nous pensons savoir que la plupart des constructeurs américains jugent que la justification en fatigue des pignons par le calcul est suffisante. Nous ignorons les marges de sécurité prises mais en tout état de cause, l'homogénéité de leur fabrication peut expliquer une telle approche. Cette homogénéité risque de ne pas se retrouver dans des programmes conçus à l'échelle européenne et faisant simultanément appel à la technologie développée dans plusieurs pays.

Par contre leur philosophie en ce qui concerne les essais d'endurance est tout à fait semblable à la nôtre, du moins en ce qui concerne les tendances d'avenir.

Justifications britanniques.

Les constructeurs britanniques pensent qu'il faut appliquer aux boîtes de transmission, considérées comme un tout, les mêmes règles, en ce qui concerne la justification de leur tenue en fatigue, que pour toutes les autres pièces mécaniques.

Ceci conduit à des essais en surcharge extrêmement sévères, un facteur multiplicatif d'environ 1,7 étant appliqué aux charges de vol mesurées pendant une centaine d'heures, ou plus exactement l'équivalent de 5×10^6 cycles sur le pignon le plus lent. (On peut, à titre de comparaison, indiquer qu'un essai d'endurance effectué avec un facteur d'accélération de 4, considéré comme un maximum, (100 heures au banc = 400 heures en vol) signifierait qu'on applique un facteur multiplicatif sur les charges de vol de 1,6. Cela situe la rigueur de l'essai de fatigue anglais. La question qu'il pose est bien sûr celle de sa conformité. Il est probable qu'un ensemble travaillant dans ces conditions extrêmes peut donner lieu à des ruptures dont l'origine pourrait être imputée à la déformation statique d'un carter ou au mauvais réglage d'une portée de denture. Aussi doit-il probablement faire l'objet de réglages ou de rigidifications spéciaux, interventions dont la nature pourra toujours paraître suspecte.

Si les britanniques ont leur conception propre relative à la tenue en fatigue des pignons, ils sont plus proches de nous en ce qui concerne leur comportement en endurance. Ils ne semblent pas cependant disposés à utiliser la possibilité d'accélérer leurs essais par des taux de surcharge appropriés.

Notre tour d'horizon serait incomplet si nous ne mentionnions pas brièvement les problèmes spécifiques aux ensembles de transmission arrière.

Le dimensionnement et les essais qui leur sont relatifs devraient, à notre sens, tenir compte des 2 exigences suivantes :

- Le cas de vol permanent le plus endommageant, qui est la plupart du temps le vol stationnaire au niveau de la mer avec vent latéral, doit conditionner le dimensionnement du point de vue de la tenue en fatigue des pièces.

- Les manoeuvres de lacet, lorsqu'elles sont réalisables par le pilote sans restriction artificielle sur leur brutalité d'application ou sur la course maximale des palonniers peuvent provoquer la rupture en fatigue temporaire, c'est à dire à faible nombre de cycles, d'un élément de la transmission. Il est nécessaire dans ce cas de vérifier au sol qu'un nombre de manoeuvres bien supérieur (5 à 10 fois) à celui pouvant être réalisé en vol pendant la durée d'utilisation de l'appareil, ne provoque pas la rupture d'un élément de la chaîne.

- CONCLUSION -

Dans la première partie de cet exposé nous avons essayé de montrer qu'une pièce mécanique réputée sûre en fatigue n'est pas à l'abri d'une rupture en vol et que sa conception peut être considérablement améliorée sur ce plan, en particulier par l'utilisation de matériaux à haut module d'élasticité. Nous avons illustré cette possibilité par la description d'un moyeu semi-rigide en fibres de verre.

Dans une deuxième partie, nous avons souligné les insuffisances des méthodes actuelles de qualification des ensembles de transmission mécanique. Nous avons proposé une démarche plus soucieuse de s'adapter aux exigences particulières de la fabrication lorsque celle-ci prend l'échelle européenne. Elle nous paraît de plus, mieux pouvoir s'adapter à un caractère essentiel de la réussite d'un appareil : la justification d'un potentiel initial entre révisions dépassent le cap des 1 000 heures.

Rappelons pour conclure que ces réflexions ne prétendaient qu'aborder les problèmes évoqués plus haut. Si nous sommes parvenus à vous faire sentir que la plus grosse partie de travail restait encore à accomplir, nous aurons atteint le principal objet de cet exposé.

— TABLEAU N° 1 —

REPARTITION DES CAUSES D'ACCIDENTS MORTELS SUR HELICOPTERES ALOUETTE (à fin 1969)

— § —

Cables non balisés et ligne à haute tension :	24 %
Conditions météorologiques :	22,8 %
Non respect des consignes de pilotage :	16,5 %
Non respect des consignes d'entretien :	11,4 %
Collision en vol :	6,3 %
"Mission de surveillance" :	6,3 %
Personnel transporté :	5,1 %
Défaillance de la turbine :	5,1 %
Incident mécanique ou structural :	2,5 %

— § — § —

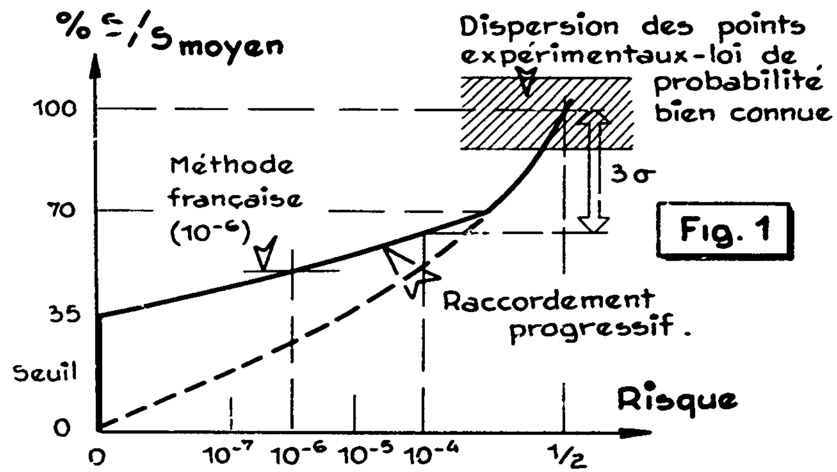


Fig. 1

COMPARAISON DE LA METHODE "3σ" ET DE LA METHODE FRANCAISE.

EFFORTS SUR BRAS DE PLATEAU MOBILE

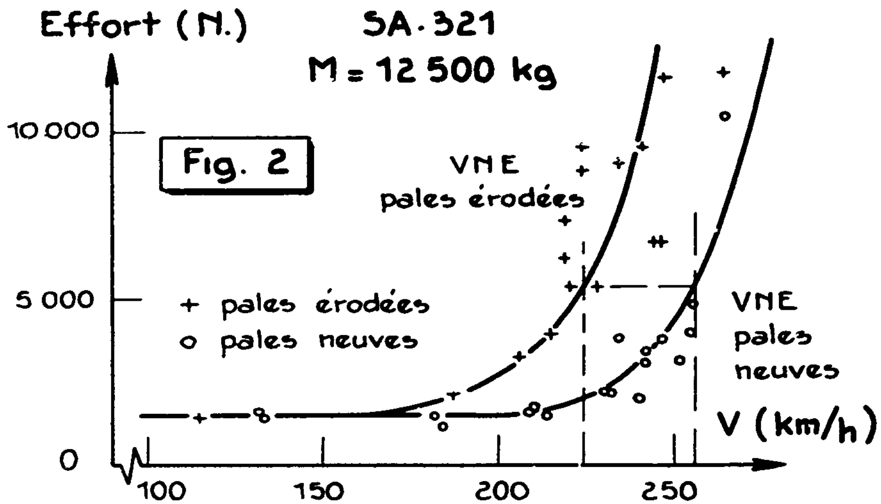


Fig. 2

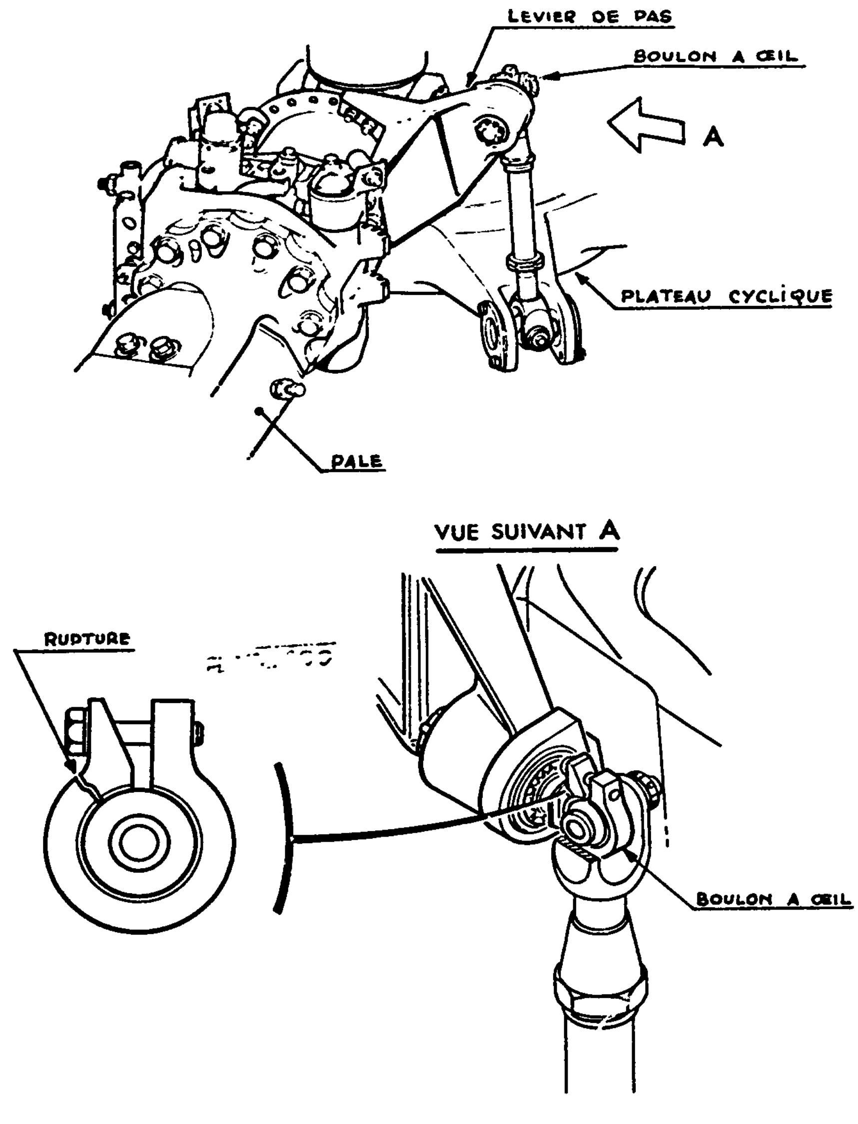
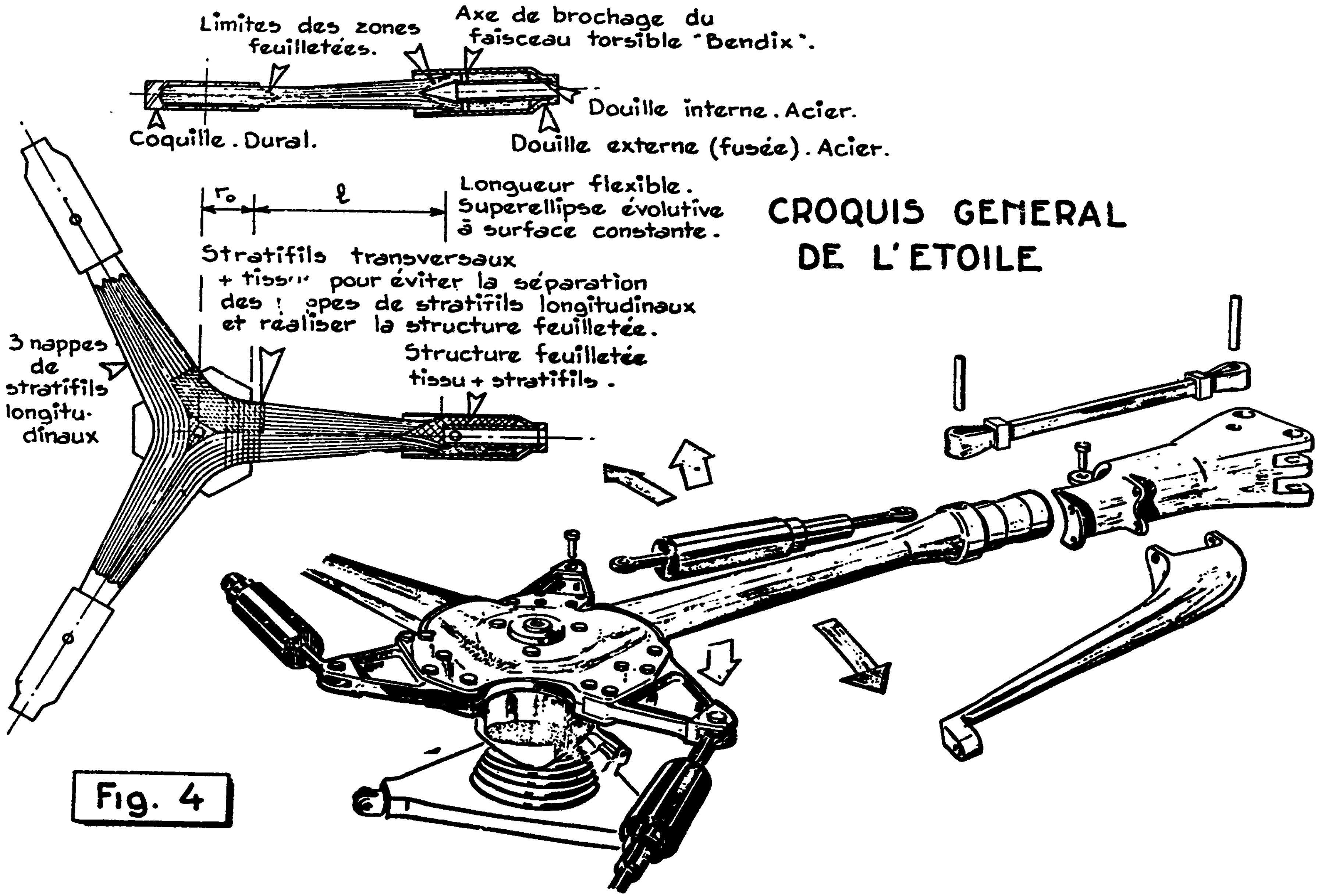


Fig. 3



Limites des zones feuilletées.
Axe de brochage du faisceau torsible "Bendix".
Douille interne. Acier.
Coquille. Dural.
Douille externe (fusée). Acier.
r_o
ℓ
Longueur flexible.
Superellipse évolutive à surface constante.
CROQUIS GENERAL DE L'ETOILE
Stratifils transversaux + tiss… pour éviter la séparation des … pes de stratifils longitudinaux et réaliser la structure feuilletée.
Structure feuilletée tissu + stratifils.
3 nappes de stratifils longitudinaux

Fig. 4

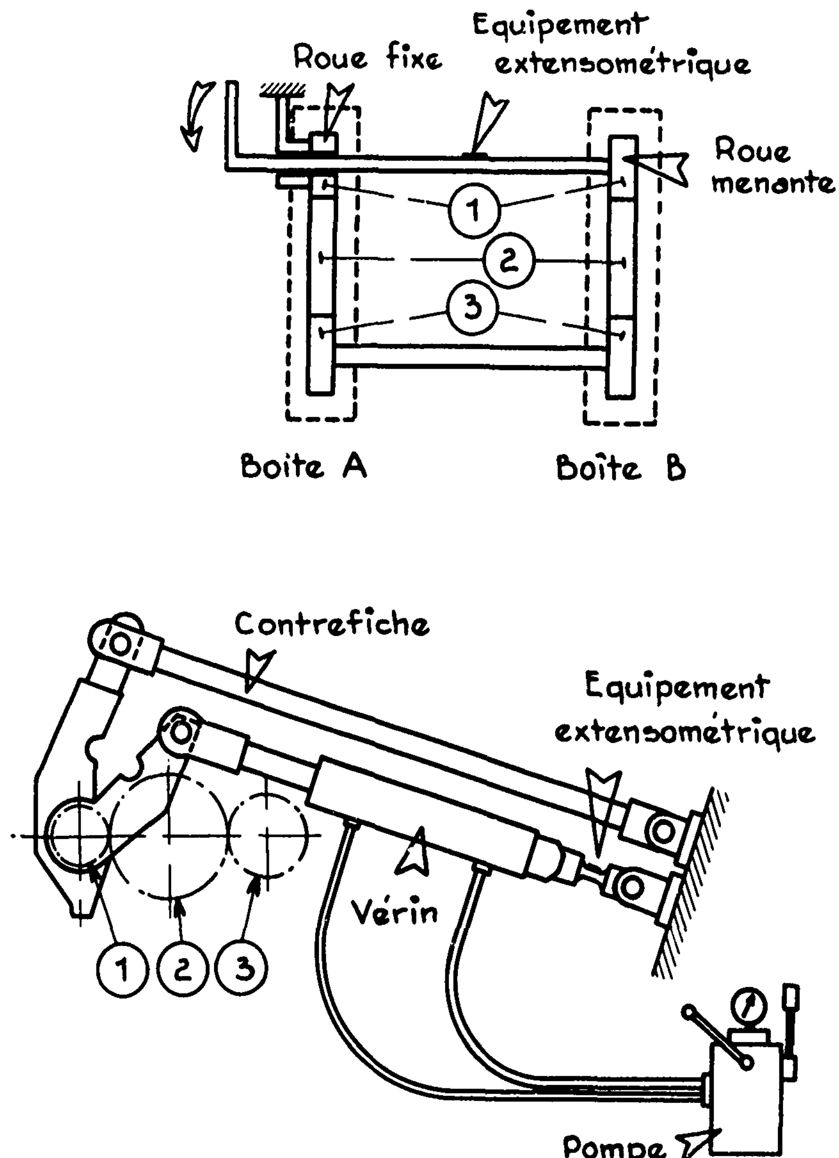


Fig. 5

BANC DE FATIGUE A ENGRENAGES UNIVERSELS

GREATER SAFETY, MAINTAINABILITY, AND RELIABILITY THROUGH IMPROVED HELICOPTER FLIGHT TESTING

by

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SUMMARY

This paper presents significant data obtained from flight test programs recently and presently being conducted by the US Army Aviation Systems Test Activity (USAASTA) at Edwards Air Force Base, California. Greater safety, maintainability, and reliability are being assured through constantly improved flight testing techniques and the use of state-of-the-art instrumentation, data acquisition, and data reduction equipment. More stringent helicopter performance criteria are placing greater demands on the test agencies to devise new methods and procedures for collecting and analyzing data.

Included are such programs as the AH-1G (Cobra) helicopter and a simplified approach to finding height loss during dive recovery from throttle chops; recommendation of limiting AH-1G tail rotor control; OH-6A "g" loads experienced at high frequencies during weapons firing; recommended pilot cues to define a safe AH-1G envelope following engine failure; investigation of large sideslip and pitch excursions following throttle chops in the TH-55 helicopter; identification of requirements for AH-1G instrument-flight-rule (IFR) evaluation; results from OH-58 and AH-1G helicopter height-velocity (H-V) (autorotational) testing with discussion of application to operational use; and AH-1G maneuvering limits from return-to-target profiles.

Tests conducted with the AH-1G helicopter determined time return-to-target time, height lost during pullout from a dive, and other maneuvering characteristics. The concept of energy maneuverability has been established, and significant data have been added to the literature. Test data have been obtained from the OH-58 instrument panel vibrations in the frequency range between 50 and 2,000 Hertz (Hz). During normal flight and while weapons are fired, significant "g" loads have been observed at high frequency, which may contribute to aircraft instrument fatigue and human hearing loss. These test results have indicated the need for sophisticated instrumentation to obtain high-frequency data.

INTRODUCTION

Today, greater emphases are being placed on research, development, testing, and ultimately on the production cycles of combat hardware to obtain the most "bang per buck."

As it develops within the state-of-the-art, USAASTA must seek constant improvements in flight testing techniques and must explore areas heretofore touched only too lightly by the flight test community, especially in the developmental testing of helicopters and light conventional aircraft. These considerations become even more significant as we move into the advanced V/STOL category which includes the AH-56 (Cheyenne) helicopter as well as other advanced aircraft design concepts.

THE US ARMY AVIATION SYSTEMS TEST ACTIVITY

Mission

The official USAASTA mission is "to plan and conduct engineering flight tests of air vehicles developed and/or procured as integrated systems, and those proposed or considered for Army application or incorporating advanced concepts having potential military application; produce test data on basic air vehicle performance, handling qualities system/subsystem interface and integrated system performance, maintainability and reliability."

History

As originally identified, the US Army Aviation Test Office was established on 2 May 1960 at Edwards Air Force Base, California. The late Lieutenant General William B. Bunker, then Commander of the Transportation Materiel Command, St. Louis, Missouri, has been recognized for his vital role in establishing the Test Activity. Its creation marked a significant step forward in the Army's aviation growth and movement toward eventual control of aviation materiel procurements.

The US Army Aviation Systems Test Activity has expanded from a handful of personnel performing liaison and monitoring of Air Force flight tests on Army aircraft to an organization with 193 military and civilian personnel with the full capability of performing engineering flight tests and publishing the test results. During the interim, progress has been marked by organization restructuring, personnel increases, and name changes from which the organization has evolved into USAASTA as it is today. Currently, USAASTA is directed in the accomplishment of its mission by the US Army Aviation Systems Command (AVSCOM), St. Louis, Missouri.

Organization

The US Army Aviation Systems Test Activity is currently engaged in the engineering flight test of all aircraft introduced into the US Army inventory. The organization has been structured to permit relatively independent operation as a tenant activity on Edwards Air Force Base, California. A manning level of 193 military and civilian personnel is now required to manage an average workload of 40 to 45 flight test projects. These test projects are conducted by the use of the team concept which includes a Project Officer who operates across organizational lines to obtain support through coordination with USAASTA subelements and outside agencies as required. The team specialists are the test pilots, engineers, data technicians, instrumentation technicians, support pilots, aircraft mechanics, and assorted administrative/logistical personnel.

Project test teams are "tailored" to meet work requirements and will vary in composition throughout the various phases from project start to completion. Test projects range through the classical phases of planning, instrumentation, flying, data reduction and analysis, report writing, and publication of the flight test report· the ultimate mission objective of documenting flight test results. The USAASTA reports receive varied and broad distribution which includes appropriate government and commercial agencies.

The US Army Aviation Systems Test Activity operates from 136,000 square feet of hangar and office space at Edwards Air Force Base. Annually, the mission budget has been allocated between \$2.3 and \$2.5 million, with direct test project expenditures reimbursable through airframe program managers – our primary customers.

Additional funds (approximately \$2.0 million) were obtained over the past 2 years to procure an advanced instrumentation and data acquisition system (AIDAS) which provides a quantum improvement in data acquisition and reduction. Currently, the onboard aircraft fleet consists of 10 assigned support aircraft and 12 attached test aircraft. Types and models represent the total US Army inventory of air vehicles, with some additional nonstandard types such as the T-28B (Trojan), the C-47 (Gooney-Bird), and the F-51 (Mustang) – all of which were developed during the 1940's.

RESULTS OF RECENT USAASTA FLIGHT TEST PROGRAMS

General

As previously mentioned, USAASTA has constantly searched for new methods and test techniques from which test data may be obtained to confirm requirements for improved safety, maintainability, and reliability in aircraft design. Previously, the criteria for safety were verified by airworthiness and qualification test programs based on the determination of structural integrity of the airframe, propulsion system reliability, aircraft performance, and handling qualities characteristics. Recently, however, the criteria have been verified by collecting and analyzing data to substantiate critical hover ceilings, autorotational capabilities, maneuvering capabilities, and the effects of vibration on aircraft systems and components. To date, USAASTA has explored many of these areas by developing test techniques that produce high-quality data and has also successfully analyzed the results.

Hover Capability

Traditionally, the assessment of hover capability has been included in helicopter performance flight tests and a great deal of effort is spent in achieving precise steady-state hover test conditions. Today's requirements, which include elevated density altitudes and assurance that adequate control margins be available at the critical hover ceiling, place even more stringent demands on the evaluator to develop new techniques which must often be devised to test for new criteria.

As an example, the AH-1G (Cobra) was found to have inadequate directional control at high referred gross weights (W/σ) during early Army evaluations. A test program was devised to measure directional control requirements in translational flight at all azimuths and at a variety of skid heights within ground effect (reference 1). The results are illustrated in figures 1 and 2. It was found that critical combinations of skid height and azimuth could be established which, when combined with a criterion requiring at least 10-percent remaining directional control, would define maximum hover capability. The resulting hover ceilings, as limited by installed power and adequate directional control, are contrasted in figure 3.

It is suspected that the hover flight envelopes of other helicopters should be similarly restricted in order to provide adequate controllability. The US Army Aviation Systems Test Activity completed tests of the UH-1H for the Air Force during August 1971 which indicated that both longitudinal and directional control requirements may limit hovering in winds. Analysis of these data is continuing, and a report will be published in early 1972.

The next generation of Army aircraft (HLH, UTTAS, etc.) will have even more stringent hover condition specifications. Application of the US Army Combat Development Command criteria (reference 2), for example, will require new test methodology to confirm vertical climb capability at the hover design point. The USAASTA Project No. 68-55 is presently underway to develop this methodology.

Height-Velocity

One of the most difficult and dangerous types of flight testing involves the determination of height-velocity (H-V) envelopes. These profiles define safe operating regions in the low-speed, low-altitude portions of the aircraft's flight envelope. The US Army Aviation Systems Test Activity has actively investigated H-V characteristics since 1968, and has developed significant new approaches to H-V testing.

A flight test program involving the UH-1C helicopter (reference 3) was conducted in attempts to predict, in advance, the H-V envelope of a given test aircraft. The program proposes that the H-V maneuver be considered in several discrete segments: throttle chop, pushover, stabilized autorotation, flare, and touchdown. Characteristics of each of these portions of the maneuver can then be analyzed and tested separately – away from the ground – to gain insight into the total maneuver performance.

The concept of H-V testing which has evolved at USAASTA is one demanding preflight analysis, rigorous test procedures, and the extension of test results which are considered reasonable for use by the operational Army aviator. Existing analyses of H-V performance (references 3 and 4) have been found to be of limited accuracy in predicting either maximum achievable or operationally recommended H-V envelopes. Consequently, an extensive analytical effort (USAASTA Project No. 68-25) is underway to improve prediction capability.

Two recent H-V test programs have examined the OH-58A (reference 5) and AH-1G (reference 6) helicopters. Several procedural developments from these programs include the following:

1. Entry condition verification and control via ground-based theodolite operator.
2. A trained test pilot acting as ground observer to augment pilot comments.
3. Video recording with instant playback capability.
4. Observation of the performance of the standardization instructor pilot (SIP) in order to define probable operational techniques.
5. Point-by-point analysis of aircraft energy states, handling qualities parameters, and test results.

During the OH-58A (Kiowa) H-V test program, the maximum H-V performance was established for two gross weights and three density altitudes. The results, shown in figure 4, indicate that the H-V profile varies in magnitude with gross-weight/density ratio. The data reflect a 2-second delay between throttle chop and initiation of pilot recovery, essentially utilizing uniform pilot technique. Figure 4 is considered by USAASTA to reflect "maximum" H-V performance for the OH-58A.

To derive operational H-V curves, the performance of two SIP's was observed. They were asked to perform school maneuvers as well as entries from unfamiliar speed and altitude combinations. Data were recorded during these flights, and typical SIP responses and performance were recorded. Additional flights were performed by the USAASTA experimental test pilot to apply SIP constraints and to derive an operationally realistic H-V curve (figure 5).

Similar H-V tests were performed on the AH-1G (Cobra). The H-V curves were obtained at different weights and altitudes and appear to generalize with gross-weight/density ratio (figure 6). The recommended operational H-V profile requires substantially higher airspeeds at the knee than does the present handbook curve (figure 7).

One consequence of the H-V testing is a recommendation to correct presently inadequate flight handbook data. The flight handbook narrative discussions of recommended pilot procedures are in particular need of revision for several aircraft. It is anticipated that inclusion of USAASTA data will result in improved training, pilot awareness, and operational safety.

Before leaving the subject of height-velocity, it is worthwhile to mention one-engine-inoperative (OEI) H-V tests. Historically, these tests have proven to be the most dangerous of the H-V programs – this is possibly due to:

1. Pilot overconfidence.
2. Engine response peculiarities.
3. Vortex ring state encounter.

The US Army Aviation Systems Test Activity has successfully performed OEI H-V testing on the CH-47B (Chinook) helicopter (reference 7). At the present time, plans are being finalized for the testing of the CH-47C (USAASTA Project No. 66-29) and the CH-54B (USAASTA Project No. 71-01) helicopters. It is intended to develop realistic operational envelopes suitable for operational Army aviator use for both of these aircraft.

Autorotational Entry

Early test experience as well as reports from operational users of the AH-1G helicopter indicated severe aircraft responses associated with engine power failure at high speeds. The USAASTA Project No. 70-25 testing was conducted to quantitatively evaluate the response of the AH-1G helicopter to simulated sudden engine failures (reference 8). Aircraft response to simulated sudden engine failure was characterized by rapid roll attitude changes and rapid main rotor rpm decay rates, both of which were unacceptable at the maximum engine torque settings. The severity of the aircraft response at a given speed is primarily a function of engine torque at the time of failure, as is seen in figures 8 and 9.

It was found that the AH-1G helicopter failed to comply with both the present and proposed military specifications for flying qualities of helicopters with regard to autorotational entry at high-airspeed/high-torque conditions. However, the present Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For,* does not fully or realistically prescribe a safe operating limit for this class of helicopter. Analysis of the test results produced a relationship between entry airspeed and engine torque, and the maximum delay time which can be tolerated prior to corrective pilot input (figure 10).

It was recommended that the operational envelope for the visual-flight-rule (VFR) daytime flight conditions for the AH-1G be limited to those combinations of engine torque and airspeed where the pilot recognition and reaction time is at least 1.5 seconds, and the bank attitude change would not exceed 40 degrees in 2 seconds. It was also recommended that the night and low-visibility operating envelope for the AH-1G be further limited to those conditions which would safely allow a 2-second control delay. Application of these recommendations should result in increased operational safety for the AH-1G.

Autorotational entry problems have also been reported by the Army Primary Helicopter School at Fort Wolters, Texas, on the TH-55A (Osage) helicopter. Abrupt pitching responses were being experienced in addition to the more classical yawing and rolling following a throttle chop. Several accidents had been attributed to this undesirable characteristic, and USAASTA was asked to investigate.

Very early in the test program, it became apparent that the pitching phenomenon was not a result of engine power failure/airframe dynamic response. Rather, it was an inherent static stability characteristic which was aggravated by stabilizer dihedral and weak aircraft directional stability. Because of the weak side-force characteristics, it was very easy to fly with as much as 15 degrees of right sideslip while the pilot thought he was in trimmed flight. The result was a significant change in tail effectiveness and angle of attack when the helicopter yawed outside the rotor wake following a throttle chop. This condition caused enormous changes in tail angle of attack (from -10 to +30 degrees) resulting in rapid nose-down pitching – an extremely unhealthy situation at low altitudes. The Fort Wolters training profile is predominantly flown at 500 feet. Time histories of this maneuver are shown in figure 11.

The US Army Aviation Systems Test Activity modified its test program to include development of a more suitable horizontal stabilizer for the TH-55A. It was determined that an upper surface spoiler would be of some benefit in reducing upload, and reduced area could be of further benefit. After several configurations, a proposed "fix" was finalized which is shown in figure 12. It is seen to incorporate both an upper surface spoiler and a 7-inch spanwise reduction.

To further cope with the weak inherent side-force characteristics, a sideslip indicator was also desired. Yaw strings proved ineffective because of the poor flow characteristics on the TH-55A canopy. Boom-mounted yaw vanes were rejected as excessively complex and expensive. Finally, Lieutenant Colonel William R. Benoit, the Director of Flight Test and project test pilot for the TH-55 at USAASTA, observed that the aircraft has a very shallow pedal position variation in airspeed in zero sideslip. Applying this observation, he devised a pedal position indicator which could be used to indicate proper pedal position versus power setting (figure 13). When used, the indicator provides a method of minimizing sideslip errors for all flight conditions.

Presently, both the modified TH-55A stabilizer and the pedal position indicator have been retrofitted to the TH-55A fleet at Fort Wolters, Texas. Initial reaction is enthusiastic, and considerable savings in personnel injury and aircraft damage are anticipated.

Maneuvering

The AH-1G has given the US Army greatly increased operational capability resulting from both high speed and maneuverability. However, several undesirable characteristics associated with maneuvering flight have been reported by operational users. One of these characteristics, inadequate altitude margins to recover from high-g pullouts at high airspeeds, was investigated during USAASTA AH-1G Phase D testing (reference 9).

It was originally suspected that the information presented in the operator's manual might be in error since it was computed on the basis of the simplified assumptions of a 1.0-second delay prior to normal load factor buildup and a constant normal load factor thereafter. However, test results showed that these simple assumptions predicted the actual aircraft performance accurately. Test data were obtained for a number of flight-path angles and are shown in figure 14. It is important to recognize that different assumptions will be required for articulated and rigid-rotor helicopters because of their inherently faster response times.

Another comprehensive maneuver test program was conducted on the AH-1G during USAASTA Project No. 69-11 (references 10 and 11). Three important areas were investigated:

1. Basic stability, control, performance, vibration, and structural loads characteristics of the AH-1G during controlled, steady-state maneuvers.

2. Kinematic relationships of the aircraft during maneuvering flight. Both steady-state maneuvers and free-form return-to-target tasks were included.

3. Potential problem areas peculiar to operational maneuvering flight, particularly in the areas of stability, vibration, loads, and total vehicle energy.

To determine maximum sustained maneuver capability, each pilot was asked to execute windup turns holding both altitude and airspeed constant while incrementally increasing bank angle for each new test point until maximum allowable engine torque was reached. Two gross weights (8,300 and 9,500 pounds) were studied to establish dependence of maneuvering parameters on the product of normal acceleration and gross weight, *ie*, main rotor thrust demand.

The pilots were next asked to exceed these maneuvering limits first by decreasing airspeed while holding altitude constant, and second by decreasing altitude while holding airspeed constant. The resulting load factor capability is shown in figure 15.

Equivalent power was calculated for all of the above data to determine if energy maneuverability is a valid concept for helicopters. Equivalent power is the sum of engine power and the rates of change of kinetic, potential, and rotor inertial energy.

Equivalent power data for the AH-1G during the above maneuvers are presented in figure 16. The following observations were made:

1. Equivalent main rotor thrust, equivalent power, and advance ratio appear to be valid scaling parameters for the family of data obtained.

2. Rotor stall, as reflected in the spacing of adjacent equivalent thrust increments, is quite gradual.

3. The characteristic trends of nonmaneuvering performance data continue to hold for maneuvering performance.

Another portion of the AH-1G maneuvering test program concentrated on return-to-target maneuvers. These maneuvers were flown by three pilots at each of five entry airspeeds, and the results are shown in figure 17. It was found that substantial variation in aircraft performance results from differing pilot technique.

As a result of the AH-1G maneuvering studies, a great deal of knowledge has been gained regarding rotary-wing maneuvering characteristics. A number of recommendations made in the previously referenced report (reference 11) are expected to improve operational safety and utility. Several of the technical concepts established as a result of this test are already being applied in the design of future US Army rotorcraft.

The UTTAS system specification contains new requirements for helicopter maneuverability. The US Army Aviation Systems Test Activity is presently evaluating (Project No. 71-32) the UTTAS terrain-avoidance maneuver (pull-up to achieve 1.75g's at 150 knots within 1.0 second, sustain 1.75g's for 3.0 seconds, pushover to achieve 0.0g, sustain 0.0g for 2.0 seconds) on three different aircraft. The OH-58A (teetering), OH-6A (articulated), and BO-105 (hingeless) helicopters will permit assessment of rotor system type on symmetrical maneuvering performance.

Vibration and Noise

Increasing importance has been assigned to the areas of helicopter vibration and noise. To date, very little has been done to require compliance with existing specifications in these areas, and US Army test programs have inadequately defined vibration and noise characteristics. The US Army Aviation Systems Test Activity is presently engaged in a significant test program (Project No. 70-15) to gather vibration data on all standard US Army helicopters. Participating with USAASTA in this effort are the Eustis Directorate of AMRDL and WECOM Weapons Laboratory, Rock Island, Illinois. Data are being obtained throughout the airframe with particular emphasis on areas and components with known high failure rates. Environmental temperatures are also being recorded. Tests on the OH-58A have been completed, and data are being analyzed at this time. High-frequency data (up to 2kHz) are being recorded using piezioelectric accelerometers, and FM/multiplex onboard tape recorders. The data are spectrally decomposed using real time analysis equipment.

A forerunner to USAASTA Project No. 70-15 was the 5.56/7.62mm weapons comparative evaluation on the OH-6A helicopter (reference 12). The objectives of that test were to qualitatively evaluate and compare the 5.56mm and the 7.62mm automatic guns for firing accuracy and the effect of the weapons firing on flight characteristics, and also to quantitatively evaluate instrument panel and gun-mount vibration levels, cockpit gun-noise levels, and gun-gas levels created by weapons firing. Only minor differences were noted in firing accuracy at ranges of 500 meters or less. A pronounced loss of accuracy was noted with the 5.56mm automatic gun at ranges of 1,000 meters and greater. Helicopter reactions were most apparent, for both weapons, during hover firing at a high rate of fire and were more severe with the 7.62mm automatic gun than with the 5.56mm gun.

Vibration levels of the instrument panel and gun mount were reduced significantly with installation of the 5.56mm automatic gun as opposed to the 7.62mm weapon. Significant reductions in instrument panel vibration and cockpit noise levels were also noted for both weapons when firing with the DOORS ON as compared to firing with the DOORS OFF. The lateral axis vibrations were most severe for the instrument panel and are shown in bar graph format in figure 18. Each bar graph was constructed by integrating the power spectral density for each test condition. A typical spectral plot is shown in figure 19. Significant vibration was noted at frequencies above 1kHz during firing tests.

The OH-6A test program also measured cockpit noise during firing. It was found that both weapons significantly exceeded the maximum noise exposure limits prescribed in MIL-A-8806A(ASG), for crews wearing helmets (figure 20). More disturbing was the finding that, even without firing, those maximum levels are exceeded.

A great deal of work remains to be done in the vibration and noise areas. It is clear that the relatively high levels of environmentally induced component failure and permanent hearing loss historically experienced by aircrews cannot be tolerated.

CONCLUSION

The purpose of this paper is to emphasize the existence of the US Army's engineering flight test facility, its resources and capabilities, and some significant results of helicopter flight testing to date. Greater safety, maintainability, and reliability are being assured through constantly improved flight testing techniques and the use of state-of-the-art instrumentation, data acquisition, and data reduction equipment. More stringent helicopter performance criteria are placing greater demands on the test agencies to devise new methods and procedures for collecting and analyzing data. Finally, the environmentally induced equipment failures attributed to structural vibrations and permanent loss of hearing experienced by aircrews are areas requiring additional work by the test agencies. A test program is currently in progress at USAASTA to more fully define the vibration characteristics of all standard US Army aircraft.

REFERENCES

1. Finnestead, Rodger L., et al., USAASTA PROJECT NO. 66-06, ENGINEERING FLIGHT TEST, AH-1G HELICOPTER (HUEYCOBRA), PHASE D, PART 2, PERFORMANCE, Edwards AFB, Ca., US Army Aviation Systems Test Activity, April 1970.

2. Maccabe, Richard S., VERTICAL FLIGHT PERFORMANCE CRITERIA, US Army Combat Development Command, Aviation Agency, January 1968.

3. Shapley, John J., Kyker, Robert A., and Ferrell, Kenneth A., THE DEVELOPMENT OF AN IMPROVED METHOD OF CONDUCTING HEIGHT-VELOCITY TESTING ON ROTARY WING AIRCRAFT, Edwards AFB, Ca., Presented at the 25th Annual National Forum of the American Helicopter Society, Washington, D.C., May 1969.

4. Pegg, Robert J., NASA TN D-4536, AN INVESTIGATION OF THE HELICOPTER HEIGHT-VELOCITY DIAGRAM SHOWING EFFECTS OF DENSITY ALTITUDE AND GROSS WEIGHT, National Aeronautics and Space Administration, May 1968.

5. Watts, Joseph C., Condon, Gregory W., and Pincavage, John V., USAASTA PROJECT NO. 69-16, HEIGHT-VELOCITY TEST, OH-58A HELICOPTER, Edwards AFB, Ca., US Army Aviation Systems Test Activity, June 1971.

6. Connor, William J. Jr., Condon, Gregory W., and Bailes, Edward E., USAASTA PROJECT NO. 69-13, HEIGHT-VELOCITY TEST, AH-1G HELICOPTER, Edwards AFB, Ca., US Army Aviation Systems Test Activity, February 1970.

7. Hoffman, Alan L., Davis, Wayne B., and Shapley, John J. Jr., USAASTA PROJECT NO. 68-02, CH-47B HEIGHT-VELOCITY EVALUATION, Edwards AFB, Ca., US Army Aviation Systems Test Activity, February 1970.

8. Buss, Marvin W., and Nicholson, Brock M., USAASTA PROJECT NO. 70-25, ENGINEERING FLIGHT TEST, AH-1G (HUEYCOBRA) HELICOPTER AUTOROTATIONAL ENTRY CHARACTERISTICS, Edwards AFB, Ca., US Army Aviation Systems Test Activity, March 1971.

9. Finnestead, Rodger L., et al., USAASTA PROJECT NO. 66-06, ENGINEERING FLIGHT TEST, AH-1G HELICOPTER (HUEYCOBRA), PHASE D, PART 2, PERFORMANCE ADDENDUM, Edwards AFB, Ca., US Army Aviation Systems Test Activity, March 1971.

10. Lewis, Richard B. II, HUEYCOBRA MANEUVERING INVESTIGATIONS, Edwards AFB, Ca., Presented at the 26th Annual National Forum of the American Helicopter Society, Washington, D.C., June 1971.

11. Lewis, Richard B. II, et al., USAASTA PROJECT NO. 69-11, ENGINEERING FLIGHT TEST, AH-1G HELICOPTER (HUEYCOBRA), MANEUVERING LIMITATIONS, Edwards AFB, Ca., US Army Aviation Systems Test Activity, March 1971.

12. Wing, John R., Diekmann, Vernon L., and Holasek, Ronald J., USAASTA PROJECT NO. 69-09, 5.56/7.62MM WEAPONS COMPARATIVE EVALUATION, OH-6A HELICOPTER, Edwards AFB, Ca., US Army Aviation Systems Test Activity, June 1971.

FIGURE 1

AH-1G DIRECTIONAL CONTROL MARGINS VS HOVER SKID HEIGHT

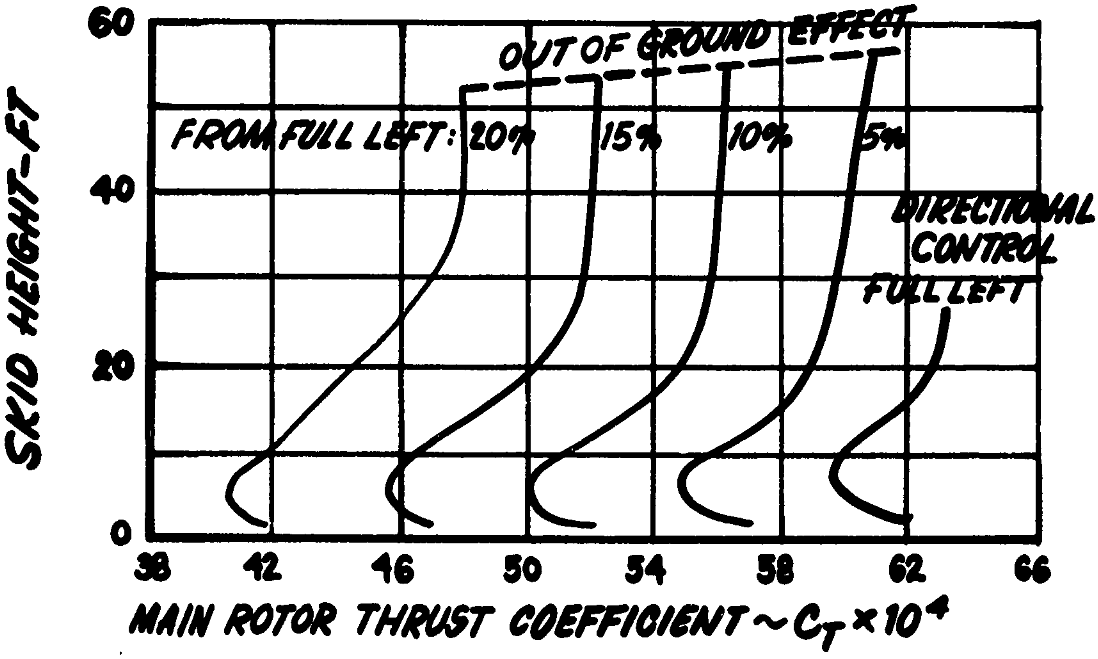


FIGURE 2

AH-1G DIRECTIONAL CONTROL MARGINS VS WIND AZIMUTH DURING HOVER

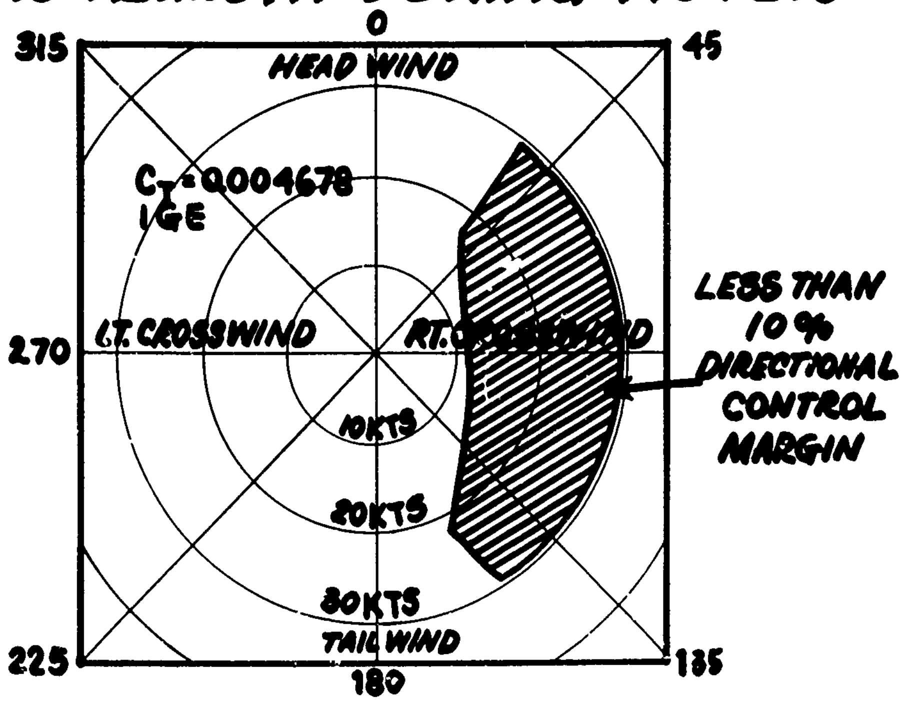


FIGURE 3

AH-1G IGE HOVERING PERFORMANCE

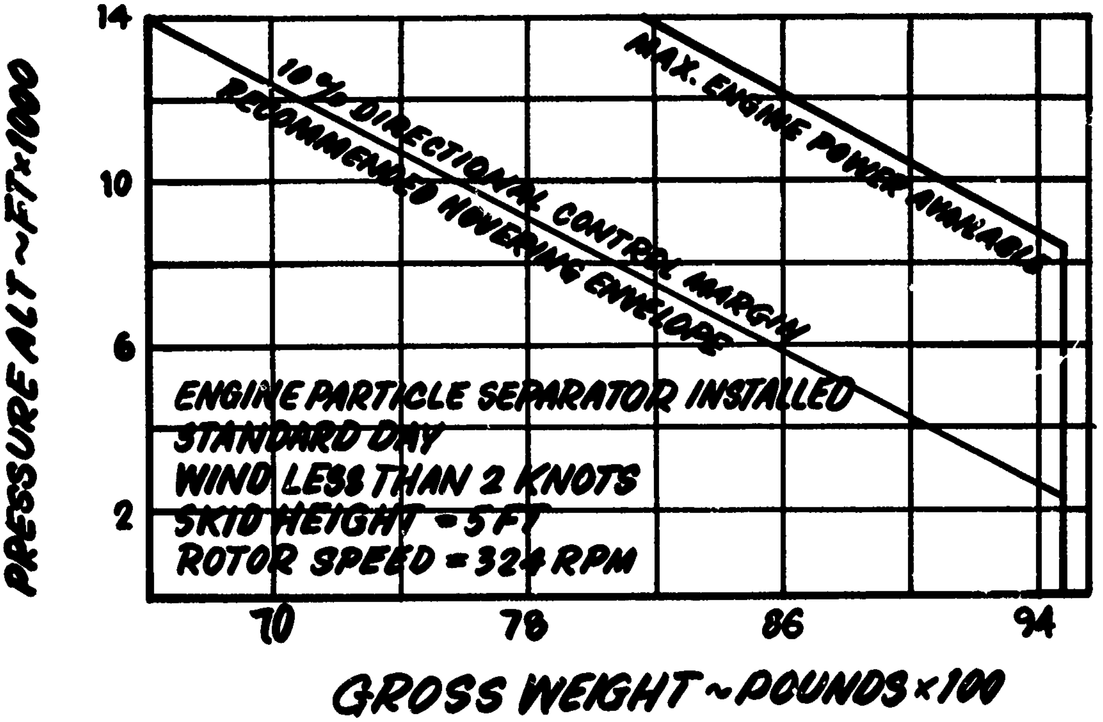


FIGURE 4

OH-58A MAX. PERFORMANCE HEIGHT VELOCITY PROFILES

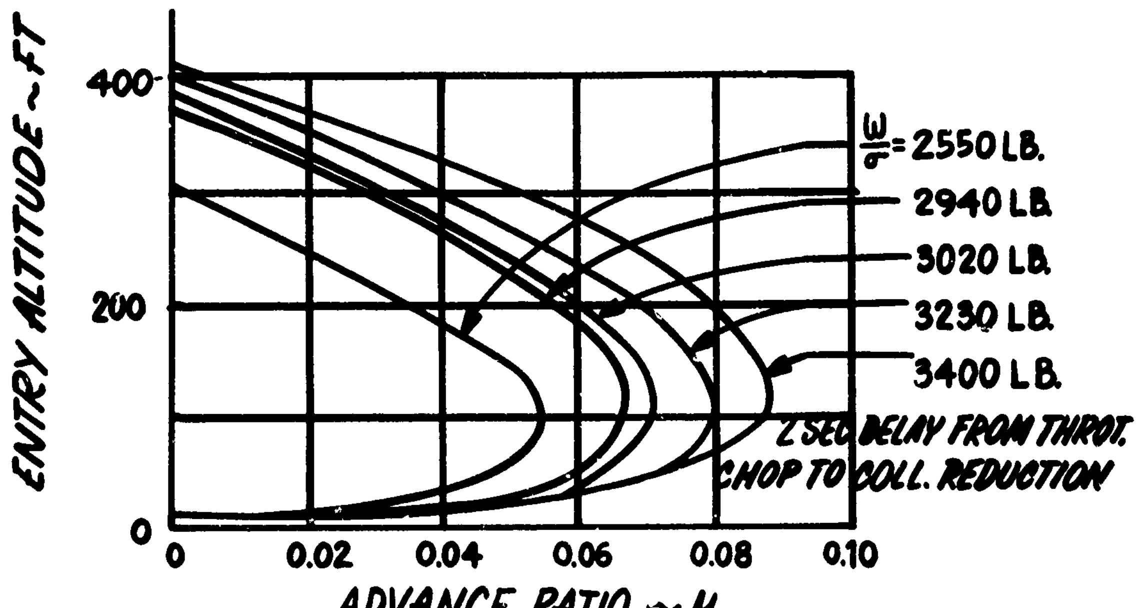


FIGURE 5

OH-58A RECOMMENDED OPN'L. HEIGHT VELOCITY PROFILE

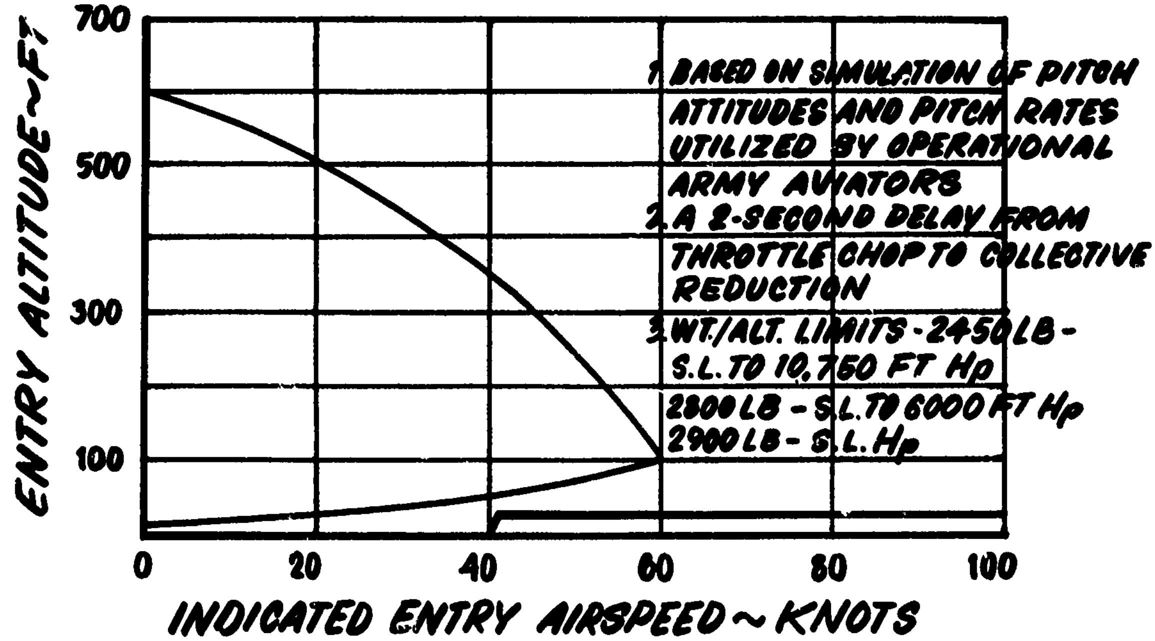


FIGURE 6

AH-1G MAXIMUM PERFORMANCE HEIGHT VELOCITY PROFILES

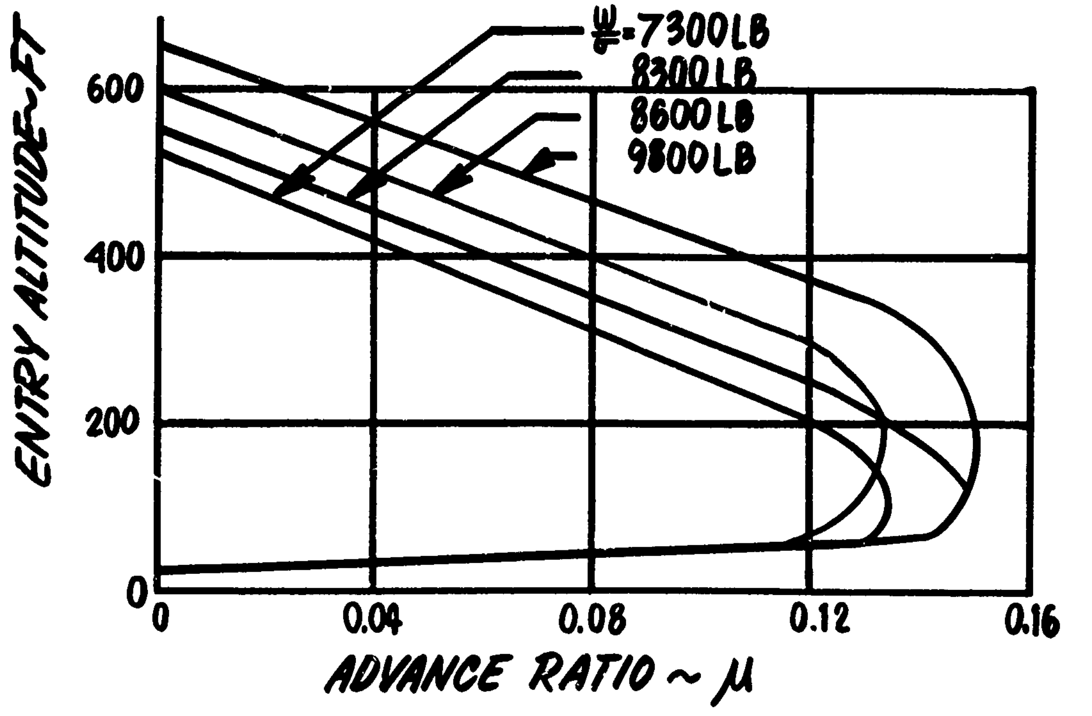


FIGURE 7

AH-1G RECOMMENDED OPERATIONAL HEIGHT-VELOCITY PROFILES

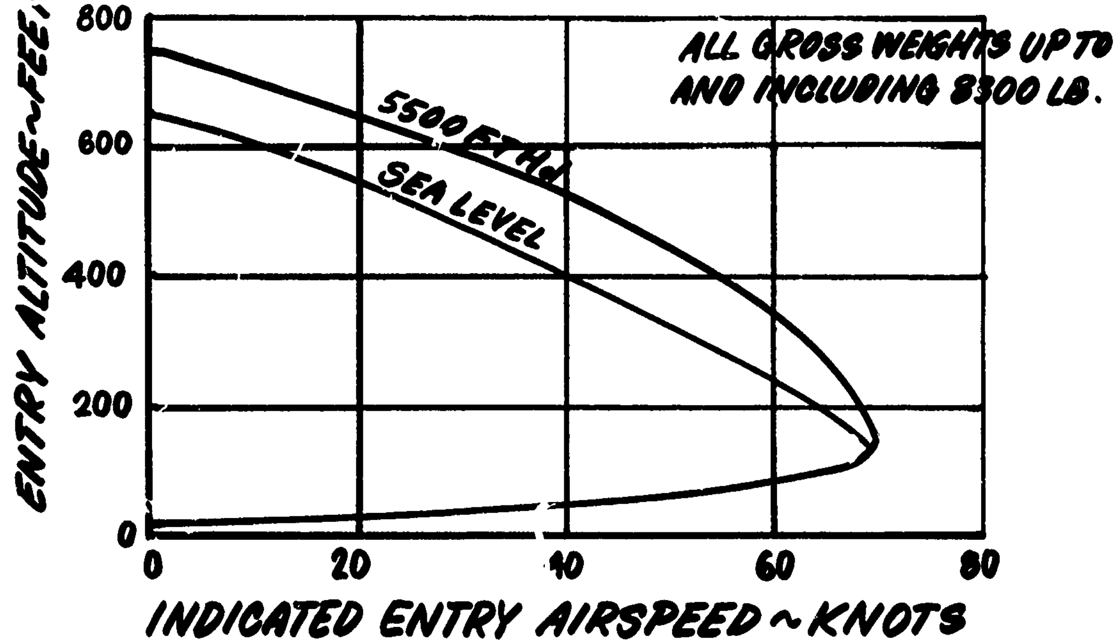


FIGURE 8

AH-1G MAXIMUM ROLL ATTITUDE VS ENTRY ENGINE TORQUE

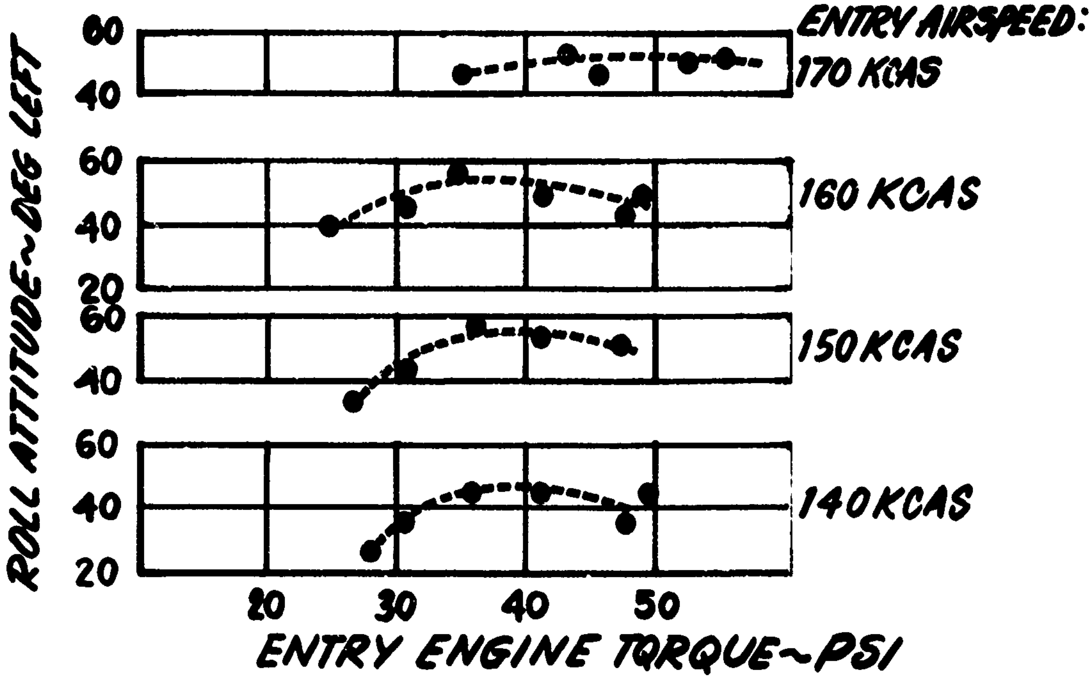


FIGURE 9

AH-1G ROTOR SPEED DECAY RATE Vs ENTRY ENGINE TORQUE

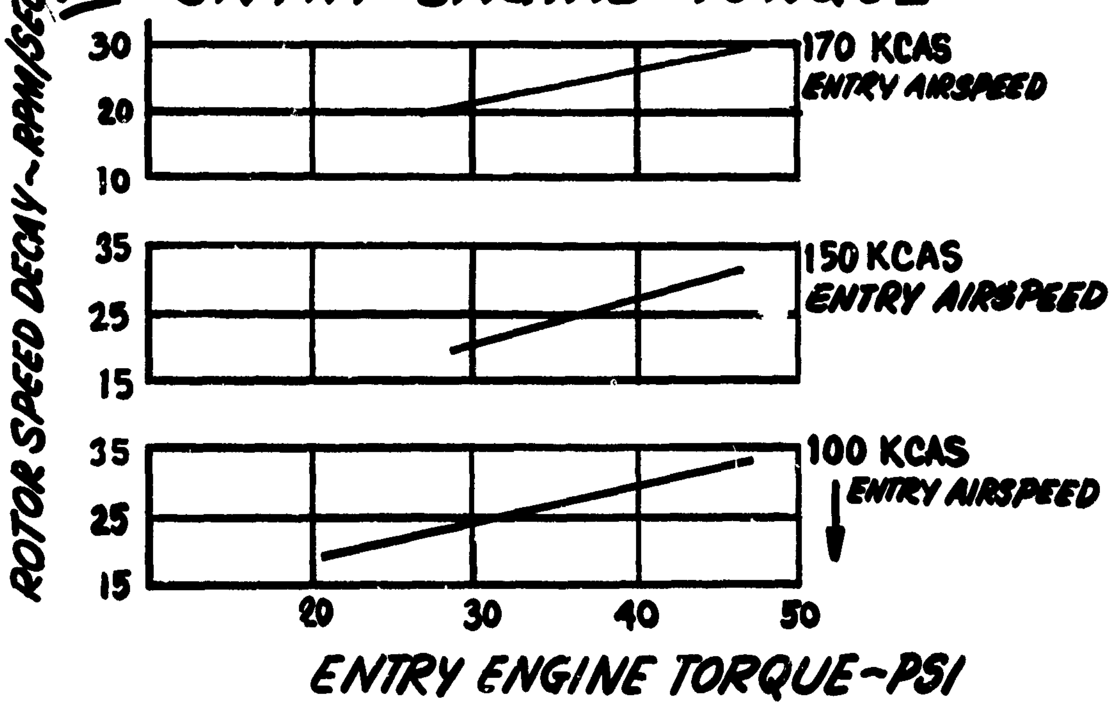


FIGURE 10

AH-1G PILOT DELAY TIMES Vs ENTRY ENGINE TORQUE & AIRSPEED

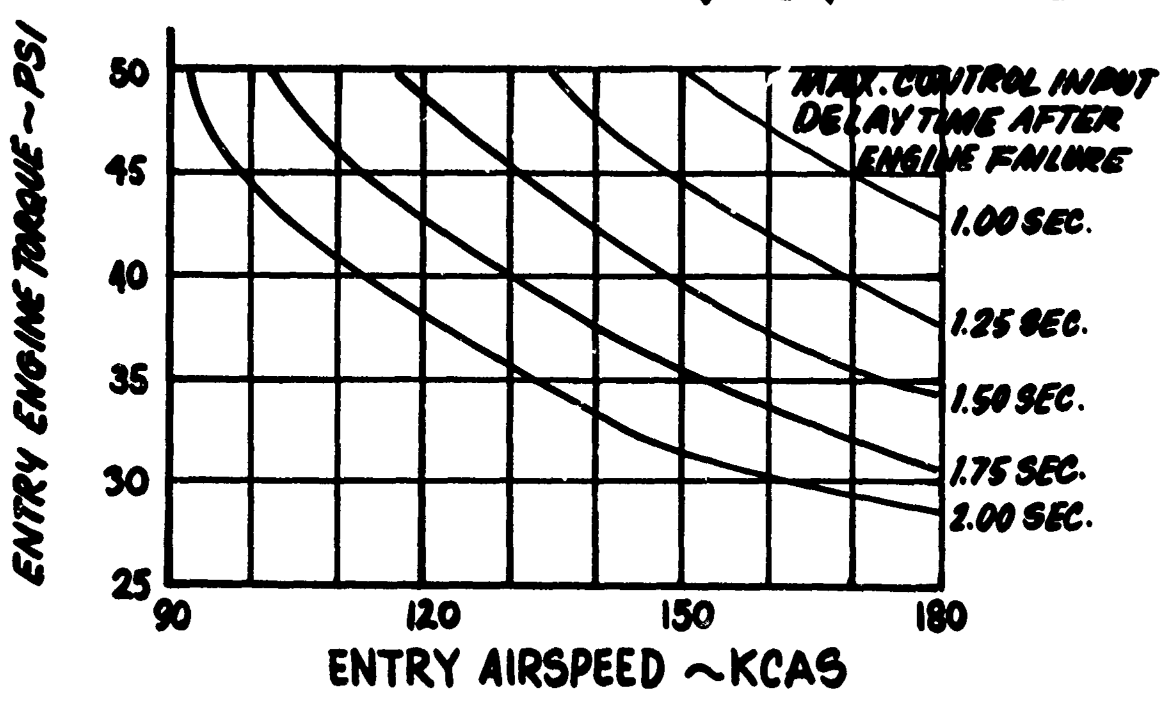


FIGURE 11

TH-55A AUTOROTATIONAL ENTRY CHARACTERISTICS

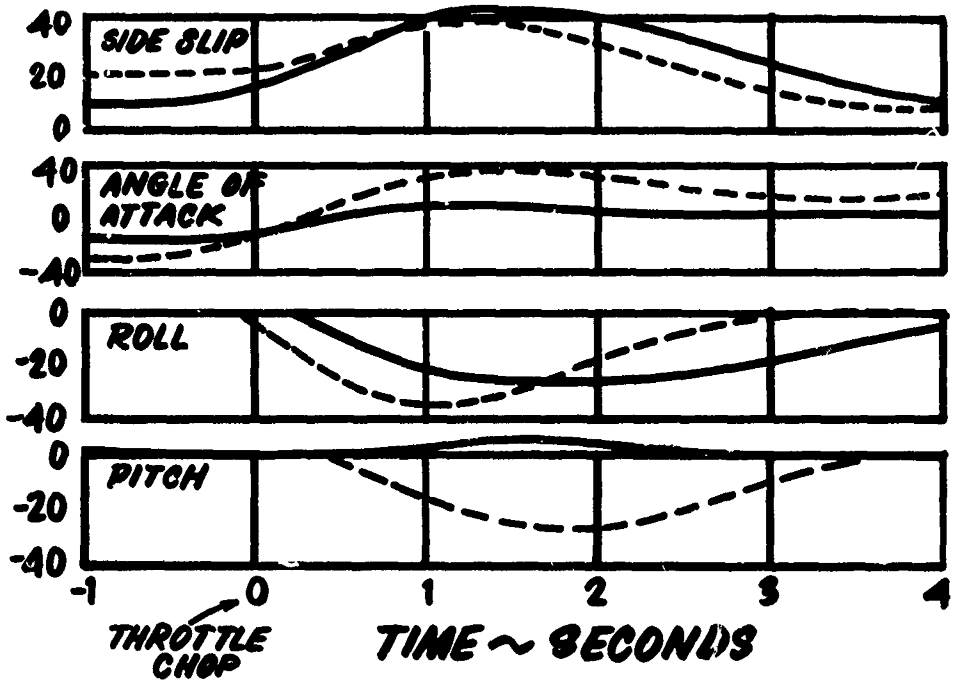


FIGURE 12

MODIFIED HORIZ. STABILIZER FOR TH-55A

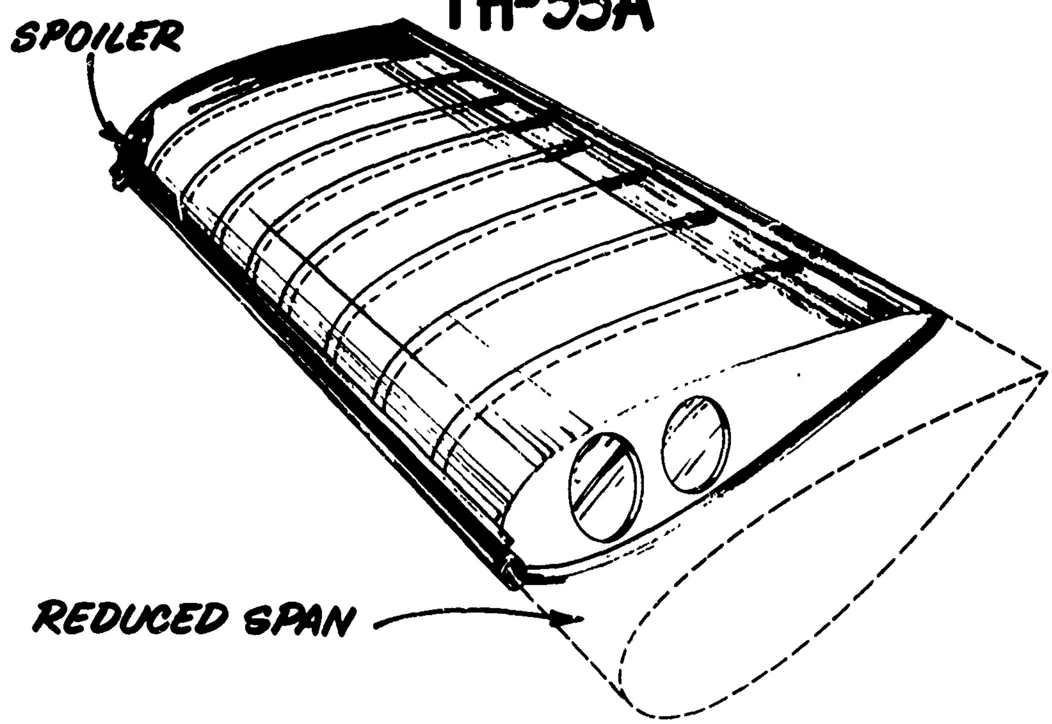


FIGURE 13

TH-55A PEDAL POSITION INDICATOR

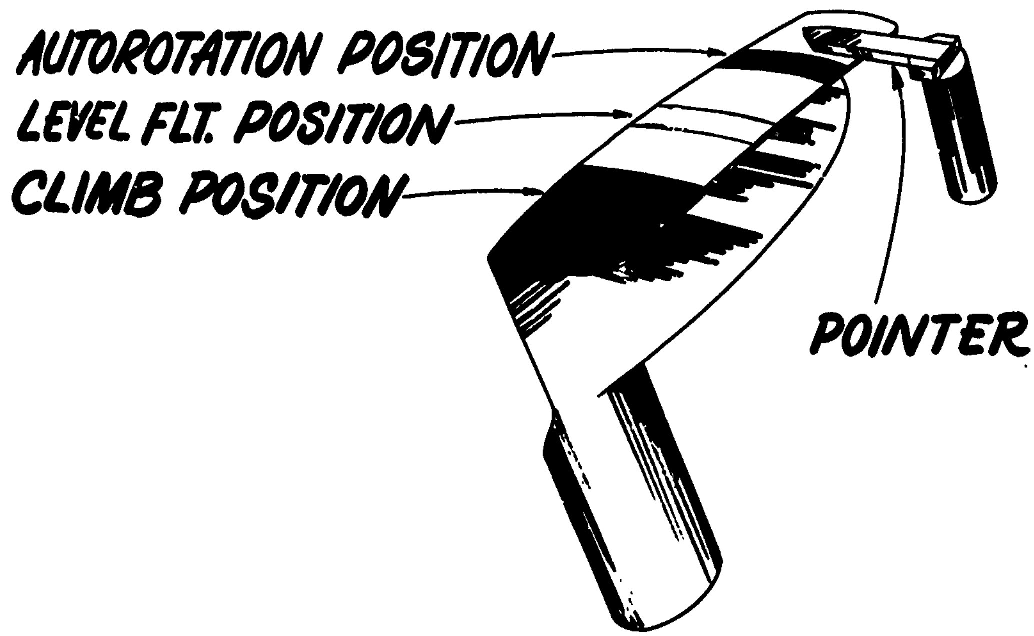


FIGURE 14

AH-1G ALTITUDE LOSS DURING RECOVERY FROM DIVE

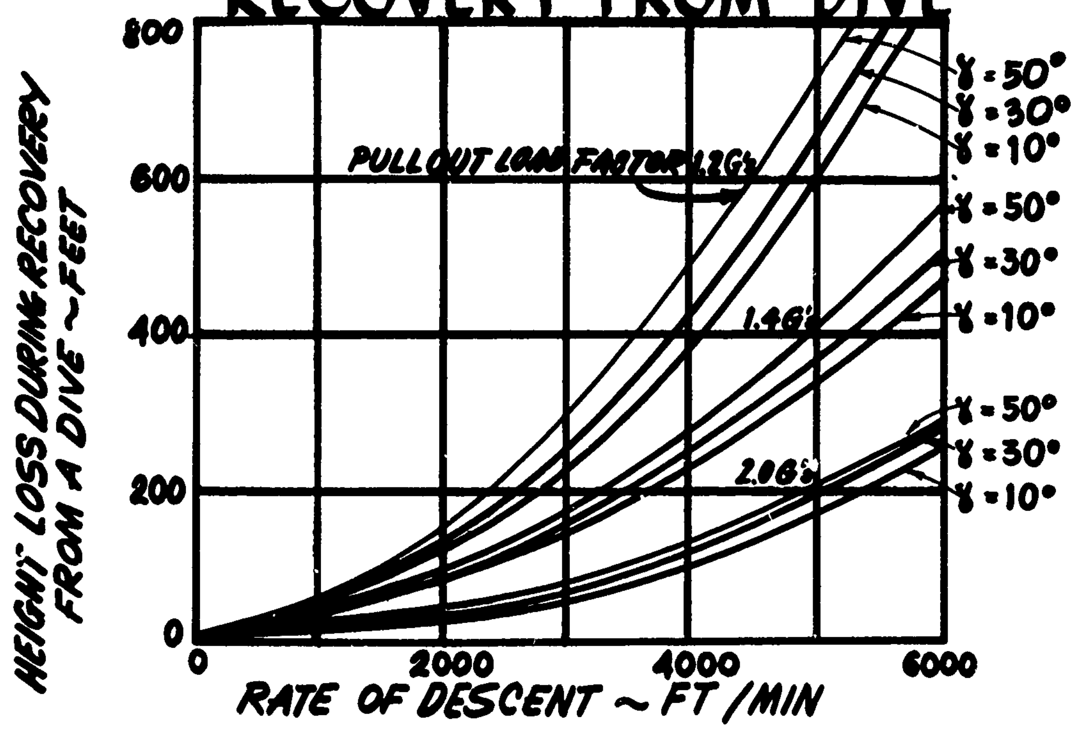


FIGURE 15
AH-1G V-n DIAGRAM

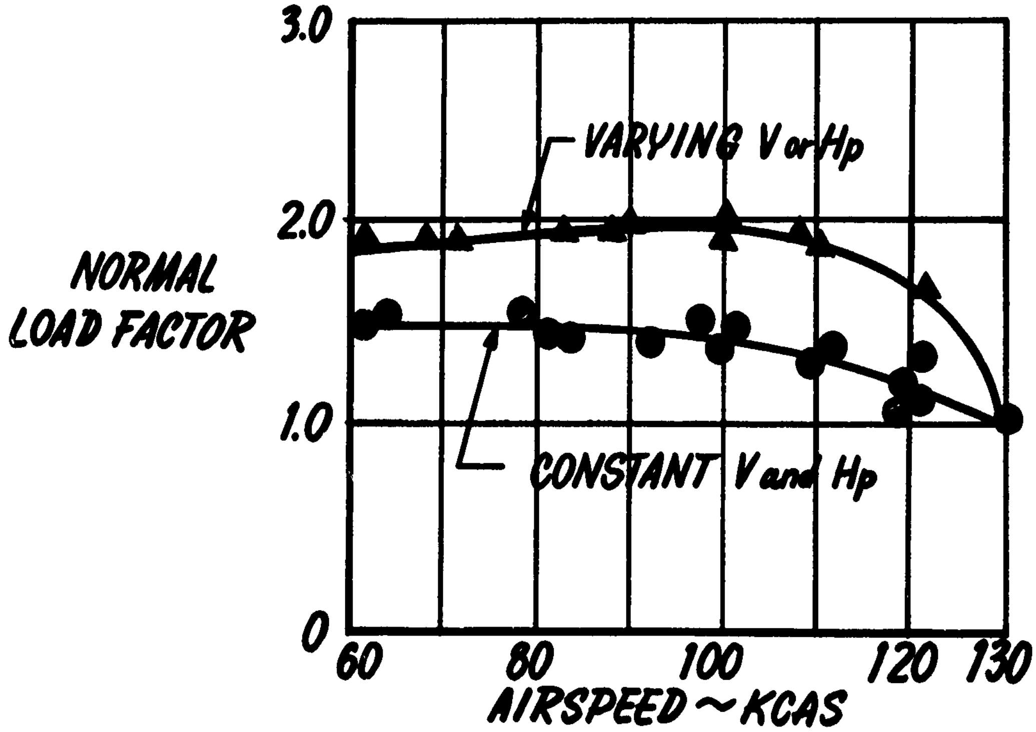


FIGURE 16
NONDIMENSIONAL MANEUVER CHARACTERISTICS

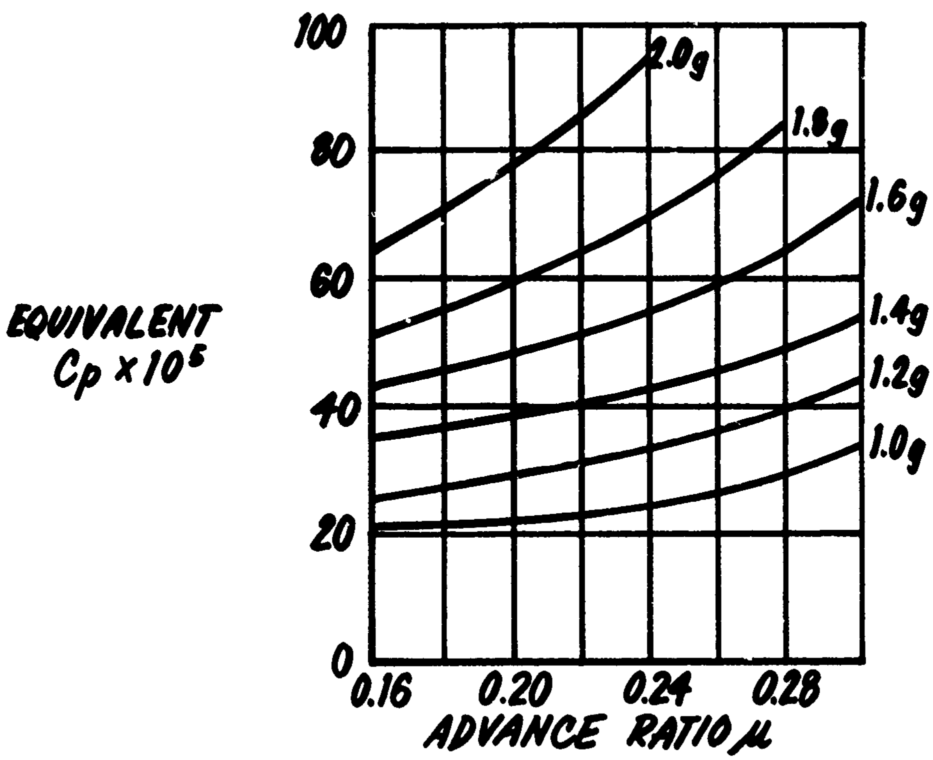


FIGURE 17

LEVEL TEARDROP TURNS

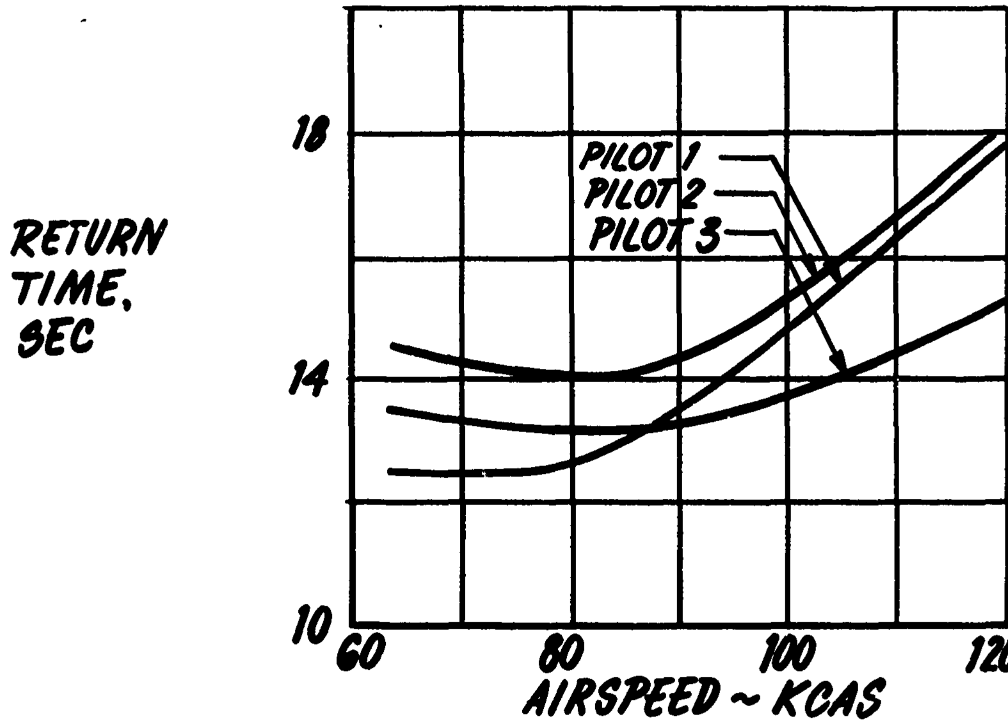


FIGURE 18

OH-6A LAT. AXIS INSTRUMENT PANEL VIBRATIONS DURING WPNS. FIRING

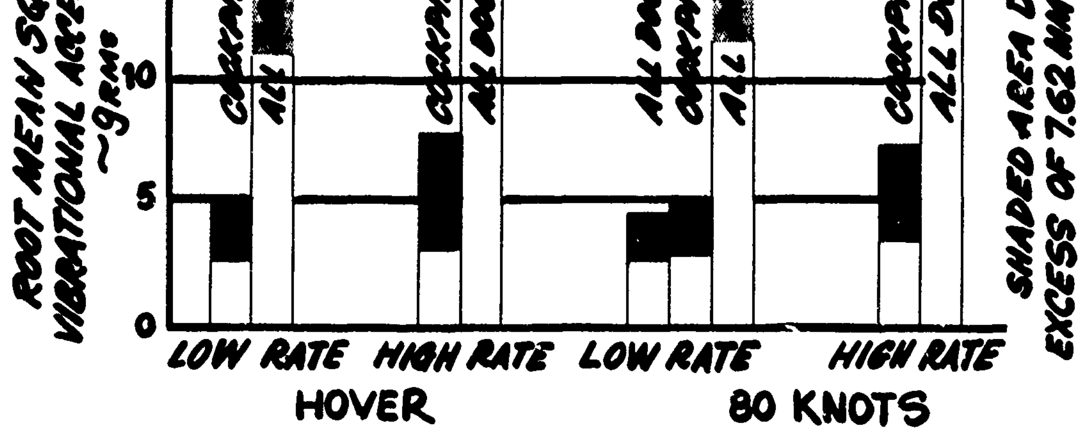
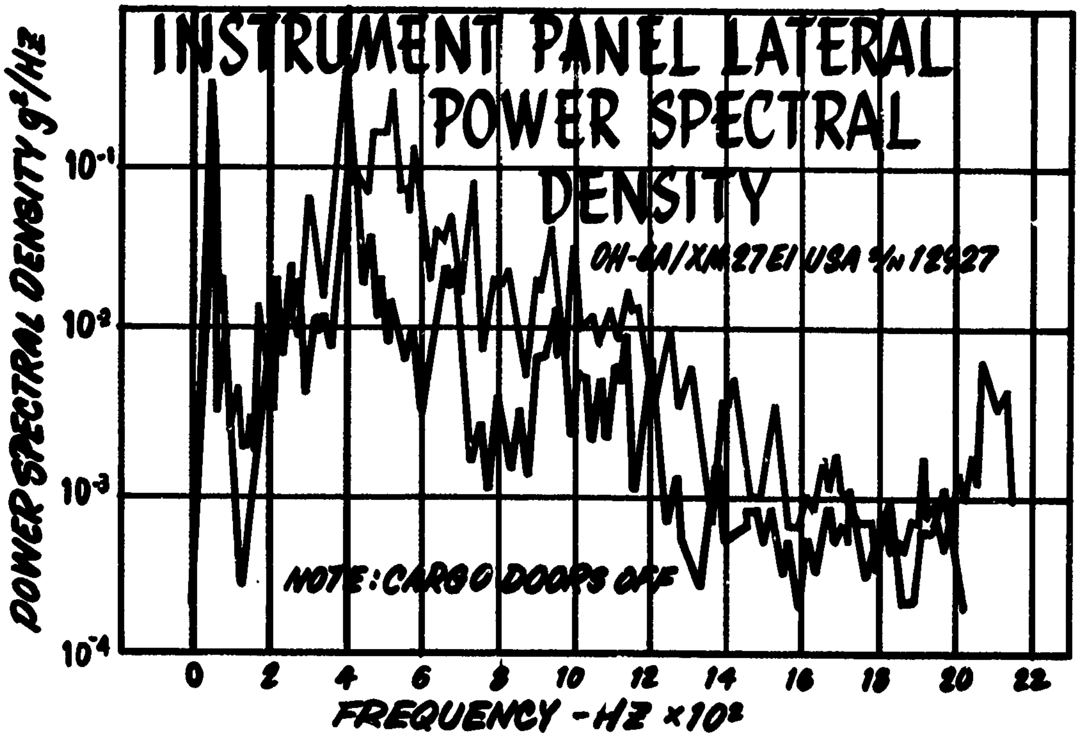


FIGURE 19

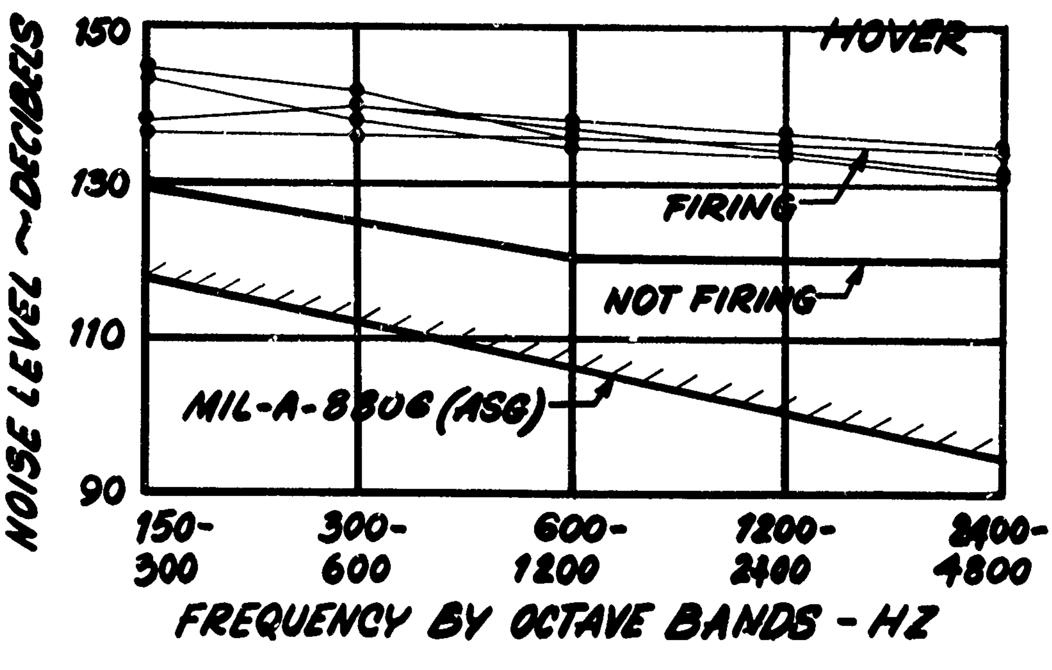


LEGEND	CONFIG.	RATE OF FIRE	AIRSPEED	ROTOR SPEED	GRWT	LONG C.G.	OAT	PRESSURE ALTITUDE
—	5.56 MM	2050 RPM	HOVER KCAS	480 rpm	2290 lb.	98.2"	33.3°C	2210'
—	7.62 MM	1950	HOVER	480 rpm	2340 lb.	97.8"	19.0°C	3400'

FIGURE 20

OH-6A COCKPIT NOISE LEVEL

COCKPIT DOORS ON - CARGO DOORS OFF



A NEW LOOK AT HELICOPTER LEVEL FLIGHT PERFORMANCE

by

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SUMMARY

Performance engineers have been ignoring test data in the area of the classic helicopter level flight power required curve just above the "bucket" from μ = 0.15 to 0.25. These data have shown a discontinuity or "ripple," but unfortunately the data analysts have elected to disregard the data and fair the curves into the shape of the familiar power required curve. The arguments supporting this practice have ranged from poor pilot technique to normal data scatter; in all cases, the rationale was backed by classical level flight performance theory and by universal acceptance.

None of these arguments, however, can withstand the rebuttal of repeatable and voluminous flight test data acquired from a full spectrum of helicopter rotor system designs. Also, the concept of the "ripple" in this area is feasible when the results of W. H. Tanner's wake flow studies are applied. These studies are examined and are compared with flight test data. Original test data acquired at the Naval Air Test Center (NATC) during the past five years are presented to illustrate the existence and magnitude of the "ripple." Data acquired from rotor systems of the two-bladed teetering, five and six-bladed articulated, four bladed servo flap, and three-bladed articulated tandem designs are included. This covers the H-1, H-2, H-3, H-46, H-53 and H-57 type helicopters. In addition, a special performance evaluation of the HH-3A helicopter was conducted at NATC to determine the existence of the "ripple." Again the test results were faired in the classic helicopter level flight power required curve.

The author has observed mixed reactions to this concept during discussions with experienced flight test engineers and data analysts throughout industry and the service Test Centers. Most agree that the trend is shown by the data; only a few have accepted its validity and still fewer have readjusted their curves and published the results. This paper is intended to serve a two-fold purpose: to lead the helicopter flight test community into acceptance of the "ripple" phenomenon; and to encourage publication of supporting data which have lain dormant throughout the years because of a low confidence level.

INTRODUCTION

Helicopter level flight performance data are presented as Power Coefficient (Cp) versus Tip Speed or Advance Ratio for a range of Thrust Coefficients (C_T). This data presentation was developed from momentum and blade element theory. The power coefficient is a non-dimensional expression for the main rotor shaft horsepower required. The main rotor shaft horsepower required consists of profile, parasite and induced power. The advance or tip speed ratio is a non-dimensional ratio of flight speed and main rotor rotational speed. The thrust coefficient is a non-dimensional expression for thrust required. It should probably be called the weight coefficient, since vertical drag is normally ignored, and thrust is replaced by gross weight. A representative classic helicopter level flight performance curve is presented in figure 1. Figure 2 is a specific range curve used in conjunction with the data from figure 1.

THE "RIPPLE"

Some years ago the author noticed what appeared to be a "ripple" in the flight test level flight power required curves just above the "bucket" in the μ = 0.15 to 0.25 range. When the question of the ripple was presented to senior flight test and performance engineers it was explained as normal data scatter and poor pilot technique. At the time this seemed reasonable since the theory did not support such a "ripple." The smooth data fairing as shown in figure 1 was universally accepted. This explanation was satisfactory until more and more data were reviewed by the author. Most of these data showed evidence of a discontinuity or "ripple." The "ripple" could not be explained so it was called scatter or poor pilot technique. It seemed more reasonable to the author to attribute the "ripple" to some instability in the flow through the main rotor.

During the Navy Evaluation of the Bell Helicopter Company HueyTug helicopter for the Navy VERTREP mission the "ripple" was found again. It was noted that while accelerating from a hover to cruise speed a directional instability was encountered at about 55 kt. The aircraft would oscillate directionally ±15 degrees. This was due to the main rotor vortex blowing back through the tail rotor. Figure 3 shows the level flight speed power as published in NATC Technical Report FT-1R-70. The "ripple" appeared to be less pronounced as the C_T increased. This was attributed to the stronger vortex at the higher gross weight. The photograph of figure 4 shows the S64 Skycrane with the trailing vortex pattern visible due to water vapor. The vortex shed by one blade tends to go below the following blade and intersect the next or second blade back. Shortly after the publication of the HueyTug report, Mr. Robert Lynn, Chief of Research and Development at Bell Helicopter, delivered a paper to the NATC Patuxent River section of the American Helicopter Society. Mr. Lynn discussed work being done by Mr. W. Tanner in rotor performance with the shed vortex taken into account. Mr. Tanner had used a single rotor blade with N trailing vortices and an iteration process in his development. He found that his program gave him a "ripple" in the level flight performance curve. Mr. Lynn took some of the NATC data back to Bell and ran it through the computer program (BRAM with vortex). The results are as shown in figure 5. The ripples appeared in the same general locations in both the flight test and theoretical data. The lower C_T agreed rather well with flight test data. As the C_T increased the curves agreed less in Cp but the "ripple" was still in the μ range. Figures 6 through 11 show NATC flight test data with curves faired to show the "ripple." Figures 6 and 7 are for two-bladed simi rigid teetering rotor systems. The FH-1100 maximum endurance speed decreases by about 9% (figure 6) and the TH-57A shows an 8% decrease. Figure 8 shows the five-bladed fully articulated SH-3 in the normal configuration. The data indicate a 12% decrease in maximum endurance speed. Figure 9 shows the six-bladed fully articulated CH-53A which shows a decrease of about 4%. The curves of figure 10 show the tandem three-bladed fully articulated CH-46D. This particular set of data is interesting in that it tends to have a second "ripple" in the area of μ = 0.275. The author has assumed that the second "ripple" is due to the overlap of the rotors. The data indicate that maximum endurance speed is decreased by about 7%. Figure 11 shows the AH-1J in the Hog configuration, again a two-bladed semi rigid rotor helicopter with a 3% decrease in endurance speed. Note that the external configuration is a "dirty" one. The two-bladed semi rigid teetering rotor system appears to display the "ripple" more than any of the other rotor systems. Figures 12 and 13 show the two-bladed AH-1G and UH-1N level flight curves with 7 and 10% decreases in endurance speed. Figure 14 is a comparison of three different types of rotor systems. The four bladed servo flap rotor of UH-2C is shown at the top. The tandem rotor CH-46 is in the center and the two-bladed teetering rotor of the HueyTug is on the bottom.

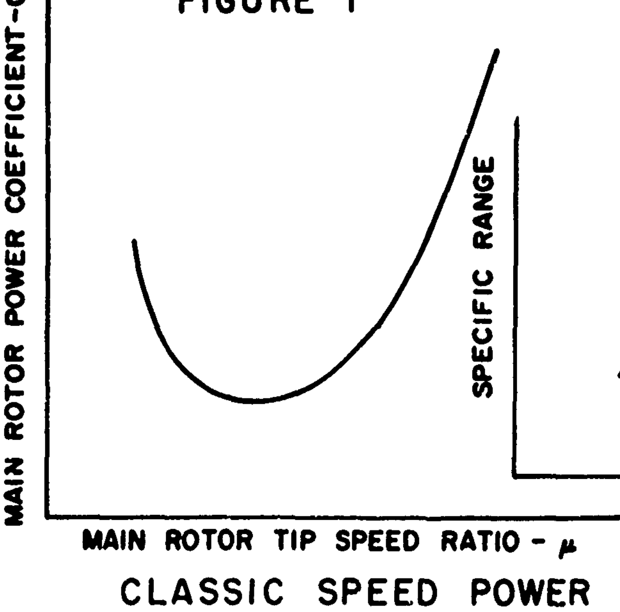
What effect does the "ripple" have on the performance data fairings? In general there is a reduction in the maximum endurance airspeed and an increase in the maximum specific range airspeed. The shift in maximum endurance speed is up to ten kt, while in some cases there does not appear to be any significant shift at all. Of course any shift in the speed power curve causes a resulting shift in the specific range curve of the same relative magnitude. The magnitude of the change in maximum endurance speed is of little consequence. The main rotor power required with and without the "ripple" are essentially the same. Maximum range data shows little or no change in power requirement. The speed change could be significant if it were not for the one percent power reduction taken in the determination of maximum range speed. The "ripple" location and shape tend to be supported by the difficulty encountered in stabilizing on speeds just above the "bucket." This difficulty is encountered in a speed range where the instrument approach is generally flown.

A recent NATC evaluation of the HH-3A helicopter showed the "ripple" in the speed power curve. The aircraft manufacturer did not agree and a special performance program was conducted to resolve the differences. The project team assigned to this program followed a procedure outlined in NATC Report of Test Results FT-21R-71 dated 24 March 1971. A constant ratio of gross weight to pressure ratio C_W/δ was maintained. This requires the rotor speed to temperature ratio $(NR/\sqrt{\theta})$ also be a constant. The advancing blade tip Mach number and tip speed ratio is held to a linear relationship. The speed po- curves from this evaluation are presented as figures 15 and 16. Three flights were conducted in each configuration with the same target C . The data from each configuration were grouped together to form one curve for that configuration. The test project team concluded that the "ripple" did not exist. The author does not agree. If the data from figures 15 and 16 are replotted using the points that fall along constant C_T lines as shown in figures 17 and 18 the "ripple" does appear. Note that figure 17 shows little or no change in the maximum endurance speed. Figure 18 indicates an 8% change in endurance speed. This is attributed to the Armed SAR configuration being a very "dirty" one. It is hoped that additional testing will be done in this area.

The author has discussed the "ripple" with experienced flight test engineers and data analysts throughout industry and the service Test Centers. Most agree that the data show the existence of the "ripple." There is some evidence of a similar "ripple" in the maximum rate of climb and minimum rate of descent curves obtained from sawtooth climbs and descents.

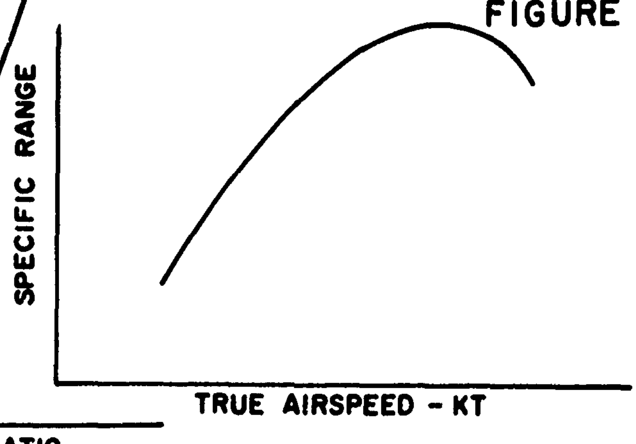
This paper is intended to serve a two-fold purpose: to lead the helicopter flight test community into acceptance of the "ripple" phenomenon; and to encourage publication of supporting data which have lain dormant throughout the years because of a low confidence level. Don't be lead to believe that all pilots develop poor data gathering techniques at the same place in the speed power curve on every performance flight.

FIGURE 1



CLASSIC SPEED POWER

FIGURE 2



CLASSIC SPECIFIC RANGE

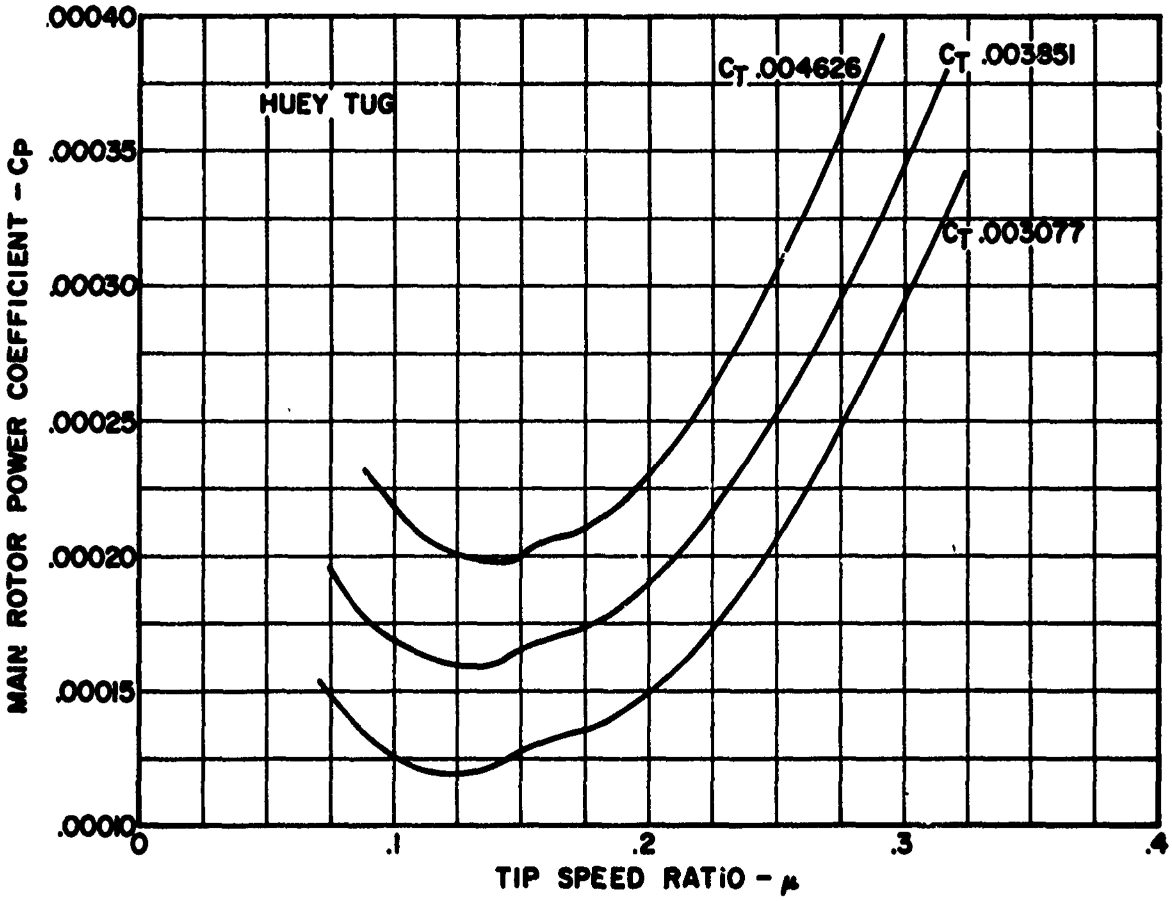


FIGURE 3

FIGURE 4 S-64 SKYCRANE

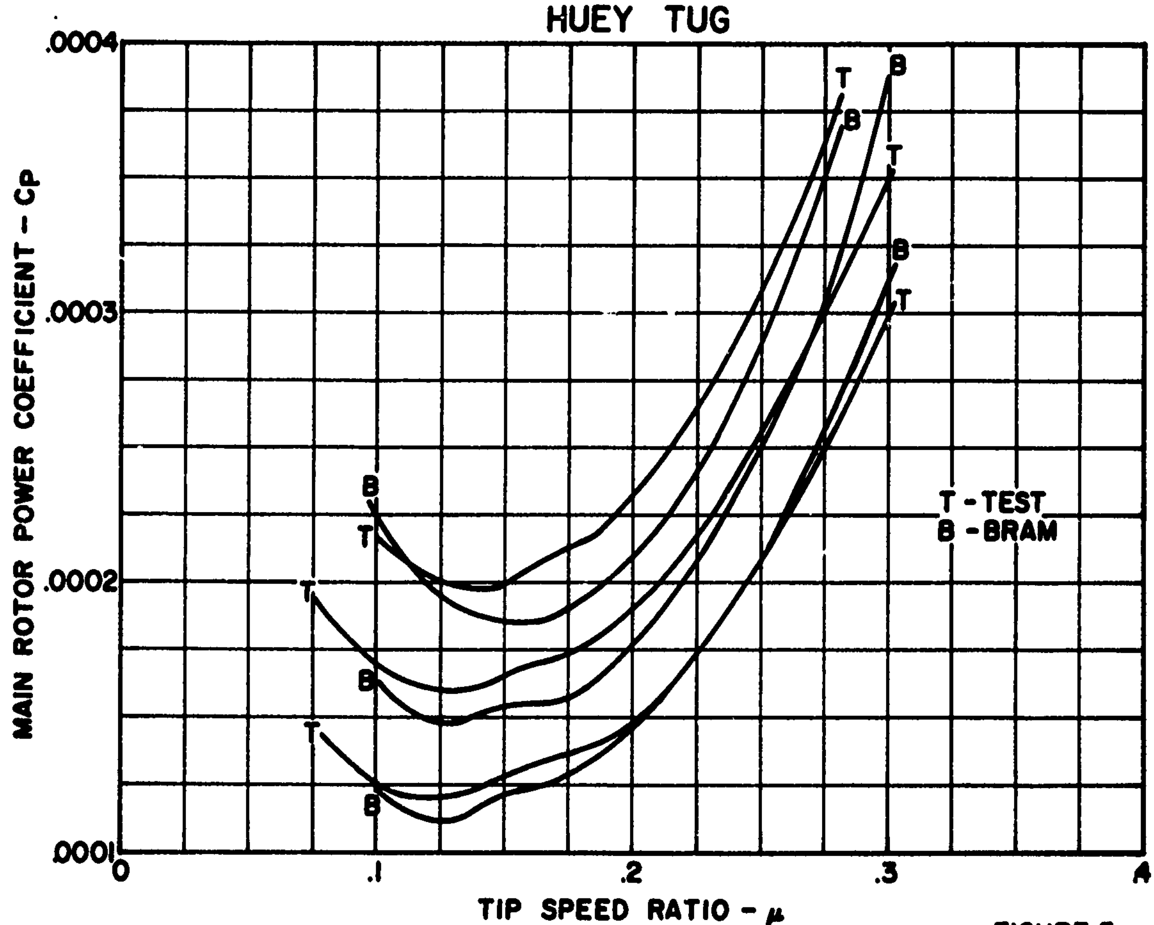


FIGURE 5

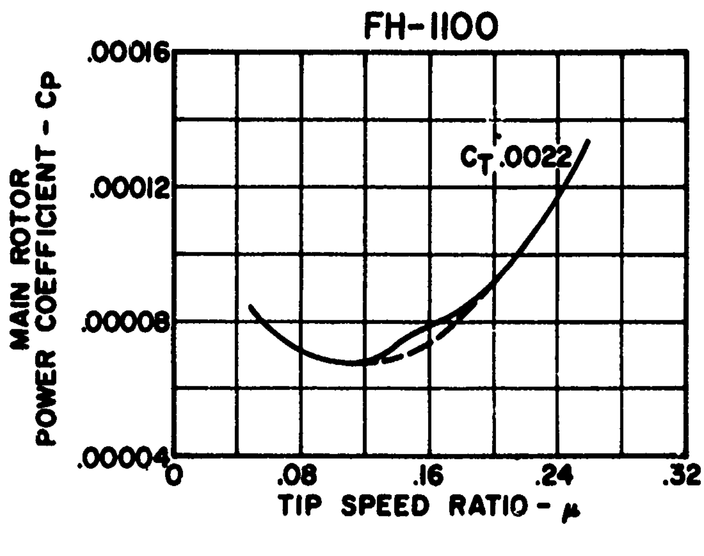


FIGURE 6

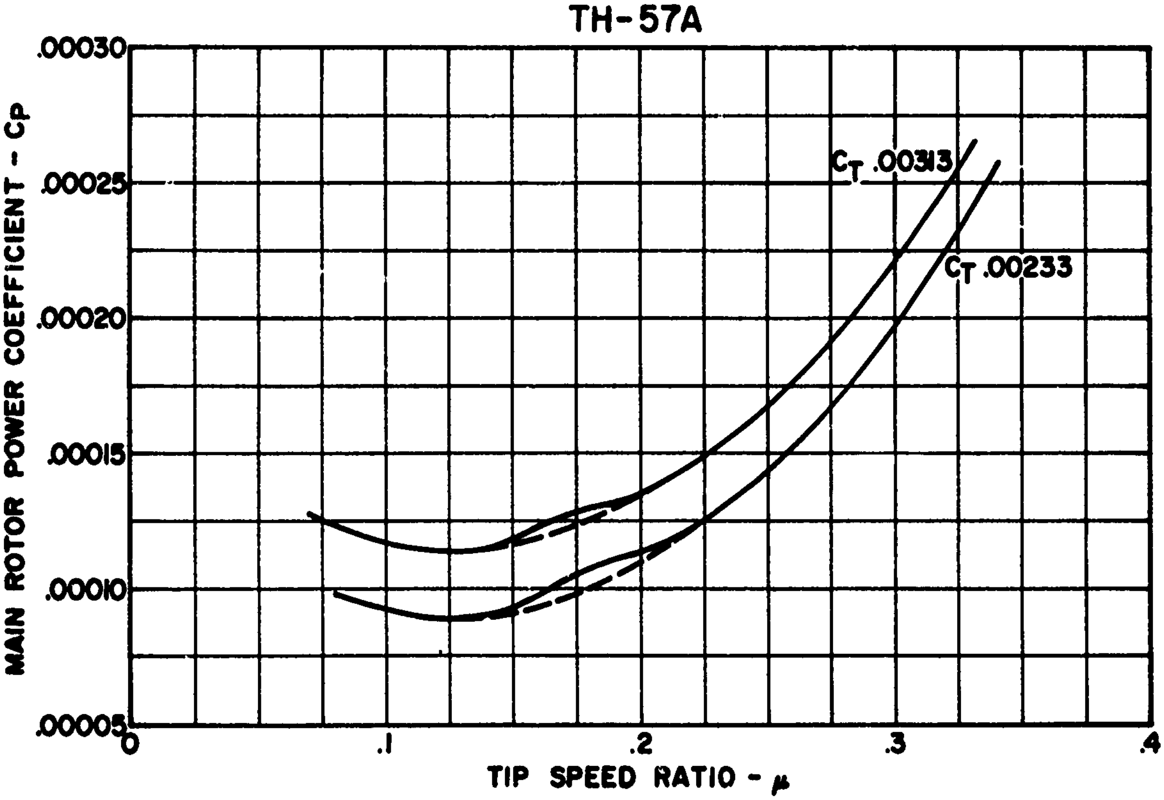


FIGURE 7

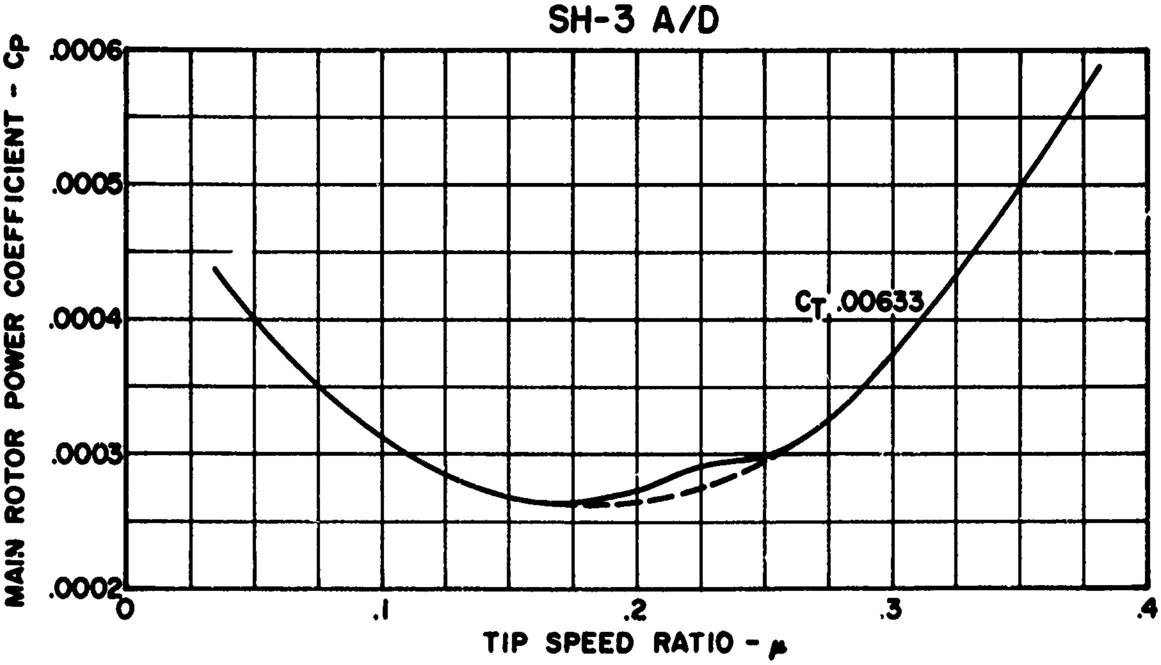


FIGURE 8

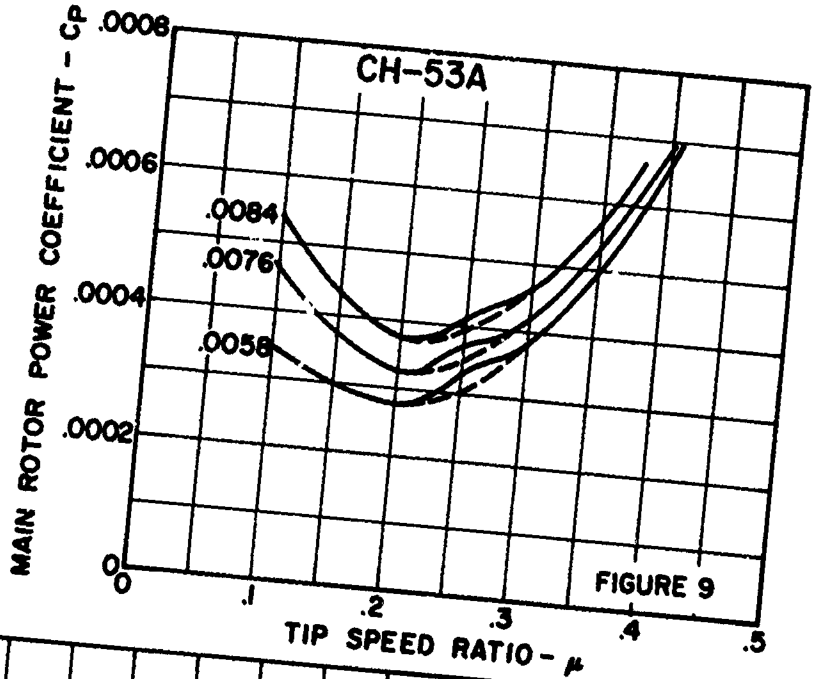


FIGURE 9

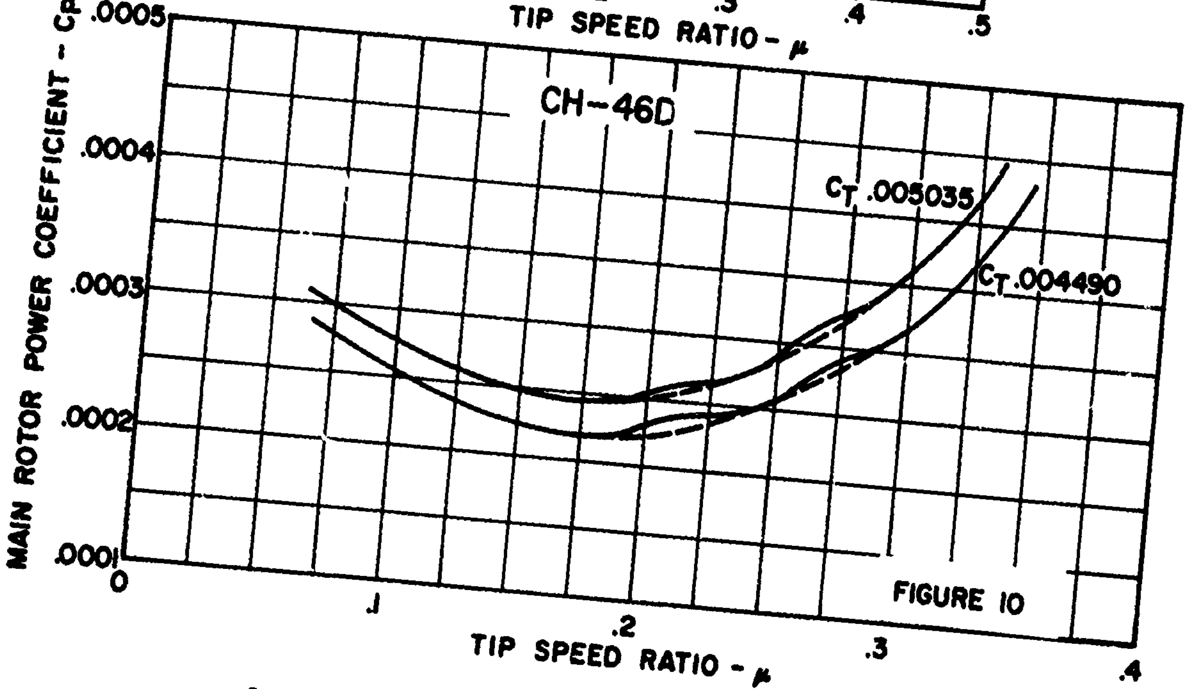


FIGURE 10

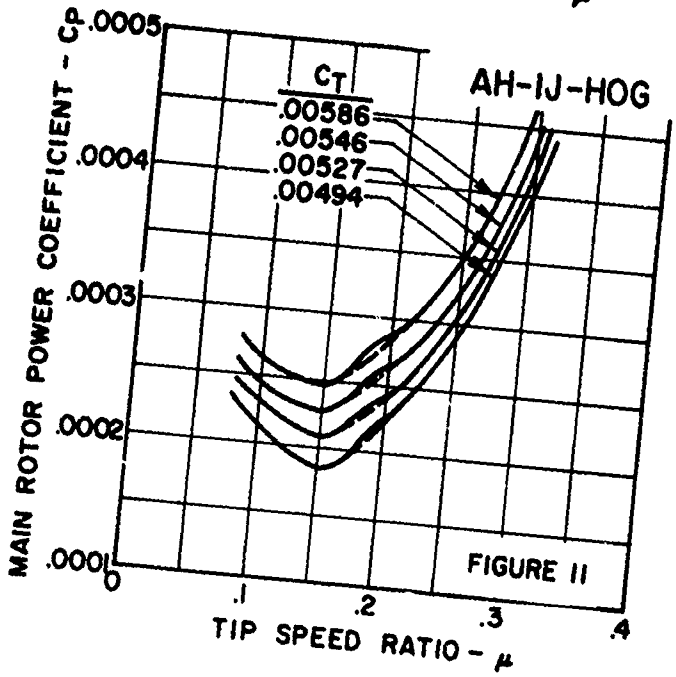


FIGURE 11

AH-1G

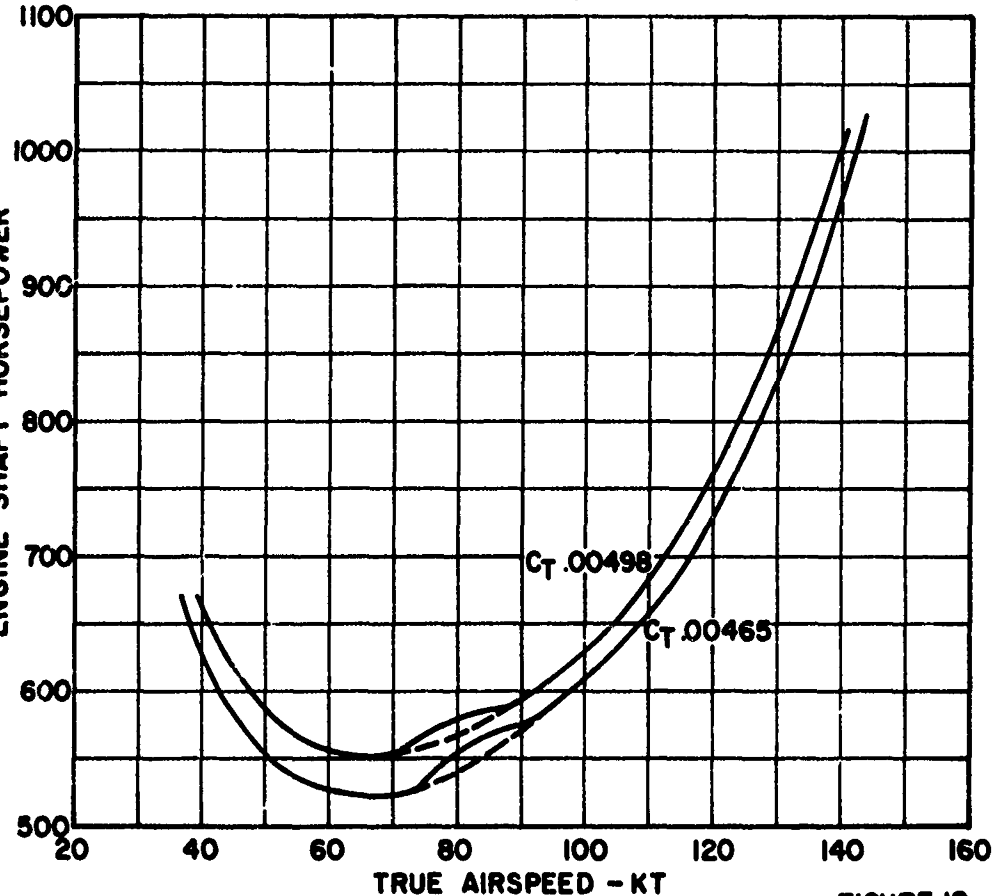


FIGURE 12

YUH-1H

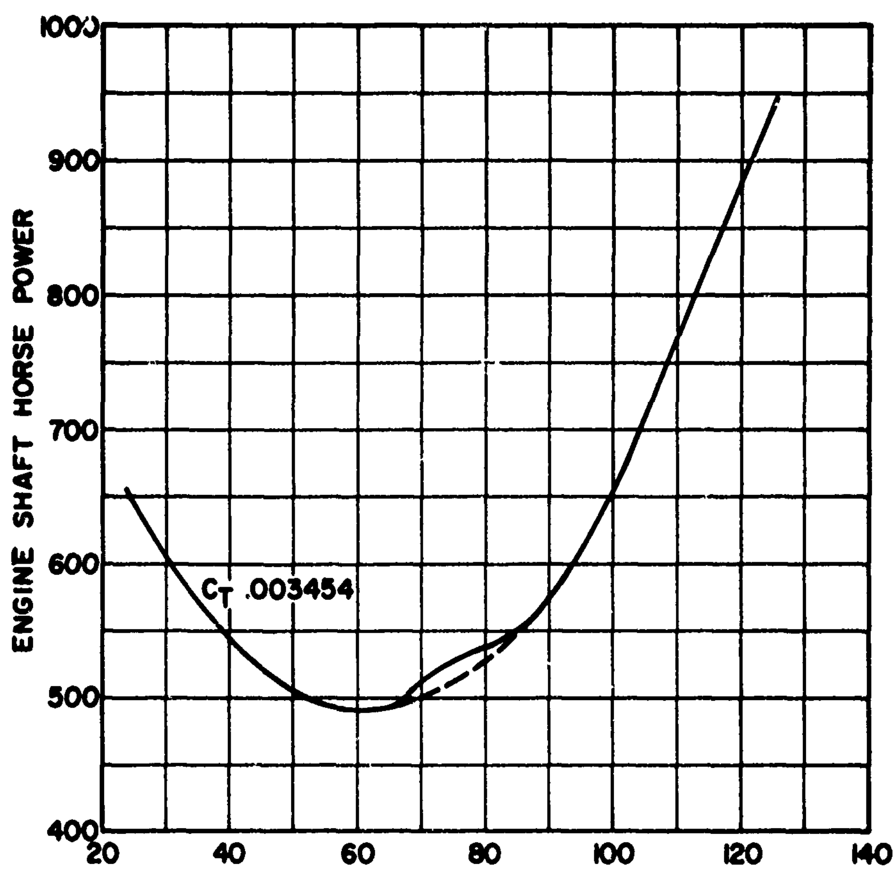


FIGURE 13

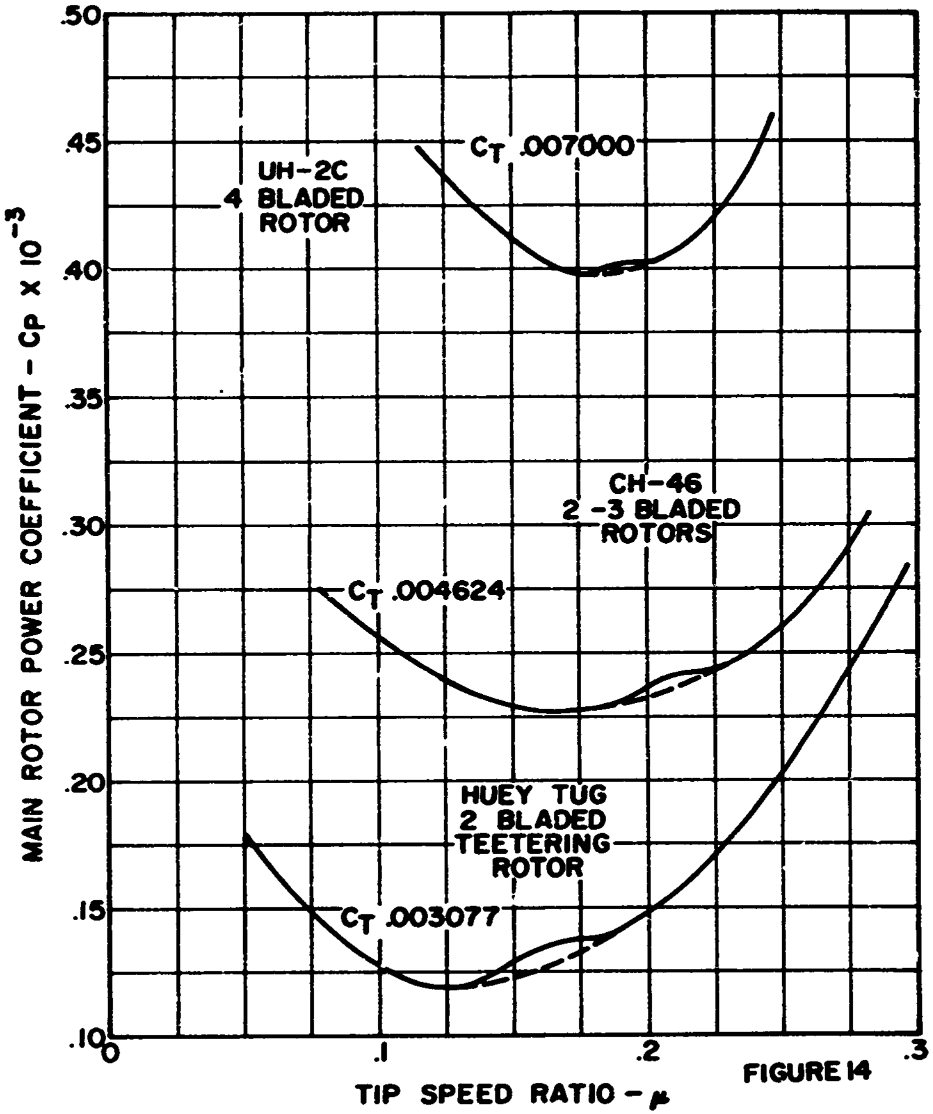


FIGURE 14

HH-3A ARMED SAR

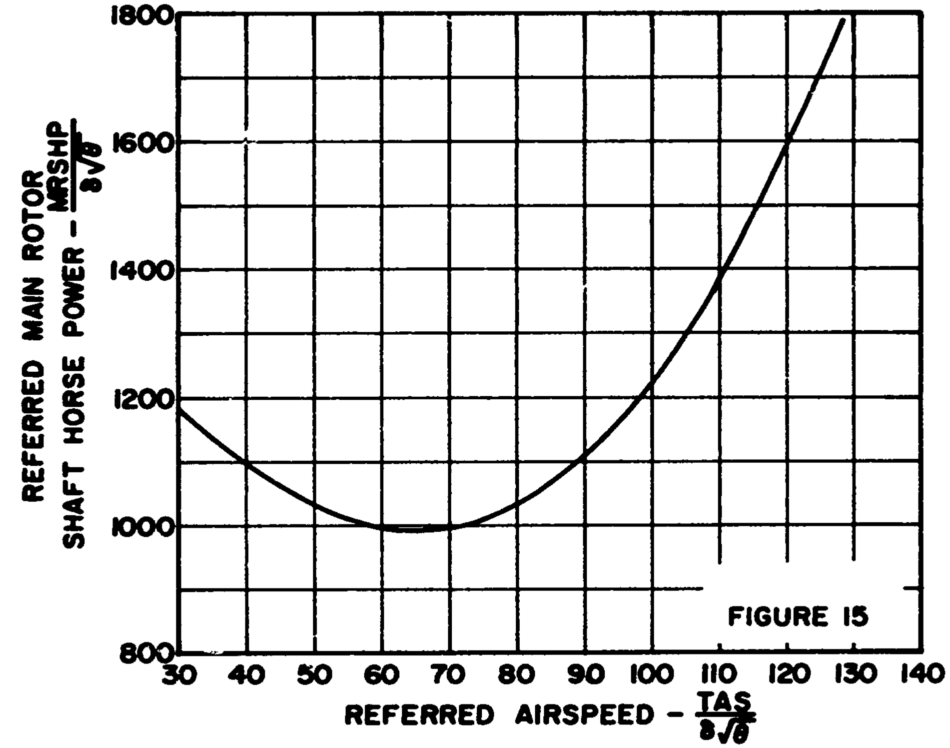
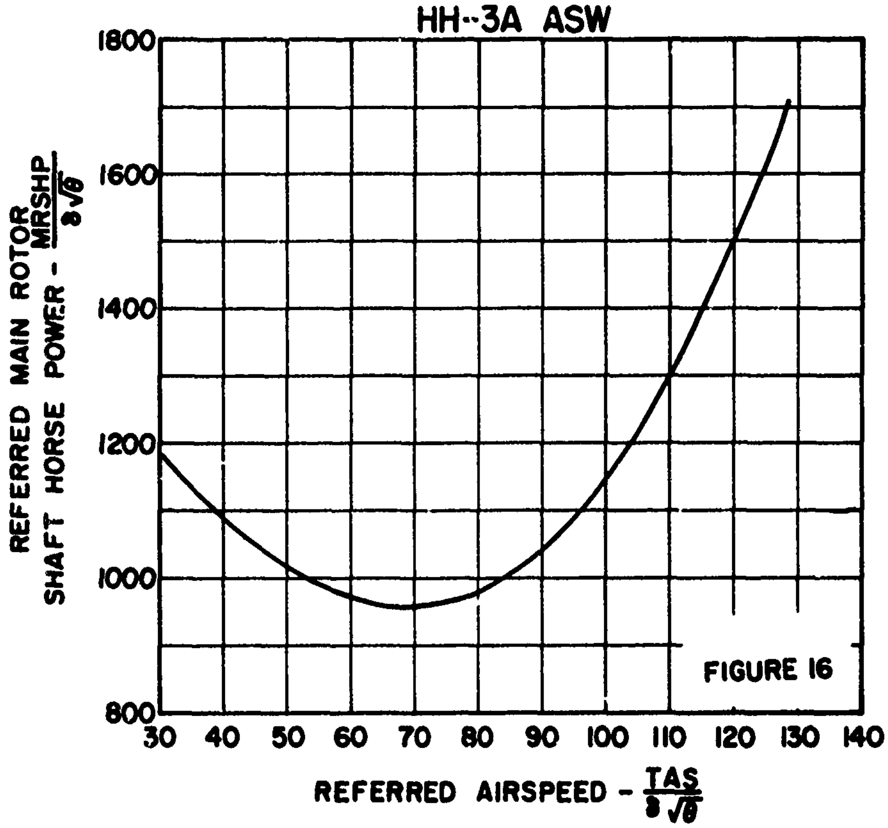
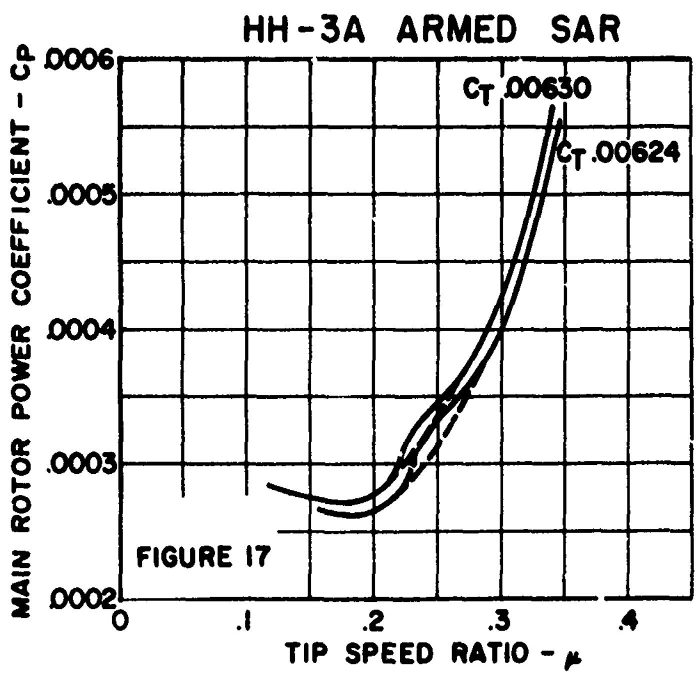
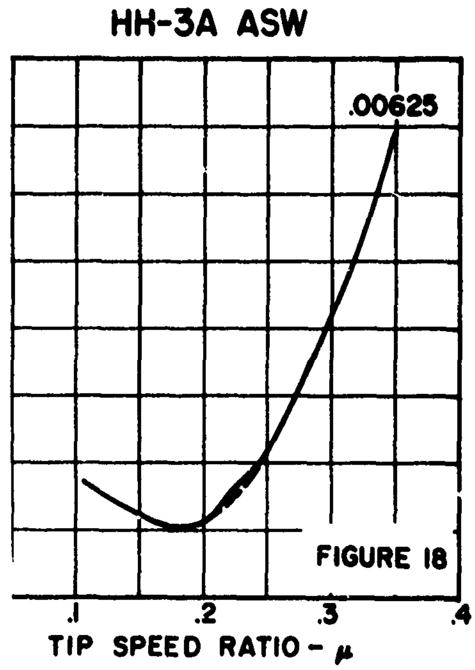


FIGURE 15


HH-3A ASW
REFERRED MAIN ROTOR
SHAFT HORSE POWER - $\frac{MRSHP}{\delta\sqrt{\theta}}$
1800
1600
1400
1200
1000
800
30 40 50 60 70 80 90 100 110 120 130 140
REFERRED AIRSPEED - $\frac{TAS}{\delta\sqrt{\theta}}$
FIGURE 16


HH-3A ARMED SAR
MAIN ROTOR POWER COEFFICIENT - C_P
.0006
.0005
.0004
.0003
.0002
C_T .00630
C_T .00624
FIGURE 17
0 .1 .2 .3 .4
TIP SPEED RATIO - μ


HH-3A ASW
.00625
FIGURE 18
.1 .2 .3 .4
TIP SPEED RATIO - μ

SOME FLIGHT EXPERIMENTS ON THE XH51 N HELICOPTER

by

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Bedford, England.

SUMMARY

This paper presents some preliminary results from a fruitful collaboration with the NASA Langley Research Center. NASA made its Lockheed XH51 N helicopter available to the Royal Aircraft Establishment for flight tests at Bedford, England, to be concentrated mainly on stability and control. Several combinations of gyro inertia and control springs have been evaluated. The principal effects of the changes in configuration were variations in control sensitivity and rotor damping; the degree of change expected on theoretical grounds has been broadly demonstrated. A variation in static stability due to a differently shaped gyro arm was also experienced. Pilot opinion on the various configurations has been assessed.

Some brief comparative flying was made in company with the Westland Research Scout helicopter which has a hingeless rotor but no autostabilisation of any kind. The results illustrate some differences in handling but the comparison of rotor blade bending moments from results so far analysed is inconclusive.

1. INTRODUCTION

This paper presents some preliminary results from a fruitful collaboration with the NASA Langley Research Center. NASA made its Lockheed XH51 N helicopter available to the Royal Aircraft Establishment for flight tests at Bedford, England. The helicopter was flown to Bedford in a Hercules (C130) aircraft of RAF Support Command, arriving on 30th September, 1970; during the next seven months the 20 hours flying allocated by NASA were completely used before the helicopter was returned by air to Langley on 10th May, 1971.

The XH51 is an experimental aircraft which has been in use at Langley since 1965 and has several interesting features. It is fitted with a hingeless rotor incorporating the Lockheed gyro-control system and an experimental cockpit suspension to provide some isolation from vibrations. The work at Bedford was concentrated mainly on stability and control for which the aircraft is particularly well suited in that some of its basic characteristics may be readily varied. For example, the effective damping of the rotor can be varied by changing the inertia of the gyro, and the control power by changing the control springs. Several gyro arms of differing inertia were available together with a range of control springs; in all, four combinations including the original were flown. The work at Langley had included investigations into handling qualities and loadings associated with nap-of-the-earth flying (ref. 1, 2) but the particular changes in gyro inertia and control springs, although originally envisaged as part of the NASA programme for the aircraft, were evaluated for the first time at RAE, Bedford.

As background to the present tests it is useful to remember that in June, 1970 when the collaborative programme was first discussed the practical first-hand experience of hingeless-rotor helicopters in Britain was very limited. Westlands advanced hingeless design for the WG13 (Lynx) was still in manufacture and the Research Scout helicopter which incorporated many features of the rotor for the Lynx had only recently flown. (Both helicopters have now accumulated many successful flying hours and a paper describing flight tests of the Research Scout is also being presented at this Symposium.) In particular, no first-hand flight-test experience of hingeless rotors then existed at the RAE, and it was against this background that NASA's loan of the XH51 N helicopter was welcomed. The loan itself and the technical support throughout the programme are gratefully acknowledged. Special mention should be made of the expert help given by Robert Huston from the Langley Center's own flight research team who spent a month with us at Bedford at the start of the tests.

Some theoretical studies of the longitudinal stability had previously been made at RAE; these had indicated the improved short-term control at low speeds but also revealed adverse effects at high speeds with helicopters having high equivalent flapping-hinge offsets. There are of course several solutions to this problem and the gyro-control system includes the ability to "de-sensitize" the rotor as one of its characteristics. Some comparative flight tests have been made with a hingeless rotor helicopter with no form of stabilisation in an effort to evaluate differences in behaviour.

Finally, it should be emphasised that the results given are preliminary; a final report will be issued later.

2. DESCRIPTION OF THE AIRCRAFT AND INSTRUMENTATION

The XH51 N helicopter (fig. 1) is a three-bladed hingeless rotor helicopter which was flown during the present tests at a nominal test weight of 4,000 lb. It is powered by a 550 h.p. free-turbine engine. Aerodynamically it is fairly clean and is fitted with a retractable skid undercarriage. The nose, including the cockpit, has a limited degree of freedom in a vertical direction and is spring-mounted relative to the rest of the fuselage for vibration isolation.

The helicopter is fitted with the gyro-controlled Lockheed rotor system. A diagram of the cyclic pitch control system is given in fig. 2. The free gyro processes to provide feathering inputs to the blades when subjected to an applied moment provided by the pilot through a system of springs. These springs are actuated through an irreversible power-boost system and are termed 'positive' springs; each of them (longitudinal and lateral) has a complementary 'negative' spring of nominally equal effective rate. They are arranged in such a way that moments are applied to the gyro by the controls, but the gyro experiences near zero spring rate as far as free oscillations are concerned. The gyro is also subject to feedback from blade moments about the feathering axis. The blades are not in line with the feathering axis but are swept forward through a small angle (2^o); in this way the blade thrust exerts a moment about the feathering hinge which is transferred to the gyro itself through the direct linkage. Blade inplane forces also contribute moments to the gyro but these are usually of a secondary nature.

Longitudinal, lateral and directional controls have trimmable springs to provide feel. The collective pitch control incorporates the usual friction device. The directional controls have no power-boost. A bob-weight is attached to the longitudinal control to provide a measure of stick force per 'g'.

The helicopter had previously been extensively instrumented for loading and handling studies (ref. 1, 2) and this instrumentation was virtually unchanged for the present tests. The quantities recorded on multi-channel trace recorders included positions and loads in the control/gyro/rotor system, fuselage attitudes and rates, main rotor flapwise and chordwise blade root bending moments, tail rotor flapwise and chordwise bending moments and tail rotor pitch link load, main rotor shaft bending moments and torque, airspeed, aerodynamic sideslip and incidence; the sensors for the last three items were mounted on a nose-boom.

In addition to the airborne recording equipment four channels of structural data (main-rotor flapwise and chordwise bending moments and tail-rotor pitch-link load) were transmitted by telemetry to the ground and continuously monitored during the flight tests.

3. CONTROL SYSTEM EVALUATION

The parameters varied during the evaluation of the control system were the gyro inertia and the control spring-rates. Four configurations were flown:-

gyro inertia, slug ft^2	spring rate, lb/in	
7.5	545	(original)
7.5	364	
5.0	545	
5.0	364	

The tests made in each configuration included:-

(a) Control to trim in level flight.
(b) Control to trim at constant collective pitch and various mean speeds.
(c) Response to step input of control at various speeds.
(d) Response to pulse input of control at various speeds.
(e) Control to trim in steady turns.

In addition, several tasks were flown which involved manoeuvring and positioning at low speed near the ground. Some take-offs and landings were also made on slopes with the original configuration.

The response of the Lockheed gyro-controlled rotor system is described in ref. 3, and the basic features which enable us to evaluate the effects of gyro inertia and of variation in positive spring-rate are discussed below under suitable headings and compared with experimental results. It should be emphasised that we shall consider first order effects only and therefore the contribution of thrust tilt to overall moments on the helicopter will be neglected.

3.1 Static Stability

With the shaft and controls fixed, any time average external moment acting on the rotor is rapidly reduced to zero. The feedback from the rotor precesses the gyro in such a way that the cyclic pitch applied to the blade eliminates the incremental flapping and consequent hub moment. The rotor therefore possesses neutral attitude and speed stability. These characteristics are confirmed by the longitudinal control to trim in level flight, when speed and attitude are varied as shown in fig. 3. It will be seen that there is virtually no change in stick position with speed when the 5.0 slug ft^2 gyro is fitted. The gyro supplies the feathering to counter the flapping with respect to the "no-feathering" axis (which is a major contribution in a rotor without gyro): the positive spring, actuated by the stick, merely provides the input necessary to generate the rotor-hub moments required to trim the relatively small fuselage moments. The curves do have a positive slope when the gyro of inertia 7.5 slug ft^2 is fitted. This is due to the fact that the arms at this gyro are flattened and form a crude aerofoil, the aerodynamic forces on which impart a degree of speed stability. Although not illustrated, the change of the longitudinal stick position with speed at constant collective pitch is also small and difficult to distinguish from the level flight trims, where of course collective pitch is varied.

3.2 Manoeuvre Characteristics

Time histories of responses to longitudinal and lateral control steps in hovering with the original configuration (gyro inertia 7.5 slug ft^2 and spring rate 545 lb/in (7.5/545)) are given in fig. 4 (a)(b). The maximum rate of roll is reached very rapidly in about 0.3 sec. including a ramp time of 0.1 sec. to reach maximum control input. Rate of pitch tends to reach an initial "knee" in about 0.9 sec. and thereafter continues to increase gradually. This feature was more noticeable with aft control movements and is possibly associated with translational and downwash effects on the fuselage. The short term response characteristics of the helicopter may be described as those of a rate demand control with a short time constant and are an attractive feature of the hingeless rotor. The general nature of the response was the same for all combinations of gyro and springs tested, but the gain and the time constant varied. The reasons for these variations are discussed below.

In response to a step input of control, before the angular rate of the helicopter has built up the angular acceleration is proportional to the hub moment due to blade flapping. A moment dependent on blade flapping reacts the moment caused by the positive spring through the action of the pilot. Thus the control sensitivity (angular acceleration per unit control displacement) is proportional to the spring rate.

For a steady rate of pitch or roll a steady precessional torque is required by the gyro. If the controls are fixed this torque is supplied through feed back from the rotor as a result of rotor flapping, and provides an overall damping moment opposing the motion of the helicopter. For a steady rate of roll (say) of the unrestrained helicopter, the hub moment is zero (if the fuselage damping is neglected) and the precessional torque required by the gyro is supplied by the spring through the displacement of the pilot's control. The required torque is proportional to the rate of roll and gyro inertia, i.e.:

$$\text{(spring rate) (stick displacement)} \propto \text{(gyro inertia) (roll rate)}$$

or,

$$\begin{matrix}\text{Maximum rate of roll} \\ \text{rad/sec/inch of control}\end{matrix} \quad \propto \quad \frac{\text{Spring rate}}{\text{Gyro inertia}}$$

and, as stated previously,

$$\begin{matrix}\text{Control sensitivity,} \\ \text{rad/sec}^2\text{/inch of control}\end{matrix} \quad \propto \quad \text{Spring rate}$$

and, assuming a first order system in rate,

$$\text{the Time constant} = \frac{\text{Maximum rate}}{\text{initial acceleration}}, \text{ sec.}$$

$$\text{and Damping} = \frac{\text{Initial acceleration}}{\text{Maximum rate}}, \text{ sec}^{-1}$$

These characteristics illustrated in the hover for the lateral step inputs of control although of course the results for longitudinal control are similar. The maximum rate of roll for the various combinations of gyro inertia and spring rates are given in fig. 5. It will be seen that, for a particular combination, maximum rate of roll is proportional to control displacement and that, as expected, the lower gyro inertia gives a higher maximum rate of roll for each spring rate. The results, which represent the mean of all suitable measurements are tabulated below:-

Lateral results in hover

Gyro inertia, slug ft^2	7.5	7.5	5.0	5.0
Spring rate, lb/in	545	364	545	364
Measured rate, rad/sec/in	0.238	0.181	0.326	0.223
Measured sensitivity, rad/sec^2/in	1.87	1.44	2.25	1.5
Rate relative to original configuration (theoretical)	1.0	0.67	1.4	1.0
Rate relative to original configuration (measured)	1.0	0.76	1.5	0.98
Sensitivity relative to original configuration (theoretical)	1.0	0.73	1.0	0.73
Sensitivity relative to original configuration (measured)	1.0	0.77	1.2	0.8
Damping sec^{-1} (measured)	6.9	7.9	6.4	6.5

It will be seen that the theoretical trends are followed broadly. The measurement of initial angular acceleration was difficult because of noise on the record and the short time constants involved; it is therefore particularly open to error.

The range of characteristics investigated is shown on a damping - control sensitivity diagram, fig. 6, for both pitch and roll, together with the criteria of reference 4. In pitch, at the smaller spring rate, the criteria for Instrument Flight (IFR) would not be met. This trend is confirmed by pilot opinion (section 6) where the lateral characteristics were preferred to the longitudinal ones which were described as "ponderous" at the lowest control sensitivity.

3.3 Characteristics in Forward Flight

A time history of the response to an aft longitudinal control displacement at 80 kt for configuration 7.5/545 is given in fig. 4(c). It will be seen that the rate of pitch rises to an initial "knee" in just under a second and thereafter rises at a uniform but lower rate. The time constant of the motion up to the initial knee is similar to that measured in the Hover. The reason for the continued rise is not explained on the simple basis of feed back to the gyro from blade thrust; inplane forces, particularly when the blades deflect under normal acceleration, are fed back to the gyro also and may contribute to this effect. It is hoped to include an assessment of these effects in the final report. The general character of the response did not change up to a speed of 110 kts, the highest speed at which tests were made.

The responses at speed in the other configurations tested, in terms of initial acceleration and maximum rate, were essentially similar to those measured in the hover in both pitch and roll. The helicopter could be flown hands off for quite long periods of time in all configurations tested, the limit being set by the lateral - directional characteristics of mild spiral instability.

4. COMPARISON FLYING WITH THE WESTLAND RESEARCH SCOUT

The Westland Research Scout has a hingeless rotor and was flown during the tests at a nominal weight of 5,000 lbs. It has no form of stability augmentation. The blade flapping and inplane stiffnesses, as indicated by the corresponding first mode frequencies, are compared below with those of the XH51 N:-

	Research Scout	XH51 N
flap	1.08Ω	1.1Ω
inplane	0.64Ω	1.4Ω

A feature of the Research Scout is the low inplane stiffness, resulting in a natural frequency less than the fundamental frequency of the rotor. The XH51 N is much stiffer resulting in a frequency greater than the fundamental and more than twice that of the Research Scout. The flapping stiffnesses are reasonably similar. One of the features claimed for the Lockheed gyro-controlled rotor is a reduction of blade root bending moments due to gusts (ref. 3) by virtue of the attenuation of transient flapping.

The main object of the comparative flight tests was to compare the behaviour of the aircraft in terms of handling and blade bending moments. The aircraft were flown in loose formation at a height of approximately 50 ft. on each side of a runway, 100 yards wide; thus it is reasonable to assume that the general level of turbulence was similar for each aircraft. The pilots were briefed to fly in line abreast with minimal station keeping activity. The length of the runway was 10,500 ft and runs were made at constant speed in each direction. At the end of each pass along the runway a tail chase was flown, including turns and reversals of direction, to return eventually for a further pass in formation along the runway in the opposite direction at the same speed. The role of lead aircraft was changed at each end of the runway. The tail chase was included to subject the aircraft as far as possible to the same manoeuvres so that handling and blade bending moments could be compared. The general pattern was repeated at speeds of 70, 95 and 110 kt. In an attempt to assess the effect of different flying techniques the pilots were interchanged for one sortie.

The flights along the runway were made at low level so that the turbulence would be of the type associated with wind shear at the surface. The degree of turbulence increases with the wind strength, the maximum during the tests being 3.6 kt, RMS associated with a mean wind speed of 18 kt. These data were obtained from an instrumented tower at a height of 50 ft, but at a distance of more than a mile from the runway; however they are thought to be reasonably representative of conditions at the runway. The aircraft were available together for one week only and unfortunately during this time the wind was less than 5 kt for the remainder of the tests.

The selection of the quantities to be compared was influenced by the noise, definition and scaling of the various trace recordings and, in the event, the following were selected:-

Longitudinal stick movement Rate of pitch	Both aircraft
Flapwise bending moment, 6 in. from hub Chordwise bending moment, 6 in. from hub	XH51 N
Flapwise bending moment, 9.5 in. from hub Chordwise bending moment, 9.5 in. from hub	Research Scout

The quantification of pilots' work load is notoriously difficult, and data in the form of paper trace records are not amenable to sophisticated analysis such as power spectra to throw light on the ease of handling of each aircraft. A method of presenting the results has been used which has the merit of simplicity and which illustrates pictorially certain differences between the aircraft and allows some tentative conclusions to be drawn. The traces of longitudinal stick movement obtained during each pass along the runway were carefully analysed and the maximum and minimum values of every excursion recorded. The time histories of cumulative stick movement irrespective of direction have been plotted for the duration of each run (figs. 7 and 9). It can be argued that the slope of the curve defines an average level of 'activity', although inspection of the number of points on the graph is also necessary to distinguish between small amplitudes at high frequency and larger amplitudes at a lower frequency.

It will be seen that the curves begin to have a more clearly defined slope 20 seconds after the start of each run. This characteristic is almost certainly consequent on a settling down period at the start of each run during which changes in activity necessary to control the relative position of the aircraft tended to occur. (This explanation is supported by pilot opinion.) In order to eliminate this effect, only events after the initial 20 seconds will be considered in the discussion that follows. The slope of the cumulative stick movement is greater in turbulence than in calm conditions for both aircraft. Each direction of pass along the runway is plotted separately and there is a noticeable difference between directions but this is insufficient to mask the difference due to turbulence. The corresponding cumulative rate of pitch is shown in figs. 8, 10 and a similar general increase in slope or 'activity' is again apparent for each aircraft. Here the activity is proportional to maximum angular acceleration, but the end point has no obvious physical significance. The slopes of the curves in corresponding conditions are also of the same order which could be taken to indicate that both aircraft were being flown with generally similar accuracy. One condition for this to be true is that the form factor of the motion is the same for each aircraft and inspection of the records makes this appear a reasonable assumption.

The curves of stick displacement do not have any direct significance unless related to the response of the aircraft, and so, to provide some measure of scaling, the longitudinal control characteristics in the hover are compared below:-

	Sensitivity rad/sec^2/inch	Damping sec^{-1}
XH51 N	0.29	1.8
Research Scout	0.32	1.8

The values are very similar and it is surprising to see (fig. 7, 9) that the slope of the cumulative stick displacement and end values for the unstabilised Research Scout are noticeably less than the corresponding values for the XH51 N. A possible explanation is that the moveable cockpit and the bob-weight on the stick cause a degree of involuntary control activity. This is supported by the pilot whose subjective impression was that no control movements were made for considerable periods during the runs.

The results for runs at a higher speed (110 kt) in calm conditions are given in figs. 11-14. Lines representing the mean slopes from the previous results at 95 kt. have been added to the results with an origin at 20 secs to eliminate activity due to positioning at the start of the runs. It will be seen that the increase in speed of 15 kt. causes a small increase in cumulative rate of pitch for both aircraft and is accompanied by a correspondingly small increase in control activity for the XH51 N. For the Research Scout, however, there is a marked increase and it is suggested that the increase is indicative of the degradation of the stability characteristics of the unstabilised hingeless rotor at the higher speeds largely due to the unstable M_w derivative. This suggestion is corroborated by pilots' assessment of the handling of the Research Scout at higher speeds when an increased tendency for pitch divergence and sensitivity to turbulence is shown.

5. COMPARATIVE BLADE BENDING MOMENTS

The analysis of the blade root bending moments is incomplete; none of the data for the tail chases and only a small part of those for the level runs have yet been examined. The initial analysis is taking the form of reading maximum and minimum bending moments, once per rotor revolution, and finding the overall means and the means and standard deviations of the once-per-revolution fluctuations. In the results so far analysed turbulence had no significant effect on either the mean or the mean of the first harmonic variations of flapwise and chordwise blade root bending moments for either aircraft. It was thought that the standard deviations of once-per-revolution fluctuations in bending moment might be a simple way of comparing overall scatter and unsteadiness. The initial results so far obtained are not particularly conclusive. There appears to be little difference in the standard deviation of blade root bending moments between calm and turbulent conditions for the Research Scout, whilst for the XH51 N the only change is an unexpected rise in the standard deviation of the flapwise once-per-revolution bending moments in turbulence, but no change in the standard deviation of the chordwise bending moments. If this result is confirmed by the analysis of further runs it may present an anomaly since one would expect the flapwise and chordwise moments to be related through the action of Coriolis forces. The increase in the standard deviation of the blade-root flapwise bending moment of the XH51 N with turbulence (if substantiated) is unexpected because of the action of the gyro in reducing first harmonic flapping.

6. PILOT OPINION OF THE XH51 N

During the trials, a total of six pilots flew the XH51 N either as first or second pilot. Their comments are presented with minimum editing and it should be emphasised that they refer specifically to an experimental research helicopter and that, although the basic principle has been retained, this unique control system has been considerably developed in later designs of helicopters.

The pilots were unanimous in their praise for the apparent attitude stability of the aircraft, although the two project pilots (who did most of the flying) commented on the poor speed stability. This was stated to be worse with the small 5.0 slug ft^2 gyro which is consistent with the zero slope of stick-position-to-trim with speed (fig. 3).

Longitudinally the stick was generally felt to be rather under geared in the basic configuration of 7.5 slug ft^2 gyro and 545 lb/in. springs, and also with the 5.0 slug ft^2 gyro and 364 lb/in. springs. Changing to 7.5 slug ft^2 gyro and 364 lb/in. springs made matters worse. The word 'ponderous' was used to describe the handling in this condition. Although the high-geared configuration of 5.0 slug ft^2 gyro and 545 lb/in. springs gave better low speed handling, the two pilots who flew in this configuration agreed that PIO tendencies were present around and above 90 kt. in turbulence. The bob-weight and spring (see below) was thought to be contributing to this.

The lateral stick gearing was considered to be generally better than the longitudinal, changing from slightly under geared with 7.5 slug ft^2 gyro and 364 lb/in. springs, through "about right" for 7.5/545 and 5.0/364, to somewhat over geared with 5.0 slug ft^2 gyro and 545 lb/in. springs.

Longitudinal stick force gradients were thought to be too heavy at all times. Although the bob-weight and spring (provided to give stick force per g) was thought to be potentially useful in forward flight and turns all pilots agreed that the force gradient had been set too high. The device was also thought to be obtrusive in certain flight conditions, notably in turbulence both at high speed and in turns, during collective adjustments at the hover, and (not surprisingly) when trying to make accurate longitudinal cyclic step inputs as part of the test programme.

All the pilots considered the collective gearing to be higher than usual, although all pilots except one stated that they adapted to it readily and after an hour or so found it no longer obtrusive.

The cyclic control was agreed to behave essentially as a rate demand system and was well liked. One pilot made the point that the effective gearing was not greatly affected by forward speed. At high speed the longitudinal response was thought to be complicated by vibratory effects, particularly bottoming of the spring cab. One project pilot commented that, once 1.5 g was exceeded there was a noticeable pitch-up tendency at the higher speeds.

All pilots commented on the heavy pedal spring forces, the inconvenience of the electric pedal trimmer, and the marked adverse yaw with roll control application. All the pilots thought the pitch-roll and roll-pitch cross coupling to be noticeable, especially near the hover. All pilots except one considered that co-ordinated turns, turn entries, and roll reversals were difficult to execute. The main reasons advanced were the cross coupling, adverse yaw, and yaw trim problems, although the bob-weight, sprung cab bottoming and sprung cab "shuffle" were also blamed.

Later in the test the two pilots most familiar with the aircraft commented that at high speed in turbulence, they could feel a slight time lag between incipient pitch divergences and (they said) the gyro controling and stabilising the rotor.

As with all hingeless-rotor helicopters, the XH51 N cyclic controls must be centered after landing (and also for start-up and take-off). This was not considered to present any piloting problem, but all of the pilots said that stick position indicators were essential for the task.

The summary of the pilot comments was as follows. The rate demand cyclic control and apparent attitude stability were well liked, although the lack of speed stability tended to be less satisfactory. Stick forces were universally considered too high, and the high speed characteristics were spoilt by excessive vibration and side effects from the bob-weight and sprung cabin. The turning behaviour was affected by the same side effects and also by control cross-coupling, adverse yaw and yaw trimmer inadequacies.

7. CONCLUSIONS

Flight tests of several combinations of gyro inertia and control springs have been evaluated on the XH51 N helicopter. The principal changes in characteristics are control sensitivity and rotor damping and the degree of change expected on theoretical grounds has been broadly demonstrated. Pilot opinion on the various configurations has been assessed.

Some brief comparative flying was made in company with the Westland Research Scout which has a hingeless rotor but no autostabilisation of any kind. The results illustrate some differences in handling but the comparison of rotor blade bending moments from results so far analysed is inconclusive with regard to gust alleviation properties of the gyro control.

8. REFERENCES

No.	Author	Title etc.
1.	Henry L. Kelly, Robert J. Pegg and Robert A. Champine.	Flying quality factors currently limiting nap-of-the-earth manoeuvrability as identified by flight investigations. NASA TN D-4931 (1968)
2.	William J. Snyder	A summary of rotor-hub bending moments encountered by a high-performance helicopter during nap-of-the-earth manoeuvres. NASA TN D-4574 (1968)
3.	G. J. Sissingh	Response characteristics of the gyro-controlled Lockheed rotor system. American Helicopter Society, Inc. 23rd Annual National Forum Proceedings. 1967.
4.	-	Helicopter Flying and ground handling qualities. MIL-H-8501A.

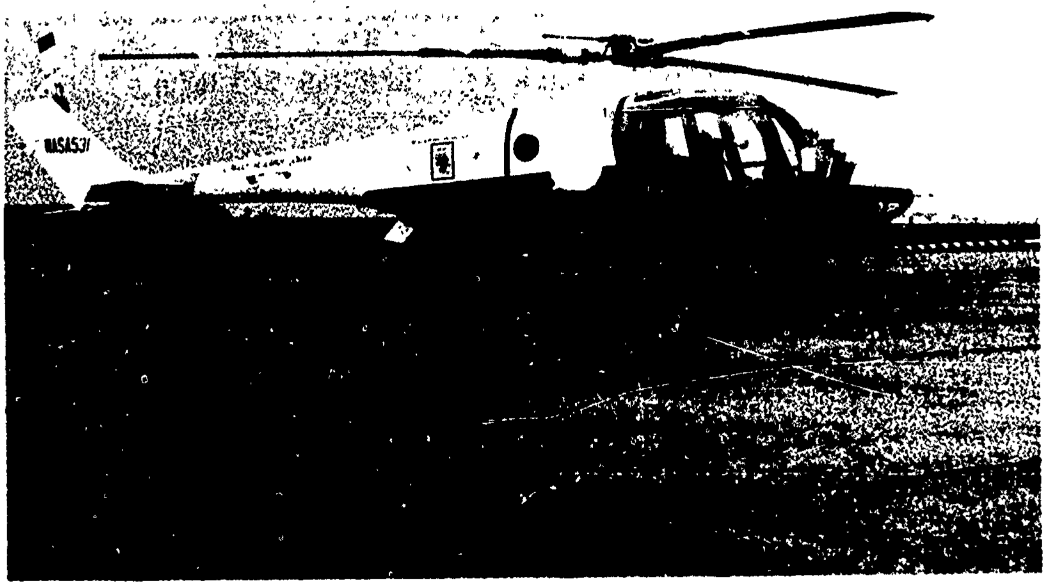


FIG.1 NASA XH51N HELICOPTER

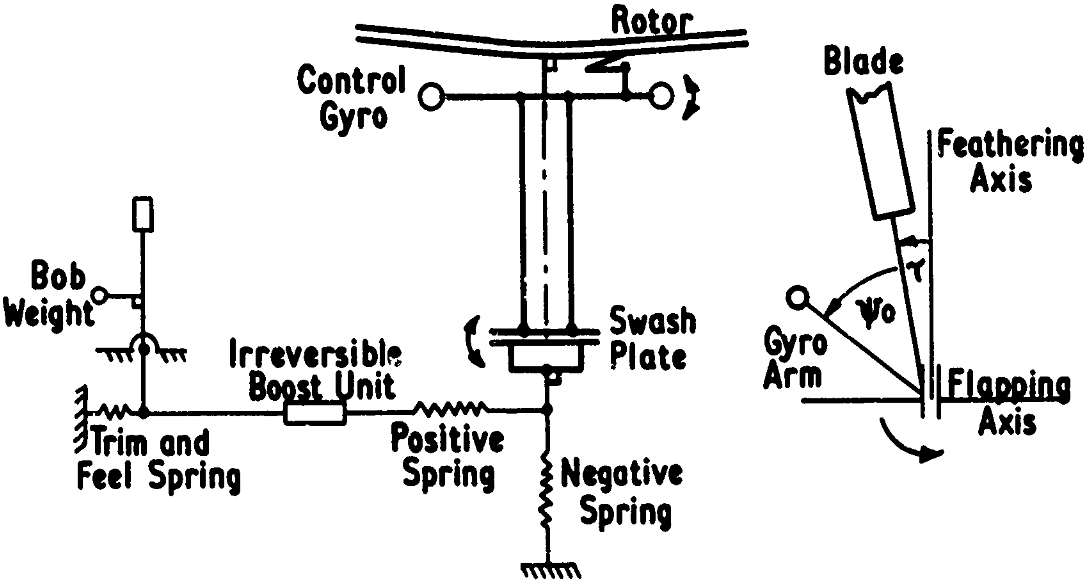
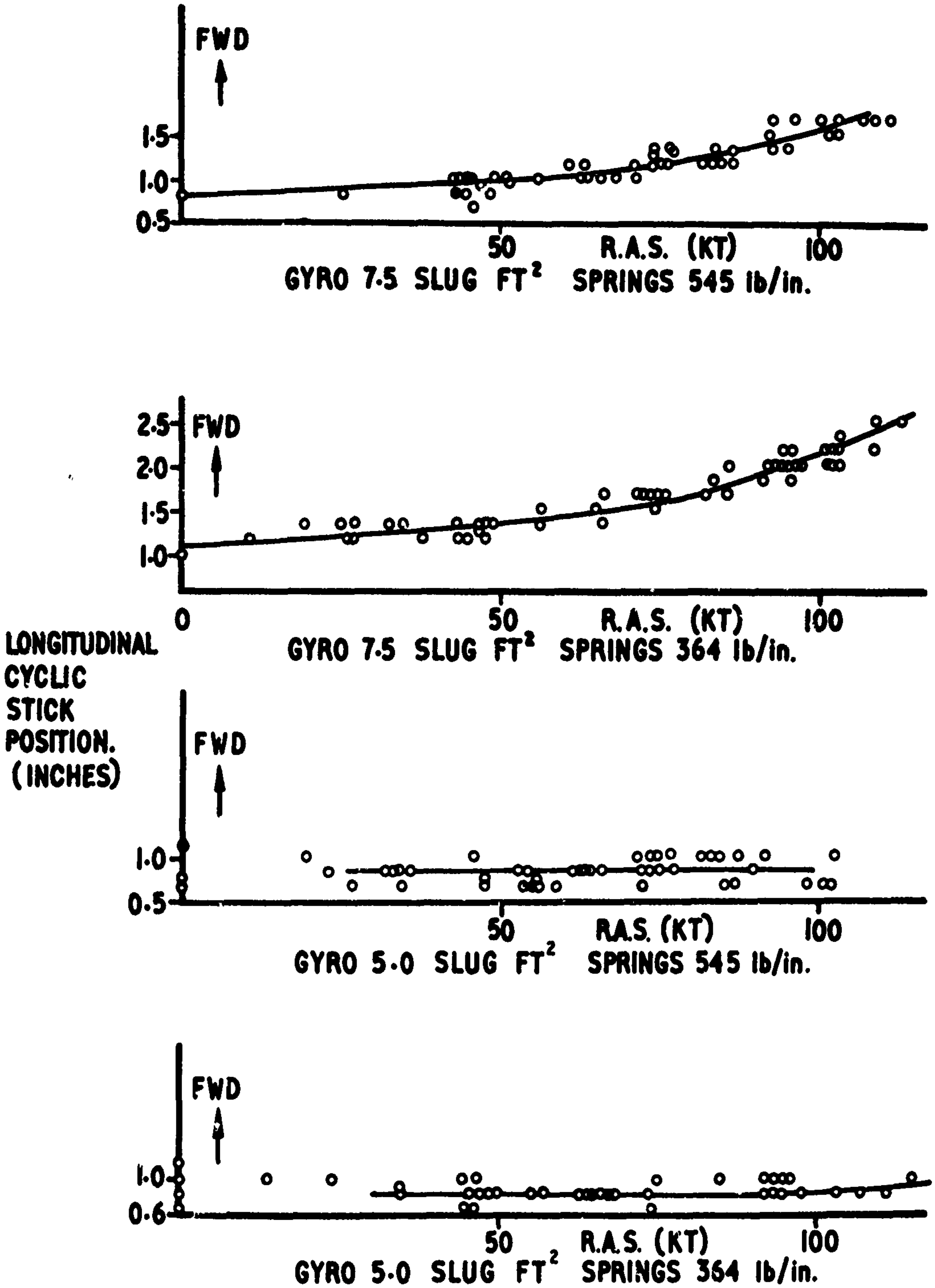


FIG.2 DIAGRAM OF CONTROL SYSTEM

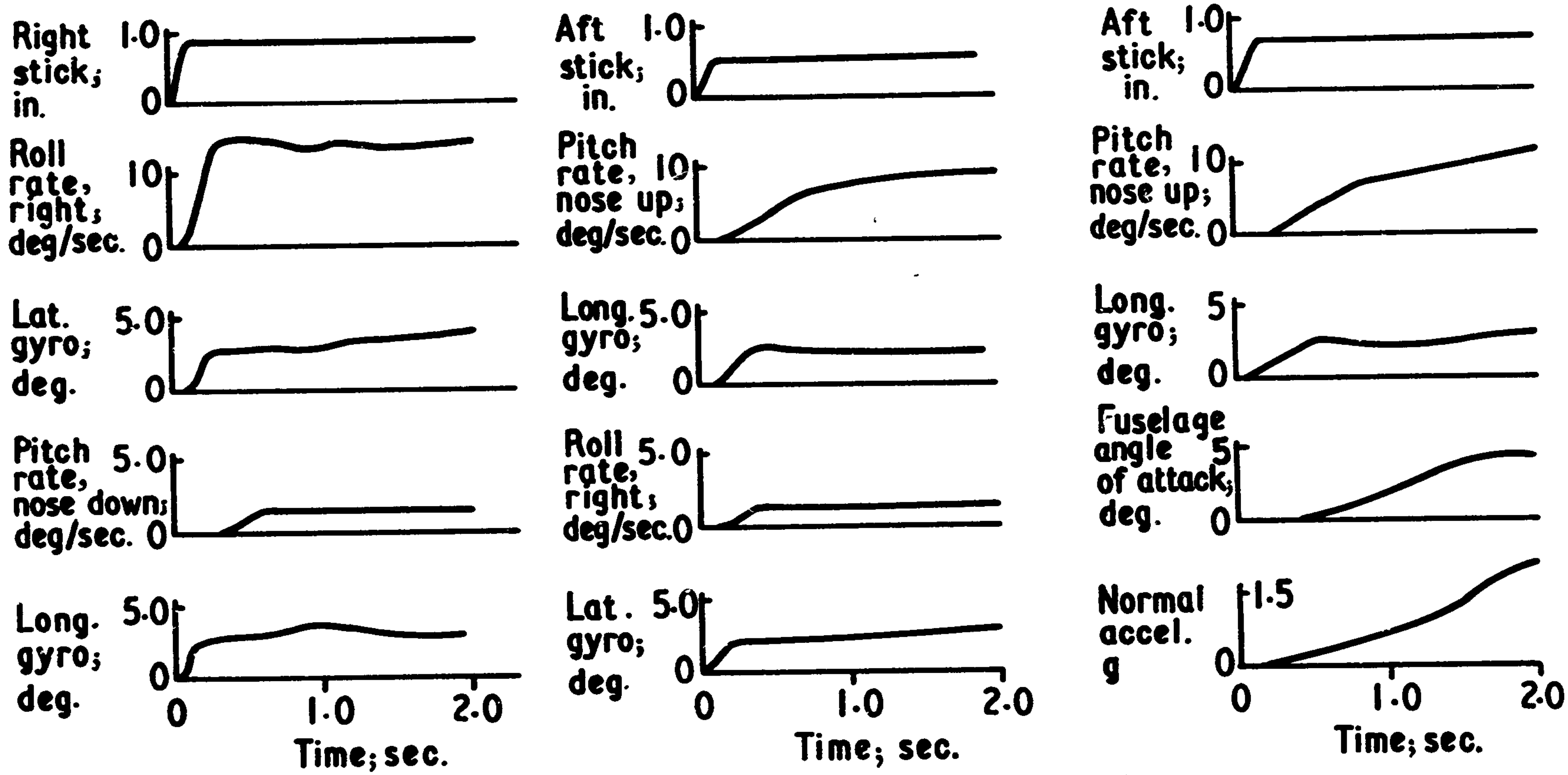


NOMINAL WEIGHT 4000lb., C.G. 0.22in. FWD. OF ROTOR SHAFT

FIG. 3

XH51N

EFFECT OF CONTROL GYRO INERTIA & SPRING STIFFNESS ON LONGITUDINAL STICK POSITION TO TRIM IN LEVEL FLIGHT.



(a)
Lateral response, longitudinal control constant. Hover.

(b)
Longitudinal response, lateral control constant. Hover.

(c)
Longitudinal response, lateral control constant. speed 80 kt.

FIG.4 CONTROL RESPONSE. GYRO INERTIA 7.5 slug ft^2, SPRING RATE 545 lb/in.

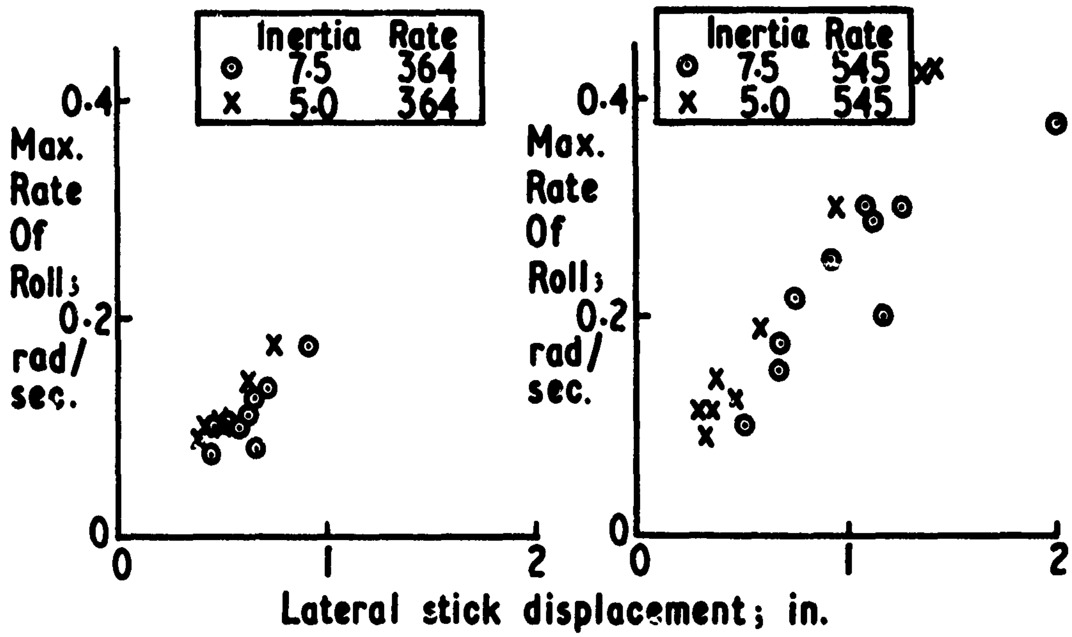


FIG.5 THE VARIATION OF MAX. ROLL RATE WITH STICK DISPLACEMENT FOR VARIOUS GYRO INERTIA & SPRING RATES IN HOVERING

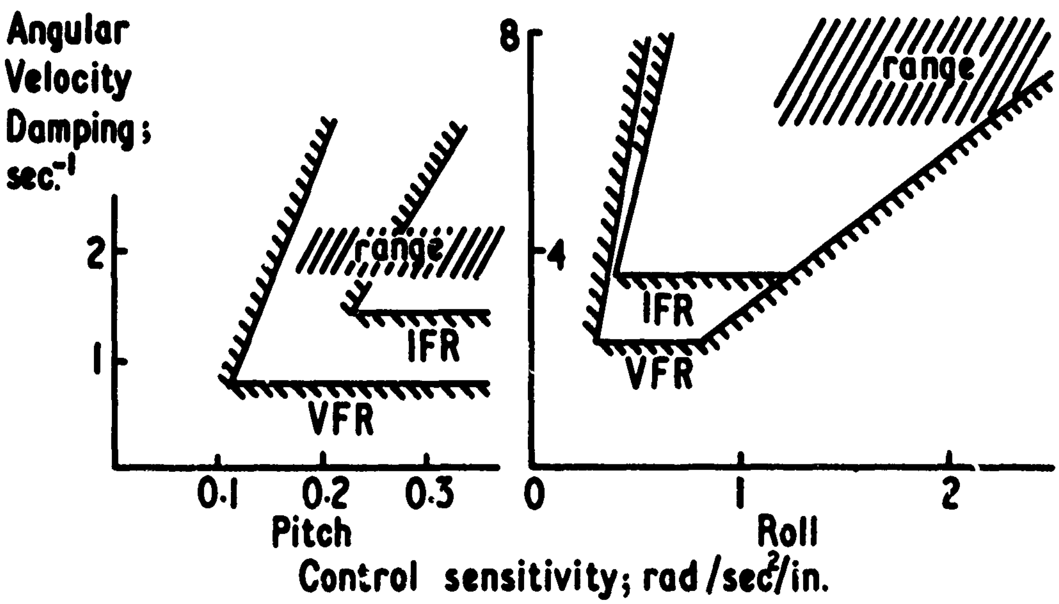


FIG. 6 RANGE OF PITCH & ROLL CHARACTERISTICS MEASURED IN HOVER & CRITERIA FROM REF. 4

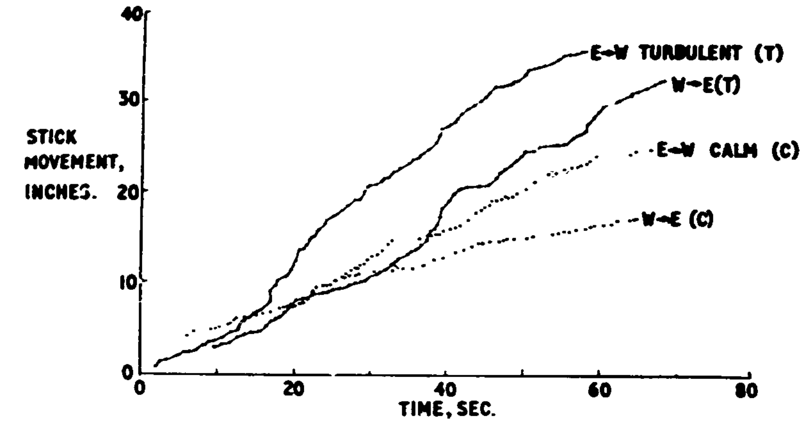


FIG. 7 XHSIN. CUMULATIVE LONGITUDINAL STICK DISPLACEMENT. SPEED 95 kt.

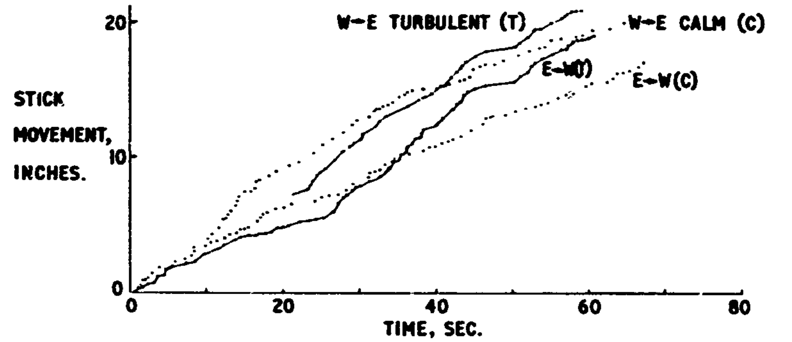


FIG. 9 RESEARCH SCOUT. CUMULATIVE LONGITUDINAL STICK DISPLACEMENT. SPEED 95kt.

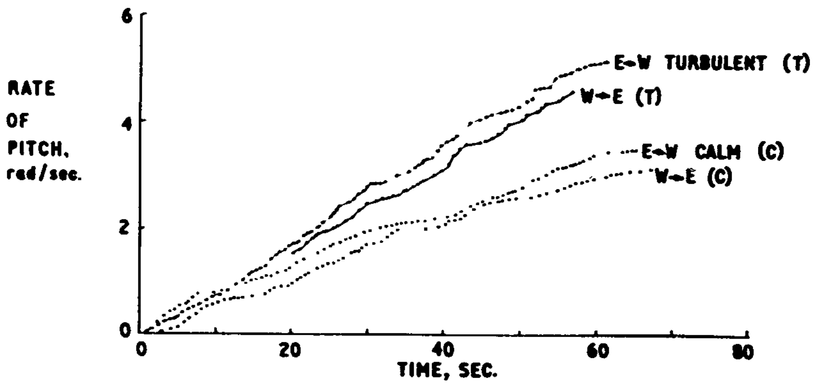


FIG. 8 XHSIN. CUMULATIVE RATE OF PITCH. SPEED 95 kt

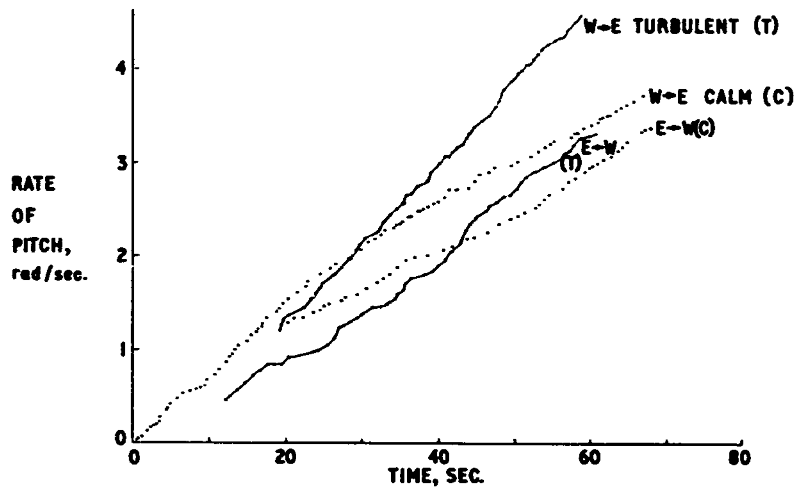


FIG. 10 RESEARCH SCOUT. CUMULATIVE RATE OF PITCH. SPEED 95 kt.

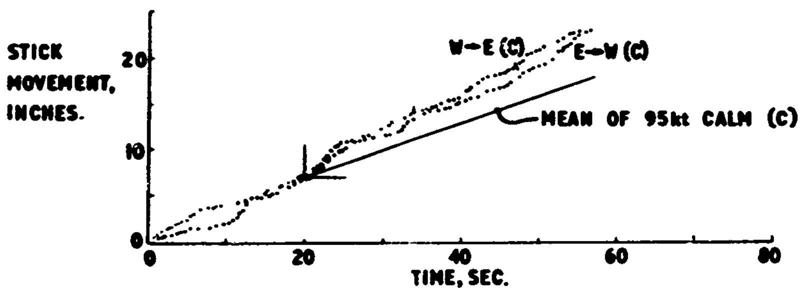


FIG. 11 XHSIN. CUMULATIVE LONGITUDINAL STICK DISPLACEMENT. SPEED 110 kt.

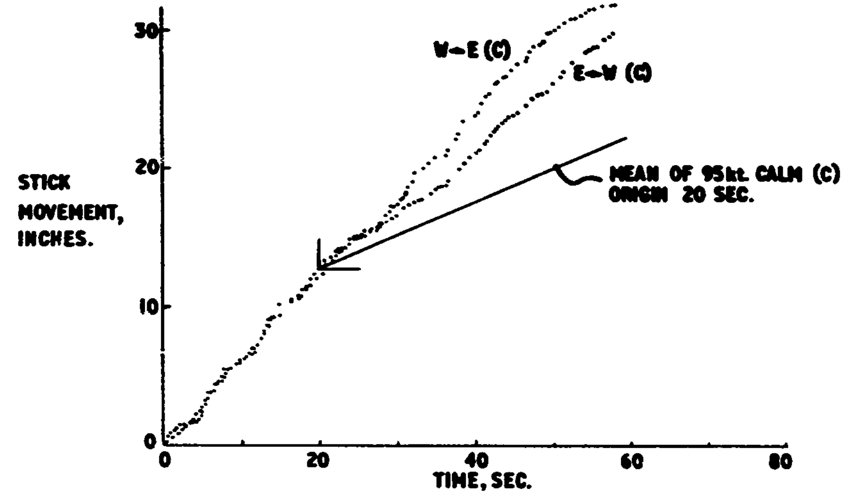


FIG. 13 RESEARCH SCOUT. LONGITUDINAL STICK DISPLACEMENT. SPEED 110 kt.

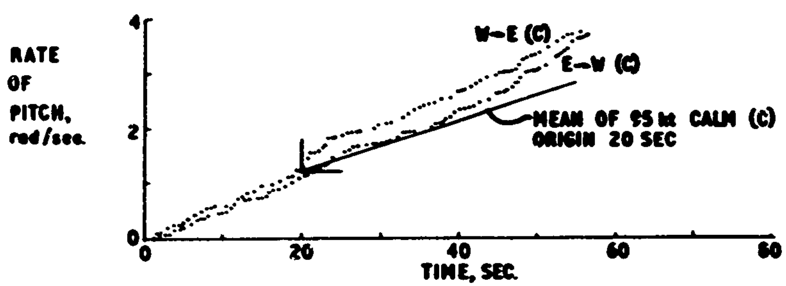


FIG. 12 XHSIN CUMULATIVE RATE OF PITCH. SPEED 110 kt.

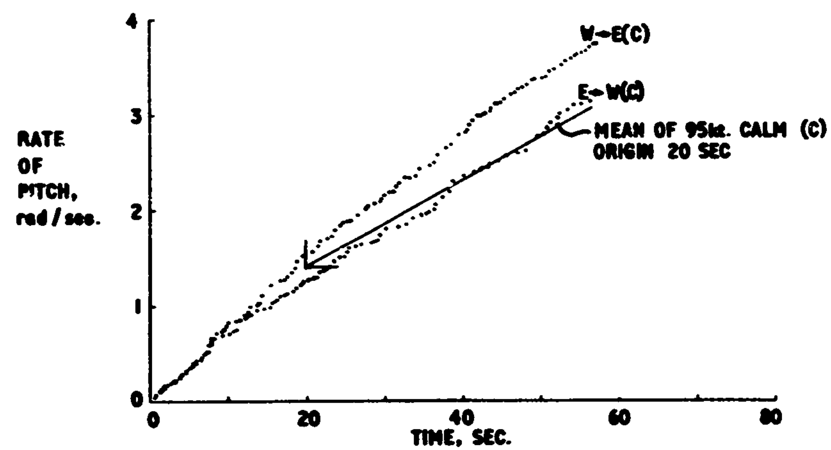


FIG. 14 RESEARCH SCOUT. CUMULATIVE RATE OF PITCH. SPEED 110 kt.

INFLUENCE OF ELASTIC COUPLING EFFECTS ON THE HANDLING QUALITIES OF A HINGELESS ROTOR HELICOPTER+

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SUMMARY

Stability and control of a helicopter with a hingeless rotor system is mainly influenced by the elastic flapping motion of the rotor blades. On a rotor with torsionally flexible blades or elasticity in the control system there can be additional aeroelastic effects, which act as control inputs on the blades.

After a short description of the rotor system and the analytical model, the paper discusses the reasons and the influences of elastic coupling effects on the stability and control behaviour of a hingeless rotor helicopter. There are effects which result from the aerodynamic characteristics and from the chordwise mass-distribution on the blade. Additional coupling effects result from flapping and inplane deflections out of the pitching control axis similar to a pitch-flap-lag-coupling. Theoretical results are compared with flight test data.

LIST OF SYMBOLS

a_s	equivalent hinge offset
c_L	lift coefficient of airfoil section
c_M	pitching moment coefficient of airfoil section
c_β	equivalent flapping hinge restraint
c_ζ	equivalent lagging hinge restraint
c_θ	torsional stiffness
c_{θ_E}	control system flexibility
D_β	longitudinal cyclic pitch
$d_{A.C.}$	distance of aerodynamic center of profile section from leading edge
$d_{C.G.}$	distance of center of gravity of blade section from leading edge
GW	gross weight
I_{yy}	helicopter moment of inertia about yy-axis
M_M	hubmoment in the rotating system
M_β	flapwise bending moment of blade root
M_ζ	inplane bending moment at blade root
M_{n_x}	control pitching moment
M_θ	pitching moment derivative due to angle of attack
$M_{\dot\theta}$	pitch rate damping moment
N	load factor in maneuvers
ΔPLL	alternating pitch link load
t	time
t_D	time to double amplitude
t_H	time to half amplitude
T	period
V	air speed
β	flapping angle
β_k	precone angle
ζ	lead-lag angle
ζ_k	sweep angle
θ_E	torsional deflection of control system
θ_C	blade cyclic pitch
Ω	rotor frequency
ω	natural frequency

+ The studies have been sponsored in part by the Ministry of Defence of the Federal Republic of Germany

1. INTRODUCTION

In research and development much work is done in order to increase the speed and maneuver range and to improve the flight characteristics of modern helicopters. With increase of physical understanding and improvement of technical facilities advanced rotor designs can be realized nowadays. Thus in recent years, the hingeless rotor system has been favoured. Figure 1 shows the helicopter BO 105 with flapwise and lagwise rigidly attached fiberglass rotorblades of high elasticity.

Theoretical and experimental work over about ten years has shown, that the rigid rotor system offers a significant improvement of handling qualities in comparison to the articulated rotor. The advantages of this rotor system are mainly caused by the flapping behaviour of the elastic fiberglass rotorblades. A further substantial gain in flight characteristics can be made by elastic coupling effects, using the coupling of the elastic blade deflections in flapping, inplane and torsional directions. Regarding the moment capability of the rigid rotor system, high efficiency can be expected for such blade integrated control systems.

As indicated in analytical studies, a proper design of blades is of great influence on rotor blade dynamics, resulting in marked improvements of flight characteristics of the helicopter. In general these elastic coupling effects increase with decreasing torsional stiffness of the blades and control system.

2. SHORT DESCRIPTION OF THE RIGID ROTOR SYSTEM

The main components of the Bölkow Rigid Rotor System (Figure 2) are a very stiff hub and fiberglass rotor blades of high elasticity. There are no flapping and no lagging hinges, the blades are rigidly attached to short hub arms. The feathering axis of the blades is fixed to these substantially rigid hub arms (1). The blade motions are pitching at the root and bending and torsional deflections.

Compared to an articulated rotor this design causes a mechanical simplification and marked improvements in handling qualities in addition. This seems to be highly desirable for modern helicopters. Theoretical investigations and intensive flight test programs have shown that these qualities are caused by the high moment capability, determined by the flapping stiffness of the rotor system. The flapping stiffness is best characterized by the rotating natural frequency of the blades for the first flapwise mode. In the Bölkow Rigid Rotor System a frequency ratio of about 1.10 to 1.15 is used.

Research and development program was started in 1960 and over more than 10 years experience has been obtained with different rotors (Figure 3) (1). After wind-tunnel model tests which had shown favourable results, a full scale rotor was built and ground tested. The first flight of a helicopter with the Bölkow rigid rotor was done in cooperation with Sud Aviation by equipping an Alouette II with a threebladed rigid rotor. After a phase of blade tuning with respect to stresses and vibrations, the first flight of BO 105 was early in 1967. With increase of theoretical understanding and with more experience the rotor system could be further refined, for example by using a very small rotor head of titanium with a somewhat increased precone angle. On this rotortype the symmetrical NACA 0012-blades were finally replaced by modified blades of cambered airfoil section, which resulted in improvements of performance and flight characteristics.

In addition a rigid rotor system with Bölkow fiberglass blades was flight tested on the SA 340 helicopter at Sud Aviation using a steel rotor head and symmetrical blades (11).

3. MATHEMATICAL MODELS FOR STABILITY AND CONTROL STUDIES

To determine the properties of a rigid rotor system in an analytical approach, a system is to be used which simulates the aerodynamic and dynamic behaviour adequately. Similar to the articulated rotor, the dynamic behaviour of the rigid rotor system is mainly determined by the flapping behaviour of the blades. For simple studies, it can be assumed that the blades have flexibility only in the flapwise direction and are infinitely stiff in the chordwise and torsional directions. In Figure 4 the main features of such a flapping equivalent system are shown. In fact, it is a conventional hinged blade with spring and damper at the hinge, simulating the first natural mode and frequency of the real blade. A question may arise if the first flapwise bending mode only will be sufficient for representing the hub moments produced by a hingeless rotor. Studies have shown that this simple equivalent system can model the rotor well enough for stability and control calculations up to medium advance ratios. This includes extreme flight conditions with high rotor loading, such as gust penetration and g-maneuvers.

In Figure 5 the moment distribution over blade span is shown for a typical flight condition. It can be seen that the moment rises mainly in the very stiff parts of root section and rotor hub. It seems well justified to represent this blade configuration by an equivalent system using an equivalent hinge with a stiff blade (2; 3; 6). Considering only the elastic flapping motion of the blades, it is assumed that there are no aeroelastic couplings of other blade motions.

A more refined equivalent system, as shown in Figure 6, is an extension of the pure flapping system and has additional degrees of freedom in chordwise and torsional directions (8). This model includes

- control system flexibility
- blade flapwise bending (first mode)
- blade inplane bending (first mode)
- blade twisting (first twisting mode).

While inplane bending of the blade has nearly no direct influence on flight behaviour, it may effect the control system dynamics, if control flexibility is considered. Torsional elastic motions of the blades are included as they act as control inputs and can thus have an influence on stability and control characteristics. Effects of different positions of the aerodynamic center, the elastic axis and the center of gravity on profile section as a function of blade span can be calculated. The aerodynamic model of the rotor is based on current blade element theory using twodimensional airfoil data, considering stall, reverse flow and compressibility effects. The mean induced velocity is calculated by momentum theory with a trapezoidal distribution in forward flight. The aerodynamic forces and the dynamic response of the rotor blades are obtained by an azimuthal step by step computation, solving the equations of motion numerically. Variable rpm can be included.

This aeroelastic rotor theory is the basis for all flight dynamics studies. It is completed by an analytical representation of the entire aircraft, including tailrotor, wing, tail planes and auxiliary thrust engines (4). A first step in all calculations is a trimming run, producing the proper control inputs on the rotors and the other trim conditions. The program is also used to find the stability derivatives of the entire aircraft as well as to calculate the dynamic response in maneuvers, gust conditions and engine failures by a step by step computation.

A somewhat modified analytical model is shown in Figure 7, which includes the flapping and inplane motions of the individual blades and rotor support flexibility (5). This complete rotor-pylon equivalent system is used for predicting more exactly the steady and vibratory forces and moments acting on the rotor system. Further possible instabilities influenced by the coupling of the various blades of the rotor are investigated with this system (9).

Three different types of analytical rotor models have been described above. In general, more degrees of freedom result in an additional complexity of the theory and in increasing computing times. Therefore, in all analytical studies the necessity of the individual degrees of freedom should be examined.

As mentioned before, the analytical method used for stability and control studies shows good correlation between theory and flight test results (6; 7). Figure 8 illustrates a comparison between measured and calculated hubmoment and root bending moments for a forward flight condition. The moments are shown as a function of rotor azimuth angle. There is good correlation for the low harmonic components. As the elastic blade is represented by a hinged equivalent blade simulating only first harmonic behaviour, no better prediction of higher harmonics can be expected. Figure 9 presents maneuver loads in forward flight (μ = .25) as a function of load factor. The longitudinal stick position, the mean flapwise bending moment due to increasing blade loads, the decreasing inplane moment due to constant collective setting and the alternating blade root moments are illustrated. The relatively good correlation between theory and test data allows the conclusion that analytical models, as described before, will be well suited for theoretical studies.

4. FLIGHT CHARACTERISTICS AS A FUNCTION OF FLAPPING FLEXIBILITY AND TORSIONAL COUPLING EFFECTS

High Moment Capability

Numerous investigations of the rigid rotor have shown that all flight characteristics are determined by the outstanding moment capacity (6; 7; 8). The reason for this high moment is the high stiffness of the rotorhub and the blade root, or - in terms of the equivalent system - the virtual flapping hinge offset of about 15% of rotor radius, which is a long moment arm for the aerodynamic shear forces.

Figure 10 illustrates this moment capacity in comparison to an articulated rotor. The dynamic flapping response of one blade and the resultant hub moment built up from all four blades due to a cyclic step input are plotted over the time. In this case, the helicopter is assumed to be fixed on the ground unable to follow the control moment. The coupled blade-pylon system (Figure 7) is used, considering each blade separately. Nearly no difference can be observed between articulated and elastic flapping response, but there is a pronounced difference in the resultant moments. The records show that the magnitude of final moment of the articulated rotor is reached in less than 1/5 of time with the rigid rotor and the final moment of this rotor type is a multiple of the value of the normal rotor. This outstanding control efficiency proves to be the most important advantage of the rigid rotor system.

The flight behaviour of this system is mainly influenced by the elastic flapping motion of the rotor blades. If there is torsional flexibility in the blade or elasticity in the control system, elastic pitching motions can be produced acting as additional control inputs to the blades. Considering the moment capability, a high efficiency on dynamic behaviour of the rotor can be expected. The rotor will be sensitive as well to all parameters influencing blade flapping such as the aerodynamic characteristics of the blade for instance.

Influence of Aerodynamic Blade Characteristics

Flight tests of the BO 105 with modified rotor blades have shown that the flight characteristics of a rigid rotor helicopter are influenced by the blade airfoil section. Theoretical studies of these phenomena have indicated that the improvements especially of performance and stability are, in general, effects of the different aerodynamic characteristics of the airfoil section in combination with elastic effects of the torsionally flexible blades. In this respect the most important characteristics proved to be

- the section maximum lift coefficient
- the section lift curve slope
- the slope of pitching moment coefficient, i.e. the a.c.-position of profile section.

First, effects resulting from maximum lift behaviour will be regarded only with respect to performance, as dynamic stall problems will not be considered here. Figure 11 shows a comparison of measured pitch link loads received with symmetrical profile section and a cambered airfoil section. There is a reduction of loads over the whole speed range, caused by a zero-moment shift of the nonsymmetric airfoil. As a still more important effect, a decrease of alternating pitch link loads is obtained in maneuver flight due to a more favourable stall behaviour of the cambered airfoil section. Additional effects resulting from increased stall limits are shown in Figure 12. With all types of rotors, longitudinal cyclic pitch to trim g-maneuvers is affected by the lift characteristics of its profile section. When essential parts of the retreating blade come up to their maximum lift coefficients, the lift of the advancing blade will be limited as well by trim conditions. This results in an increase of longitudinal forward stick position, i.e. an unstable stick travel over g. It can be seen in Figure 12 that this instability can be postponed by using unsymmetric profile section resulting in a general increase of possible g-range. These effects have been proved by trim calculations, the results of which are in good agreement with experimental data.

As is well known, the most important influence on the dynamic stability of a helicopter in forward flight is represented by the static stability due to angle of attack (pitching) (6). In this respect all rotors are more or less unstable, depending on the flapping frequency of their blades. Studies have shown, that a somewhat flattened lift curve slope will produce a less destabilizing hub moment due to angle of attack and thus will improve the dynamic stability.

Experimental data are available indicating the influence of blade modifications on the inherent dynamic stability. Figure 13 shows the stability values (stick fixed) of the BO 105 for longitudinal motion over flight speed. It is indicated by calculation and by flight test data that significant improvements in dynamic flight behaviour can be obtained with blades of cambered airfoil section. In the speed range of 100 knots, for example, the time to double amplitude is increased from 4 sec to about 10 or 20 sec. The scatter of experimental data includes different cases of gross weight, c.g.-position and altitude, all parameters which influence the dynamic stability of a helicopter. It should be noted, that the blade modification included small changes in the dynamic characteristics in addition, resulting in stabilizing aeroelastic effects.

In addition to the lift characteristics the pitching moment characteristics of the profile section are of great importance. Shifting the aerodynamic center of an airfoil will result in a change of moment curve slope. Considering a torsionally flexible blade, positive (rearward) a.c. offset is stabilizing, because the effective pitch angle of the blade is reduced, when lift is increased. If there is sufficient flexibility, the elastic deformations act as substantial control inputs (8).

In Figure 14 control power and damping of helicopter pitching are presented as a function of a.c.-position on airfoil section. For better illustration both characteristics are shown in relation to neutral a.c.-position. Rearward positions result in a decrease of control power, while rotor damping is increased. This proves to be a characteristic feature of torsional elastic coupling effects, and it is indicated that the effects are more pronounced if torsional stiffness of blade is reduced. To obtain a more direct relation of the two effects consider the diagram of control characteristics (Figure 15). For satisfactory control behaviour a definite correlation should exist between control power and damping. The recommendations for armed helicopters (9) seem to apply to modern helicopters. This range can be well obtained with rigid rotor systems, and it seems to be quite favourable that this range can be enlarged by aeroelastic effects. In this respect the increase of damping is the more important effect. The slight decrease of control power nearly is insignificant at a rigid rotor system and can be compensated by change of control ratio, if necessary.

Similar to the control characteristics, the stability behaviour of a helicopter will be influenced by aeroelastic effects. Figure 16 shows some results of a study, illustrating longitudinal dynamic stability of the BO 105 at a medium flight speed of 100 kts. The results are shown as a function of a.c. offset from quarter chord. Starting from t_D = 6 sec at neutral position, significant improvements and even stable conditions can be obtained by rearward a.c. position. These effects are again depending on the torsional flexibility of the blade, lower stiffness being more effective. As is shown in Figure 13, a decrease of stick fixed longitudinal stability is typical for rigid rotors at higher flight speeds. Earlier studies indicated that more stable conditions can be obtained by larger stabilizer areas, which however result in increasing rotor loads (6). Figure 17 shows that equivalent stabilizing effects can be obtained by aeroelastic couplings, postponing unstable conditions to higher flight speeds. Considering the normal torsional flexibility of the BO 105 fiberglass blades, the "speed of equivalent stability" is increased by about 60 knots in the high speed range with a.c. of blades rearward. It is indicated by these results that aerodynamic design work on blade section will be worthwile, especially for a hingeless rotor.

Effects of Chordwise Mass-Distribution

An analysis of the torsional equation of motion indicates that the torsion of rotor blades is strongly influenced by chordwise mass-distribution. It can be shown that both inertia forces and centrifugal forces can produce nose-up and nose-down moments, depending on blade position in relation to the plane of rotation. Considering a blade with c.g. shifted toward leading edge, centrifugal forces will produce a nose-down twist, if blade bending is upwards. The amount of twist is proportional to the blade deflection. This means that a strong coupling of flapwise bending and torsion of the blade exists, when the blade is unbalanced in chordwise direction. In fact, this is a blade integrated feedback system.

In order to give a first impression of its efficiency, the dynamic blade response to a discrete gust is shown in Figure 18. The rotor was trimmed at 100 kts and enters a 35 fps sine-squared vertical gust. The figure shows the elastic deflection of the blade tip, as an observer at the hubcenter would see it, when looking toward the blade tip and rotating with the blade. There is a strong difference in blade response between the blades with c.g. at 30%, 25% and 20% of blade chord.

In order to determine the effects on maneuvering and stability qualities of the helicopter, a variation of c.g. position fore and aft of torsional elastic axis was conducted. To show the benefits of the rigid rotor in applying elastic feedback systems, analysis was carried out for an articulated rotor too. The effects of blade c.g. position on control behaviour are indicated in the control-damping diagram of Figure 19. In contrast to a.c. effects, increasing pitch damping is now obtained with forward shift of c.g.. It should be noted that the same improvements in flight behaviour can be obtained whether c.g. or a.c. of the blade is changed. Thus the relative offset is the determinative parameter.

A general overview should consider secondary effects too. A decreasing phase angle between blade control and flapping response is caused by a c.g. forward position, resulting in more crosscoupling effects. For instance, with c.g. at 20% of chord the control phase is reduced by about 10 degrees, while the flapping response due to a nose-up pitching motion is increasing from 5 degrees to about 20 degrees nose right. This is only a slight longitudinal-lateral coupling and will not influence the flight behaviour as a whole.

As we have seen, chordwise overbalancing of the blade reduces the flapping response to all disturbances. If there is sufficient torsional flexibility in the blade, this can be a highly stabilizing effect. Figure 20 shows the pitching moment derivative due to angle of attack for the isolated rotor and for the total helicopter, including rotor, fuselage and horizontal stabilizer. Shifting the c.g. forward by about 5%, the rotor instability is reduced to about 50%, resulting in a neutral angle of attack stability of the helicopter. It is interesting to note that there are only small stabilizing effects with an articulated rotor. The reason is the low moment capability of this rotortype, which makes it nearly insensible to all small control inputs in general and especially to the small elastic inputs.

Regarding the stabilizing effects of favourable c.g. position in static stability and damping, an essential improvement of dynamic stability can be expected for the rigid rotor. Figure 21 shows the stability values of the BO 105 at medium flight speed. A corresponding picture has been shown at a.c. variation in the foregoing chapter (Figure 16). The c.g. position of the blades is very effective, stable conditions will be obtained at about 20% position. Normal torsional flexibility of the BO 105 blades is considered in this case. Therefore, these seem to be remarkable improvements. For comparison, the corresponding values of a helicopter with an articulated rotor and a small horizontal stabilizer are shown in this figure. It must be noted that its basic dynamic stability is better, but there is nearly no effect with change of blade c.g. position.

Further investigations were done for the rigid rotor to examine these coupling effects in more detail. As they are mainly produced by centrifugal forces, c.g. shifts proved to be more effective at blade sections toward blade tip. Also, positive results can be achieved by adding concentrated weights to blade leading edge. For example, by adding a 4-pound weight on leading edge at 0.7 R spanwise position, dynamic stability will improve from t_D = 6 sec to 20 sec at a 100 knots flight speed.

It is quite evident, that decreased flapping response by means of blade control is equivalent to a reduction of rotor blade and hub bending stresses. This will be illustrated by gust loads in Figure 22, where alternating bending moments at the blade root in flapwise and inplane direction are shown. For this diagram it is assumed, that the aircraft will not change its attitude with the gust, i.e. the increment of rotor moment and blade loads by pitching upward of the helicopter is not considered here. As the pitching up tendency of the rotor is increased with aft c.g. position on blade (Figure 20), the differences would be still more significant. It should be noted that alternating moments in steady flight conditions, which are mainly first rotor-harmonic loads, are nearly not affected by blade design. The elastic effects are compensated in this case by somewhat different control inputs, especially in collective and longitudinal cyclic, in order to obtain the trim conditions. It seems to be an important conclusion of these investigations, that in addition to the improvements of flight characteristics a strong reduction of blade bending stresses can be achieved. This will result in an increase of the fatigue life of blades and rotorhead.

Elastic Pitch-Flap-Lag Coupling Effects

At the Bölkow Rigid Rotor System the feathering axis of the elastic blades is fixed in a very stiff hubarm (1). As mentioned before, all blade motions, except of control input, are due to elastic bending and torsion. By means of these deflections out of pitch axis, moment arms for the aerodynamic and inertia forces are produced. Thus the blade system will be subjected to elastic pitch-flap-lag coupling, when control system flexibility is present. The physical principle is illustrated in Figure 23. The forces acting on a blade element can be replaced by the spring-moments at the equivalent hinges, resulting in the simplified relation, as shown in the figure. It is indicated that control system dynamics are influenced by flapping (β) and inplane motion (ζ). The characteristics of the coupling effects depend on flapping, inplane and control stiffness. As inplane root stiffness is normally higher than flapping stiffness, the coupling factor in the simplified relation gets positive. For example, when the blade is coned up over the feathering axis, a forward inplane deflection produces a feathering moment, which causes the blade to pitch down.

Numerous investigations were done, to examine the physical basis of these couplings and their influence on handling qualities of a helicopter. All parameters were included, which influence the flapping and inplane motion. These are mainly precone angle, blade sweep, blade flapping and inplane natural frequencies, lag damping and control system flexibility. The studies have shown that nearly all effects are similar to δ_3-effects. Its physical attribute is an "aerodynamic spring", which influences flapping frequency and thus the amplification factor between control input and flapping response. In general, these effects result in decreasing flapping with both positive and negative coupling, showing a maximum at a definitive factor (11).

The main effects of a positive coupling are: improvement of static and dynamic stability, reduction of control power and damping. Some results are plotted in Figure 24, showing the angle of attack stability and the pitch damping moment of the isolated rotor in forward flight. The destabilizing moments, increasing with speed, can be reduced, for example, by rearward blade sweep angle. Because pitch damping is reduced too, as shown in the right plot, only slight improvements of dynamic stability can be obtained by these effects. This is characteristical for δ_3-coupling. Similar results were obtained by increase of precone angle or by a reduction of inplane natural frequency, for instance. It should be noted that some of these effects could be proved during the development program of the BO 105 helicopter and the Sud Aviation SA-340 helicopter with the hingeless rotor. Improvements of maneuver characteristics and dynamic stability were obtained by corrective actions, such as increase of precone and forward shift of blade c.g. (8; 10).

Finally, the possible range of improvements in flight behaviour due to aeroelastic coupling effects is illustrated for the control characteristics (Figure 25). Increasing flapping stiffness of the blades leads to an increase of damping and control power of the rotor; elastic effects from pitch-flap-lag-coupling will change the characteristics in the opposite direction. In contrast, varying the c.g.-a.c. offset of the blades will result in an increase of damping with decrease of control power. The slope of this relationship is perpendicular to the slope of the change of stiffness. This basic effect results in a possibility of a wide range for damping and control power combinations for rigid rotor helicopters with proper design.

5. CONCLUDING REMARKS

Experience with the hingeless rotor has indicated that the capability to produce high moments is the main feature of this rotor system. The Bölkow system with its fixed control axes and the fiberglass blades of high elasticity offers in addition the possibility to utilize aeroelastic coupling effects, which are highly efficient because of the moment capacity. Theoretical studies and flight test data have shown the following results:

The flight characteristics of a rigid rotor helicopter are strongly influenced by the aerodynamic characteristics of the blade profile section. While increased maximum lift coefficient causes mainly an extension of maneuver-range and a reduction of loads and vibrations, the pitching moment characteristics of the blade are of influence on control and stability behaviour. Shifting the a.c. of torsional flexible blades rearward will result in an increase of rotor damping, improved static and dynamic stability with only slightly reduced control power.

The same effects can be obtained by chordwise overbalance of the blade, i.e. a c.g. shift toward the leading edge. Stabilizing effects can be reached in rotor damping and static stability and thereby in inherent longitudinal stability, while control power is somewhat reduced again. Besides of this, a reduction of blade bending stresses can be obtained, resulting in an increase of fatigue life for blades and rotorhead.

Additional coupling effects are caused by the blades flapping and inplane deflections out of the pitching axis. If there is torsional flexibility in the control system or in the blade root section, these δ_3-similar effects can improve the static and dynamic stability, while control and damping moments are reduced. Coning and sweep angle, as well as inplane stiffness of the blade are active parameters in this respect.

Because of these aeroelastic coupling effects and their dependence on the specific moment reaction to any kind of control inputs, the hingeless rotor system with its high moment capability is well suited to tuned elasticity-control-systems. In general, the effects increase with decreasing torsional stiffness of blades and/or control system. In respect of applying elastic blade integrated feedback systems, the fiberglass technique in blade design proves to be most advantageous. The BO 105 rotor blades can be modified to include any desirable characteristics which can be obtained through incorporation of the effects discussed in this paper.

6. REFERENCES

1. Weiland, E.: "Development and Test of the BO 105 Rigid Rotor Helicopter", AHS, 24th Annual National Forum, 1968

2. Reichert, G.: "Flugeigenschaften bei Hubschraubern mit elastisch angeschlossenen Rotorblättern", Wissenschaftl. Gesellschaft für Luft- und Raumfahrt (WGLR), Jahrbuch 1963 (Translated by NASA-Langley Research Center, NASA-TT-F-11374, 1967)

3. Reichert, G.: "Flugmechanische Besonderheiten des gelenklosen Hubschrauberrotors", Wissenschaftl. Gesellschaft für Luft- und Raumfahrt (WGLR), Jahrbuch 1965 (Translated by NASA-Langley Research Center, NASA-TT-F-11373, 1967)

4. Oelker, P., Huber, H., Winter, K.: "Berechnung von Flugeigenschaften, Stabilitäts- und Steuerungsverhalten eines Hubschraubers mit gelenkigem und gelenklosem Rotor für beliebige Flugzustände", Messerschmitt-Bölkow-Blohm GmbH, Bericht DF 55, 1966

5. Huber, H.: "Berechnung der gekoppelten Bewegungen des Masts und der einzelnen Blätter von Rotoren", Messerschmitt-Bölkow-Blohm GmbH, Bericht DF 92, 1970

6. Reichert, G., Oelker, P.: "Handling Qualities with the Bölkow Rigid Rotor System", AHS, 24th Annual National Forum, 1968

7. Reichert, G.: "The Influence of Aeroelasticity on Stability and Control of a Helicopter with a Hingeless Rotor", AGARD CP 46, 1969

8. Huber, H.: "Das Stabilitäts- und Steuerungsverhalten des gelenklosen Rotors System Bölkow", Deutsche Gesellschaft für Luft- und Raumfahrt, DLR-Mitt. 71-12, 1971

9. Woitsch, W., Weiß, H.: "Dynamic Behaviour of a Hingeless Fiberglass Rotor", AIAA, AHS VTOL Research, Design, and Operations Meeting, 1969, AIAA Paper 69-204

10. Edenborough, H.K., Wernicke, K.G.: "Control and Maneuver Requirements for Armed Helicopters", AHS, 20th Annual National Forum, 1964

11. Gallot, J.: "Effets Aeroelastiques sur les Qualites de Vol d'un Rotor Rigide", AGARD CP 46, 1969

12. Gaffey, T.M.: "The Effect of Positive Pitch-Flap Coupling (Negative δ_3) on Rotor Blade Motion Stability and Flapping", AHS, 24th Annual National Forum, 1968

Fig. 1 BO 105 with the Rigid Rotor System in Flight

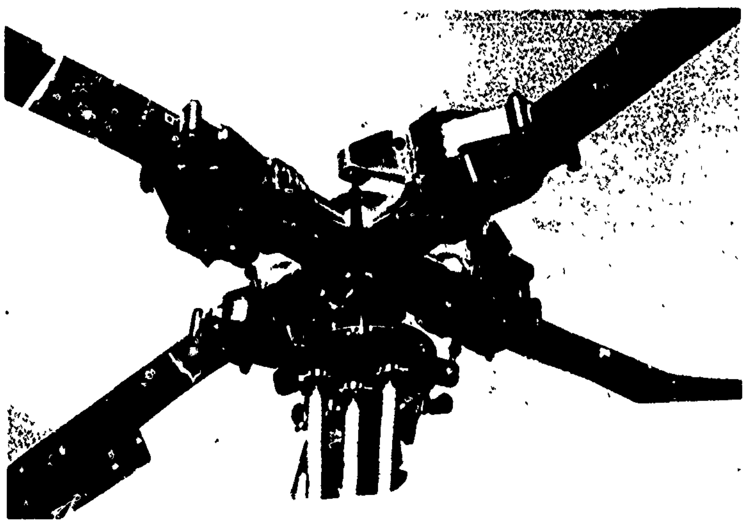


Fig. 2 Rotorhub and Blade Attachment of the Bölkow Rigid Rotor System

YEAR	ROTOR	ROTORHEAD	DIA-METER	NO OF BLADES	BLADE PROFILE SECTION
'63	WIND TUNNEL TEST	STEEL	5.5 m	3	NACA 0012
'65	WHIRL TEST	STEEL	10.2 m	4	0012
'66	FL TEST ALOUETTE II	STEEL	10.2 m	3	0012
'67	PROTOTYPE BO 105	STEEL	9.8 m	4	0012
'67	PRE-PRODUCTION BO 105	TITANIUM	9.8 m	4	0012
'70	CIVIL VERSION BO 105	TITANIUM	9.8 m	4	NACA 23012 MOD.
'69	FLIGHT TEST SA 340	STEEL	10.5 m	3	NACA 0012/0010

Fig. 3 Rigid Rotor Development History at MBB

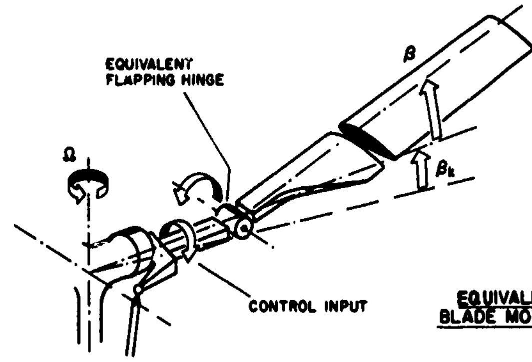


Fig. 4 Analytical Model of Rotor Blade - Flapping

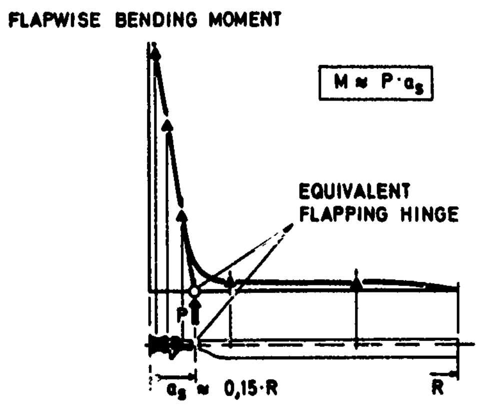


Fig. 5 Distribution of Flapwise Bending Moment on a Rigid Rotor Blade

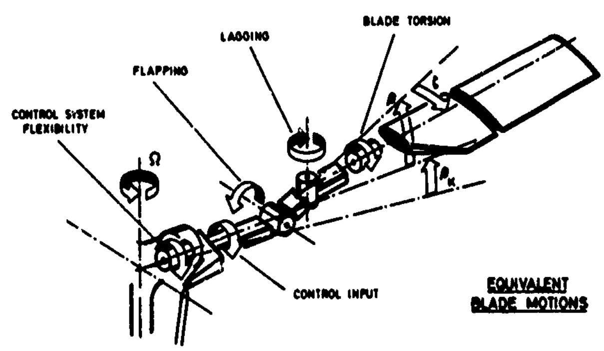


Fig. 6 Analytical Model of Rotor Blade - Flapping, Lagging and Torsion

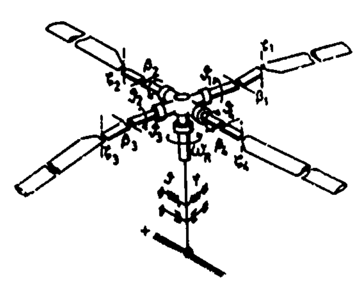


Fig. 7 Coupled Blade-Pylon System

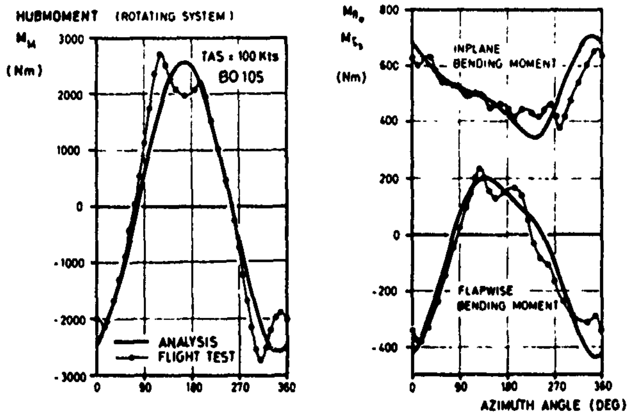


Fig. 8 Hub- and Blade Root Moments in Forward Flight

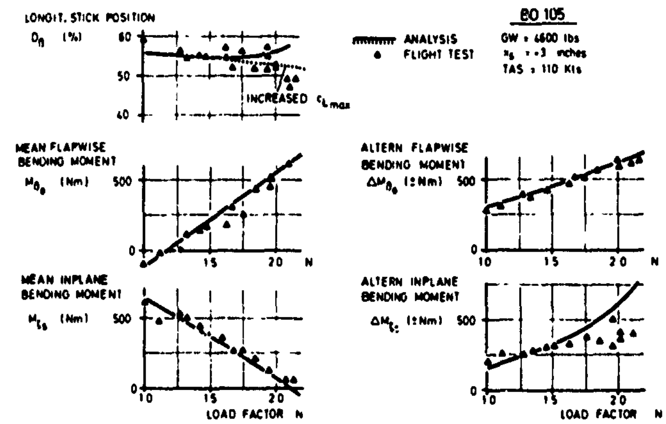


Fig. 9 Maneuver Loads in Forward Flight

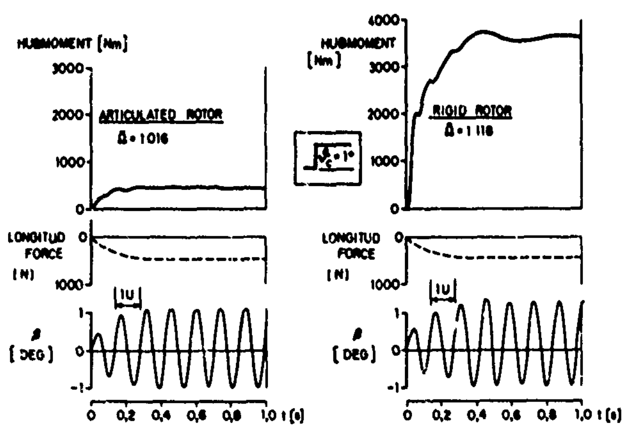


Fig. 10 Time Histories of Moment Build Up due to Cyclic Control Step Input

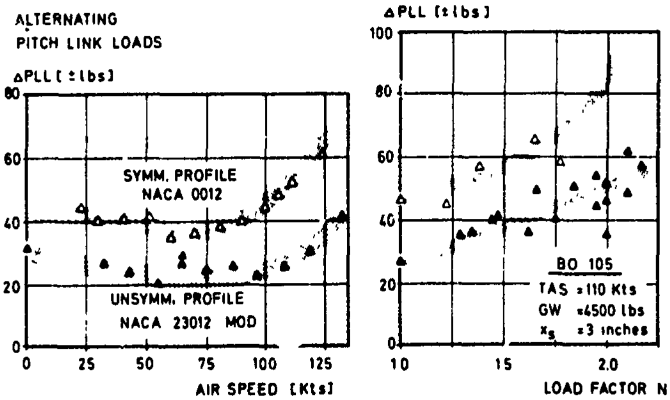


Fig. 11 Alternating Pitch Link Loads in Forward Flight and Maneuvers

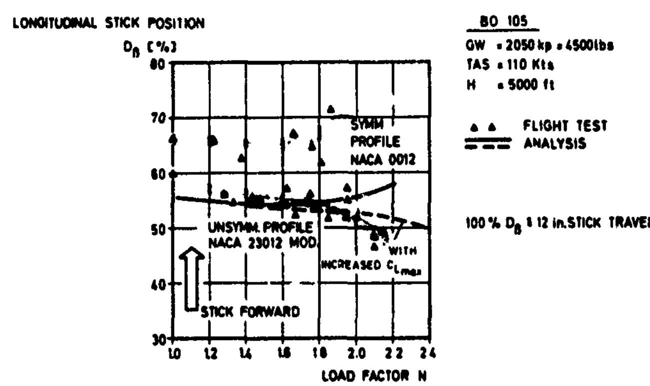


Fig. 12 Longitudinal Control in Maneuvers

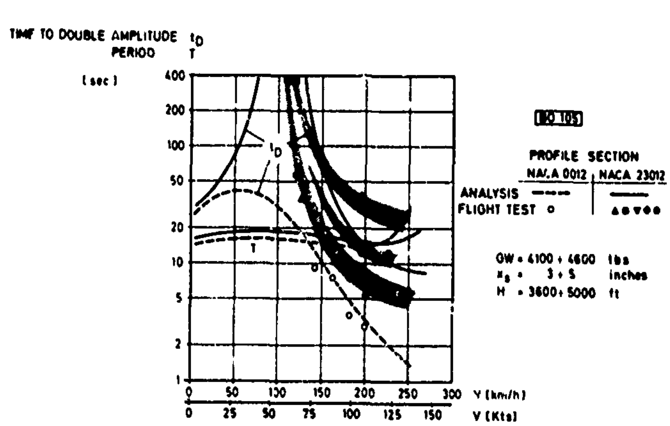


Fig. 13 Inherent Dynamic Stability in Forward Flight

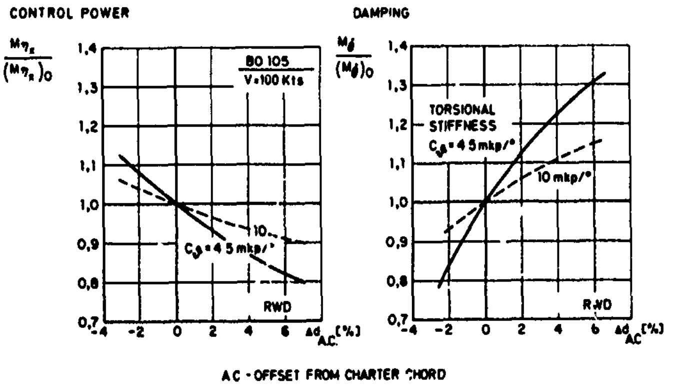


Fig. 14 Effect of A.C. Offset on Control Power and Damping

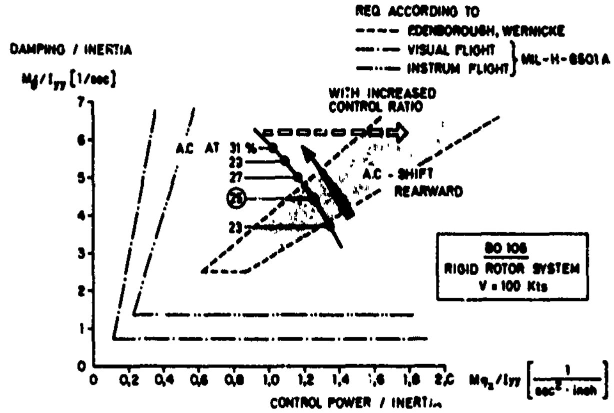


Fig. 15 Effect of A.C. Offset on Control Characteristics (Pitching)

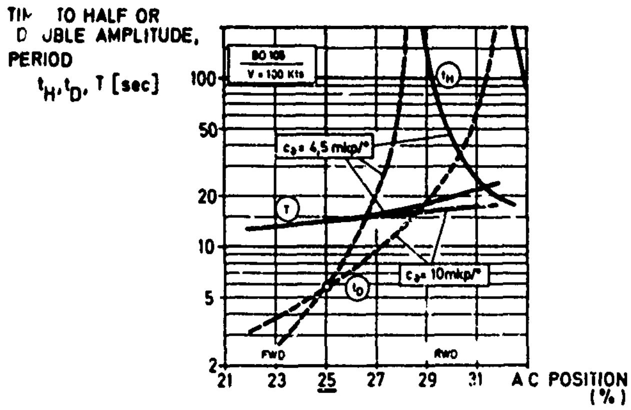


Fig. 16 Influence of A.C. Position on Longitudinal Dynamic Stability

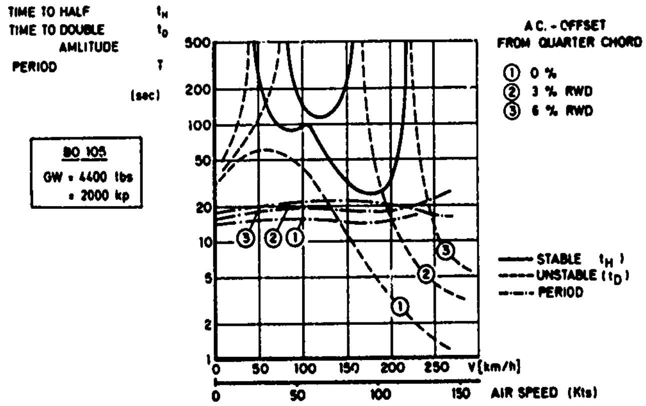


Fig. 17 Effect of A.C. Offset on Dynamic Stability in Forward Flight

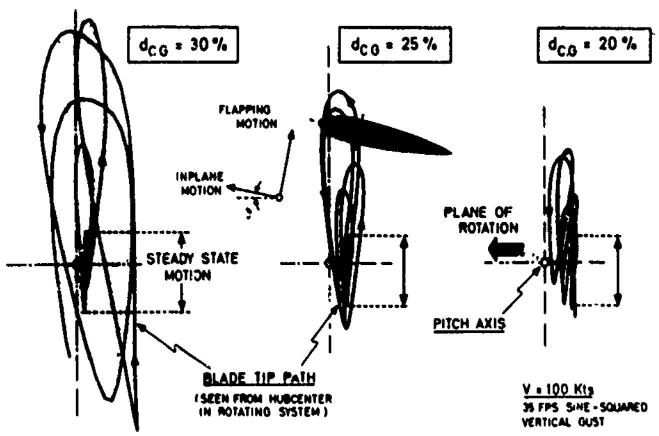


Fig. 18 Influence of Blade C.G. Position on Blade Gust Response

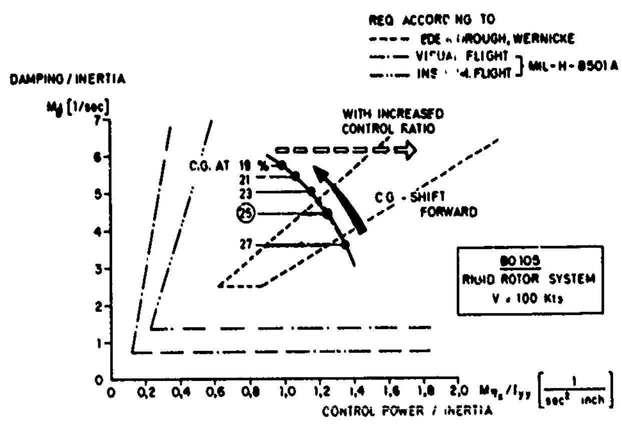


Fig. 19 Effect of C.G. Offset on Control Characteristics (Pitching)

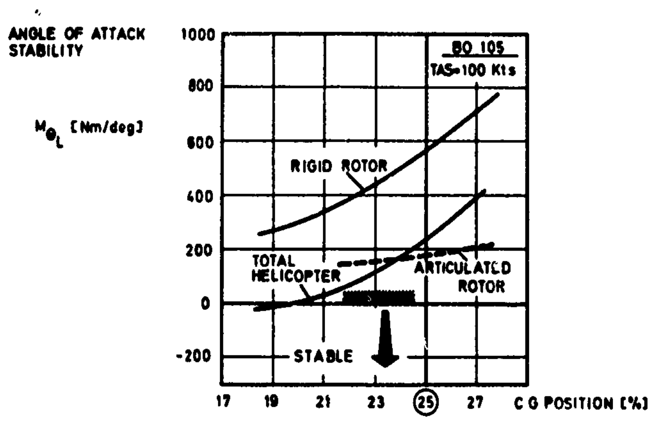


Fig. 20 Influence of C.G. Position on Angle of Attack Stability

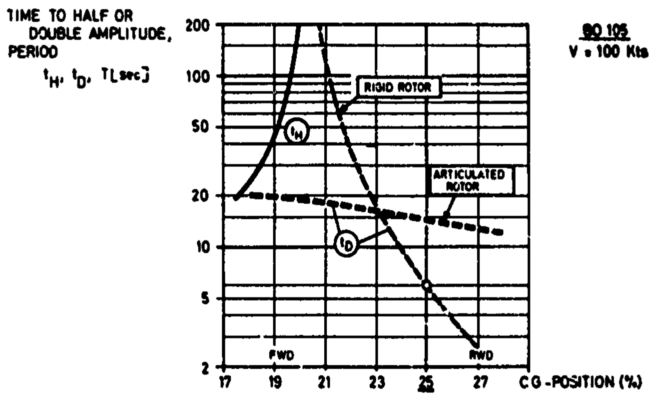


Fig. 21 Influence of C.G. Position on Longitudinal Dynamic Stability

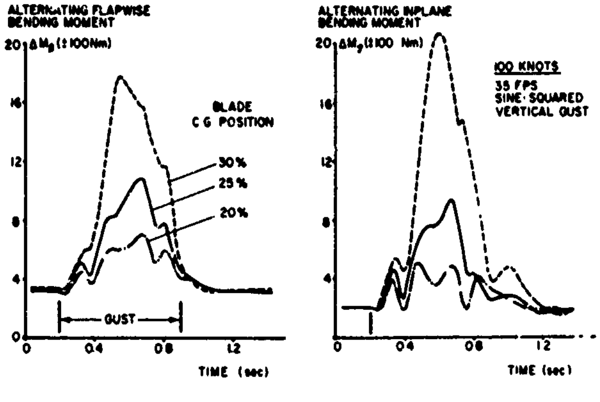
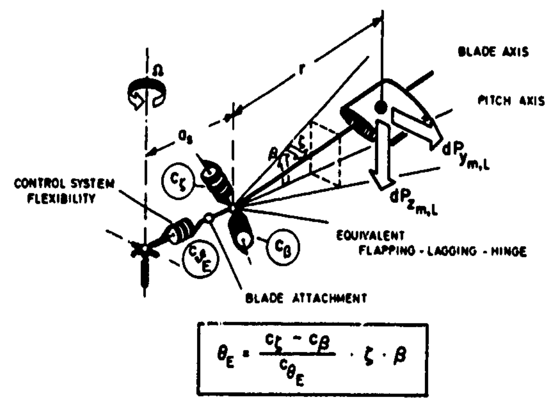


Fig. 22 Influence of Blade C.G. Position on Alternating Blade Loads



$$\theta_E = \frac{c_\zeta - c_\beta}{c_{\theta_E}} \cdot \zeta \cdot \beta$$

Fig. 23 Elastic Pitch-Flap-Lag Coupling

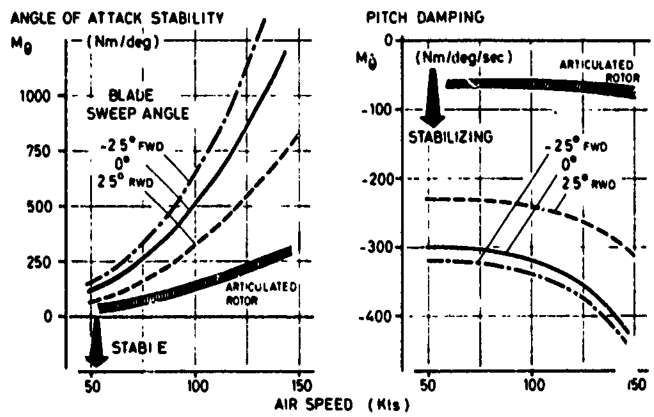


Fig. 24 Pitching Moment Derivatives in Forward Flight (Isolated Rotor)

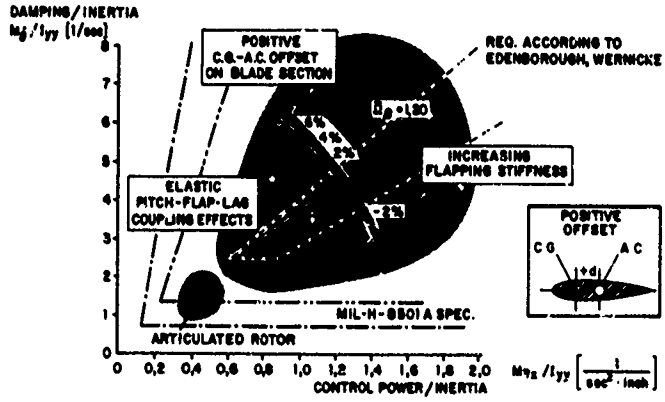


Fig. 25 Range of Control Characteristics with a Rigid Rotor System

GROUND AND FLIGHT TEST EXPERIENCE WITH THE WESTLAND SCOUT HINGELESS ROTOR HELICOPTER

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1. SUMMARY

This paper summarises the flight test experience gained during the basic clearance of a Westland Scout helicopter fitted with a reduced scale version of the hingeless rotor system designed for the Westland W.G.13 Lynx helicopter presently under development for the British Army and the British and French Navies. The basic clearance was aimed at producing an aircraft with sufficient capability to embark upon a series of research tasks and as such was devoted to investigating the airworthiness and handling of the aircraft broadly within the limit of the flight envelope of the standard production Scout fitted with an articulated rotor. Provided that these limits could be approached reasonably closely it was considered that the aircraft would be adequate for its research tasks. The paper concludes with a statement of the present status of flight testing of the Lynx helicopter.

2. INTRODUCTION

Approximately 4 years ago, the decision was made to employ a "soft" in-plane hingeless rotor system on the Westland W.G.13, subsequently named the Lynx. This decision was made following a short but intensive study of seven rotor systems by a Westland/S.N.I.A.S. study group. A fundamental design aim of the Westland Lynx is simplicity and ease of maintenance, and it was these aspects which played an important part in the final choice. Any resulting enhanced flying qualities were to be regarded as a bonus, and in fact strenuous design efforts were made during the conceptual and detail design phases to depart from the behaviour of the conventional offset hinge articulated rotor as little as possible. This attempt was centred upon two main features. These were (i) the minimisation of the control power as expressed by the head moment generated per degree of applied cyclic pitch and (ii) the reduction of feathering moment feedback due to structural coupling between flap, lag and torsional motions, and the feathering moments due to the offsets of the lift and drag forces from the feathering hinge axis.

This basic philosophy led to the concept of rotor system shown in Fig.1. The main root flapwise flexibility is provided by a tapered planform titanium element of elliptical cross-section (the "cutlet") designed to provide maximum flexibility for constant stress under the applied flapwise and lagwise design vibratory bending moments. The feathering hinge assembly uses conventional needle roller bearings, and Bendix tension-torsion bars to transmit the centrifugal loads. The feathering hinge bearings are quite modestly loaded, since the hinge is positioned just outboard of the region of steeply increasing flapwise and lagwise steady and vibratory bending moment. (See Fig. 2) The flexible head element which provides the major lagwise flexibility (the "dog-bone") controlling the fundamental lag natural frequency is positioned between the feathering hinge and the blade root, and is a titanium component of circular cross-section and slightly tapered along its length. It is manufactured integrally with the outer sleeve of the feathering hinge. Provision is made on the Lynx for blade folding at the 2-pin blade root attachment. The relative contributions to the total tip deflection in fundamental rotating flapwise and lagwise mode shapes of the main components of the rotor (the "cutlet", feathering hinge, "dog-bone" and blade) are shown in Fig. 3, which indicates the worthwhile reduction obtained in blade axis displacements away from the feathering hinge axis. However, some deflection is inevitable, and the adverse effects of their influence on feathering moments have been minimised by the adoption of the circular cross-section dog-bone which confers a very good degree of "matched stiffness" of the rotor system outboard of the feathering hinge. As a further attempt to minimise the unwanted feathering moments, a reasonably high value of torsional stiffness of the blade and control circuit between the feathering hinge and the servo-jack earth was specified on the Lynx. The relief of the relatively thin "cutlet" from torsional moment assisted in this area.

A fundamental issue was the choice of first lagwise natural frequency. This was finally chosen to be 0.64R at normal operational speed, and represented the design compromise between a high sub-critical situation which would minimise or eliminate the necessity for augmentation of the inherent structural lag damping for suppression of ground and air resonance and a low frequency which would obviate the problem of large oscillatory lag plane bending moments at 1R frequency generated during manoeuvres which caused significant in-flight rotor speed reductions, such as autorotative landing. It was considered highly probable that it would be necessary to augment the inherent structural damping in the lag plane, and lugs were provided at each end of the dog-bone so that the bending deflection of the component could generate linear motions of an hydraulic damper. That this was a wise provision was indicated when decay tests and forced response tests carried out by accelerating the rotor through its lag resonance indicated the presence of only 0.5% critical damping. The Lynx rotor has however, been run on a ground spinning rig up to the full available twin engine power without the dampers.

After the initiation of the Lynx design programme, the desire by Westlands and the Ministry of Technology for a helicopter for research purposes was growing, and it was considered that a very useful first task would be to gain ground and flight test experience of the Lynx rotor system in advance of the start of flight tests. To this end it was decided to modify two Scout helicopters by the addition of scaled Lynx rotor systems. The aim was to preserve the radial proportions of the major elements, the non-dimensional lag natural frequency and, as far as possible, the ratio of control power divided by aircraft pitching inertia. It was not intended to design a new blade for this exercise, and hence the existing Scout blade was utilised by removing 9.25 ins. from the tip in order to preserve the original total rotor radius.

It was desired to build into the rotor system the ability to vary certain critical parameters. These were cyclic gearing between stick and blade root, pitch/flap (δ_3) coupling, and phase advance angle. The range of values provided in the design are

(i) gearing

a) longitudinal 1 to 2.6 degrees per inch of stick
b) lateral .5 to 1.2 degrees per inch of stick

(ii) δ_3 $+9^{\circ}$ to -17°

(iii) phase advance angle 73°, 67°, and 59°

Fig.4 indicates the rotor head as fitted to the Scout and Fig. 5 gives general view of the aircraft in flight. This aircraft, XP 189, was of pre-production standard.

In order to obtain desirable positioning of the 2nd and 3rd flapwise mode and 2nd lagwise bending mode frequencies, modifications were made to the spanwise distribution of nose balance weight and provision was also made for the addition of a concentrated mass at the position of the inboard antinode of the 2nd flapwise mode. No attempt, however, was made to simulate the first torsional mode frequency. This was due to the high value of the flexibility of the control system to which it was intended to make no modifications.

3. DYNAMIC ASPECTS OF THE CLEARANCE

3.1. GROUND RESONANCE

The prediction of ground resonance behaviour was based on a method which combines the theoretical characteristics of the rotor system with measured natural frequencies and mode shapes of the airframe chassis modes. The measured airframe characteristics were obtained from tests of a separate airframe, ballasted to represent the actual aircraft. This test airframe was fitted with a standard production version skid undercarriage which is very similar in principle to the pre-production standard, but has certain detail differences which were considered not to be significant from a stiffness point of view.

The predictions indicated that about 1% critical blade structural lag plane damping would be sufficient to prevent instability.

The ground testing was conducted in two stages:-

(i) With the airframe braced to the ground to form a stiff support for the rotor system, the rotor was checked for aeroelastic instabilities, obviously appropriate to the hover and near-hover conditions only.

(ii) Ground resonance clearance of the rotor/airframe assembly, with the aircraft in the unrestrained condition. Should however evidence of divergence be detected, the restraining gear could be hydraulically tensioned to place the airframe impedence characteristics into an overall stable situation.

Whilst carrying out the testing described in (i) above, an instability was encountered at 60% lift and a rotor speed of 440 r.p.m. (the maximum test overspeed). An investigation was undertaken to determine whether the incident was rotor system instability, or whether it was due to ground resonance. As a result of the investigations, it was discovered that the fundamental pitch and roll frequencies of the chassis modes were considerably lower than those of the impedance test airframe, and that the lateral stiffness of the tethering gear in the tensioned condition was much lower than the design requirement, and decreased with increase of applied load. Also the rotor lag plane structural damping was lower than anticipated. When these three factors were taken into account, there was a predicted deficiency of 3% critical damping in the lag mode. In order to resolve this issue as quickly as possible so that flight trials could commence, it was decided to augment the blade structural damping by fitting hydraulic dampers of the type previously developed for the Scout articulated rotor. This damper was fitted parallel to the "dog-bone" so that bending deflections of this component would cause linear displacements of the damper. The damper has an initial V^2 characteristic with a cut-off, and provided 8% critical damping appropriate to an equivalent blade oscillatory lag angle of $\pm 3^{\circ}$, i.e. well in excess of the predicted minimum requirement for the suppression of ground resonance. Fig. 6 and 7 indicate the imaginary and real parts of the solution of the relevant differential equations. Fig. 7 has been plotted in terms of the reciprocal of the time to half or double amplitude, and demonstrates that there is no possibility of instability up to a rotor speed well in excess of the maximum overspeed value. This prediction was borne out by the fact that testing of the aircraft revealed no tendency to instability throughout the full rotor speed range and for values of rotor lift up to 1g. Practical tests with the aircraft tied down using normal production Scout centre tie-down facility also demonstrated complete freedom from aeroelastic instability in the near-hover condition and ground resonance up to a thrust state equivalent to the maximum available engine power.

3.2. AIR RESONANCE

Having considered it expedient to fit lag plane dampers in order to remove any possibility of ground resonance, air resonance predictions were carried out assuming the availability of 8% critical damping. Figs. 8 and 9 indicate the imaginary and real parts of the solution of the air resonance equations, and it can be seen from Fig. 9 that complete stability is predicted throughout the full operational rotor speed range with wide margins in both directions. The theoretical predictions are only appropriate to hovering flight, but it is considered that forward flight effects are not essentially de-stabilising. Fig. 8 is drawn for a thrust of 5000 lbs., whereas Fig. 9 shows the results for 5000 and 8000 lbs. of thrust. Flight throughout the speed, altitude, C.G. and manoeuvre ranges have not shown any incipient tendency to air resonance, and the aircraft is stable in this respect throughout the declared flight envelope, including full engine-off autorotative landings.

It should also be stated at this stage that flight tests have not revealed any actual or incipient form of instabilities which are not primarily dependent on coupling between the rotor and airframe motions e.g., blade pitch/flap flutter and pitch-lag instability.

3.3. ROTOR LOADING

3.3.1. Derivation of design loading

Design spanwise distributions of flapwise and lagwise vibratory bending moments were established under which the titanium elements should posses infinite life. This design loading was obtained from a non-dimensionalising procedure using measured data obtained from flight tests of the S.N.I.A.S. Alouette II fitted with the Bolkow hingeless rotor, and the design loadings predicted for the Lynx using a sophisticated iterative performance and loading derivation procedure, where the dynamics of the rotor system are represented by the first 8 coupled normal modes. Aircraft weight, number of blades, rotor diameter, rotor speed and fundamental lag frequency were the basic parameters used. This loading was then used to design the flap and lag flexible elements for infinite life.

3.3.2. Estimation of rotor loading spectrum

Whilst the design loading enabled the datum rotor design to be established, it did not assign a working life to the rotor which must be sufficient to enable the basic characteristics of the rotor system to be verified. Hence an attempt was made to estimate a loading spectrum for the rotor compatible with the envisaged experimental programme.

Comparisons of the design loading, estimated loading spectra, and measured loads are made in Figs. 10 and 11. Fig. 10 shows the flap plane design loading and spectral loading distribution as a function of radial station. Also shown are the oscillatory loadings for 1 g steady flight in moderate turbulence in the speed range 10 to 115 kts. for both neutral and 5 ins. aft C.G. position, and the peak value of oscillatory loading measured during a wide range of manoeuvres for various C.G. positions. These points are the transient maxima for the manoeuvres and represent only a few cycles of loading per manoeuvre. Fig. 12 represents the same comparison for the lag plane, and the two figures show that the measured 1g loading is less than the design loading in the case of the flap plane, and a similar situation is true for the case of lag loading at neutral C.G., whereas the design and measured lag loadings are the same for the 5 ins. aft C.G. case.

3.3.3. Control loads

The concept of the rotor is such that feathering moment feedback due to blade flap and lag deflections is reduced to a minimum compared with the more conventional type of hingeless rotor. This is due to the "matched-stiffness outboard of the feathering hinge" concept. With the feathering hinge encastre at its inboard end the ratio of non-rotating flapwise to lagwise frequency is 0.6. Therefore it was considered that the vibratory control loads should not be substantially increased over those experienced by the articulated rotor. That this was the case is indicated by comparison of steady 1 g flight and manoeuvre loads for the two rotor systems shown in Figs. 12 and 13. In fact, the steady flight loads are generally a little less than those for the articulated rotor Scout, with the manoeuvre loads being substantially the same. It should be noted however, that the cyclic gear ratios are substantially different. (Fig. 16 refers)

3.4. AIRFRAME VIBRATION

The natural frequencies of the rotor system were positioned so that the amplification of the basic aerodynamic forcing loads at frequencies contributing to 4R airframe vibration were constrained to an acceptable level. Fig. 14 is an interference diagram for the datum design standard of blade balance weight, and does not include the effects of any additional antinode weight which the design modifications to the existing blade were required to accommodate.

The major sources of airframe vibratory loading at 4R frequency for a hingeless rotor helicopter are considered to be :-

(i) Vertical 4R shears at the rotor head.
The proximity of a flapwise bending mode to 4R excitation will amplify the basic loading. The major mode of consequence is the 3rd flapwise mode at 4.73R which is well positioned from this point of view.

(ii) Horizontal 4R shears at the rotor head.
4R in-plane excitation arises from the proximity of "hub-fixed" lag plane modes to 3R and 5R excitation. The only mode of consequence is the 2nd lag mode which at 6.54R is very well positioned for minimum amplification of this source of excitation.

(iii) Pitch and roll moments at the rotor head.
4R pitch and roll excitations of the airframe arise from 3R and 5R moments rotating with the rotor. These moments will give rise to significant magnitudes if there are rotor flapwise modes close to 3R and 5R excitation. The 2nd flap mode at 2.[illegible]5R and the 3rd flap mode at 4.73R have adequate separation to avoid excessive amplification in this respect. Fig. 15 indicates the comparison between cabin area vibration levels in the 3 directions compared

with average levels for the production Scout helicopter.

4. AERODYNAMIC ASPECTS OF THE FLIGHT CLEARANCE

The main design features affecting the handling characteristics of the aircraft are those arising from the considerably increased rotor head control moment and damping provided by the hingeless rotor compared with that of the articulated rotor normally fitted to the Scout helicopter. The essence of this is summarised in Fig. 16 which shows comparative "hover" control characteristics for the two rotors. It is apparent that the ratio between the rotor "moment" control power and "force" power is substantially altered. Also indicated for comparative purposes are the corresponding values for the Lynx.

The flying controls of the Research Scout were designed with provision for some latitude in the selection of longitudinal and lateral cyclic gearing both individually and collectively. By this means some freedom of choice in the selection of control sensitivity and harmonisation was available.

The principal objective of the initial flying programme was to establish an engineering and air-worthiness clearance for the aircraft prior to embarking on a programme of specific research tasks. Recordings of appropriate basic aircraft parameters and strain gauges were obtained throughout the flying. However, the general nature of the flight clearance programme precluded the acquisition of systematic flight data to the level of accuracy necessary for detailed technical study, and accordingly only limited trace analysis has been carried out as necessary for the flight clearance. Detailed studies in specific areas will form the scope of future research programmes.

4.1. ESTABLISHMENT OF INITIAL DATUM STANDARD OF CONTROL GEARING

The rotor head moment contributed by the hingeless rotor system is approximately four times that of the articulated rotor of the standard Scout. Since such increased control moment power could have produced excessive control sensitivity with attendent piloting difficulties, and possibly critical structural loads, it was decided to adopt a desensitised cyclic gearing for the initial flights.

The selection of the control gearing was based on the provision of an adequate cyclic control range for aircraft trim and manoeuvre purposes over an initially limited flight envelope. This control range was associated with an available stick travel "box" of 11 ins. fore and aft by 10 ins. laterally, measured at the top of the stick. This resulted in a longitudinal gearing of 1.55 degrees per inch and a lateral gearing of 1.0 degrees per inch of stick. Also a symmetrical lateral cyclic range was initially selected.

Phase advance angle was retained at the same value (73°) as used on the articulated Scout, and nominally zero δ_3 was adopted.

Longitudinal and lateral cyclic stick positions in trimmed level flight up to V_{MAX} (115 kts. I.A.S. at 2000 ft) for 5000 lbs. A.U.W. at neutral C.G. were measured and are shown in Fig. 17. The form of the longitudinal stick position curve is similar to the standard Scout and shows positive static stability above 45 kts. The characteristic "hump" is apparent in the 20-30 kts. region. The lateral stick position also shows this feature but is otherwise unremarkable. The "hump" characteristic is associated with regions of low translational velocity and is attributable to the combined effects of complex rotor induced velocity distributions and the interaction of rotor wake and airflow effects upon the airframe pitching moment characteristics. It was found that the stick displacements required for low speed trim, particularly in the longitudinal sense, were somewhat greater than those required for the articulated rotor Scout. This effect is mainly due to the alteration in cyclic gearing. The initial datuming of zero cyclic resulted in the stick being further forward and to port than on the standard Scout. As a result, the manual cyclic stick forces at high speed, although no heavier than on the articulated rotor, were less easily managed due to the position of the stick relative to the pilot. Also, because of the longitudinal stability characteristics of the aircraft, substantial fore and aft control displacements were required for satisfactory recovery from longitudinal control inputs at speeds greater than 90 kts. where proximity of the forward control stop may have been limiting. However, early flight tests having confirmed the acceptability of the basic control gearing and harmonisation from the handling viewpoint, it was considered that the only change required was a re-datuming of the cyclic stick aft by $1\frac{1}{2}$ inches and starboard by 1 inch.

This control arrangement proved satisfactory for the remainder of the clearance tests. Stick displacements during manoeuvres for all flight and loading states were acceptable, and at no time did available cyclic control range limit the manoeuvre within the required flight envelope of clearance.

All flight tests were carried out at a nominally neutral lateral C.G. position and the stick margins available were considered adequate for the lateral C.G. range of $\pm 1\frac{1}{2}$ inches proposed for the clearance. Occasionally, longitudinal dynamic stability tests at and above V_{MAX} required the application of full forward cyclic for recovery.

4.2. FLIGHT CLEARANCE

The flight envelope objective tested and cleared is shown in Fig. 18. This envelope is the same as that for the articulated rotor Scout for a weight of 5000 lbs. up to a density altitude of 6000 feet, but with the C.G. range restricted to 2 inches forward to 4 inches aft and $\pm 1\frac{1}{2}$ inches laterally. This was considered to be entirely adequate for the research tasks envisaged for this aircraft.

The clearance was divided into 3 main areas :-

(i) Low speed manoeuvres

(ii) Normal flight manoeuvres up to 80% V_{MAX} and

(iii) V_{MAX}

In the course of the clearance, the following positions of longitudinal C.G. were tested :-

1.66" fwd; Neutral; 3" aft 5" aft

An assessment of handling and operation in high winds was also carried out.

4.2.1. Low speed manoeuvres

Evaluation of the aircraft in this area was largely on the basis of the pilot's subjective reports. In addition to the normal low speed manoeuvres, the following were also used to assess low speed handling:-

(i) Advancing "S" turns

(ii) Down wind approaches with late turn to hover

(iii) Sideways "sheep dogs"

(iv) Climbing reverse turns with large power change

The scope and vigour of these manoeuvres were aimed at covering all that is implied by the phrase "normal manoeuvres of an Army tactical helicopter".

No handling problems were encountered during any of the manoeuvres at any loading tested, although stick displacements in some cases were fairly large. However, the available cyclic stick range was always adequate. Large movements of the stick were particularly apparent in the 20 kts. region where control appeared rather sluggish. Basic control characteristics were rated satisfactory and no adverse pitch/roll couplings were apparent.

The aircraft was tested in steady wind speeds up to 25 kts. with gusting to 32 kts. No difficulties in this respect were experienced. Rotor accelerations and decelerations on the ground in these conditions were reported to be superior to the corresponding articulated rotor.

Control response in the hover was assessed by the application of cyclic "pulse" inputs. Figs.19 and 20 indicate the responses to a nose-down input and a port input respectively in the hover.

4.2.2. Normal flight manoeuvres up to 80% V_{MAX}

The four loading conditions were tested through the normal range of flight test manoeuvres firstly at a pressure altitude of 2000 ft. and with the exception of the forward C.G. at a pressure altitude of 6000 ft. The forward C.G. and neutral C.G. characteristics at 2000 ft. were so little different that it was considered unnecessary to test this condition at 6000 ft. In the course of testing at 6000 ft. the aircraft was flown at pressure altitudes up to 8000 ft.

In the speed range up to 80% V_{MAX} the aircraft was easy to fly with satisfactory control characteristics. During rolling manoeuvres there was a nose-up pitching tendency which was readily counteracted with forward stick. Entries to, and recoveries from autorotation were straight forward. Very rapid entry (1 to 2 secs) to autorotation from a high collective pitch setting showed a strong pitch-down tendency. This feature was dependent upon the rate of pitch reduction, and became more marked with increased speed.

The aircraft was initially longitudinally dynamically stable with a tendency to become dynamically unstable as speed increased above 50 kts., the effect becoming more marked with aft C.G. movement. However, in comparatively smooth air the aircraft could be flown for prolonged periods without difficulty. In conditions of moderate turbulence the effects of instability in pitch became more apparent requiring considerably more attention from the pilot in order to achieve accuracy.

Stability in roll was substantially dead-beat throughout the speed range, but at the 3" and 5" aft C.G. loadings the unstable pitch mode tended to intrude at the higher speeds, requiring eventual intervention by the pilot. Corresponding tests at 6000 ft. produced similar results for equivalent airspeed based on a normal density altitude relationship.

The manoeuvres tested included vigorous rolling reversals of $\pm 45^\circ$ bank, wing-overs and turns through 180° up to 60° bank. The tests were carried out at 100% torque at 50 and 80 kts. and were typical of tests to which the standard Scout is subjected. Normal accelerations up to ± 1.7 g were achieved.

4.2.3. Clearance to V_{MAX}

Substantiation of V_{MAX} as defined by Fig. 18 entailed flight assessment of the aircraft progressively up to 1.1 V_{MAX} (130 kts. at 2000 ft; 112 kts. at 6000 ft.). At each speed increment, 30° banked turns were executed to assess controllability and rotor loadings. Above 120 kts. it was necessary to carry out this manoeuvre in a slight descent due to power limitations. Control and stability were also assessed by means of longitudinal and lateral cyclic "pulse" inputs.

In conditions of calm to light turbulence the aircraft was fully controllable to 130 kts. but became progressively more sensitive in pitch as speed increased above 90 kts. Longitudinal stick "pulse" input showed the aircraft to be dynamically unstable in pitch, the rate of divergence above 115 kts. being so rapid as to require almost immediate pilot intervention. Substantial forward stick displacements were frequently required to effect satisfactory recovery from high nose-up attitudes. Tests also confirmed the effectiveness of a reduction in collective pitch as a means of recovery from excessively high nose-up attitudes.

Flight tests to 130 kts. in moderate to severe turbulence illustrated the influence of the pitch instability on the level of pilot work load required to achieve accuracy.

Whilst not wishing to divert attention from the issue of the de-stabilising influence of increased control power, it should be noted that the standard Scout with its 2% offset articulated rotor is dynamically unstable in pitch and displays a vigorous response to longitudinal cyclic stick inputs compared with many other helicopters. In fact, the overwhelming impression obtained by the pilots from the testing described was that the handling characteristics of the Scout fitted with the semi-rigid and the articulated rotors were very similar.

5. GENERAL STATUS OF THE TEST PROGRAMMES AND ITS RELEVANCE TO THE LYNX

The test programme outlined above represented the work carried out to demonstrate that the aircraft had a capability sufficient for it to embark upon a useful research programme. Subsequently, the aircraft has completed a programme of run-on and full autorotative landings with no handling, stress or ground/air resonance problems, the rotor speed at touch-down being allowed to drop to a minimum of 280 r.p.m. (70% N_R). Take-offs and landings on slopes of up to 10° have been accomplished on dry soft turf. The technique used has been entirely normal, and no instability or stress problems were encountered.

It should be emphasised that although variability of cyclic gearing, phase advance angle, and coupling are built-in to the design, the full basic flight clearance was accomplished with the initially chosen values, and no changes were required in order to achieve the flight envelope indicated. It was originally intended to duplicate the hydraulic supply to the single channel servo jacks. This was because it was known that the manual reversion characteristics at high speed of the standard Scout were such that any increase in the basic cyclic control loads would not have been acceptable. However, the duplicated supply system had not reached an acceptable stage of development for it to be incorporated at the commencement of flight testing. Therefore it was decided to undertake manual reversion at frequent stages of progression through the speed and manoeuvre range. This experience, together with the accumulating data on measured pitch link loads convinced the pilot that loss of the hydraulic control system would be no more of an embarassment than that of a standard production Scout, and it was considered fully acceptable to continue the research programme with the basic single channel, single supply system.

The aircraft made its first flight at the end of August 1970 and to date has accumulated 45 hours flying time. It is currently engaged on a programme designed to investigate the influence on handling of the longitudinal control gearing. The second Scout modified to this standard first flew in June 1971 and accumulated 7 hours flying time before delivery to the Royal Aircraft Establishment, Bedford, at the beginning of August.

The Lynx made its first flight in March 1971, and to date (July) has accumulated 20 hours flying time, during which it has achieved a maximum forward speed of 160 kts., an altitude of 6000 ft. and has flown at a maximum A.U.W. of 8,600 lbs. No major problems of handling, stress, and vibration have been encountered, and there have been no signs of any form of ground and air resonance, or any form of rotor aeroelastic instability.

Fig. 21 shows the Lynx in flight, and Fig. 22 shows the main feature of the production rotor head.

It is certainly the case that the background of experience provided by the hingeless rotor Scout programme enabled a high rate of progress to be achieved in the initial programme of Lynx development. In particular it gave the pilots great confidence in the handling qualities of the Lynx when they observed the similarity with the Scout at the initiation of Lynx flight testing. It was recognised early in the design programme and confirmed by simulator and Scout experience that the degradation of the longitudinal stability characteristics due to the increased control power could represent a problem on a high speed pure helicopter. Therefore the decision was made to incorporate the device known as the C.A.C. (computer acceleration control). This device provides an input to the collective pitch servo control proportional to aircraft normal acceleration. The aircraft is currently flying at a C.A.C. gearing of 1.5° of collective pitch reduction per unit increase of normal "g" and this has reduced the pitch instability to entirely acceptable proportions up to the presently achieved maximum speed of 160 kts.

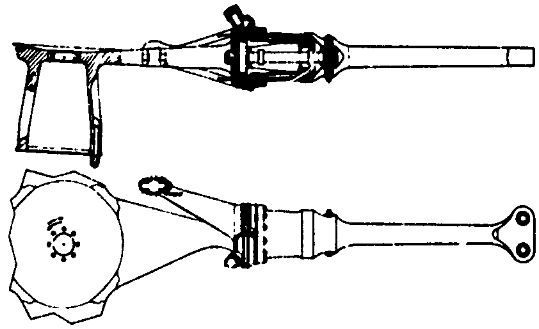


FIG. 1. LYNX ROTOR HEAD

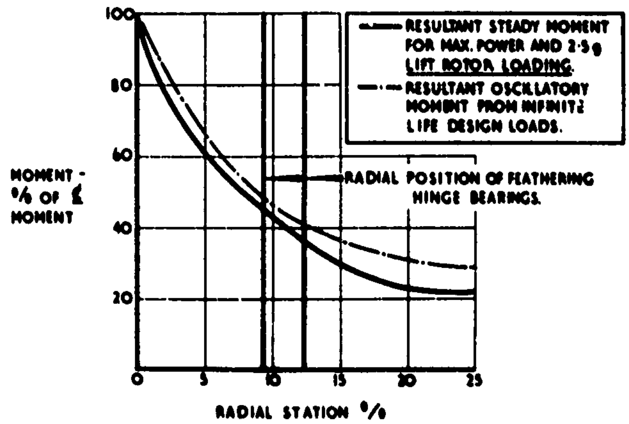


FIG. 2. RADIAL DISTRIBUTION OF BENDING MOMENT

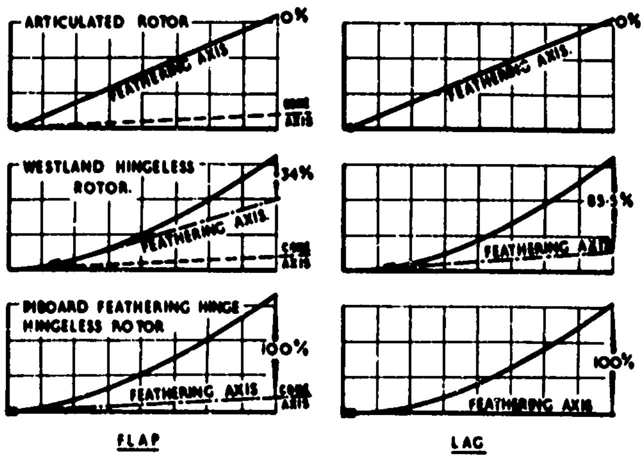


FIG. 3. COMPARISON OF ROTOR SYSTEMS

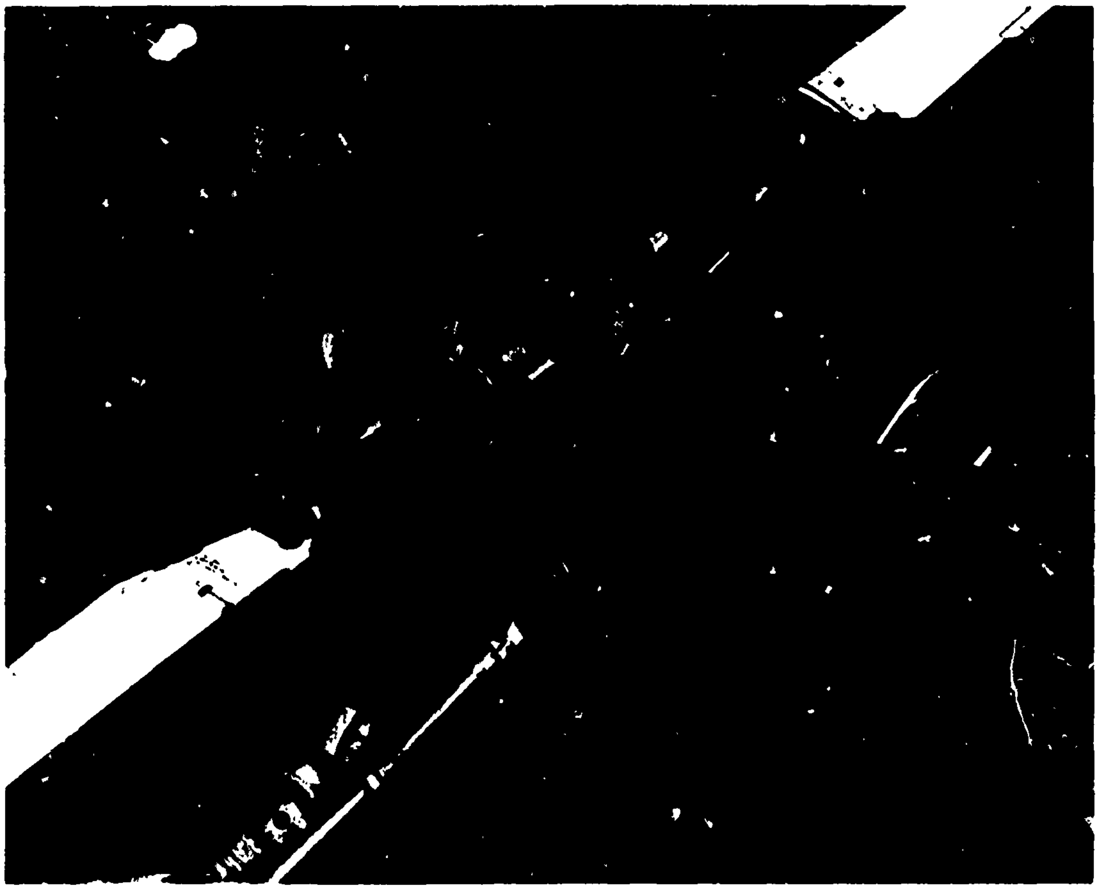


FIG. 4. SCOUT HINGELESS ROTOR HEAD

FIG. 5. SCOUT HINGELESS ROTOR HELICOPTER

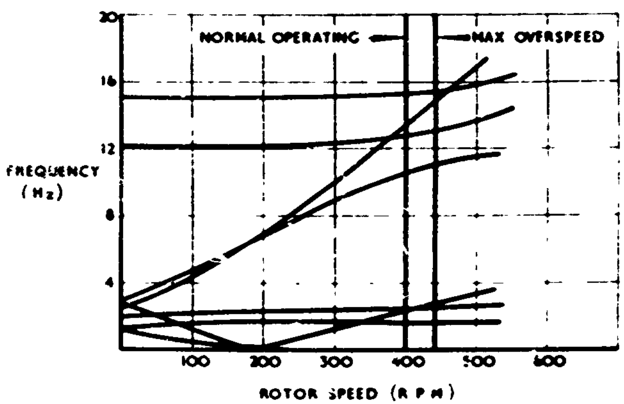


FIG. 6. GROUND RESONANCE CHARACTERISTICS

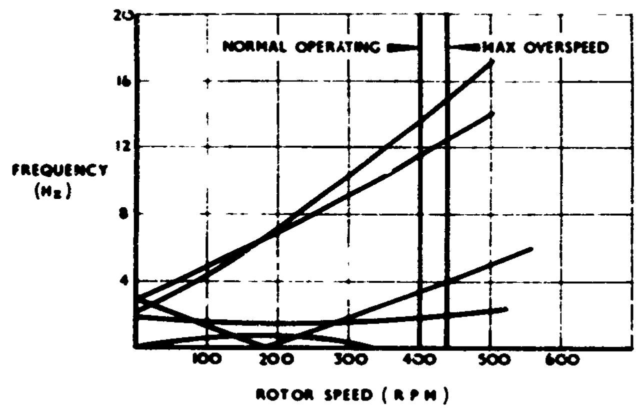


FIG. 8. AIR RESONANCE CHARACTERISTICS

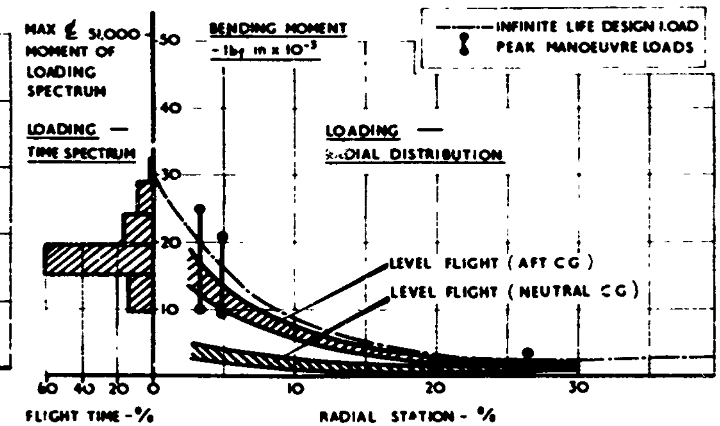


FIG. 10. FLAP ROTOR LOADING. DESIGN AND FLIGHT COMPARISON

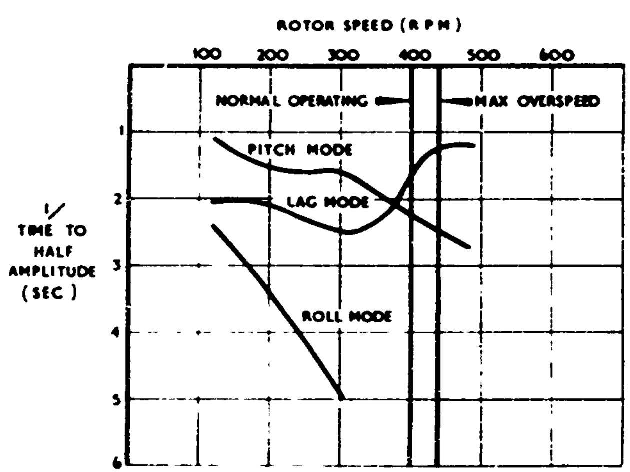


FIG. 7. GROUND RESONANCE CHARACTERISTICS

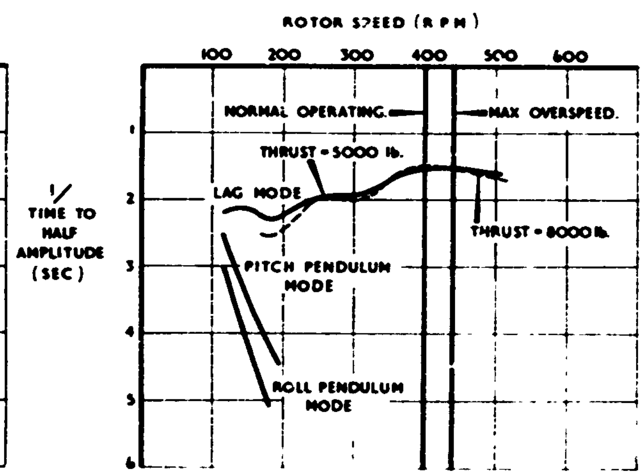


FIG. 9. AIR RESONANCE CHARACTERISTICS

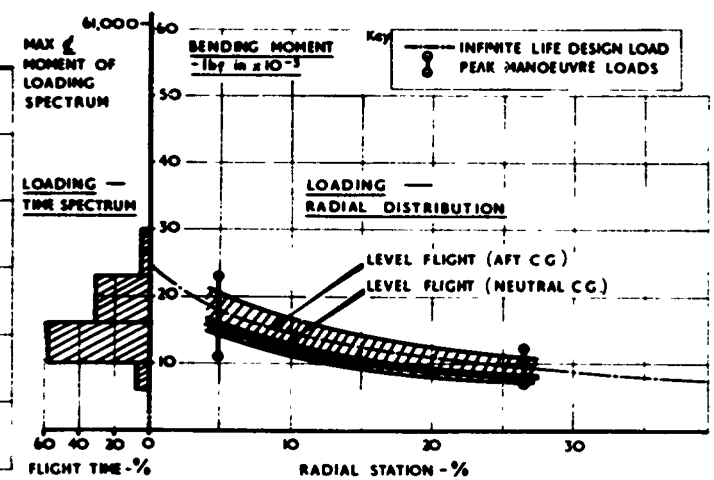


FIG. 11. LAG ROTOR LOADING. DESIGN AND FLIGHT COMPARISON

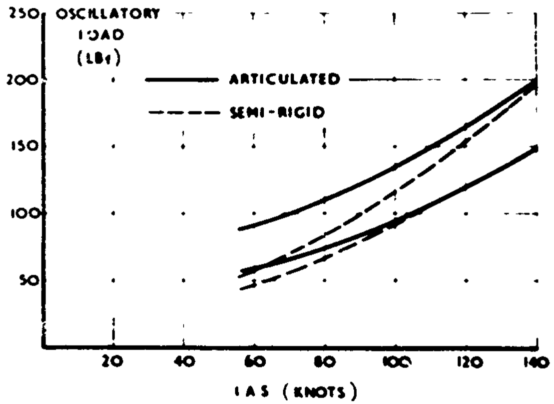


FIG. 12. PITCH CHANGE ROD OSCILLATORY LOAD COMPARISON IN STEADY FLIGHT

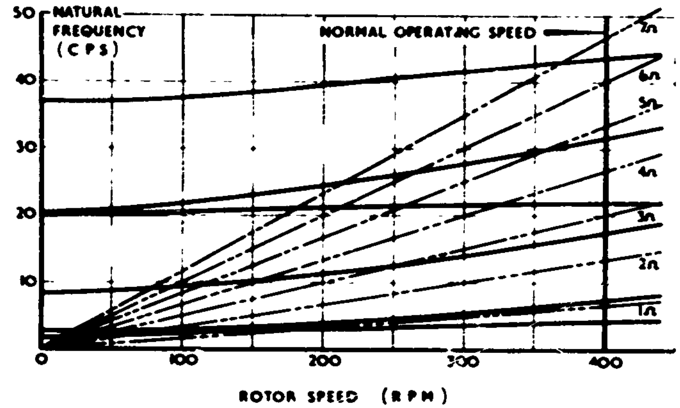


FIG. 14. NATURAL FREQUENCIES OF ROTOR SYSTEM

	SCOUT		WG 13
	STANDARD	INITIAL RIGID	NOMINAL DESIGN
ROTOR HEAD MOMENT lbft/deg	400	1640	3560
TILT MOMENT lb ft/deg	364	315	535
TOTAL MOMENT ABOUT CG lbft/deg	764	1955	4095
LONGITUDINAL CONTROL			
CYCLIC GEARING Deg/Inch	2 6	1 55	1 65
TOTAL MOMENT GEARING lbft/Inch	2000	3030	6760
MOMENT PER INCH / INERTIA	0 552	0 835	0 663
KNOTS PER INCH (TRANSLATION)	18	11	12
LATERAL CONTROL			
CYCLIC GEARING Deg/Inch	1 2	1 0	1 25
TOTAL MOMENT GEARING lbft/Inch	916	1955	5100
MOMENT PER INCH / INERTIA	1 415	3 02	2 5
KNOTS PER INCH (TRANSLATION)	5	7	9

FIG. 16. COMPARATIVE HOVER CONTROL CHARACTERISTICS

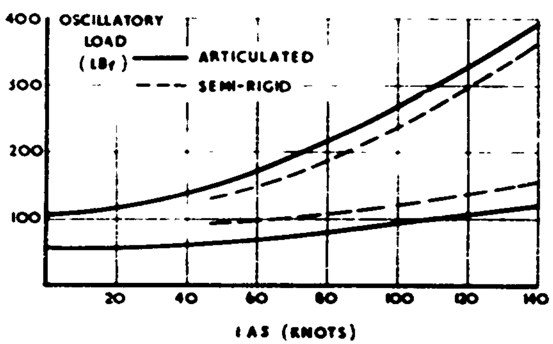


FIG. 13. PITCH CHANGE ROD OSCILLATORY LOAD COMPARISON FOR MANOEUVRE FLIGHT

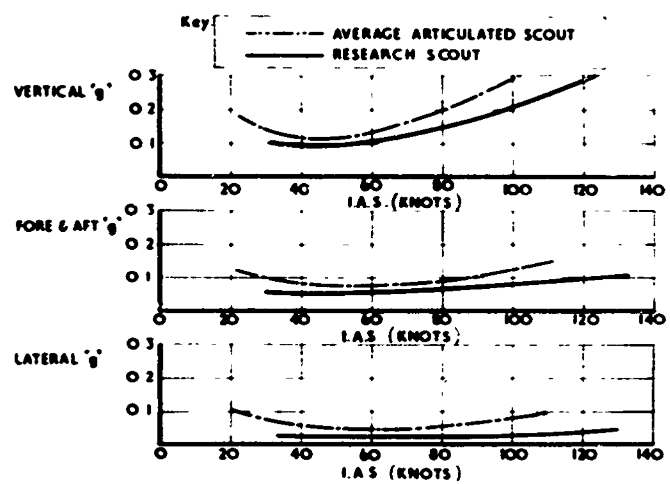


FIG. 15. CABIN VIBRATION LEVELS

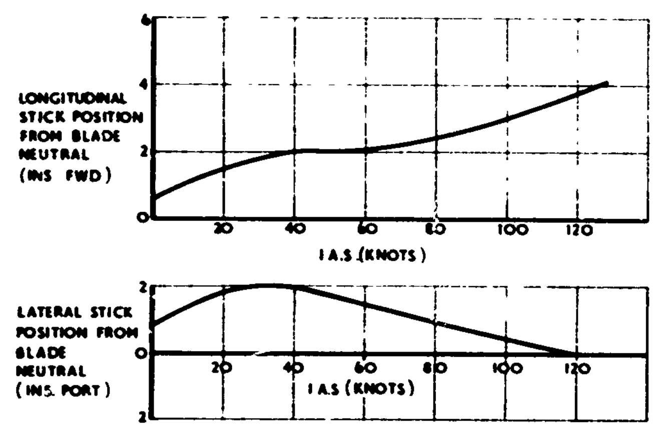


FIG. 17. CYCLIC STICK POSITIONS

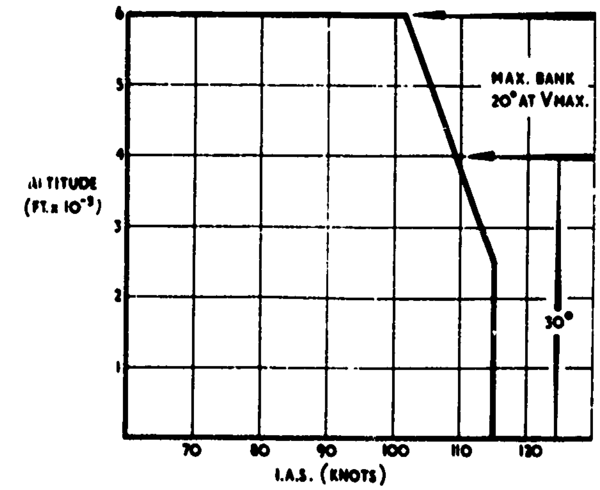


FIG. 18. FLIGHT ENVELOPE

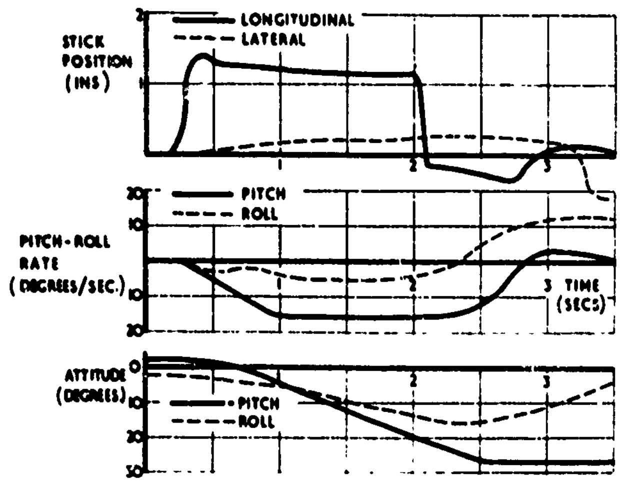


FIG. 19. LONGITUDINAL INPUT IN HOVER

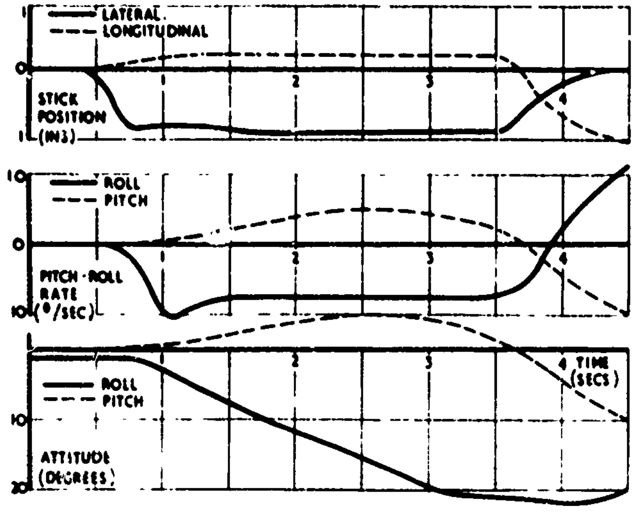


FIG. 20. LATERAL INPUT IN HOVER

FIG. 21. LYNX IN FLIGHT

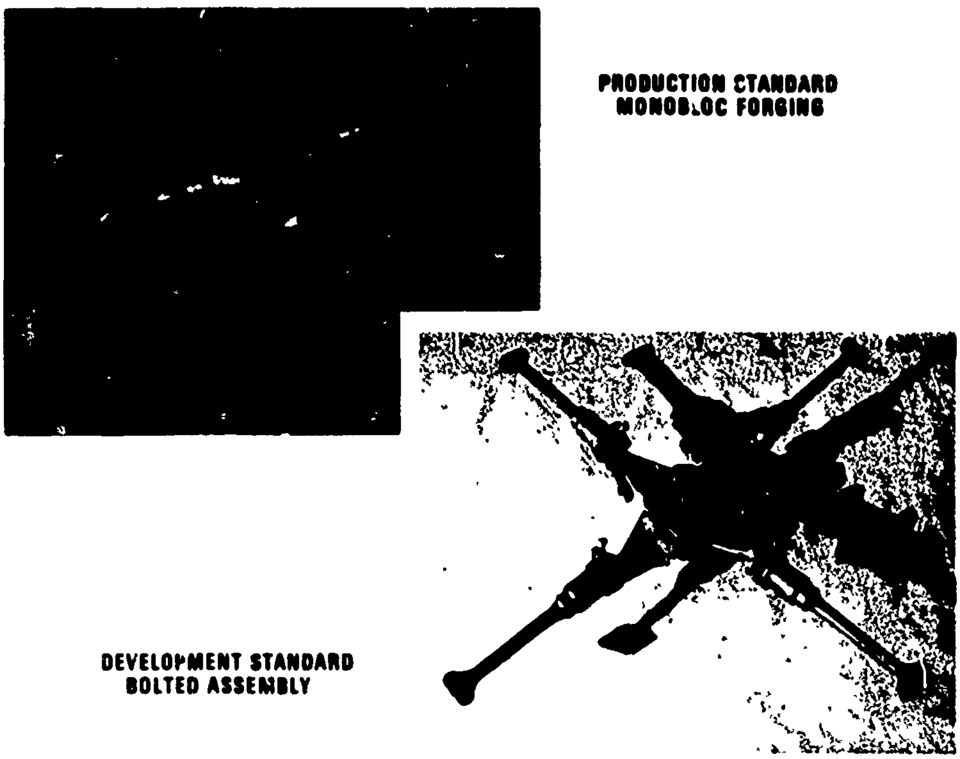


FIG. 22. LYNX ROTOR HEAD

SOME THOUGHTS ON THE SA 341 "GAZELLE" SPEED RECORD

J. SOULEZ-LARIVIERE
Deputy Director for R. & D.
"AEROSPATIALE - Helicopters"

Ladies and Gentlemen,

While speaking from this tribune, I can see amongst the attendance numerous friends already met during the years I have been working on helicopters and V.T.O.L's. I would say they are getting older in the same manner as myself, and this ineluctable fact brings to my mind the idea to try to make us all younger by recalling the pionneering years before the second world war, then the immediate post-war period which has seen, thanks to the efforts of so many enthusiasts, the helicopter coming out of "dreamland" to enter into the field of industry and operation on a wider and wider basis. And, this was the time when it was of good form in a great number of aeronautical spheres, particularly amongst our colleagues manufacturing or operating fixed wing aircraft, to consider with an amused condescension this "toy" just good enough to amuse people but unable to have, one day, any practical interest due to its nature-imposed limitations. Speed was one of these and speed limits were predicated to the helicopter generally in round figures of hundreds of kilometres per hour (at least in the metric countries !). Hundred kilometres per hour, then quickly two hundreds, were considered as being insurmountable barriers. Then, by the force of things, it had to be admitted that the limit had been pushed up to 300 km/hr and now that this barrier has been jumped by numerous record breaking helicopters many people are talking of 400 km/hr as being the new unrealizable performance.

It is in such a context that it seems opportune to evaluate the record just broken by the "last-born" of the Aerospatiale, the "SA 341 - GAZELLE" flying at a speed of 300-310 km/hr.

After having commented it as a fact, I will take advantage of the occasion to say a few words on these famous speed limits and give you, on this subject, my personnal feelings.

The record :

It is the 13th of May 1971, we are at Istres, the airfield of the "Flight test Centre", near Marignane. The weather is fine and a light wind blows at 8-10 knots. Four performances will be recorded during the outbound and inbound flights over the speed base or the closed circuit :

- 3 km at specified low altitude : 312.174 km/hr
- 15 to 25 km at selected altitude : 312.616 km/hr
- 100 km in closed circuit : 295.889 km/hr

then, on the following day, a fourth performance will be made over a 15 km distance after the special fairings, designed for the record breaking flights, have been removed. In the production configuration, with 5 persons on board, a speed of 293.400 km/hr will be recorded.

First, it should be noted that these speeds are not the highest reached by helicopters. The SA 321 "Super-Frelon", as soon as 1959, has exceeded a speed of 350 km/hr and the present record, held by the "Blackhawk", stands at 355.340 km/hr, but these speeds are those reached by heavy helicopters and the large size is a favourable factor as the maximum cross-section increases more slowly than the weight and power. This fact justifies the existence of several categories of records and, particularly a category for small helicopters.

Further, it must be said that the "large helicopters" records were broken by much lightened aircraft and so, if we are considering the fineness parameter "$\frac{Mg \times V}{W}$", which has an economical interest as great as the speed, it can be noted that the record-breaking "Super-Frelon" has a ratio of about 2, but for the SA 341 this figure was 3 to 3.5.

On the other hand, the last performance noted which is not really a record as such, should, in our mind, have a value at least as great. In fact, it is the configuration, very close to the production standard and at the production all-up weight, which has allowed a speed close to 300 km/hr. The only change made for the record and included in this build standard, is relative to the undercarriage fairings. It has proved to be sufficiently beneficial, while simple, to be incorporated, after the record breaking flight, in the production build standard.

The conclusion to the above is that we can, without being boastful, state in the SA 341 commercial litterature the following values : maximum speed = about 300 km/hr; cruising speed : 275 km/hr. This is not a sales argument or a performance out of the reach of production aircraft, it is truly the performance which can be achieved practically by any aircraft sold to our customers.

The preparation for the record attempt

How did we get there ? I feel that the description of the efforts which we have made will surprise many of you. The record attempt was prepared in one month and made in 8 days from the moment the decision to break the standing records was taken. This means that it was an "easy" record, but the most difficult was to take the decision i.e. to admit, for the helicopter, the interest of the speed factor and there, it was all a history and a lot of ideas which had to be changed and that was not accomplished in one day.

In fact, by a quite curious paradox, the blemish assigned, once for all, to the helicopter by the outside world which considers it as being a slow aircraft, unable of speedy flight and full of many other unacknowledged defects, is so well anchored in our minds that even the helicopter operator and manufacturers, amongst whom we are, seem to be fully convinced of such a state of things. This is well marked by the lack of effort in this field, as well from the research aspect as for the design of really fast helicopters.

However, from the technical aspect, an explanation may be found. In fact, up-to-now the problem of the helicopter manufacturers was not to "produce speed" but give to their customers an aircraft capable of vertical take-off, and this type of machine is still nearly the only one to have such a capability, and then transport the greatest possible payload at the lowest cost. But, the consolidated weight of power was very great. When talking of large power installation, looming in the background are the weight of the engine, fuel, transmission components, the reduction gear complexity, the fragility of the whole, the costly maintenance, etc.. and in this litany of complaints there is sufficient work for the designer so that he is not tempted to increase his work load by contemplating a faster machine.

With the piston engines, then the first-generation turbine engines, the helicopter was essentially a V.T.O.L. machine, as well in its design and operation as in the progress required.

Its problems were questions of mechanics, vibration, fatigue, etc... . Speed, aerodynamics, fineness ..? Yes, later, when the other problems would have been solved !.

The speed record broken by the SA 341 is a slight sign that this "later" has arrived and before analyzing the meaning of this fact, let us examine the details of the preparation for the record attempt.

1. First, "Gazelle" has the advantage of having benefitted from the AEROSPATIALE's initial general approach to the helicopter field, that is, the reserve of power. In fact, the experience acquired on piston engined helicopters had shown that the helicopter engines were always great sufferers. Operated close to their power limit, poorly cooled, they were subjected to quite a heavy burden. Further, when the take-off altitude or the temperature was increasing, the engine performance and, hence, the helicopter's was rapidly decreasing. So, when, appeared the "Turbomeca" turbine engines (between 1950 and 1955) which developed more power for a given weight, the AEROSPATIALE's policy was to install an overpowerful engine on a given helicopter while trying to improve the payload by taking advantage of an engine of equal power but lighter. The bad-minded people are saying that, in fact, it was the only way to have satisfactory engines, but this is very biased and, truly, the "Turbomeca" engines has proved to be excellent and reliable light weight engines.

   The piston-engined "ALOUETTE I" had an installed power of 150 KW for an all-up-weight of 1150 kg; the ALOUETTE II ARTOUSTE, 260 KW for 1350 then 1500 kg. The ALOUETTE II ASTAZOU disposes of 400 KW and "GAZELLE" has 450 KW for 1700 kg. If, to go fast, power is not the only requisite, it is certainly necessary. This overpower condition, which is not found on any other helicopter in the world, is therefore well a primary condition.

2. Further, in the same spirit, the rotor characteristics are very favourable to speed; the blades, sufficiently wide to allow taking-off at great altitude, gave on the ground a good margin relative to the retreating blade stall. The blades, made of glass fiber and partly of carbon fiber, are withstanding exceptionally well alternate loads and thus can accept the strong excitations generated at high speed. It is to be noted that the rotor head is not fully rigid, but only semi-rigid. To follow the fashion and also for its well-known inherent advantages, the machine had been designed and built with a rigid rotor head, but this configuration was found to be unacceptable as soon as the speed was exceeding that of present helicopters and it has been necessary to come back to a conventional flapping hinge and be content with a rotor head rigid in the drag plane only.

3. The general shape of the aircraft was, without any doubt, a progress relative to the former generation. But, we must admit that esthetics and fashion, which were no longer to the "string bag" helicopter of the heroic era, had much to do in the decision to design a pleasant looking aircraft. For example, it is to be noted that in spite of our aerodynamists, who where loudly claiming that "the undercarriage drags awfully", the general reaction was to think that it did not matter. Think of it ! a faired undercarriage, that will be heavy and costly !.

In the same way, the engine was well along the centre line of the aircraft resting on its skids, but, unfortunately it was not at all oriented along the local flux, deviated by the general nose down attitude of the aircraft and the deflection in this rear section of the fuselage.

4. At last, with the shrouded tail rotor it was possible, in cruising flight to transfer the anti-torque function to a fin which is much quieter and free from all kinds of limitations. M. Gallot has come here to make a lecture on this subject.

We are now at the beginning of 1971. When finally it was admitted that "Gazelle", after all these detailled arrangements, was no longer an aircraft so ugly nor so slow as first thought. In 1970, dive trials had shown that the behaviour of the aircraft at high speed was perfectly sound and the interest of an attempt to break the speed record for the relevant category began to be realized. The rest was easy ; order was given to the aerodynamists to extract from their files the results of the drag tests. They were granted a small testing period in the wind tunnel to determine how the fuselage drag could be reduced; manufacture, test, record attempt, all was quickly done. On the figure it can be seen that at the usual negative incidence angles, the parasite drag was practically reduced by half (0.9 down to 0.45) by some simple modifications. The remainder of the aircraft was not changed.

First, the undercarriage having a considerable effect due to its own drag and also its moment tending to increase the nose-down attitude and, hence, the overall drag.

Then the engine and main gear box assembly for which the conventional aerodynamic procedures are fully applicable. A "turtle deck" was installed around the main gear box and blending with the engine. The rotor head was faired in by a "tray" covered by a "sea-urchin", whirling with the rotor and thus, a substantial reduction was obtained. It is to be noted that the wind tunnel tests, confirmed in flight, had shown that it was necessary to blank off the fairing holes to obtain a good efficiency.

5. Lessons to be drawn from this record for the helicopter future

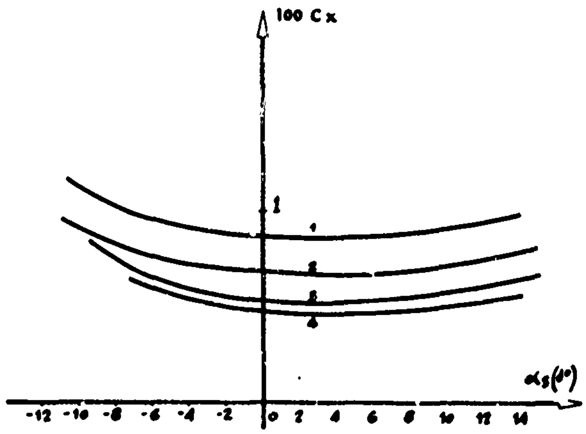
Will I fall in the trap I denounced at the beginning of this paper and will I predict a speed limit for the pure helicopters (400 km/hr ?); no I refuse although such a limit exists surely. But, we are forced to admit that this SA 341 has brocken a speed record nearly "without wishing it". It is an aircraft, designed as in the past to do some hovering and look nice, which has been capable to realize such a performance. But, there are so many aerodynamic resources which are still unexplored if we wanted really to go faster !.

But, do we really want it ? are the operators wishing it ? Instinctively, I would tend to give an affirmative answer, as these operators are beginning to think in terms of "tons/kilometres" instead of flying hours. I would answer also that the SA 341 materialize the boundary between the helicopters designed for hovering and those which shall be optimized in the future to have an equilibrium between hovering and cruising. But, I feel also that speed alone is not sufficient to interest the customers if the price to be paid is too high, particularly for the fineness. In fact, if the fineness is not improved, the increase in speed will be paid by a power installation which will grow as the cube of this speed and a range which will decrease in the same manner, except if all the payload is fuel. In the future, therefore any speed increase must be linked to an improvement in fineness. And this is also a trend amongst the customers, the faster is the flight, the farthest one wants to go and vice versa. And on this point, there is much to do, as, once again the parasitic drag increases at the cube of the speed if nothing is done to reduce it.

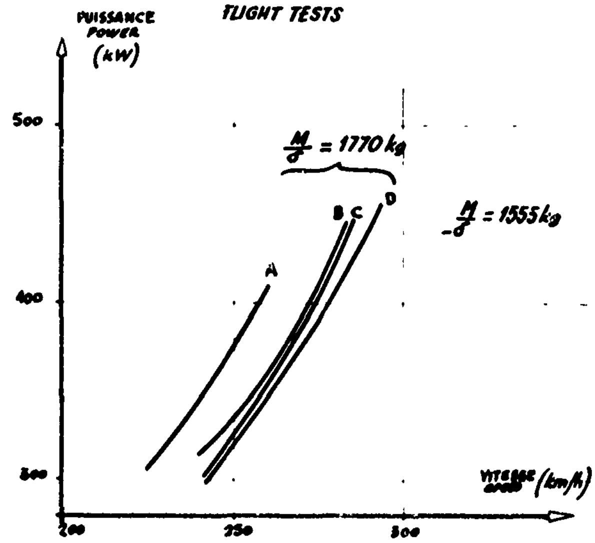
But, there is some hope. First, I do not see any reason for the helicopter fuselage drag being inherently much greater than that of the equivalent fixed-wing aircraft; carefully work-out the shapes, retract the undercarriage, maintain a level attitude in cruising flight, and these are some obvious remedies not however always easy to realize. Then, the propulsion efficiency of the rotor is excellent and better than any other propulsion means (propeller or jet) of the fixed wing aircraft. At last, there remains the rotor, the aerodynamics of which are in 1971 nearly the same they were in 1935, that is the aerodynamics of a rotor designed for hovering. There is a wide field of research open to us tc find rotors capable to work efficiently and with a good fineness at high tip speed ratios. The first results obtained in France from systematic tests run in the Modane large wind tunnel have shown that we are still far away from the barrier.

Certainly, for the helicopter in its present form with its nearly-vertical axis, this barrier exists, would it be only for the disc inclination required to propulse a body at high speed. But on this point, we have in front of us the derived formulae, the compound and particularly the convertible aircraft, which will be capable to take over and which, for us, are still helicopters. But, this is an other story.

SA341 RECORD
MESURES de TRAINÉE en SOUFFLERIE
DRAG MEASUREMENTS IN WIND TUNNEL



SA.341 RECORD
ESSAIS EN VOL
FLIGHT TESTS



1 – Appareil actuel
Aircraft in present configuration

2 – Appareil actuel sans train
Aircraft without landing gear

3 – Appareil actuel sans tête rotor
Aircraft without rotor head

4 – Appareil actuel sans train, avec cheminée à plateau et carénage moyeu
Aircraft without landing gear, with rotor head fairing and rotor hub fairing

A – Appareil de série
Production aircraft

B – Appareil de série avec train caréné
Production aircraft with landing gear fairing

C – Appareil de série avec train caréné et cheminée à plateau
Production aircraft with landing gear fairings and rotor head fairing

D – Appareil de série avec train caréné + carénage moyeu (trous ouverts)
Production aircraft with landing gear fairings + rotor hub fairing (open holes)

PROGRESS IN ROTOR-BLADE AERODYNAMICS

by

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SUMMARY

There has been a continuing effort over the past few decades to develop the technology of rotary-wing aircraft. However, these efforts have been, for the most part, directly oriented toward the operational evaluation of a particular concept or configuration rather than the acquisition of future technology. Relatively little consideration has been given to research aimed at understanding the phenomena and to development of analytical methods for predicting the aerodynamic forces and moments. At an AGARD meeting in 1967, Professor J. P. Jones assessed the status of the rotorcraft and concluded that only the very best configurations could survive in direct competition with fixed-wing aircraft. He suggested that inadequate understanding of the aerodynamics of the rotor constituted the primary factor limiting improvements in rotorcraft performance. With Professor Jones' presentation as a point of reference, this paper discusses those areas of rotary-wing aerodynamics which still pose the most perplexing problems and those areas where aerodynamic improvements are likely to have the largest pay-off in terms of improved design and performance of new aircraft.

In the authors' opinions, the primary factors inhibiting the performance of current rotary-wing aircraft are the following:

1. Aerodynamic limitations leading to excessive power demands, loss of lift and propulsive capability, and saturation of control range,

2. Unsatisfactory stability characteristics and handling qualities, and,

3. Restrictions imposed by vibration and fatigue considerations.

These problems are examined and discussed with respect to developments, particularly during the past four years, in the aerodynamics of the rotor, the mathematical modeling of its wake, and the prediction of dynamic airloads and their effects on flying qualities. An attempt is made to correlate the statement of these problems with research recently conducted and currently underway. The paper reviews recent developments in rotor flow studies, rotor-blade pressure distributions, rotor-blade boundary-layer analyses, airfoil behavior in rotors, rotor-blade dynamic stall, unsteady aerodynamics of rotors, and the aerodynamics of new rotor configurations. Progress over the past four years towards solution of the problem areas is discussed in the light of comparisons between theoretical and experimental data. Results of past and current research efforts are presented as foundations upon which projections are based. The paper concludes with a suggestion for a coordinated treatment of aerodynamic research of rotary-wing aircraft aimed at improving the overall performance potential of this class of vehicles.

INTRODUCTION

Four years ago (1967) at an AGARD meeting jointly sponsored by the FMP and FDP, Professor J. P. Jones presented a most scholarly assessment of the rotorcraft titled "Rotor Aerodynamics - Retrospect and Prospect" [1]. This paper concluded that, except for the ability to rise vertically, most of the advantages of VTOL had been eroded away by sheer engineering progress and only the very best configurations would survive in direct competition with fixed-wing aircraft. The prominent use today of the helicopter, in the absence of other operational VTOL aircraft, has proven that the ability to rise vertically and hover efficiently is a most effective attribute in maintaining the helicopter's competitive position.

However, this efficient hover capability is obtained at the expense of translating the aircraft through the air with the plane of its low disc-loaded rotor nearly parallel to the freestream. This condition, as is well known, leads to a myriad of problems which, formerly, were either generally ignored or evaded by empiricism and engineering approximation. Many compound and composite configurations have been tried to improve the speed and performance of the helicopter to approach those of fixed-wing aircraft and a few new concepts are still subjects of research. It is of interest to note that these use either very lightly loaded or very heavily loaded rotors (e.g., tilt propeller or lift fan/lift engine) for vertical flight indicating the difficulty of combining efficient vertical flight and efficient forward flight.

It is evident that the helicopter not only enjoys a prominent place in current operational aircraft inventories, but that it will continue to be prominent, if not in simple form, certainly as a compound. Therefore, it is logical to ask what is being done to maintain or improve the competitive position of helicopters.

Without repeating the detail of Professor Jones' eloquent treatise, a summary of the major points of the paper is in order to establish a point of reference. Perhaps his most significant conclusion was that the field of rotor aerodynamics, including, of course, dynamic and aeroelastic effects, offers the most productive area of research. This fact arises primarily from the advent of the shaft-turbine engine which permitted fuselages to be designed in an efficient aerodynamic shape, thus reducing the large parasite drag resulting from earlier configurations. It is encouraging to see many rotorcraft emerging that clearly show the efforts of good aerodynamicists who have finally prevailed over the cut-and-try design engineers in reducing appendages and appurtenances which long have shocked and dismayed fixed-wing-aircraft designers and aerodynamicists.

Having concluded the importance of rotor aerodynamics, Jones assessed the basic problem areas as emanating from the following: non-uniform spanwise loading resulting in heavily loaded blade tips; need for small blades to minimize profile drag, resulting in high loading coefficients; variations in relative wind magnitude and direction with azimuth resulting in fluctuations in both lift and pitching moment; compressibility effects and flow separation on both the advancing and retreating blades; and reverse flow on the inboard sections of the retreating blade in forward flight. It was concluded that these problems were extremely difficult to resolve because the available theories were inadequate. This inadequacy was attributed to the highly nonlinear properties of fluid flow through the rotor making not only formulation difficult, but solution so complex that even the larger high-speed computers were taxed beyond capacity. Theories of 1967 were simplistic in nature and incapable of sophistication to account properly for the nonlinear flow and wake interactive effects. Thus, a new comprehensive theory which calculated simultaneously nonlinear aerodynamic, aeroelastic and dynamic effects was indicated as required if real progress was to be made in pushing forward rotor speed and efficiency.

It was also indicated that new facilities would be required, along with new experimental techniques, to validate the theories generated. Jones felt that wind-tunnel testing, restricted to small scale and fixed base, was incapable of scaling and simulating the dynamic response of flight and, therefore, he envisioned new flight techniques, ground-test vehicles and highly sophisticated instrumentation. Larger, specialized computers were projected, some of hybrid design, to permit simultaneous solution of nonlinear, interactive mathematical formulations and for coupling simulations to actual flight-vehicle components.

Finally, Jones sagely observed that research for research sake would not always be valuable; research should only be pursued as a buttress for engineering improvement, not simply for a new generation of vehicles or a new speed range.

What, then, four years later is the state of rotor aerodynamic technology? Has the aerodynamicist, and his kinsman the aeroelastician, risen to the challenge? Has the designer been provided with new criteria formulated by new, sophisticated mathematical models which adequately account for nonlinear flow parameters, wake interaction and boundary layer characteristics? Have the fundamental arguments of two-dimensional versus three-dimensional models, short versus long bubble transition characteristics, lifting-line/momentum models versus lifting-surface/trailing-edge-vortex models been resolved? Can the aerodynamics at or near the separation boundaries, where the rotor blade usually works be predicted? Have new, "transonic" airfoil sections been developed to offset the penalties of higher forward speeds? What is the hope for more efficient, faster helicopters, simple or compound? These questions form the basis for this paper.

In 1926, Glauert developed the fundamental rotor theory based on the classical blade element concept coupled with simple momentum equations. A year later Lock added the supporting analysis that provided the foundation for performance predictions of rotors. Even so, after 45 years we still find that rotor static thrust cannot be well predicted, that the operational limits of rotors are still not clearly defined, and that the vibration and acoustic excitations of rotors cannot be accurately quantified. The fact is that not until the last ten years, and particularly the last four, has any real progress been made in the basic understanding of the rotor performance problem.

Over twenty approximations entered into the establishment of the original blade element concept. As is indicated in Figure 1, these constraints were removed very slowly over the years until the introduction of high-speed computers which provided the capability to model the problem in the required detail and to process the vast amount of data necessary to understand the test results and to correlate them with analyses. Although there has been substantiated improvement in the theory for predicting rotor performance, much remains to achieve a satisfactory level of understanding. The limitations still remaining in the mathematical representation of the problem appear to be associated largely with the inadequate model of the rotor wake and with the lack of consideration for the effects of three-dimensionality of the flow field and unsteady viscous effects in the boundary layer. The complex array of parameters is still too difficult to consider in its entirety, even with modern high-speed computing equipment. Computer programs are available or are being developed which separately represent considerations such as free wake, variable inflow, unsteady aerodynamics, and tip effects. However, each of these is a large and time consuming program to run. At present, there is not a single program which combines all of those elements required for satisfactory prediction of performance or maneuvering. The evolution of computers such as ILLIAC IV now being developed, capable of simultaneous solution of nonlinear equations, might offer the needed capability.

With the helicopter becoming larger and more complex and the cost of development testing growing correspondingly, it is becoming increasingly imperative that analytical tools be made available to design engineers to enable them to make reliable predictions. Consequently, simple actuator-disc methods of predicting performance have been replaced by complex wake models attempting to represent the details of the wake and the vortex trajectories. Lifting-surface representations are currently under development for the blade aerodynamics. Many studies are being pursued in an attempt to understand the fundamental mechanisms of the rotor-blade boundary layer. However, most efforts still are directed toward the operational evaluation of a particular concept or configuration rather than the acquisition of future technology. Applications requirements have driven the work that did not await technological development, with relatively little consideration given to research aimed at understanding the phenomena and to development of analytical methods for predicting the aerodynamic forces and moments.

The continued lack of understanding of the basic mechanisms largely is responsible for current limitations on the operational capabilities of rotors.

In the opinions of the authors, the most serious constraints on the performance of rotary-wing aircraft today are the following:

1. Aerodynamic limitations leading to excessive power demands, loss of lift and propulsive capability,

and loss of maneuverability due to saturation of control range.

2. Unsatisfactory stability characteristics and handling qualities, and,

3. Restrictions imposed by vibration and fatigue considerations.

These problems all emanate from rotor aerodynamic phenomena. Today's rotor systems still are all limited in their operating weight/speed/altitude envelope by inadequate propulsive force, loss of controllability or, most frequently, by some manifestation of aerodynamic stall effects. It is not possible to understand these limitations without an accurate definition of the airflow passing through the rotor system and an understanding of the characteristics of the airfoil operating in that environment. Although many important studies directed toward this goal have been made in the past four years, much remains to be done. Some of the more significant work is reviewed below.

THE ROTOR

A helicopter blade in the process of making one complete revolution, encounters a rapidly varying aerodynamic environment due to the varying relative magnitudes of the forward flight and rotational velocities, a complicated induced flow field resulting from the rotor wake system, spanwise flow due to blade yaw as well as Coriolis and centrifugal forces, and the control and elastic motions of the blade. The prediction of this flow field is difficult because of the complexity of the wake and the unsteady effects. The recent development of variable inflow methods, using classical lifting-line analysis, represent a considerable improvement in the state of the art, but they still do not permit airload distributions and associated bending moments to be predicted with the degree of accuracy desired for detailed blade design. One of the principal assumptions limiting the accuracy of these theories regards the geometry of the wake. In the prescribed-wake analysis, a particular wake model is assumed, generally a helical sheet made up of discrete vortex filaments or a system of vortex rings, and the circulation strengths and inflows at the disc are determined consistent with this wake model. There is, however, considerable experimental evidence that it is not enough to assume a semi-empirical wake shape for a particular blade, but that, if accurate prediction of inflow velocity and performance is desired, the precise path of the tip vortex, particularly its separation and radius as it passes under the following blade, must be known (see, e.g., Ref. 1).

The accuracy of the airloads in current calculations is limited, in part, by the use of lifting-line theory which is not valid for the large variations of the downwash along the span associated with a nearby vortex. The more accurate lifting-surface theory is needed to obtain the vortex-induced airloading. This problem has been investigated by Johnson [2]. Although the lifting-surface solution has been shown to be superior to the lifting-line theory in the calculation of vortex-induced loads on a simplified blade-vortex configuration, the use of the lifting-surface solution in the calculation of helicopter rotor airloads is of doubtful value until many other problems are resolved A very accurate wake geometry model is required in order to make full use of the accuracy of the lifting-surface solution.

Even if the flow field induced by the rotor-wake-vortex system is assumed to be known, we are still confronted with predicting the characteristics of lifting elements operating in compressible flow and executing complex unsteady motions into stall. Unsteady effects arise not only from the angle of attack variations, but also from variations in blade sweep angle and local velocity [3,4]. Before the airfoil designer can go to work, he needs to know the requirements to which he should design his new airfoil, and he has to know what features are most desired in case he cannot achieve all the goals simultaneously. This puts the burden on the rotor aerodynamicist who is soon embarrassed by these questions. Airfoil design will be a compromise to meet conflicting requirements imposed by hovering performance and figure of merit, "g" capability the potential stall environment of the retreating side and the high-speed environment of the advancing side.

Considerable evidence exists that two-dimensional, static airfoil data are totally inadequate for purposes of predicting rotor system behavior. Moreover, the substantial gain in rotor system performance capability produced by the introduction of cambered airfoils provides one positive piece of empirical data to indicate the magnitudes of the performance improvements that might be achieved by developing airfoils specifically tailored to the rotor performance requirements. The main effect of introducing leading-edge camber is that the maximum lift coefficient is increased without adversely affecting any of the other characteristics. Another recent and important development has been the introduction of spanwise profile variation and its combination with planform variation. Thin-tip blades and blade tip sweep have markedly improved performance as shown by Spivey [5,6], mainly because the formation of shock waves on the advancing blade tip is delayed. Airfoil design methods have recently become available which permit analytical optimization of performance at selected operating conditions with one airfoil. Given this capability, the helicopter designer can specify airfoil requirements in relation to the aircraft's performance and operation [7]. However, these techniques cannot yet address the usual limiting factor on speed and lifting capability of contemporary helicopters - blade stall.

As aircraft speed or weight is increased until the retreating blade exceeds its stall limit. rapid increases in blade loads, control loads, or rotor horsepower occur. Attempts to predict these effects by using steady flow airfoil characteristics in strip theory analyses have been unsuccessful. The onset of stall is predicted too early. Under conditions where rotor-blade-section angles of attack are predicted to exceed steady-state stall values, the theories have invariably predicted conservative stall characteristics; that is, the blade develops far more lift and has a very different value of drag than would be obtained for static two-dimensional flow conditions. Figure 2 illustrates some of the more recent data. The "stall limit" lines are actually limits of validity of the theory, even though stall and Mach number are included in the calculations. Both the lift for a given angle and the power for a given lift are, in this case, much more favorable than is theoretically predicted. There is no significance in the angle-of-attack shift between the theoretical and measured data below the stall limits in Figure 2. Only the slope differences above the stall limit are of interest here.

It is now well established that classical rotor theories are incapable of predicting overall rotor performance characteristics in stall (see, e.g., Ref. 8). The ability of the rotor to experimentally disregard the classical presumptions of blade stall is shown in Figure 2. Since blade stall limits the performance of the helicopter because of increased power requirements, aircraft roughness, vibration, and control loads, a knowledge of the flow separation mechanism by which rotor blade stall develops is needed; however, blade stall depends on the nature of the boundary layer which exists on the airfoil, and rotor boundary layers are so complex that only recently have investigations into their nature been undertaken [9,10,11].

Current efforts to improve the performance of rotary-wing aircraft have resulted in an increased interest in understanding the role of viscous effects in these applications. The US Army Air Mobility R&D Laboratory has undertaken a long range study concerning the fundamental nature of boundary-layer flows on rotors and propellers, and how these boundary layers can differ from the more familiar two-dimensional viscous flows around translating bodies.

McCroskey [12] measured the laminar separation, transition to turbulence, and surface streamline directions on helicopter rotor blades in a variety of rotating and nonrotating configurations. The results of these tests indicate that the centrifugal effects of rotation do not significantly alter the boundary-layer development for most operating conditions. Laminar separation bubbles were observed near the leading edge of the upper surface of the blades at moderate and large angles of attack and this phenomenon triggered a sudden transition to turbulent flow. Surface streamline patterns were found to be the same for both laminar and turbulent flows.

McCroskey's boundary-layer experiments indicated that, for the suction surface of typical helicopter rotors, the transition from laminar to turbulent flow is dominated by the chordwise adverse pressure gradient and a conventional laminar separation bubble rather than by Reynolds number, rotational, or crossflow effects. However, this investigation completely ignored any unsteady effects and little was learned about the actual mechanisms of stall on the rotating blades.

More recently, Dwyer and McCroskey [13] addressed analytically and experimentally the problem of boundary-layer flow over a helicopter rotor. The limitations of their results are the following:

1. Separation results have been obtained for rotating, three-dimensional steady flows and for two-dimensional, unsteady flows, but not for the complete helicopter rotor problem of three-dimensional, unsteady flows.

2. Complete solutions for the potential flow which serve as boundary conditions at the outer edge of the boundary layer are not available at this time and therefore the solutions of Sears [14] and McCroskey and Yaggy [15] for infinite blades with constant circulation had to be used, and

3. The flow has not been calculated for blades with oscillating changes in angle of attack.

Nevertheless, within these limitations, the roles of the various physical effects have been identified and the laminar flow on rotating blades is now well understood. The centrifugal force effect appears to be the least important for helicopter rotor blades. The most important effects appear to be time derivatives and crossflow derivatives. Also important are Coriolis forces and apparent pressure gradients induced by the potential crossflow. For a helicopter blade in forward flight there is a substantial inviscid crossflow due to translation and, therefore, the boundary layer generally resembles the viscous flow over a swept wing. It appears quite possible that the major three-dimensional and unsteady influences that affect the separation and stall characteristics of actual rotors occur in the turbulent regions of the boundary layer rather than in the laminar flow. The correct flow model probably is a small region of essentially quasi-steady, two-dimensional laminar flow followed by a three-dimensional, unsteady flow that has its initial conditions in a chordwise direction determined by a classical separation bubble (see Figure 3).

Of course, the boundary-layer characteristics that actually prevail depend upon the flow field in which the rotor is operating. In forward flight, the flow environment of the blade results from the combined effects of the angular velocity of the rotor, the forward-flight velocity, the induced velocity of the wake, and the downwash vorticity due to the wake of the previous blade. The angular velocity causes the rotational effects on the blade such as the Coriolis and centrifugal forces that arise in blade-fixed coordinates. The forward velocity combined with the rotational motion causes the flow over a rotor to be unsteady. The inflow velocity influences the local angle of attack and the slipstream contraction and the trailing vorticity in the wake from the preceding blade can produce locally large changes in the flow direction and angle of attack that are important in the boundary layer as well as in the potential flow.

THE ROTOR WAKE

In all the areas discussed above, the rotor wake has been the single item that has occurred repeatedly in discussions of the ability to predict rotary-wing behavior. The rotor wake has a major influence on almost all aspects of rotary-wing aerodynamics. Until the details of the rotor wake are well understood, it will not be possible to predict satisfactorily the behavior of rotary-wing aircraft.

The importance of the vortex shed by rotor blades in determining the performance of the rotor has long been recognized. For the three-dimensional wing in rectilinear motion, the vortex extends downstream from the trailing edge to infinity. The induced flow from the sheet is small relative to the freestream and its exact location is not too important in calculating the downwash at the wing. However, the wake from a helicopter remains close below the rotor disc and a large portion of the velocity at the rotor disc is induced by the wake. The first attempt to calculate the vortex wake was made by Goldstein [16] who assumed that the vortex wake was a regular helical surface and obtained a solution for the velocity potential of a cylindrical, helical vortex sheet. His analysis gives good results for very lightly loaded rotors or propellers. Lock [17] extended Goldstein's solution to take account of the variation of load with radial distance on the rotor blade. However, the classical wake theory used in predicting the behavior of lightly loaded rotors is not applicable to the more highly loaded, higher-speed modern rotors. Modifications of the classical wake have been devised in order to make the wake models conform more to the actual rotor wake.

When the loading for a rotor varies with azimuth angle, the vortex sheet contains vortices shed parallel to the trailing edge as well as trailing vortex lines in the flow direction. Early attempts to account for the unsteady aerodynamic effects assumed a rigid wake; that is, a wake convected with the free stream velocity and the average induced flow through the rotor computer by momentum theory. For a helicopter in horizontal flight, this assumption leads to a helical surface skewed by the angle whose tangent is equal to the ratio of the average flow through the rotor disc and the speed of horizontal flight. The normally skewed, helical-type vortex wake for a helicopter in forward flight was replaced by spaced rectangular vortex sheets by Willmer [18] and also by Molyneux [19].

To treat the helicopter in hover over a ground plane, Brady and Crimi [20], represented the tip vortex wake by a series of Helmholz finite-core ring vortices issuing periodically from a reference plane above the ground plane. To satisfy the ground-plane boundary conditions, images of the ring vortices were added to the solution. Unlike the wakes described by previous investigators, the effect of the induced velocity of the wake on its own motion was taken into account. Parameters of the ring-vortex systems such as the time between the issuing of adjacent rings and the circulation were identified with parameters of the rotor such as the rotational rate of the rotor and the rotor blade lift.

All of these techniques were attempts to represent the true vortex wake below the helicopter rotor disc by models which yield simple integrals requiring, at most, numerical quadrature. These theories predict many of the phenomena observed in helicopter rotor response and yield approximations to thrust and to the trends in the variation of certain parameters. With the development of high-speed, high-capacity computers, more realistic approximations to the rotor wake were attempted. Piziali and DuWaldt [21,22] assumed that the shed and trailing vortices retain the individual velocities with which they leave the trailing edge. Since this procedure neglects interaction of the vortices with their own motions, the rolling up of the vortex sheet is accounted for by arbitrarily replacing the vortex sheet by two trailing vortices in the far wake. This prescribed-wake analysis yields somewhat improved results for non-uniformly loaded rotors, particularly for forward flight where the free stream velocity is large compared to induced velocities and the wake is convected away from the rotor. However, for a helicopter hovering or ascending in a small crosswind, this wake model does not predict the performance with satisfactory accuracy. It was assumed that the wake is well behaved and any possibility of direct interference between a near wake and the blades was ignored. Figure 4 shows the large discrepancy between the actual power required for a given thrust of a hovering rotor and that predicted by either the Goldstein-Lock analysis, the momentum strip theory, or the prescribed-wake model. Jones and Noak [23] reported on a study that was made to determine the paths followed by the vortices trailing from the blade tips of a hovering rotor using continuous smoke emission for flow visualization. The positions of the vortex cores were measured and indicated that the wake contraction of not a smooth process. The trailing vortex remains substantially in the plane of rotation until the close approach of the following blade and there is clear evidence showing significant departures from the traditionally assumed helical tip trail.

Crimi [24] attempted to develop a more realistic wake using finite-core vortices with the core size determined by conservation of kinetic energy assuming a linear distribution of vorticity in the core. From observations of smoke studies which indicate that the vortex sheet rolls up into a strong tip vortex in a few chord lengths beyond the trailing edge, and that the inboard vortex rises through the rotor disc and dissipates, Crimi assumed that the wake could be represented with sufficient accuracy by a single tip vortex. A classical wake of straight-line segments is first calculated using momentum theory for predicting the motion of tip vortices. The vorticity of each tip vortex segment is assigned a strength equal to the maximum circulation on the rotor blade at the instant it is shed. The wake is then allowed to move with the combined velocity of the free stream and the total induced velocity including that due to the wake. The calculations of the wake motion are continued until the wake becomes periodic. Crimi's predicted wake trailing behind a two-bladed rotor in forward flight (Figure 5) shows substantial distortion relative to the skewed helix which he attributes partially to the perspective and the rest to the integration increment chosen. This wake model was also presented by Jones in his 1967 review [1].

The single tip finite-core vortex model of Crimi does not adequately describe the flow at the rotor disc because it neglects the effect of rolling up of the vortex sheet at the trailing edge of the rotor.

The most recent attempt to develop a method for computing the vortex wake was made by Clark and Leiper [3]. They represent the rotor by bound vortices made up of two-dimensional segments. At each discontinuity, single vortex filaments trail behind the rotor at the local velocity. A classical wake is constructed similar to that of Piziali and DuWaldt [21] which is allowed to move with its own induced velocity as well as with the free stream and the induced velocity of the wing bound vorticity. The rolling up of the vortex sheet to form a strong tip vortex is not considered as all filaments shed from the rotor blade trailing edge are retained throughout the calculations. However, because the wake interaction is taken into account in the calculations, the wake is observed to roll up along the outside spiral. The helical-type sheet moves downward very slowly near the axis and appears to reverse direction to move upward through the rotor disc.

This free-wake analysis was used to predict the hovering performance of the Sikorsky CH-53A rotor at high thrust level. The results are summarized in Figures 4 and 6. Figure 4 shows a comparison between measured performance and the performance calculated using the free-wake analysis and other existing techniques. The results of the free-wake analysis shows considerably improved correlation with the experimental data.

A clear indication of the significance of the close passage of the tip vortex from the preceding blade is found in the distribution of local angle of attack along the blade (Figure 6). The interfering vortex induces a large decrease in angle of attack inboard of the interference radius, and a corresponding increase outboard. The interference effect is significant when compared with the distribution of angle of attack predicted using blade element theory. The large peak in the distribution of profile power shown in Figure 6 is associated with the vortex-induced angle of attack increase and is characteristic of an airfoil operated well beyond the critical point. The blade outboard of the interference radius is obviously stalled and the flow separated. This conclusion was supported by tuft pictures taken during whirl tests.

The free-wake analysis of Clark and Leiper appears to be the method which most nearly takes adequate account of the influence of the wake-induced velocity on the position of the wake itself. However, the extremely costly calculation times required by this method preclude its extensive use in the design of helicopter rotors. Research is needed to develop a method that is computationally simpler to that of Clark and Leiper but more general than that of Crimi.

DYNAMIC AIRLOADS

The wake-induced velocities at the rotor disc are primary factors in the establishment of the unsteady flow environment of the rotor blade and, hence of its dynamic airloads. This is indicated by the theoretical local angle-of-attack distribution based on a variable inflow model compared with that based on a uniform inflow model shown in Figure 7. Not only do these effects manifest themselves directly as contributions to the higher harmonic airloads but more importantly, they provide the environment for dynamic stalling of the rotor blade.

Rotor blade stall is a dynamic phenomenon associated with the rapidly changing angle of attack that characterizes a helicopter rotor blade as it traverses the rotor disc, particularly on the retreating side of the disc at high advance ratios (Figure 7). The importance of the concept of dynamic stall for a rotor operating at high forward speed is that the blade stalls at a time when the rate of change of angle of attack is very high (Figure 7) and the lifts and moments initially are several times larger than the corresponding static stall values. As indicated in Figure 8, the inability of the quasi-static stall theory to predict rotor lift for a given pitch control, shaft tilt, and advance ratio is responsible for the poor correlation between theoretical and measured performance when blade stall is prevalent. Not only does the nonsteady penetration of stall alter the lift characteristics of the airfoil but it also alters the moment characteristics. Ham and Young [25] have shown that negative aerodynamic damping of blade torsional vibration can occur during dynamic stall. Such torsional vibrations increase the torsional stresses in the blade to the point where they are sufficiently severe to reduce the fatigue life of rotor mechanical components.

Recently several investigators have studied the problem of predicting the aerodynamic loads on a blade section undergoing dynamic stall. Ham [26] has represented the dynamic stall process by the shedding of a concentrated vortex from the airfoil leading edge and has developed a theory based on this model. Ericsson and Reding [27], on the other hand, have developed an analytical method by which airfoil static aerodynamic data might be used to predict the unsteady variations in airfoil lift and moment.

While methods such as these might be useful in making predictions of unsteady aerodynamic effects, they do not go to the root of the problem. Basically, stall is a phenomenon which is related directly to boundary layer separation and any complete analysis of stall, either steady or unsteady, must include the effect of boundary layer separation.

Recently acquired oscillatory aerodynamic data such as those presented by Gray and Liiva [28] are illustrated in Figure 9. The comparison of the static and dynamic characteristics shown in this figure demonstrates very clearly that proper lift and moment behavior of the airfoil can only be determined dynamically. The dynamic stall characteristics of symmetrical and cambered eleven-percent and six-percent-thickness-ratio helicopter rotor blade airfoils have been determined in a two-dimensional wind tunnel. The airfoils were oscillated in pitch about the quarter chord. Mach number, Reynolds number and the reduced frequency corresponded to those of a full-scale helicopter rotor blade on the retreating side of the rotor disc. All the airfoils exhibited dynamic increases in the maximum normal-force coefficients compared to their static values. The dynamic increase in the maximum normal-force coefficient is highly dependent on Mach number and becomes very small at Mach numbers greater than 0.6. There are regions of negative damping for angles of attack near stall and for Mach numbers below 0.6. Symmetrical airfoils have a wider range and an earlier inception of instability than the cambered airfoils.

The negative aerodynamic damping is caused by time lag (hysteresis) effects in the blade pitching moment versus angle-of-attack relation and can result in large torsional blade deflections and large control loads. The existence of excessive torsional loads feeding into the control system is a primary limitation on rotor operation. It is a direct result of the dynamic stalling of a large portion of the blade and is commonly called "stall flutter". Tarzanin [29] has developed a semi-empirical aeroelastic theory that clearly demonstrates the fundamental dependence of stall flutter on the dynamic stall delay. The effects of varying stall delay are illustrated in Figure 10. Calculated moment-coefficient hysteresis loops, with and without dynamic stall delay are shown for an airfoil in a constant flow field oscillating with an amplitude of five degrees about a mean angle of attack of fourteen degrees. These results show that the regions of negative damping (shaded area) decrease as the stall delay is eliminated. A comparison of the calculated pitch-link-load waveform, with and without stall delay in the lift and pitching moment, is also shown in Figure 10. The introduction of dynamic stall delay not only greatly increases the loads but also dramatically changes the waveform with the appearance of stall spikes.

Harris, Tarzanin, and Fisher [8] formulated an empirical approach to evaluate the unsteady, three-dimensional aerodynamic effects on the rotor stalling process. This entailed a pure analog computer simulation that was an extension of work performed by Jeffrey Jones [30]. The simulation contained rigid blade flapping, coupled first modes of flap bending-torsion, uniform downwash, reverse flow, and several small-angle assumptions. Figure 11 shows considerable improvement in the correlation with the engineering approximation to unsteady, three-dimensional blade stall effects in predicting rotor lift characteristics for a given pitch control and shaft tilt. Not only were they able to show improvement in the prediction of rotor lift, even though extensive regions of stall were present, but also good correlation was achieved in the prediction of the rotor lift-effective drag polar as is evidenced in Figure 12. The improvement in prediction capability apparently is attributable to both spanwise flow effects and unsteady aerodynamics.

Although this engineering approach resulted in improved rotor performance prediction, the double integration of the local airloads involved tends to average out competing factors in the approximations to the sectional characteristics. Therefore, it is reasonable to ask whether the blade element airload characteristics predicted by this method, such as those shown in Figure 13, do indeed represent the sectional characteristics in the actual rotor environment. The most unusual features of the empirically predicted sectional airloads are in the regions where classical airfoil data would exhibit stall phenomena. Measurements made at MIT of the differential pressures on a typical model rotor blade as it experienced stall during hovering tests revealed that a large suction peak originated near the blade leading edge as the blade approached its maximum nose-up position and then moved aft toward the blade trailing edge as the blade started its nose-down motion. It was suggested that the motion of this large suction peak was indicative of the shedding of intense vorticity from the vicinity of the blade leading edge as dynamic stall occurred. Subsequent experiments with a two-dimensional airfoil undergoing high linear rates of change of angle of attack substantiated that the maximum dynamic lift achieved was considerably higher than the maximum static lift. In addition, the results of these tests were compared with the corresponding dynamic lift variations synthesized by Carta from experimental oscillating airfoil data. The significant difference suggested that the use of oscillating airfoil data to determine blade airloads during transient or non-oscillating angle-of-attack changes could be quite unconservative, at least for large pitching rates. Ham recommended that transient dynamic stall data should be obtained from representative transient motion tests such as those of Ham and Garelick [31] rather than synthesized from data for airfoils oscillating through the stall.

Ham suggested that the separation mechanism is that shown in Figure 3 and that an understanding of the dynamics of the separation bubble is essential if the airfoil dynamic stall angle of attack is to be determined theoretically.

Fisher and McCroskey [32] undertook to examine the validity of such speculations on the basis of data obtained from a highly instrumented model rotor. They found that for the advancing blade, that is, in the first and second quadrants, the measured surface streamlines essentially followed the ideal local sweep angles. In the second quadrant, the measured blade element circulation per unit span increases rapidly. However, the measured pressure and skin-friction distributions remain similar to those measured statically as is shown in the left half of Figure 14. This means that large amounts of vorticity are being shed into the wake of the blade in a classical manner. In the third quadrant the stall onset region is entered (see Figure 7) and a heavily loaded rotor blade experiences a sequence of several distinct events that occur prior to complete blade stall. The similarity to static data disappears and subtle changes begin to take place in the pressure and skin-friction distributions as the blade passes an azimuthal angle of 200 degrees. A separation-like phenomenon appears in the boundary layer and the surface streamlines on the upper surface suddenly start to turn radially outward with respect to the ideal local sweep angle. The center of pressure begins to shift aft from its location near the quarter chord. The pressure distribution still exhibits a strong suction peak even though the angle of attack is well past the static stall value. At an azimuth of 210 degrees, the lift coefficient is still increasing and the center of pressure moves aftward and the nose-down pitching moment increases more rapidly, as indicated in Figure 15. Figure 14 shows a substantial increase in the negative pressure coefficient at the midchord region on the upper surface while the leading-edge suction peak is still clearly present. The suction peak at the leading edge starts to collapse at an azimuthal angle of 215 degrees but the midchord suction continues to grow. Both the negative pitching-moment and the lift coefficients grow rapidly. The normal-force coefficient and the negative pitching-moment coefficient attain their maximum values at approximately 240 degrees (see Figure 15), both at values considerably higher than are achieved statically. Shortly thereafter, the lift stall occurs because of the reduction in the suction at the midchord. The blade now enters the region of deep stall. In the region of azimuthal angles between 270 degrees and 345 degrees, variations occur in the pressure distribution but the suction at midchord on the upper surface remains the dominant feature. At 345 degrees, the blade passes in close proximity of a vortex shed by the preceding blade and then passes through the wake of the hub. At this time, the pressure distribution quickly changes to that typical of two-dimensional static data with an established leading-edge suction. The attached boundary layer characteristics are reestablished and the center of pressure is stabilized as the blade begins another revolution.

The rate of change of local blade angle is the predominant factor in this flow mechanism. The unusually high adverse pressure gradients experienced dynamically by the airfoil at its leading edge, exist as long as the angle of attack is increasing at a large rate. When this rate of increase becomes small, the flow field becomes unstable and the suction peak collapses, at which time a vortex apparently is formed and shed from the leading edge of the airfoil. This vortex sweeps back across the chord of the blade at a velocity significantly less than the local freestream velocity and causes the various stall events to occur. The basic mechanism of the dynamic stalling process of the rotor is increasingly dominated by dynamic vortex shedding from the leading edge as advance ratio increases. However, at low advance ratios in maneuvering flight, blade-vortex interactions, or blade elastic excitations can play significant roles if they produce rapid changes in the angle of attack of the blade as it traverses the rotor disc. It is now suspected that separation bubbles similar to those found on an airfoil during dynamic stall occur during the interaction of a rotor blade with the returning vortex from its own tip or that of another blade. Recent tests at MIT have indicated that during such a blade-vortex interaction, the chordwise pressure distributions on the blade, when it is in close proximity to a vortex, resemble those found on a dynamically stalling blade.

Contemporary empirical theories that account for three-dimensional, unsteady effects can adequately predict blade element forces and moments on the advancing blade and during the delay to the onset of stall, but not after the blade stalls completely.

FLYING QUALITIES

The dynamics of the rotor airloads also establish the stability and control characteristics and the flying qualities of the helicopter. Basic controls of the rotary-wing aircraft are the collective pitch and the cyclic pitch. In hovering flight, the collective pitch changes the total thrust of the rotor and the power required, whereas cyclic pitch changes the tilt of the thrust vector without significantly changing its magnitude or the torque required. However, in forward flight, changes in collective pitch will also change the thrust vector tilt and changes in the cyclic will affect both the magnitude of the thrust vector and the power required. This nonlinear coupling of the controls is the result of the structural dynamics of the rotor and the azimuthal variations of the airloads. Cross-talk effects such as these become even more significant to aircraft flying qualities with increasing size of the vehicle.

Flight through gusty air poses another special problem in the flying qualities of rotary-wing-type aircraft. It is considerably more complex than the corresponding problem for a fixed-wing aircraft. For example, the detailed response of a helicopter rotor to a sharp-edged gust will depend on the azimuthal position of the first blade to enter the gust. Another consideration at high forward speed is that the helicopter might have some part of the rotor disc stalled or, at least, very close to stall. Entry into the gust might change the rotor operating condition from one in which only a small portion of the swept area is stalled to one in which a large portion is stalled.

In general, rotary-wing aircraft experience milder reactions to gusts than do most fixed-wing aircraft, but this is not substantiated by simple theoretical considerations. It has been customary to determine the normal rotor forces due to sharp-edged gust by using charts such as those developed over twenty years ago by NACA [33]. Since this method does not take into account rotor limits at high advance ratios, it yields very conservative results. The development of helicopters with higher forward speeds and higher disc loadings and the addition of wings and auxiliary propulsion in the case of compound helicopters have made the present methods of determining gust response inadequate. At high speeds and for disc loadings greater than about six pounds per square foot, the computer load factors are too high. When, in addition, gusts are super-imposed on maneuver loads, an unrealistic design situation is created. Drees and Harvey [34] did an analytical study of the response of helicopters to discrete gusts. Their results indicate that the added considerations of gust profile, non-steady aerodynamics and gradual penetration have a primary effect on the gust loads. Recent studies by Hohenemser (see, e.g., Ref. 35) have added to the understanding of the response of rotors to random gusts.

It has been accepted generally that the flying qualities requirements should be tailored to the particular prime mission of the vehicle under consideration. For example, it is unlikely that the flying qualities which are optimum for a weapon platform will also be optimum for a transport aircraft. However, it has not been possible, thus far, to translate this intuitive knowledge into a cohesive set of handling qualities specifications. Although, work is underway in that area, the moving-base simulation technique most likely offers the best approach to this current dilemma. Simulation techniques can be used to arrive at a set of specific stability and control requirements to assure that the handling qualities of the vehicle are satisfactory for accomplishing the aircraft's mission and that the stability and control characteristics meet the flying qualities objectives. Fundamental to the problem of developing a satisfactory simulator is the fidelity with which it simulates the real world and a very important part of this representation for a helicopter is the mathematical model for the aerodynamics of the rotor. The success with which a simulator can be used to establish flying qualities requirements depends upon understanding the aerodynamic mechanisms significant to the generation of forces and moments and the accuracy with which they are represented.

NEW ROTOR CONFIGURATIONS

Research and development efforts to overcome the limitations of the conventional helicopter rotor have usually concentrated on providing a wing to share the load with the rotor in order to achieve higher speed and maneuverability. However, in recent years, a variety of new rotor configurations have been investigated in an effort to overcome the aerodynamic limitations of the conventional rotor while retaining the operational advantages of low disc loading. Research in rotor concepts pursued by the US Army Air Mobility R&D Laboratory and its co-participants has included design studies (and in some cases wind-tunnel tests) of the hot-cycle rotary wing, the tilt prop-rotor, hingeless stopped/stowed rotor, shaft-drive and warm-cycle heavy lift rotor systems, matched stiffness/ flexure root rotor system, stopped, folded and/or stowable rotor systems, the rotating-wing concept and the jet-flap rotor system. Wind-tunnel tests have been completed successfully on the forty-foot diameter Advancing Blade Concept (ABC) rotor system, and on a twenty-five-foot diameter tilt prop-rotor system. Both of these rotor systems used in the wind-tunnel tests were full-scale flightworthy systems. In addition, scale-model tests are currently being conducted of a telescoping rotor system capable of forty-percent diameter reductions in flight. Model tests and analysis have been conducted on the Controllable Twist Rotor (CTR). The US Naval Ship Research and Development Center has tested a six-foot diameter circulation controlled rotor and plans to test a twelve-foot diameter, dynamically scaled rotor in the NASA-Langley Sixteen-Foot Pressure Tunnel. The Navy is also studying the Reverse Velocity Rotor (RVR) in which a cyclic pitch change is introduced in addition to the basic cyclic pitch change so that the retreating blade presents a positive angle of attack in the region of reverse velocity. The NASA-Langley Research Center is currently studying variable geometry rotors with the objective of achieving some direct control over the dynamic interaction of the blades and the tip vortices. The US Army and NASA are also planning the development of a rotor test vehicle, a flying test bed which will permit conceptual feasibility demonstration of new rotor systems.

RESEARCH REQUIREMENTS

It is interesting that, although the helicopter rotor is an aerodynamic device, progress in rotary-wing technology has come more from advances in propulsion, structural dynamics and vibration control, materials and fatigue life, and stability and control than from improvements in the aerodynamics of the rotor. Nevertheless, as discussed above, current limitations on operational capabilities of helicopters are intimately related to the rotor aerodynamics and a better understanding of the fluid dynamics of the rotor could lead to significant improvements in the performance of future helicopters. A review of the state of the art such as this reveals the necessity for a coordinated treatment of aerodynamic research of rotary-wing aircraft and a unification and integration of the results of the many studies of the various aspects of this complex problem. Figure 16 represents the helicopter rotor aerodynamics problem and indicates interrelationships of its diverse facets. The figure can be considered an information flow diagram similar to one presented by Joglekar and Loewy [36]. Each of the rectangular blocks corresponds to a part of the total problem that is all too frequently treated as though it were independent of the others. There is much work to be done in each of these areas, but it is important that the interactions among the results of these studies should not be overlooked.

The need for comprehensive, steady and unsteady two-dimensional airfoil data of a representative range of applicable airfoil sections is widely recognized and there is a steady flow of information from a variety of sources. However, uncritical acquisition of data is not enough and the technique of measurements should be determined by the requirements for their application. A systematic program is needed to develop and evaluate airfoils showing potential for specific rotor missions. Suitable evaluation criteria are needed for rotor airfoils that are expected to operate in the mixed unsteady flow of continuously varying angle of attack and Mach number. The analytical techniques for airfoil design require further development to permit the selection of promising families of airfoils prior to wind-tunnel testing. There is considerable evidence that airfoils designed with recognition of the flow field can produce improved rotor performance.

Systematic gathering of quantitative data on the characteristics of the boundary layer on typical rotors in hover and forward flight should be emphasized. These data should include mean and instantaneous velocity profiles, flow direction, separation points, transition from laminar to turbulent flow, and the nature of reattachment as well as the usual pressure distribution measurements. These data are needed to verify the analytical boundary-layer prediction techniques currently in the early phases of development.

In the course of reviewing the literature reporting theoretical and experimental studies of rotor aerodynamics, it becomes very apparent that experimental measurements are scarce, conflicting and incomplete. Correlation between experiment and theory is generally poor and there are significant discrepancies among the various theoretical methods. New and better experimental techniques need to be developed utilizing instrumentation and flow visualization in the rotating frame of reference.

The new rotor concepts such as those discussed above need further investigation to establish their feasibility. For those categories of rotary-wing aircraft using fixed lifting surfaces, such as compound helicopters and tilt-wing-rotor vehicles, to achieve optimum performance over the whole flight envelope there must be careful harmonization of all the main aerodynamic components. Performance characteristics of typical rotors, isolated and in wing/rotor and wing/rotor/body combinations, need to be investigated using idealized models.

On the basis of what we have learned recently regarding the importance of boundary-layer effects, we can speculate that some sort of boundary-layer control should be beneficial for rotors. In contrast to fixed-wing practice, no production helicopter uses any sort of boundary-layer control device. It might be possible to obtain performance gains through application of boundary-layer control to helicopter rotors, however, this requires further research to expand our understanding of the characteristics of the rotor boundary layer.

Research in the area of rotor aerodynamics should include the accumulation of experimental data which demonstrate the effects of camber, twist, number of blades, aspect ratio, variable geometry and Mach number on forward-flight rotor performance and the associated characteristics of the wake geometry. These data, along with inputs from such programs as the experimental and theoretical investigations of rotor aerodynamics environment and boundary layer analyses, should be used to evaluate the existing performance theories and to develop new ones as required which will accurately predict these effects. In order to achieve the desired accuracy, the wake model has to be improved and methods for analytically treating unsteady aerodynamics and compressibility effects have to be developed and incorporated in a workable and usable performance theory.

REFERENCES

1. Jones, J. P.: "Rotor Aerodynamics-Retrospect and Prospect", AGARD Advisory Report 13, Aspects of V/STOL Aircraft Development, Sept 1967.

2. Johnson, Wayne: "Application of a Lifting-Surface Theory to the Calculation of Helicopter Airloads", Presented at the 27th Annual National V/STOL Forum of the American Helicopter Society, May 1971, Preprint No. 510.

3. Clark, D. R. and Leiper, A. C.: "The Free Wake Analysis - A Method for the Prediction of Helicopter Rotor Hovering Performance", Journal of the American Helicopter Society, Vol. 15, pp. 3-11, 1970.

4 Clark, D. R. and Landgrebe, A. J.: "Wake and Boundary Layer Effects in Helicopter Rotor Aerodynamics", AIAA Paper No. 71-581.

5. Spivey, R. F.: "Blade Tip Aerodynamics - Profile and Plan Form Effects", 24th Annual National Forum, American Helicopter Society, May 1968.

6. Spivey, W. A. and Morehouse, G. G.: "New Insights Into the Design of Swept-Tip Rotor Blades", Amer. Helicopter Society, Preprint No. 420, June 1970.

7. Wortmann, F. X. and Drees, J. M.: "Design of Airfoils for Rotors", Proceedings of the 3rd CAL/AVLABS Symposium, Buffalo, New York, June 1969.

8. Harris, F. D., Tarzanin, F. J., Jr., and Fisher, R. K., Jr.: "Rotor High Speed Performance, Theory Versus Test", Journal of the American Helicopter Society, Vol. 15, No. 3, July 1970, pp. 35-44; also: Proceedings of the V/STOL Technology and Planning Conference (AFFDL), Las Vegas, Nev., 23-29 Sept 1969.

9. Blaser, D. A.: "An Analytical and Experimental Investigation of Helicopter Rotor Boundary Layers", Ohio State Univ., Ph.D. Dissertation, July 1971.

10. Velkoff, H. R., Blaser, D. A., and Jones, K. M.: "Boundary-Layer Discontinuity on a Helicopter Rotor Blade in Hovering", AIAA Journal of Aircraft, Vol. 8, No. 2, Feb 1971.

11. Clark, D. R. and Arnoldi, D. R.: "Rotor Blade Boundary Layer Calculation Programs", USAAVLABS, TR 71-1, March 1971.

12. McCroskey, W. J.: "Measurements of Boundary Layer Transition, Separation and Streamline Direction on

Rotating Blades", NASA TN D-6321, April 1971.

13. Dwyer, H. A., and McCroskey, W. J.: "Crossflow and Unsteady Boundary-Layer Effects on Rotating Blades", AIAA Journal, Vol. 9, No. 8, August 1971.

14. Sears, W. R.: "Potential Flow Around a Rotating Cylindrical Blade", Journal of the Aeronautical Sciences, Vol. 17, No. 3, March 1950.

15. McCroskey, W. J. and Yaggy, P. F.: "Laminar Boundary Layers on Helicopter Rotors in Forward Flight", AIAA Journal, Vol. 6, No. 10, October 1968.

16. Goldstein, S.: "Vortex Theory of Screw Propellers", Proceedings of the Royal Society, London, A1-3, (792) pp. 440-465, 1929.

17. Lock, C. N. H.: "The Application of Goldstein's Theory to the Practical Design of Airscrews", Aero Research Council, R&M No. 1377, 1931.

18. Willmer, M. A. P.: "The Loading of Helicopter Rotor Blades in Forward Flight", Aero Research Council, R&M No. 3318, 1963.

19. Molyneux, W. C.: "An Approximate Theoretical Approach for Determination of Oscillatory Aerodynamic Coefficients for a Helicopter Rotor in Forward Flight", The Aeronautical Quarterly, Vol. 13, pp. 235-254, 1962.

20. Brady, W. G., and Crimi, P.: "Representation of Propeller Wakes by Systems of Finite Core Vortices", CAL Report BB-1665-S-2, Feb 1965.

21. Piziali, R. A. and F. A. DuWaldt: "A Method for Computing Rotary Wing Airload Distributions in Forward Flight", TRECOM Report No. TCREC TR 62-44, November 1962.

22. Piziali, R. A.: "Method for the Solution of the Aeroelastic Response Problem for Rotating Wings", Journal of Sound and Vibration, Vol. 4, pp. 445-489, 1966.

23. Jones, J. P. and Noak, M.: "Some Problems in the Numerical Solution of Vortex Motion Equations", Proceedings 3rd CAL/AVLABS Symposium, Buffalo, New York, June 1969.

24. Crimi, P.: "Theoretical Prediction of the Flow in the Wake of a Helicopter Rotor - Part I: Development of Theory and Results of Computation" and "Part II: Formulation and Application of the Rotor-Wake-Flow Computer Program", CAL Reports BB-1994-S-1 and S-2, September 1965.

25. Ham, N. D. and Young, M. I.: "Limit Cycle Torsional Motion of Helicopter Blades Due to Stall", Journal of Sound and Vibration, Vol. 4, No. 3, November 1966.

26. Ham, N. D.: "Some Recent Research on the Nature of Airfoil Dynamic Stall, With Application to Airfoil Design", MIT Aeroelastic and Structures Research Laboratory, ASRL TR 156-2, September 1971.

27. Ericsson, L. E. and Reding, J. P.: "Dynamic Stall of Helicopter Blades", American Helicopter Society, 26th Annual National Forum, June 1970, Preprint No. 422.

28. Gray, L. and Liiva, J.: "Two-Dimensional Tests of Airfoils Oscillating Near Stall", USAAVLABS TR68-13, 1968.

29. Tarzanin, F. J., Jr.: "Predictions of Control Loads Due to Blade Stall", American Helicopter Society, 27th Annual National Forum, Washington, D. C., May 1971, Preprint No. 513.

30. Jones, J. P.: "The Use of an Analog Computer to Calculate Rotor Blade Motion", Journal of the American Helicopter Society, April 1964.

31. Ham, N. D. and Garelick, M. S.: "Dynamic Stall Considerations in Helicopter Rotors", Journal of the American Helicopter Society, Vol. 13, No. 2, April 1968.

32. Fisher, R. K., Jr. and McCroskey, W. J.: "Detailed Aerodynamic Measurements on a Model Rotor in the Blade Stall Regime", Presented at the 27th Annual National V/STOL Forum of the American Helicopter Society, May 1971, Preprint No. 521.

33. Amer, K. B. and Gustafson, F. B.: "Charts for Estimation of Longitudinal Stability Derivatives for a Helicopter Rotor in Forward Flight", NACA TN 2309, March 1951.

34. Drees, J. M. and Harvey, K. W.: "Helicopter Gust Response at High Forward Speed", AIAA Journal of Aircraft, Vol. 7, No. 3, May-June 1970.

35. Gaonkar, G. H. and Hohenemser, K. H.: "Flapping Response of Lifting Rotor Blades to Atmospheric Turbulence", AIAA Journal of Aircraft, Vol. 6, No. 6, Nov-Dec 1969.

36. Joglekar, M. and Loewy, R.: "An Actuator-Disc Analysis of Helicopter Wake Geometry and the Corresponding Blade Response", USAAVLABS TR 69-66, Dec 1970.

PROGRESS IN REMOVING ASSUMPTIONS FROM ROTOR PERFORMANCE THEORY

PROGRES OBTENUS EN SUPPRIMANT DES HYPOTHÈSES DANS LA THÉORIE DE PERFORMANCE DU ROTOR

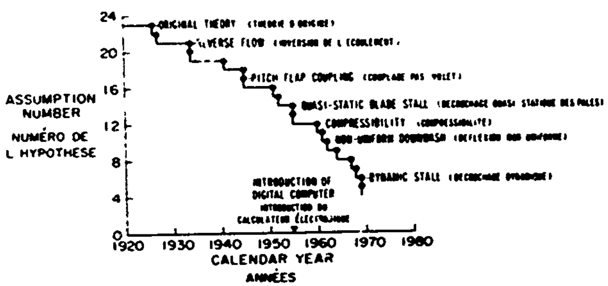


FIGURE 1

THEORETICAL ROTOR PERFORMANCE COMPARED TO EXPERIMENT

PERFORMANCE THÉORIQUE DU ROTOR COMPARÉE À L'EXPÉRIENCE

--- MEASURED (MESURÉ)

—— THEORETICAL (THÉORIQUE)

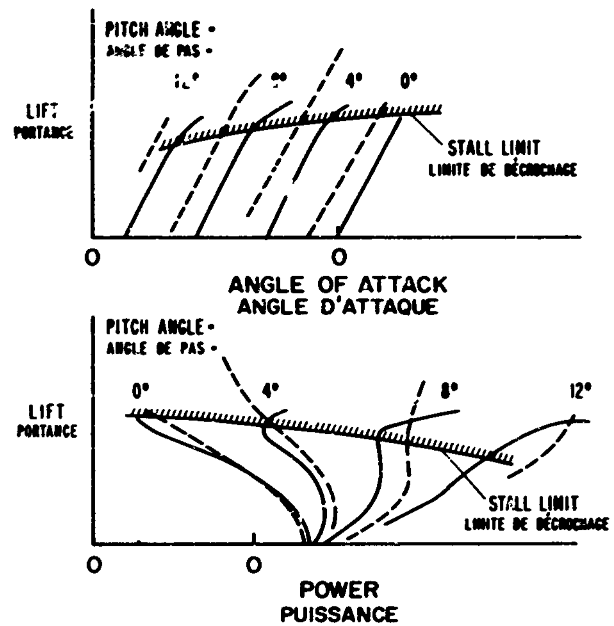


FIGURE 2

BUBBLE STRUCTURE ON A ROTOR BLADE

STRUCTURE EN BULLE SUR PROFIL D'AILE MINCE

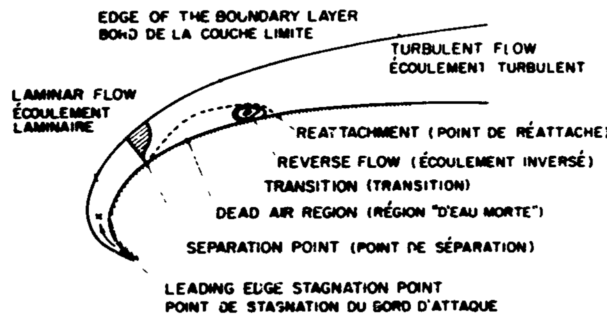


FIGURE 3

COMPARISON OF MEASURED AND PREDICTED HOVERING PERFORMANCE

COMPARAISON DES PERFORMANCES SUSTENTATOIRES MESURÉES ET PRÉDITES

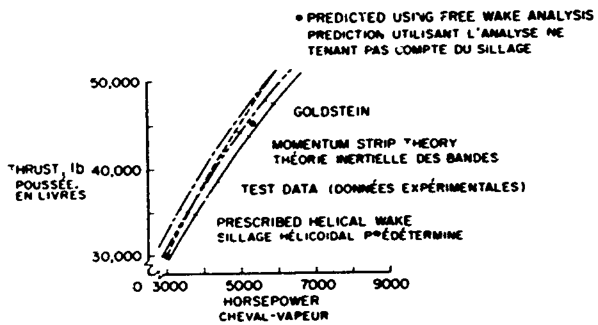
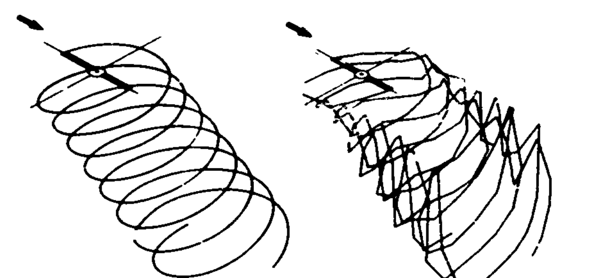


FIGURE 4

CRIMI'S DISTORTED WAKE

SILLAGE DÉFORMÉ DE CRIMI



INITIAL WAKE CONFIGURATION, SKEWED HELIX
CONFIGURATION INITIALE DU SILLAGE, HÉLICE GAUCHE

FINAL WAKE CONFIGURATION, PERIODICITY ESTABLISHED
CONFIGURATION FINALE AVEC PÉRIODICITÉ ÉTABLIE

FIGURE 5

THEORETICAL RADIAL DISTRIBUTION OF NORMALIZED PROFILE POWER AND ANGLE OF ATTACK

DISTRIBUTION RADIALE THÉORIQUE DE LA PUISSANCE NORMALISÉE DU PROFIL ET DE L'ANGLE D'ATTAQUE

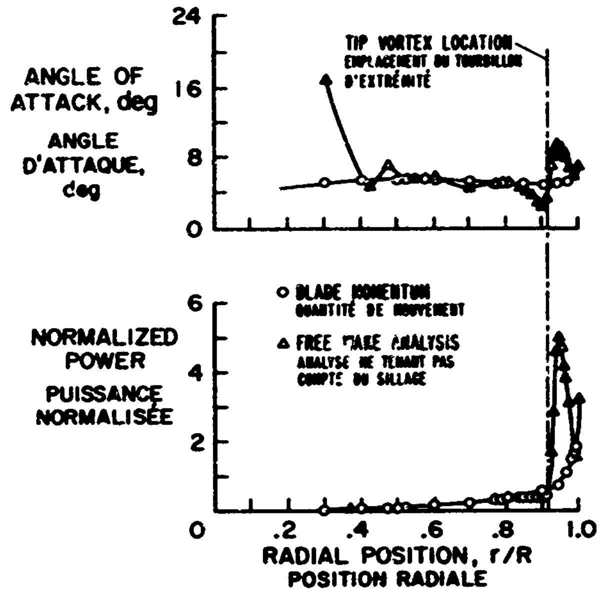


FIGURE 6

THEORETICAL LOCAL ANGLE OF ATTACK DISTRIBUTION IN TRIM FLIGHT

DISTRIBUTION TYPIQUE DE L'ANGLE D'ATTAQUE THÉORIQUE EN VOL COMPENSÉ

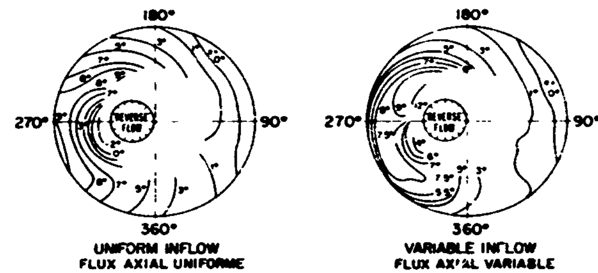


FIGURE 7

ROTOR PERFORMANCE THEORY COMPARED TO EXPERIMENT

PERFORMANCE DU ROTOR-LA THÉORIE EN COMPARAISON DE L'EXPÉRIENCE

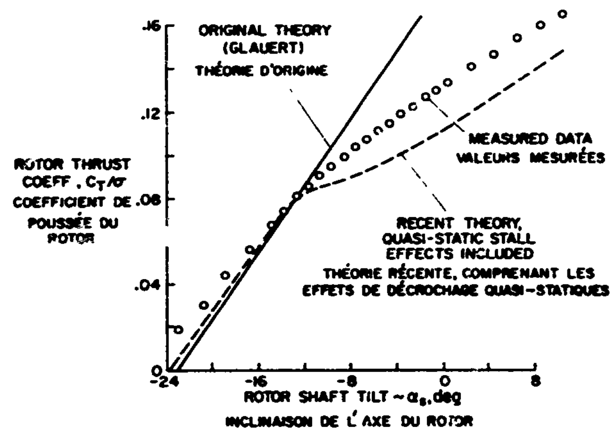


FIGURE 8

STATIC & DYNAMIC AIRFOIL BEHAVIOR
COMPORTEMENT STATIQUE ET DYNAMIQUE D'UN PROFIL D'AILE

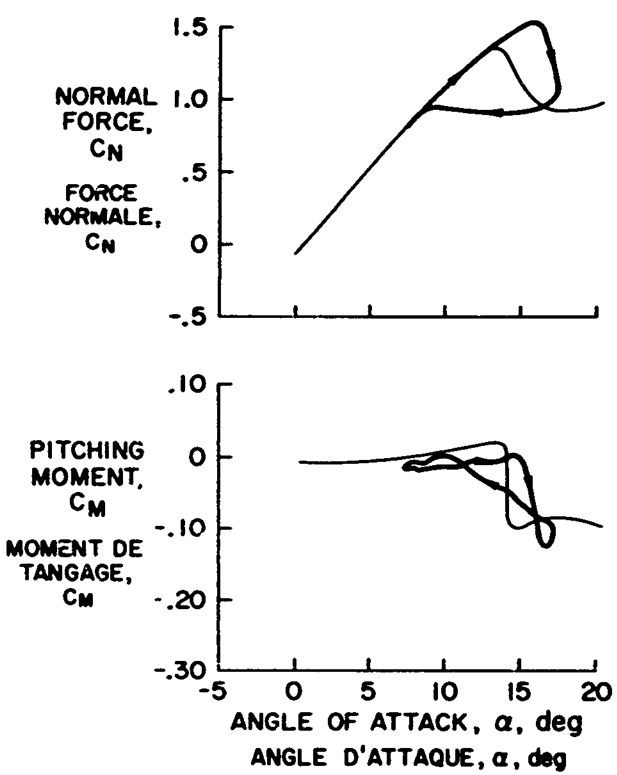


FIGURE 9

EFFECTS OF STALL DELAY ON PITCHING MOMENT AND PITCH LINK LOADS
EFFETS DU DÉLAI DE DÉCROCHAGE SUR LE MOMENT DE TANGAGE ET LES CHARGES DANS LA COMMANDE DE PAS

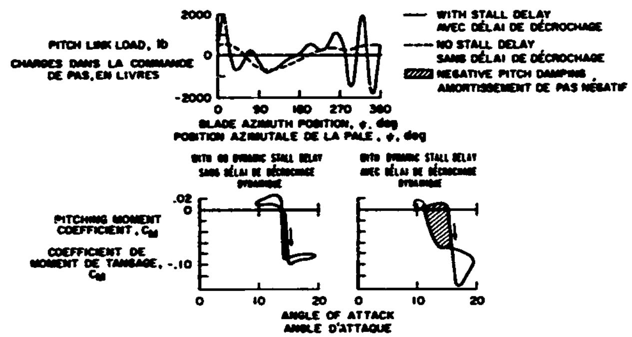


FIGURE 10

ROTOR PERFORMANCE—UNSTEADY, THREE-DIMENSIONAL EFFECTS ON THEORETICAL PREDICTIONS
PERFORMANCE DU ROTOR - EFFETS DE LA NON-STATIONNARITÉ ET DE LA TRIDIMENSIONALITÉ SUR LES PRÉDICTIONS THÉORIQUES

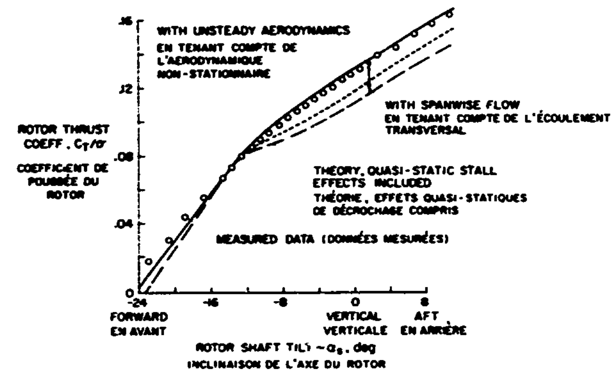


FIGURE 11

ROTOR DRAG POLAR—UNSTEADY, THREE-DIMENSIONAL EFFECTS ON THEORETICAL PREDICTIONS

POLAIRE DE TRAÎNÉE DU ROTOR—EFFETS DE LA NON-STATIONARITÉ ET DE LA TRIDIMENSIONALITÉ SUR LES PRÉDICTIONS THÉORIQUES

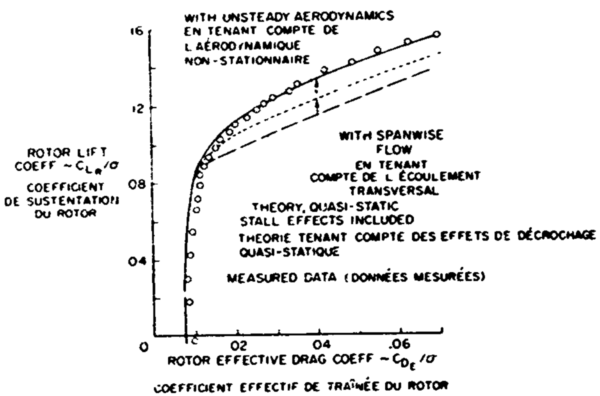


FIGURE 12

SECTIONAL CHARACTERISTICS DERIVED FROM MEASURED INTEGRATED ROTOR LOADS

CARACTÉRISTIQUES DE LA SECTION DÉDUITES DE MESURES DE LA SOMME DES EFFORTS SUR LE ROTOR

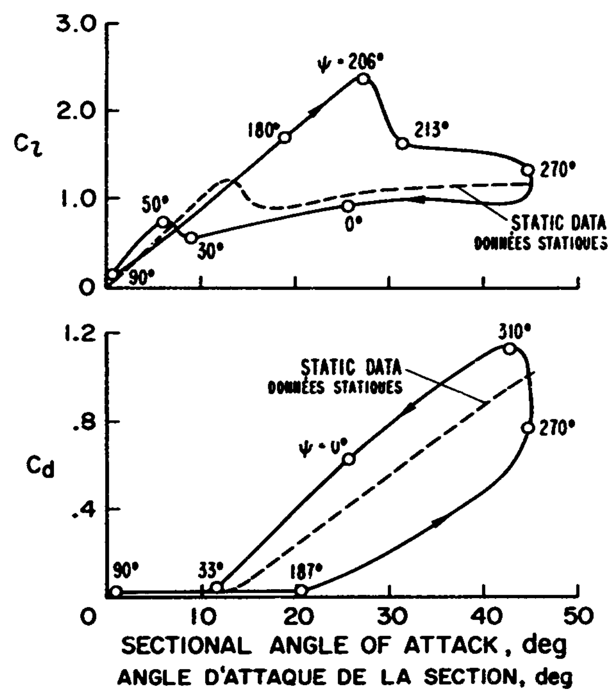


FIGURE 13

PRESSURE DISTRIBUTIONS AT VARIOUS AZIMUTH LOCATIONS

DISTRIBUTIONS DE LA PRESSION À CERTAINS POINTS AZIMUTAUX

○ FORWARD FLIGHT DATA
VOL EN TRANSLATION

△ HOVER OR NON-ROTATING DATA
POINT FIXE OU ROTOR STOPPÉ

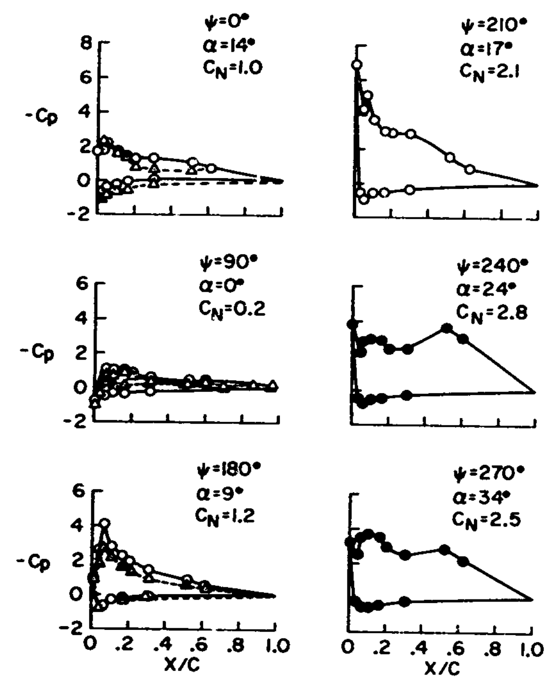


FIGURE 14

COMPARISON OF ROTATING AND NON-ROTATING LIFT AND MOMENT DATA
COMPARAISON DE LA PORTANCE, TOURNANTE ET NON-TOURNANTE ET DES DONNÉES DU MOMENT

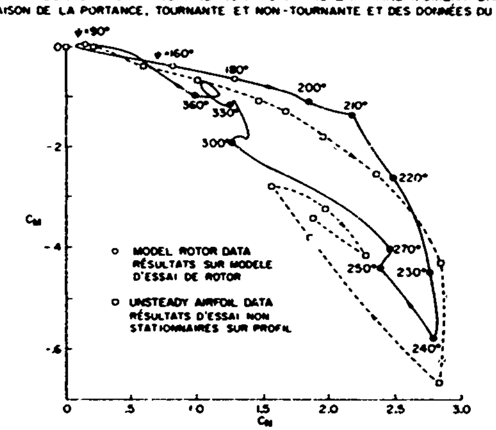
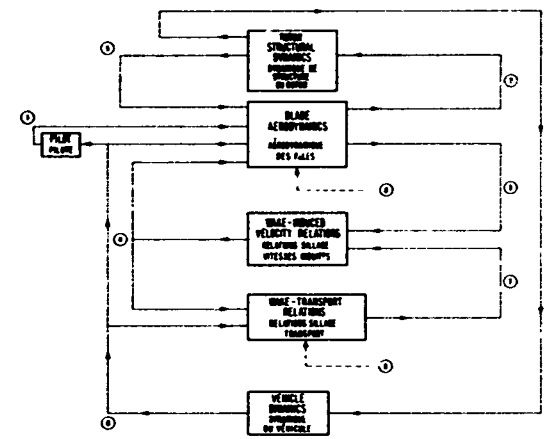


FIGURE 15

THE HELICOPTER ROTOR AERODYNAMICS PROBLEM
LE PROBLÈME DE L'AÉRODYNAMIQUE DU ROTOR D'HÉLICOPTÈRE



QUANTITIES OF INTEREST
QUANTITÉS ENTRANT EN JEU

1 CONTROL SETTINGS
AJUSTEMENT DES CONTRÔLES
2 AIRLOADS
FORCES AÉRODYNAMIQUES
3 VORTEX STRENGTH, POSITION DUE TO UNPERTURBED FLOW
INTENSITÉ TOURBILLONNAIRE, POSITION DUE À L'ÉCOULEMENT NON PERTURBÉ
4 INDUCED VELOCITY FIELD
CHAMP DES VÉLOCITÉS INDUITES
5 BLADE DISPLACEMENTS AND MOTIONS
DÉPLACEMENTS ET MOUVEMENT DES PALES
6 VEHICLE ATTITUDE AND MOTION
ATTITUDE ET MOUVEMENT DU VÉHICULE
7 WAKE VORTICITY POSITION (WAKE GEOMETRY)
POSITION DU SILLAGE TOURBILLONNAIRE (GÉOMÉTRIE DU SILLAGE)
8 COMPRESSIBILITY, STALL, AND REVERSE FLOW EFFECTS MIGHT ENTER HERE
DES EFFETS DÛS À LA COMPRESSIBILITÉ AU DÉCROCHAGE ET À L'INVERSION DE L'ÉCOULEMENT PEUVENT ÊTRE INTRODUITS ICI
9 INTERFERENCE, WAKE INTERACTION, AND VISCOSITY EFFECTS MIGHT ENTER HERE
DES EFFETS DÛS À L'INTERFÉRENCE, L'INTERACTION ENTRE SILLAGES ET LA VISCOSITÉ PEUVENT ÊTRE INTRODUITS ICI

FIGURE 16

SURVEY OF ROTARY WING LOADS AND STABILITY ANALYSIS PROBLEMS

by

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1. INTRODUCTION

A survey of some of the problems encountered in the prediction of structural design loads and aeroelastic stability margins during the development of rotary-wing aircraft is presented.

The accurate prediction of structural design loads and aeroelastic stability margins during steady and transient maneuvers is essential during the development of a new flight vehicle. In order to preclude major delays in deployment of new aircraft, an accurate definition of the aircraft detail configuration is desirable prior to initiation of flight testing. Early, accurate analysis is needed to reduce the extent of both planned and unplanned flight testing required to develop a satisfactory final configuration. Accurate methods are also necessary to reduce reliability and maintenance problems in the operational environment, deficiencies in performance due to flight envelope restriction arising from unanticipated structural problems, and deficiencies in handling qualities due to changes made in the configuration to solve loads or aeroelastic stability problems.

At present, each airframe developer employs several analysis methods of varying complexity to determine loads and aeroelastic stability for the critical flight and ground conditions. These methods, particularly the more simplified ones, are applicable only to the type and size of rotor system in which the airframe developer has specialized. Furthermore, most developers have limited capability to account for the effects of coupling of advanced flight control systems, fuselage motions, and inadvertent high-frequency pilot inputs (pilot-coupled oscillations). Accordingly, loads and aeroelastic stability of rotary-wing aircraft employing larger rotors, advanced rotor concepts, and advanced flight control systems cannot be predicted with sufficient accuracy.

The purpose of this paper is to provide the reader with some understanding of the complexity involved in the prediction of rotary-wing loads and aeroelastic stability, to discuss the impact of this complexity on the cost and accuracy of these predictions, and to suggest areas of investigation where the more complex, expensive analysis methods now under development can be effectively used.

2. COMPLEXITY

Accurate computation of loadings on a helicopter rotor blade clearly involves one of the most complex mathematical models encountered in today's technology. This complexity is attributable to the rotation and flexibility of the rotor blades. It has been said that in forward flight during one revolution of a rotor, a segment of a rotor blade encounters an experience comparable to the wing of a very flexible, high subsonic speed, acrobatic airplane which is also designed to fly backwards.

Accurate computation of loadings during transient maneuvers requires the simultaneous determination of fuselage rigid body motions, blade motions, airflow distortions in the blade wake, blade aerodynamics, and control system response. Each of these is coupled with the other, and unfortunately many nonlinear and time-variant terms appear in the differential equations which describe the total system response. This requires the use of iterative procedures to achieve a simultaneous solution of the system of equations for a steady-state condition.

Figure 1 presents a flow chart depicting a possible approach to obtaining an iterative solution of the equations leading to a determination of loads in a steady-state flight condition. This figure will be used as a road map for the following discussion of some aspects of rotary wing-loads analysis. This discussion is intended primarily to acquaint the reader with the complexity of the problem rather than to provide technical detail.

The solution is initiated by supplying input data which describes the steady-state operating condition of the aircraft, including such parameters as airspeed, rotor speed, altitude, gross weight, and center of gravity. First, a rather simplified analysis is conducted to estimate the cyclic and collective control positions necessary to trim the aircraft in steady flight, i.e., to provide rotor lift, thrust, and pitch and roll moments. Such an analysis might be based on quasi-static linear aerodynamics, rigid blades, and uniform inflow.

Next, the wake geometry can be determined using a complicated analysis that treats the wake as a helical web of vortex elements left behind and beneath the rotor as shown in Figure 2. This helical web consists of shed and trailing vortices. The strength of the shed vortices is related to the time rate of change of lift on the blade segment which is generally large at high airspeeds. The strength of the trailing vorticity depends on the difference in lift on adjacent radial blade segments, the tip vortex being the strongest vortex. The spacings between the shed vortices and between the trailing vortices are chosen arbitrarily.

The vertical velocity of the wake is determined in accordance with momentum theory, and the horizontal velocity is based on the velocity of the vehicle. The induced velocity on each blade due to each vortex is derived using the Biot-Savart Law.

Combining these induced velocities with the velocities associated with the blade motions determined in the initial trim analysis provides the distributions of translational and rotational velocities and their time rates of change over the rotor disk.

These provide the initial inputs to the aerodynamic loads analysis. The blade can be considered to consist of an arbitrary number of spanwise segments. The aerodynamic lift, pitching moment, and drag on each segment can be determined by including static and unsteady aerodynamic effects.

Static aerodynamic data, which include the effects of static stall, can be stored in a table as a function of Mach number and angle of attack. At blade angles of attack below the static stall angle, unsteady effects due to the rates of change of pitching velocity can be computed using the Theodorsen functions. The effects of dynamic stall on the retreating blade, encountered at combinations of high airspeed and rotor lift, can be determined usually as a function of the time rate of change of angle of attack and its derivative. Of course, compressibility and viscosity effects on the dynamic stall characteristics of the airfoils must be accounted for in some manner.

Two approaches to representing the rotor blades to determine their dynamic response and net loads are available. One approach is the lumped mass method which involves the division of each blade into an arbitrary number of masses interconnected with springs. The motion of each of the many segments can be determined by subjecting the system to its previously obtained aerodynamic loading. The other approach uses the orthogonal vibratory modes, including linear coupling terms. The response of the system is obtained by superposition of the response of each blade in each mode when subjected to the aerodynamic loads obtained above in combination with any non-linear inertial coupling loads associated with the blade motion. The solution would proceed as a step-by-step integration around the azimuth to determine the transient response of the blades. This process is repeated until the loads during two consecutive revolutions of the rotor are nearly equal.

The average lift, thrust, and pitching and rolling moments acting on the rotor shaft during one revolution can then be compared to the values of these loads obtained in the initial or prior trim calculation. If significant differences in these shaft loads are present, then a retrim analysis can be used to estimate the changes in the cyclic and/or collective control positions necessary to trim the aircraft. Iteration within this loop would continue until trim is established.

Next, the velocity distribution along one blade during one revolution due to the blade motion and the initial variable wake can be compared to the velocity distribution inherent in the initial variable wake analysis. If these velocity distributions differ significantly then a revised variable wake description can be derived.

Satisfactory comparison of these velocities would result in culmination of the steady-state loads analysis.

Although not indicated in Figure 1, the aerodynamic loads and flexible blade load analyses can be used to estimate the transient response of the system to simulated pilot-induced maneuvers. Angle-of-attack increments due to control position changes can be input to the aerodynamic loads and flexible blade loads analyses to excite the rotor.

Major additional difficulties arise in the prediction of loads during transient maneuvers. Transient changes in the variable wake geometry cannot be readily calculated. This limitation is quite significant for large transient changes in rotor thrust at high forward airspeeds which clearly involves large changes in wake geometry. Large changes in control position may produce significant changes in the characteristic frequencies and mode shapes for blade motion, thus requiring additional computation. Another difficulty is the simulation of changes in rotor speed due to a large change of time rate of change in power required by the rotor. The rotor tends to overspeed due to a nose-up pitch maneuver induced by cyclic control input or due to a suddenly reduced collective control setting. Increased complexity in the loads prediction methods presently under development will be required to account for these difficulties.

3. COST

The complexity of predicting loads and aeroelastic stability discussed above has obvious impact on the cost of conducting such analysis. Some indication of the computational cost associated with three analysis programs under development is presented in Figure 3. Each of these programs provides for estimation of loads during transient maneuvers and rigid body representation of the fuselage.

These programs differ primarily with regard to the representation of variable wake geometry and unsteady aerodynamics. Program C uses the most complex variable wake representation. Each program provides for the variation of the airfoil characteristics along the blade and includes aerodynamic section lift, moment and drag data as a function of Mach number and angle of attack. For Programs B and C, unsteady effects are included in both the stalled and unstalled angle-of-attack ranges. Program A includes only the real part of the Theodorsen function and omits stall delay characteristics.

The computational time required to trim the aircraft in a steady-state flight condition is shown to vary from 4 to 30 minutes, with Program C requiring the most time primarily due to the more complex variable wake analysis.

For transient response analysis, the ratio of computational time to real time (actual time on the vehicle) is shown to vary from 40 to 200.

As might be expected, the cost of these computations is quite high compared to most engineering analyses. Based on a computer storage capacity of from 60,000 to 110,000 words, the cost of computation is assumed to be $1000 per hour. The cost to trim the vehicle analytically in a steady-state flight condition ranges from $30 to $250. The cost to simulate the transient response during 1 second of flight of the vehicle ranges from $1 to $6.

Assuming that experimental flight test of a fully instrumented helicopter is in the range from $10,000 to $30,000 per hour and further that useful strain and motion data is obtained during only 10% of actual flight time, the cost of 1 second of flight-recorded data is in the range from $30 to $90.

Based on the foregoing cost estimates, which contain some assumptions (perhaps gross assumptions), it is evident that the cost of conducting loads and aeroelastic stability analyses using these methods is quite high on an absolute basis. Indeed, these costs are seen to be appreciable even when compared to the notoriously high cost of flight testing a fully instrumented rotary-wing vehicle.

Another practical problem arising from the complexity of these analyses needs to be considered. A large collection of mass, geometric, stiffness and aerodynamic data must be included in these computer programs. This requires very systematic data handling procedures to avoid errors while storing data in the computer. Detectable errors will delay the start of the analysis of a new vehicle and undetectable errors will contribute to inaccurate prediction. This tendency toward errors due to the large amount of data to be handled could have an adverse effect on both cost and prediction accuracy.

4. PREDICTION ACCURACY

Due to their complexity, these methods of predicting structural design loads and aeroelastic stability margin are not sufficiently accurate for those flight conditions which produce the highest loads and fatigue damage rates. Good correlation between predicted loads and flight measured loads for a lightly loaded rotor operating at a moderate airspeed under steady flight conditions can now be achieved using available methods; however, good correlation has not been demonstrated for either steady or transient operation of a highly loaded rotor at high forward airspeeds where stall flutter tends to develop on the retreating blade.

Recent correlation of loads due to stall flutter shows good qualitative agreement of the control system loading time history. This demonstrates that the essential mechanisms causing stall flutter can be included in the analysis methods. At present, no analysis method is available which provides good quantitative correlation simultaneously for control loads, net rotor forces and blade loads. At present, good simultaneous correlation cannot be obtained although the test data are available to provide guidance to the analyst.

The capability to achieve good quantitative correlation between test results and analysis conducted prior to testing is required; i.e., accurate prediction methods are needed.

5. UTILIZATION

In view of the high cost and lack of good correlation it is clear that effort must be made to improve accuracy and reduce the cost of calculation. In addition, criteria for efficient utilization of these complex methods must be developed.

Unlimited use of such a program as outlined here is not appropriate. Simplified analysis methods applicable to a specific phenomenon, e.g., ground resonance, are generally more economical than the complex methods. Model testing can also provide a method for load and stability analysis. Obviously, experimental flight testing can never be eliminated. Complex load and stability analysis methods should be used as appropriate to reduce development risk and cost.

The use of these complex methods can be expected to increase as their cost and accuracy are improved. As improvements are made, it will be possible to greatly reduce the cost associated with the many structural modifications that now must be made during development flight testing prior to achieving a satisfactory final configuration.

There appear to be four specific areas in which such complex analysis will be used extensively. One of these areas is determination of loads during penetration of the stall flutter region during both steady and transient maneuvers at high airspeeds.

The second area is stability margin analysis for rotor/vehicle concepts that exhibit significant nonlinear behaviour. Significant nonlinearities may lead to unacceptable risk during tests to determine the neutral stability boundaries if the nature, i.e., the qualitative behaviour of the vehicle near these boundaries is not understood. Furthermore, the extent of test required may be greatly increased by the existence of nonlinear characteristics. The ability to analyze the system response can reduce the risk and cost of these tests significantly.

The third area is the development of simplified loads and stability analysis methods useful for preliminary and detailed analyses. Once the validity of these complex analysis methods has been demonstrated by extensive correlation with flight test results, then simplified analysis methods hopefully can be developed for new structural problem areas with a minimum requirement for flight testing. Improved, simplified performance and handling qualities methods also can be derived.

The fourth area is the calculation of coefficients to be used in simplified linearized analyses.

6. CONCLUSIONS

Accurate loads and aeroelastic stability prediction is very complex. This complexity leads to high cost and poor accuracy. Improved prediction accuracy and lower cost are required before extensive efficient utilization of these complex methods. Criteria should be developed for selection of problems to be analyzed in order to assure efficient utilization of the complex methods. Improved accuracy and cost can provide for reduced development cost, early deployment of vehicles, and increased reliability and maintainability in the operational environment.

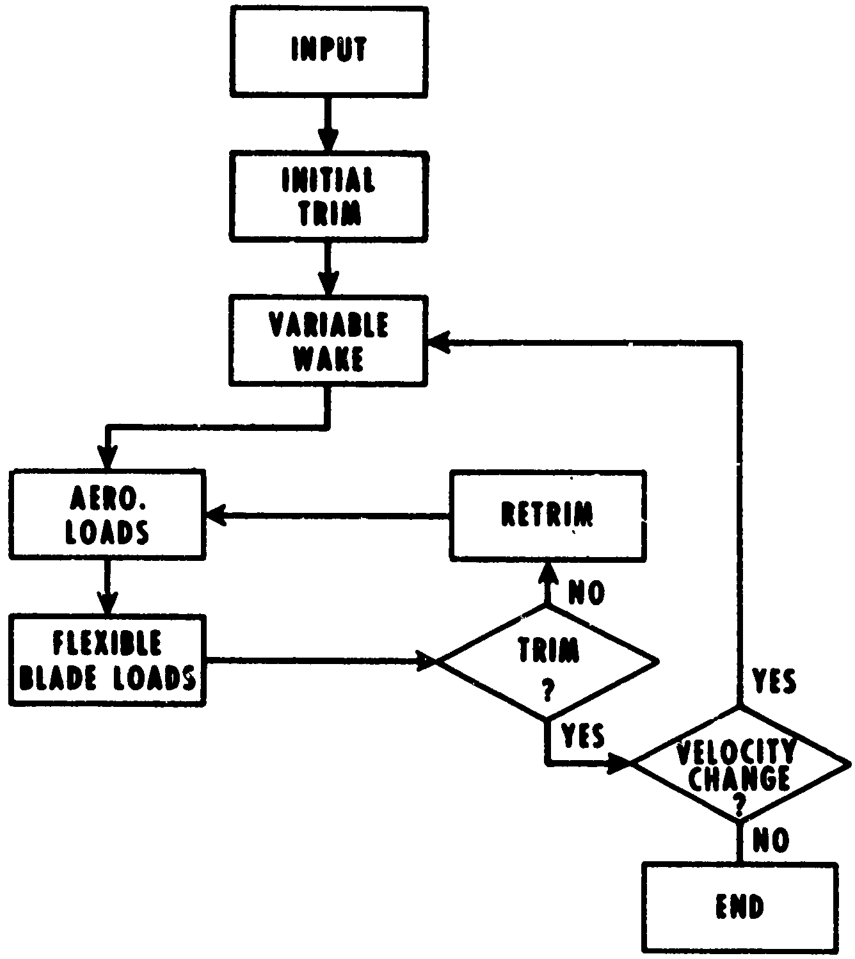

Fig.1 Rotor loads flow chart

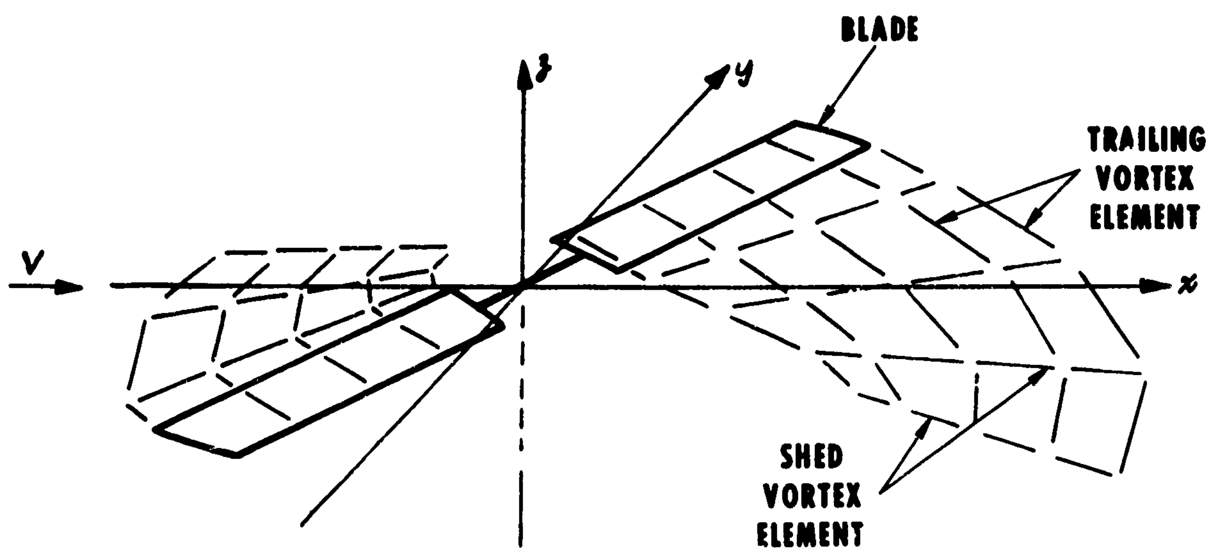

Fig.2 Variable wake geometry

PROGRAM / ITEM	A	B	C
VARIABLE WAKE	SIMPLIFIED EMPIRICAL	SIMPLIFIED ANALYSIS	MORE COMPLEX ANALYSIS
UNSTEADY AERODYNAMICS	REAL PART THEODORSEN	THEODORSEN PLUS STALL DELAY	THEODORSEN PLUS STALL DELAY
MINUTES TO TRIM	4	5	30
COMP. TIME / REAL TIME	120	200	40
COMP. COST TO TRIM	30	40	250
COMP. COST PER SECOND	3	6	1
FLIGHT DATA COST PER SECOND	30 - 90		

Fig.3 Analysis cost

IMPACT OF NEW STRUCTURAL CONCEPTS ON SYSTEM CAPABILITIES

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SUMMARY

V/STOL Systems unquestionably offer the most fertile field short of space application for exploitation of advanced structural concepts primarily because there is so much room for improvement. Though payload fractions of helicopters have become quite respectable thanks to the gas turbines and titanium, large improvements are still mandatory before high speed configurations can truly justify themselves economically. We have reached the point where the required improvement can now be projected with advanced materials and structural concepts, but the full exploitation of these advances can only be achieved by parallel advances in Aerodynamic, Dynamic and Control Technology.

INTRODUCTION

This paper attempts to take an "outside-in" look at the impact of Structural Concepts on System capabilities. It will not attempt to trace in any depth the roots of the many new ideas for solving structural problems or quantify the structural parameters. It will also not attempt to draw a line between structural concepts and the application of new materials. The two are inseparable. Instead we will back off and try to overview the sort of impact new ideas can have and attempt to assess what this can mean in terms of top level measures of performance and effectiveness. In addition, it will, I hope illustrate a principal thesis: although new structural concepts are in many cases achieving the obvious benefits of weight reduction by direct part substitution, and producing numerous side benefits in terms of survivability, safety, reliability and dynamic forgiveness, the full potential can only be obtained by increased competence in applying the new freedom these techniques give us to form our aerodynamic surfaces and specify elastic requirements.

As will be quickly apparent, this paper draws heavily on recent publications both domestic and foreign on the subject. In addition, I must acknowledge the assistance of Sikorsky experts in the field: Bill Coronato, Sikorsky Chief of Aircraft Design; Bill Paul, Rotor Design Supervisor; Mike Salkind, Chief of Structures and Material and their staffs.

Current System Capabilities

By way of introduction to the potential of advanced structural concepts, Fig. 1a from Ref. (1) summarizes our current VTOL system capabilities in terms of payload to gross weight ratio. While helicopters today can achieve 200 nautical mile PL/GW ratios of 27% at speeds of 150 to 175 knots (up from about 19% at 120 knots 10 years ago) current technology is inadequate to provide increased productivity i higher speed VTOL configurations because payload lost to weight empty increases offsets the speed advantage. Fig. 1b, also from Ref. (1), illustrates this point in terms of typical transport efficiencies for a military mission. While the dollar figures will vary drastically with assumptions concerning utilization and maintenance efficiency (commercial operations have significantly better productivities) the relative picture is probably valid for most applications. The corollary to all of this is that any reduction in weight empty has tremendous leverage on payload. A 5% reduction in weight empty can easily produce a 20%-40% increase in payload and productivity for 200 mile VTOL missions.

Similiarly the maintenance burden of current VTOL's exceeds desirable norms for economic operation, accounting for about 40% of the per seat mile costs, so there is a tremendous potential for greatly improving VTOL economics by reducing the maintenance burden.

Finally the nature of the VTOL mission, particularly the military mission, puts a very large premium on the ability to survive both enemy action and the inevitable mishaps of nap of the earth operations, so that any contribution advanced structural concepts can make to survivability will have extra significance for VTOL.

Potential Structural Concept Impact

1. Rotor Blades

Rotor Blades, be they helicopters' or direct lift fans', offer the most specialized challenges to structural technology for VTOL applications. In most other cases, the VTOL developer can expect some rake-offs from other aeronautical system developments, but with rotors, the industry is strictly on its own for the concept brainstorms, the R&D funds to develop them, and the service experience to prove them out. It is also in the rotor that the interdependence of structural technology and the aeromechanics disciplines are best illustrated.

Fig. 2 illustrates a typical situation. The original CH-53A had been optimized with a low twist blade to provide relatively high cruise speeds. Research to improve static performance had shown that by modifying twist and tip planform significant improvement could be made in hover G.W. capability. However, with existing aluminum spar technology the stresses induced by the higher twist significantly reduced allowable cruise speeds. By applying titanium spar technology it has been possible to design a blade which will not only tolerate the stresses resulting from higher twist, but also provide more chord for the same blade weight and a more sophisticated airfoil and tip planform, thus increasing even further hover efficiency and high speed capability. The result is an 8½% increase in payload and 10% increase in speed for a productivity increase of 19% all with no increase in Weight Empty.

The most advanced example to date of the application of metal spar technology to provide a blade tailored to non uniform requirements is the "ABC" blade. Fig. 3 shows the completed blade. The tapered spar is titanium and the skin is fibreglass. Ref. 2 reports on the development of this blade and its fabrication techniques. The chord tapers from 10.8" at the root to 5.5" at the tip, while the airfoil section tapers from 30% to 6%. However, the most novel structural requirement of the advancing blade concept is the necessity for the blade to transfer a significant portion of its lift to the hub by its cantilever stiffness rather than depending solely on centrifugal force as is the case with articulated rotors and even the less rigid "hingeless" types

This structural capability opens up the potential application of rotors on two fronts. First, it becomes possible to free the lifting rotor from the retreating blade stall limits of speed by removing the requirement for balanced first harmonic flapping moments and allowing the lift to be carried almost entirely on the advancing blade. This, of course, requires a counterrotating coaxial rotor configuration to provide symmetrical lift, but the stiff blades with suppressed flapping decreases the vertical separation requirement which has been a principal disadvantage of coaxial configurations. This is, of course, the ABC or Advancing Blade Concept described in Ref. (2). The system capability this provides is best illustrated in Fig. 4 where the ability of the ABC to carry lift at high advance ratios is compared with conventional rotors.

A second potential advantage of a "cantilevered" blade has yet to be exploited. As numerous investigators have pointed out, weight reduction in conventional rotors becomes ultimately limited by the high coning angle which results as centrifugal force is reduced. This produces a square/cube law relation which says in effect that as GW is increased, if disc loading, blade cg, tip speed and allowable coning angle are held constant, the rotor blade weight relative to gross weight must inevitably increase with size. Fig. 5 illustrates the trend, which is fortunately only a square root growth. So far, and even through forseeable HLH rotor sizes, this trend has been offset by disc loading growth, some increase in coning angles and redistribution of weight in the blades (tip weights will help) to maximize centrifugal force. And, in any case, structural technology until recently hasn't offered much opportunity to build blades for less weight. But as we look ahead to composites and much larger rotors, we can clearly see the day where coning suppression and cantilevered lift transfer will be necessary to hold down the rotor weight fractions of very large helicopters or to use the full potential of composites to reduce the weight fraction of current sizes. Thus the ability to treat the aerodynamic and dynamic consequences of a "very rigid" rotor must be developed side by side with the structural technology if we are to realize the weight reduction potential of advanced materials in rotors.

The application of fiber composite technology to rotors is an immense and extremely active subject that we can only touch upon briefly here. The performance pay off through the use of composites is particularly promising because of the ability to mold almost any aerodynamic shape and tailor the elastic properties especially to take advantage of their anisotropic possibilities. Actually materials with fibrous and anisotropic characteristics have been used in rotors since the earliest days of the helicopter. The earliest Sikorsky, Bell and Piasecki helicopters all started with molded wood blades, and fibreglass has been the objective of a good deal of research for at least 15 years. Probably the most sophisticated use is found in the Boelkow BO-105 (Ref. (3)) and in the Aerospatiale SA-341. But the most advanced application of composites so far reported is undoubtedly the Boeing CH-47 blade developed under U. S. Air Force sponsorship and described in Ref. (4) and (5). Ref. (4) points up nicely the many constraining considerations which must be defined by the dynamicist or the aerodynamicist before the composite design can be optimized and illustrates an interesting approach to identifying those design parameters which can meet all these requirements. Because of coning angle limits, weight reduction was not a major goal, but presumably the major system level pay off will be in higher cruise speeds resulting from alleviation of stall flutter boundaries, thus pushing out speed capabilities at higher gross weights much as reducing 1 per rev stresses has improved the CH-53 envelope as illustrated in Fig. 2. Ref. (5) points up the further benefits which accrue from the use of composites in more exotic VTOL where rpm variation increase the demands on aeroelastic tuning.

2. Tail Rotors

Tail rotors also stand to benefit from the application of composite technology. Although TR stress limitations are less apt to constrain the aircraft system envelope, tail rotors have conventionally been over designed and should have significant potential for weight reduction which is especially important since weight reduction at the tail of a single rotor helicopter can have a big impact on easing the design balance problem. Also tail rotors can provide a good proving ground for new blade construction concepts. In fact, the first flight to our knowledge of a VTOL dynamic component fabricated from high modulus composite materials, was the testing in December of 1968 of the Sikorsky Boron Epoxy spar tail rotor shown in Fig. 6. This blade was designed to give higher chordwise stiffness than the production blade. Fig. 7 illustrates the stiffness improvement that was achieved along with 5% weight reduction.

The tail rotor developed by Kaman for the Bell UH-1 is an excellent example of the utilization of the anisotropic properties of composites. (Ref. (6) and Fig. 8). By using a single continuous unidirectional fibreglass epoxy spar, running through the hub from tip to tip, all the centrifugal and bending loads are reacted by the high strength and stiffness of the unidirectional fibers. Feathering action was provided by the torsional deflection of the basic spar which has low torsional stiffness. The primary pay off here was the reduction of maintenance requirements, although the program also made important contributions to establishing the producibility of this sort of design.

3. Rotor Heads

The rotor heads themselves can also benefit from advanced materials and concepts. The steady increase in the application of titanium through the 1960's achieved up to 20-25% weight reduction in rotor heads while increasing the safe life and retirement times. Initial studies of the application of composites to rotor heads such as those reported in Ref. (4) have suggested only 12% weight reduction relative to conventional steel and aluminum constructions; in rotor heads the local load and attachment problems become all important so whole new concepts of rotor head design may be required to fully exploit the high stress potential of composites. Current programs directed toward development of a composite rotor head may well provide the answer. However, the most significant advances coming up in the hub area attributable to structural concepts probably is the reduction of maintenance requirements resulting from the application of elastomerics. Recent studies of the potential of a spherical bearing elastomeric rotor head for the CH-53, see Fig. 9, have suggested a reduction of field maintenance costs of 50% and extension of TBO's by a factor of 8 which adds up to a 3 to 1 reduction in Life Cycle Cost contribution of the rotor head.

Looking to the future, more component and systems application development should reduce the bulkiness of elastomeric designs or flexbeam composites technology applied to main rotors may provide the maintenance benefits of elastomerics with less weight and frontal area. On the size frontier, new fabrication techniques such as diffusion bonding will be exploited to economically produce the large monolithic elements demanded by very large rotary wing aircraft.

4. Variable Geometry Concepts

Perhaps the most challenging goal for novel structural concepts would be the development of simple and feasible variable geometry concepts. Even more than the option to arbitrarily select rotor geometry unencumbered by constant section requirements, the aerodynamicist and dynamicist would like to vary his geometry from flight condition to flight condition. Cries of "down with complexity" have so far constrained the rotary wing designer from variable geometry solutions although his fixed wing counterpart has been making it a standard part of his bag of tricks. One has only to contemplate the 727 wing with its high lift devices to realize how dependent CTOL system effectiveness has become on variable geometry. VTOL must eventually be allowed the privilege to do such things as alter blade twist distribution or airfoil camber between hover and cruise or even perhaps with blade azimuth position, to vary shaft inclination (the tilt rotor is, of course, the ultimate product of this degree of freedom) and to alter rotor diameter. A possible concept for rotor diameter retraction is shown in Fig. 10 and described in Ref. (7). The benefits of 40% diameter variation are illustrated in Fig. 12 for a compound helicopter and for a tilt rotor. Although work on this sort of thing is progressing systematically, the VTOL designer will probably not be allowed to exploit these possibilities until overall VTOL system weight fractions and maintenance burdens have been reduced to the point where the extra complexity can be afforded. For this we must look for all of the structural concept advances we can get and we must also find simpler means of achieving variable geometry such as the compliant airfoil concept proposed by Fred Gustafson of NASA some years ago.

5. Drive Systems

Virtually any VTOL aircraft, (except direct lift engine configurations) must use connecting drive shafts of some sort to synchronize power distributions to multiple lifting rotors or to counter torque tail rotors. Generally this shafting must operate at high speeds necessitating high fundamental frequency, and transmit as much torque as possible for minimum weight. The resulting requirement for high torsional strength, buckling stability and lateral stiffness can be met particularly well by the proper application of composite technology once the designer can define what he wants.

The potential advantage of the application of composites to the tail rotor drive shaft problem is illustrated in Fig. 12. By using two concentric tubes with a low density core of foam to overcome torsional elastic instability problems, a drive shaft for the S-61 helicopter was designed with significantly better failure allowables than was achievable with either aluminum or single shell composites. A major and unexpected advantage of this design turned out to be ease of dynamic balancing. Apparently the higher damper capacity of the 45° composite material offset any tendency towards less uniform mass distribution.

Of course, the most important attribute of composites applied to shafting is the ability to tailor stiffness distributions to optimize shaft length and supports for minimum weight, by using 45° laminae to react torque and axial layers for bending. A good example of this is contained in Ref. (8) describing a 92 inch graphite epoxy drive shaft built by Bell Helicopter and the Whittaker Corporation. In this design in which the rpm, diameter, span and number and orientation of plies was optimized to meet critical speed, torsional stability and torsional strengths requirements, an overall weight reduction of 28% was realized and studies indicated this might extend to 34% with carbon eopxy. Ref. (4) is even more optimistic citing weight reduction as large as 52% for Drive System Shafting.

6. Transmissions

Material improvements in transmissions will of course reduce weight. Titanium for gears and short shafting and composites for gearbox housing and for large ring gears will both be used but advanced steels will also have their place. Typical figures quoted for the substitution of composites for magnesium/aluminum technology in housing and support assemblies suggest up to 25% weight reduction.

However, structural weight reduction will combine with many other conceptual improvements to greatly enhance systems effectiveness. New bearings should yield 3 to 7 times the lives of current technology and gearing material and forming improvements should reduce the cost of fabrication as well as improving life.

Conformal gear techniques offer the potential of 3 to 4 times the load capacity for a given size and weight if critical tolerance and rigidity requirements can be met. The Westland WG-13 gearbox will be a key development in the practical implementation of this approach.

A basically different structural approach to gearbox design is the roller gear concept. This concept, originated by TRW, has been extensively studied at Sikorsky and currently a 3700 horsepower transmission for the H-3 helicopter is being built under AAMRDL sponsorship and reported in Ref. (9). Fig. 13 shows the H-3 design which is projected to yield a 7% weight reduction over conventional designs and perhaps a ½% improvement in transmission efficiency (a 12% reduction in transmission losses which typically run about 4%).

7. Control System

Stiffness is all important in control systems especially as high speeds are often limited by the torsional response to aeroelastic excitations such as stall flutter in modern high speed rotary wing aircraft. The increase in blade torsional stiffness achievable will help but the control system, and swashplates in particular, can also benefit by stiffening.

8. Airframe

Time and space won't permit more than the most cursory overview of the impact of advanced structural techniques on airframes. Much of the gains will mirror comparable improvements in fixed wing aircraft. However VTOL, and rotary wing aircraft in particular, have special requirements for dynamic response to preclude resonances with rotor excitation frequencies and to optimize model distribution of the response that does occur. Here again, the dynamicist must be prepared to specify the stiffness distribution he wants which requires that he develop the technology to take advantage of the new freedom highly efficient stiff structural concepts can provide him.

A good example of the benefits which can be achieved is illustrated in a recent requirement to stiffen the tail cone of the CH-54B to increase the separation of first mode frequency from one per rev. Fig. 14 illustrates the problem. A conventional solution of beefing up the aluminum skin weighed 160 lbs. The bonded boron-epoxy strings reinforcement solution compared in Fig. 15 with the present solution weighed only 48 lbs. for a 115 lb. weight savings. Such a saving could justify the composite solution with Boron Epoxy priced at significantly more than the current market price of $300/lb. Ref. (10) reports on the design, analysis and test of this airframe modification which is to be delivered for service evaluation early in 1972.

Truss construction techniques may be especially attractive particularly for cranes, because they again can take advantage of the anisotropic properties of fibers and the "Captive Column Concepts". Ref. (11) cites projected weight savings of more than 50% for composite compression tubes compared to metal tubes as illustrated in Fig. 16. One specific approach, the "captive column" concept, summarized in Ref. (12), utilizes 3 or more parallel column elements between a central core structure and a surrounding skin filament structure to enhance the strength/weight ratio in buckling. A design study of a truss fuselage for the CH-54B, shown in Fig. 17, suggested a savings of 41% in the center section, 59% in the tail cone and 50% in the landing gear. Combined with vertical drag reduction of the open truss this could yield more than a ton of payload for low speed missions. Similar studies of weight savings potential for the compression boom of a twin lift system (see Fig. 18) suggested better than 2 to 1 weight reductions. This problem is ideally suited to the "captive column concept". Here the benefit is not only in weight empty but also is easing of ground handling problems and simplifying the logistics of boom availability (a major concern with regard to the twin lift concept). This, then is another example where structural technology combined with other technology, in this case the advanced control technology which can make twin lift feasible, together can produce a significant advance in system capability.

No discussion of airframe technology can overlook the large variety of sandwich construction concepts. Honeycomb has been making a big contribution for years, particularly in the area of cost savings of curved elements of semi structural skin. An interesting marriage of these concepts with high modulus materials is illustrated in Fig. 19. Studies of this cargo floor promised 20 to 40 percent weight savings over conventional construction by using a sandwich structure of high modulus boron epoxy bottom skin combined with a low density core to resist deflection, a balsa wood layer to absorb impact energy and a titanium skin to provide a puncture and wear resistant surface.

It would be impossible to do adequate justice to all the other diverse concept improvements that can be expected to enhance the many other aspects of airframe design. A particularly intriguing possibility is the more sophisticated structuring of anisotropic fibers into specially oriented space frame structures. A good example of this is the Tetracore concept under development by AAMRDL, which not only tries to utilize unidirectional fibers in an optimum manner but also offers inherent redundancy. Ref. (13) describes this concept. The basic Tetracore element is illustrated in Fig. 20, along with the AVCO "infiltrated extrusion" concept, another approach to using fiber composites to maximum advantage.

Other conceptual innovations deserving mention as representative of areas of future contributions of structural technology to airframes are: 1) Structural armor to decrease vulnerability and allow the battlefield helicopter to stay in the nap of the earth and allude larger intensity threats, see Fig. 21 and Ref. (14); 2) The selective use of frangibility to increase crash worthiness and survivability (see Fig. 25) and, 3) the use of ingenious means of providing flexibility or damping for vibration reduction and control such as the SUD "Barbecue Grill" (Ref. (16)), the Sikorsky Rotor Head Bifilar Absorber, (Ref. (17)), the Kaman "DAVI" antiresonant absorber, (Ref. (18)), and the various transmission isolation schemes, Ref. (19). Fig. 23 illustrates these concepts.

Recently Sikorsky completed an assessment of the total weight reduction potential of all these promising developments as they might apply to a large crane helicopter airframe, when consideration was given to cost factors. Fig. 24 sums up the results in terms of reduction in airframe percent of gross weight. These studies suggested the feasible range of reductions in airframe weight should be between 22 and 26%. More optimistic forecasts are contained in Ref. (3), where airframe weight reduction as high as 39% are forecast. The difference is undoubtedly related to the question of how far the application of high modulus fiber composites can be carried on a cost effective basis; only time will answer that question. Certainly the major technical challenge that fiber technology must respond to is reduction of cost, both the cost of the basic material and the cost of fabrication into the spatial orientation that takes advantage of the anistropic characteristics of fibers.

CONCLUSIONS:

System Sum Up

What does all this mean in terms of system capabilities at the system level? There is obviously no clear way to forecast to what degree economic consideration will allow the exploitation of the capabilities, we have, but the following general trends seem to be clearly attainable within the next 10 years, if the necessary development support is forthcoming:

1. Weight reductions running as high as 25 to 40% in certain key subsystems should combine to produce about 15% overall reduction in weight empty, so that helicopter EW/GW ratios now typically running close to 62% should drop to about 53%, with comparable weight empty reductions in more exotic rotor VTOL configurations.

2. Aerodynamic efficiency improvements made by the combination of a better understanding of rotor aerodynamics and a greater freedom to specify the rotor geometry desired made possible by structural advances will combine with improved power plant efficiencies to reduce the fuel ratios for 200 nautical mile mission from 11% of GW to about 9%.

3. These aerodynamic improvements combined with the judicious use of some variable geometry will make significant increases in cruise speeds achievable, though parasite drag of vertical lifting systems will remain the ultimate constraining factor.

4. These improvements will combine to produce payload to gross weight fraction improvements ranging from 40% for the pure helicopter to 80% for the more exotic low disc loading composites (tilt or semi-stowing types).

5. Transport efficiency (Productivity) factors will rise even more provided the materials and fabrication costs of utilizing these advanced concepts can be constrained to the point where they are offset by a higher degree of automation.

Table I summarizes in a quantitative way, these "crystal call" trends. The dramatic impact on transport efficiency is illustrated in Fig. 25, if we assume cost factors remain a constant function of weight empty.

TABLE I

IMPACT OF IMPROVED TECHNOLOGY

ON SYSTEM EFFICIENCIES

	1970			1980		
	Helo	Compound	Composite	Helo	Compound	Composite
WE/GW	.62	.67	.72	.53	.57	.61
Fuel/GW	.11	.12	.12	.094	.10	.10
PL/GW	.27	.21	.16	.38	.33	.24
Velocity	175	235	350	190	275	400
PV/WE	63	73.6	78	136	159	190
Ton-Miles / $	0.9	.92	.96	1.61	2.0	2.34

Of course this is only a part of the story. Advanced structural concepts, particularly in rotor heads and transmission elements can greatly reduce day to day maintenance burdens and TBO's to the point where Maintenance Manhours per flight hour are reduced by a factor of 2 or 3 to 1. Vulnerability will be decreased and survivability increased not only by new structural concepts but also by new configuration concepts they will make possible. Ride comfort will be increased and airframe fatigue problems virtually eliminated by advanced concepts of vibration control. The size of commercial VTOL will increase, producing the DOC benefits which go with a larger vehicle.

Structural technology will undoubtedly pace these improvements, particularly in the evolution of techniques to take advantage of the anisotropic characteristics of composites. But, as these illustrations show, the system designs and the aerodynamics and structural dynamic disciplines must come up with comparable concepts improvements before the full potential can be realized.

REFERENCES

1. Leroy H. Ludi, Composite Aircraft Design, American Helicopter Forum Proceedings, May 1967.
2. R. Burgess, Development of the ABC Rotor, American Helicopter Forum Proceedings, May 1971.
3. E. Jarosch and A. Stepan, Journal of the American Helicopter Society, Vol. 15, 1970, 33-41.
4. Stratton and White - "Structural Application of Advanced Composite Materials". AIAA 5th Annual Meeting, October 1968 (AIAA Paper No. 68-1039).
5. Warren Stratton, "The Potential of Advanced Composites for V/STOL Propellers". V/STOL Planning Technology Conference, Sponsored by USAF Flight Dynamics Laboratory, Las Vegas, September 1969.
6. Maloney, Clark, McIntyre: "Study and Evaluation of Directed Glass Fiber Reinforced Plastic Tail Rotor Assembly", USAAVLABS Technical Report 69-43, October 1969.
7. Segel and Fradenburgh, "Development of the TRAC Variable Diameter Rotor Concept", AIAA No. 69-221, February 1969.
8. H. Zinberg and M. Symonds, "The Development of an Advanced Composite Tail Rotor Driveshaft", AHS Preprint 451, AHS Forum, Washington, D. C., June 1970.
9. L. R. Burroughs, "Roller Gear Transmission Development Program, American Gear Manufacturer's Association Aerospace Gear Committee Meeting, February 1971.

10. R. T. Welge, "Application of Boron/Epoxy Reinforced Aluminum Stringers for the CH-54B Helicopter Tail Cone", NASA-CR-11929.

11. D. Collings and F. Steinlein, "The Design of Composite Compression Tubes", AIAA Paper 70-898, Los Angeles, July 1970.

12. "Captive Column Structure Optimizes Strength/Weight Ratio", Design News Annual, 1970.

13. J. E. Figge, Tetracore, A Three Dimensional Space Structure, Shell Aviation News, Number 391(1971).

14. G. R. Stack, Aerial Armored Reconnaissance Vehicle Armor, U. S. Armor Association, May-June 1971.

15. M. J. Rich, Vulnerability and Crashworthiness in the Design of Rotary Wing Vehicle Structures, SAE No. 680673, October 1968.

16. F. L. Legrande, "Structural Solutions Investigated in Connection with Vibration Problems in the SA 330", American Helicopter Society, May 1968.

17. W. F. Paul, Development and Evaluation at the Main Rotor Bifilar Absorber, American Helicopter Society Paper No. 354, 1969.

18. W. F. Flannery, "Dynamic Antiresonant Vibration Absorber", Kaman Aircraft Corporation Report No. RN 631 November 1963.

19. P. W. Von Hardenberg and P. B. Saltanis, "Ground Test Evaluation of the Sikorsky Active Transmission Isolation System", American Helicopter Society Paper No. 514, May 1971.

Salkind: Application of Composite Materials; Edited by M. Salkind and G. Hollister, to be published by ASTM, Chapter III; VTOL Systems

FIGURE CREDITS

1.	Current VTOL Capabilities	-	U. S. Army Air Mobility Research and Development Lab, Ref. (1)
2.	Typical Improved Rotor Technology Payoff	-	Sikorsky Aircraft
3.	ABC Titanium Blade	-	Sikorsky Aircraft, Ref. (2).
4.	Payoff from the Advancing Blade Concept	-	Sikorsky Aircraft, Ref. (2).
5.	Coning Angle Limited Blade Weight Trend with Gross Weight	-	Sikorsky Aircraft
6.	Sikorsky Boron Epoxy Tail Rotor	-	Sikorsky Aircraft
7.	Stiffening Effect of Composite Construction	-	Sikorsky Aircraft
8.	Kaman "Directed Glass Fiber" Tail Rotor	-	Kaman Corporation, Ref. (6)
9.	Three Axis Elastomeric Rotor Head Design	-	Sikorsky Aircraft
10.	Telescoping Rotor Concept	-	Sikorsky Aircraft, Ref. (7)
11.	Payoff from Retractable Rotor	-	Sikorsky Aircraft, Ref. (7)
12.	Effect of Sandwich Construction on Torsional Elastic Stability Limits	-	Sikorsky Aircraft
13.	Roller Gear Drive Assembly	-	Sikorsky Aircraft
14.	Crane Fuselage Stiffness Requirements	-	Sikorsky Aircraft
15.	Alternate Stiffening Solution to CH-54B Requirement	-	Sikorsky Aircraft
16.	Weight Saving Achievable by Composites for Compression Tubes	-	Sikorsky Aircraft, Ref. (7)
17.	Truss Airframe for CH-54B	-	Sikorsky Aircraft
18.	Spreader Bar for Twin Lift	-	Sikorsky Aircraft
19.	Composite Cargo Floor	-	Sikorsky Aircraft
20.	Novel Structural Concepts Applying Composite Fibers	-	AVCO Systems Division & Sikorsky Aircraft Model of AAMRDL Concept
21.	Armored Airframe Resistance to Small Arms	-	Sikorsky Aircraft
22.	Typical Structural Design Features for Reduced Vulnerability and Increased Crashworthiness	-	Sikorsky Aircraft, Ref. (17)
23.	Vibration Suppression Concepts: Rotor Head Absorber	-	Sikorsky Aircraft, Ref. (17)
	Active Transmission Isolation	-	Sikorsky Aircraft, Ref. (19)
	"Barbecue Grill" for Transmission Tuning	-	Aerospatiale, Ref. (16)
	"Davi" Seat Isolation	-	Kaman Corporation, Ref. (18)
24.	Range of Potential Weight Reduction with Composite for Typical Large Crane Helicopters	-	Sikorsky Aircraft
25.	Productivity Benefits from Application of Advanced Structural Concepts	-	Sikorsky Aircraft

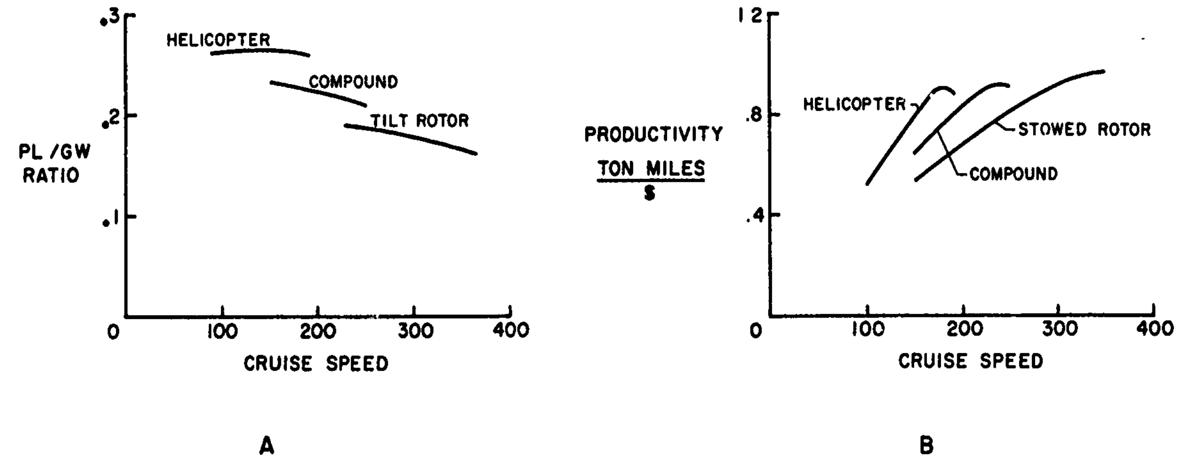


Figure 1. Current VTOL Payload Capabilities and Productivity.

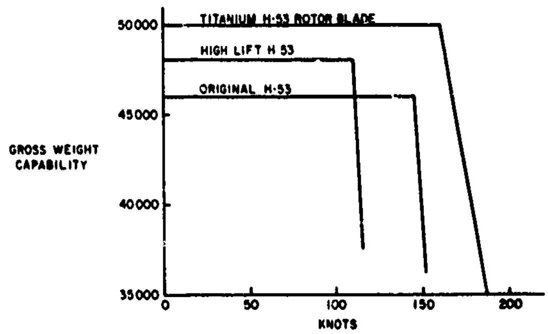


Figure 2. Typical Improved Rotor Technology Pay-off.

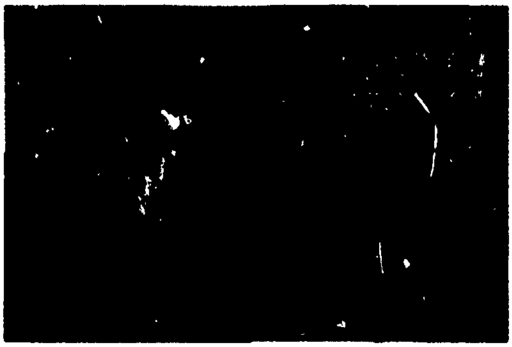


Figure 3. ABC Titanium Blade.

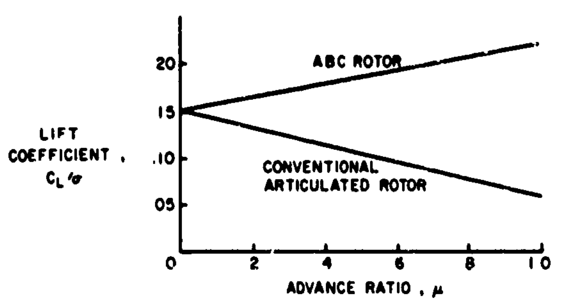


Figure 4. Pay-off From "Advancing Blade Concept".

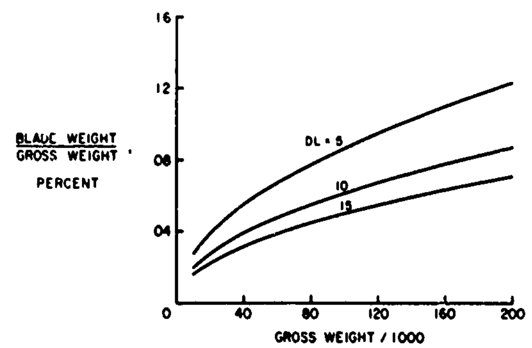


Figure 5. Coning Angle Limited Blade Weight Trend With Gross Weight.

Figure 6. Sikorsky Boron Epoxy Tail Rotor.

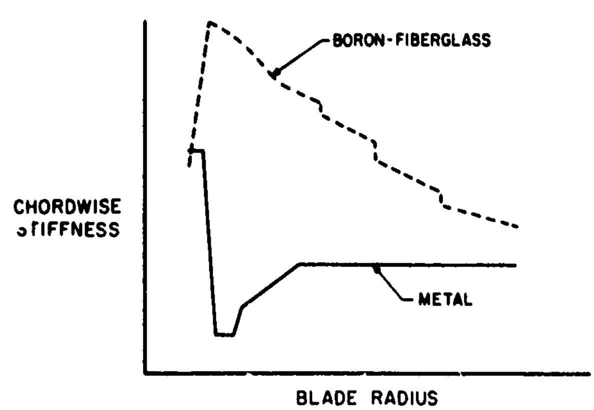


Figure 7. Stiffening Effect of Composite Construction.

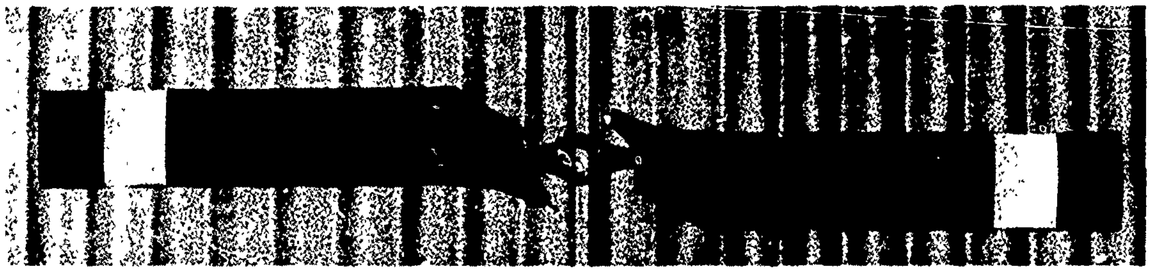


Figure 8. Kaman "Directed Class Fiber" Tail Rotor.

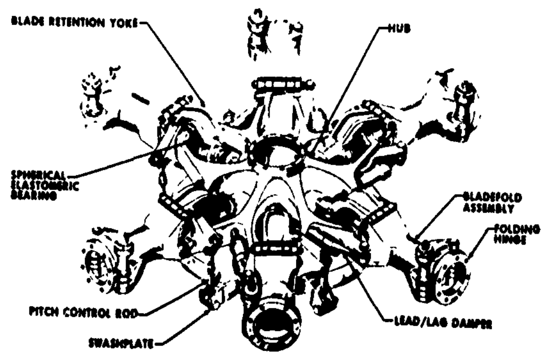


Figure 9. CH-53A/D Main Rotor Head Spherical Elastomeric Bearing Concept.

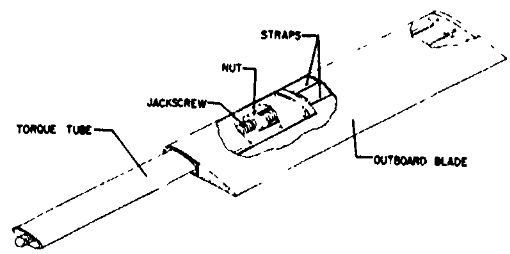


Figure 10. Telescoping Rotor Concept.

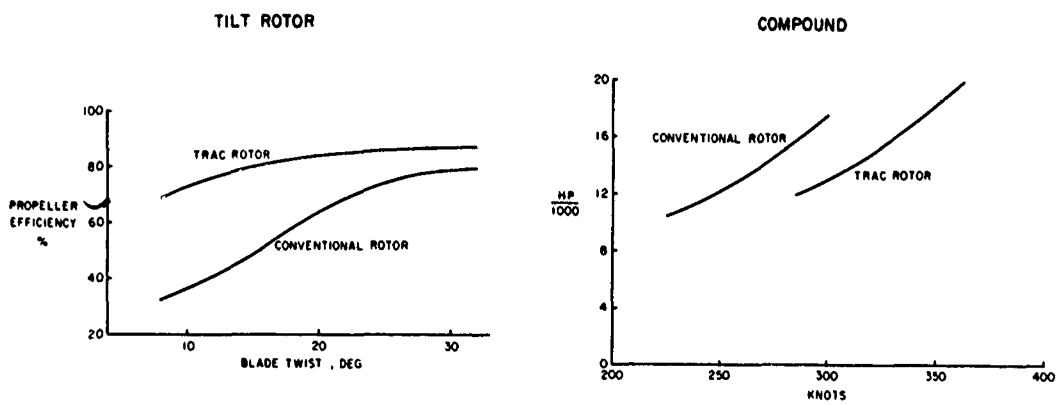


Figure 11. Pay-off From Retractable Rotor.

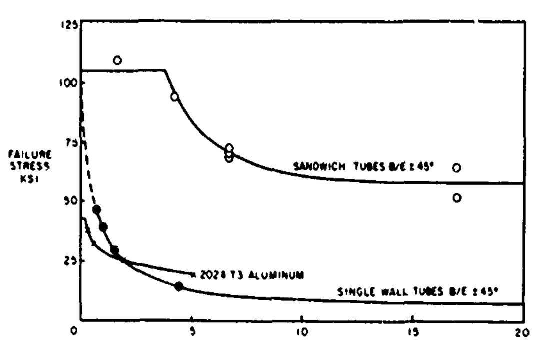


Figure 12. Effect of Sandwich Construction on Torsional Elastic Stability Limits.

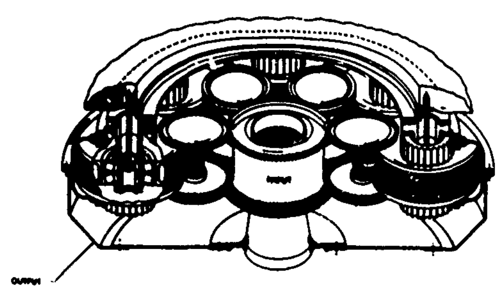


Figure 13. Roller Gear Drive Assembly.

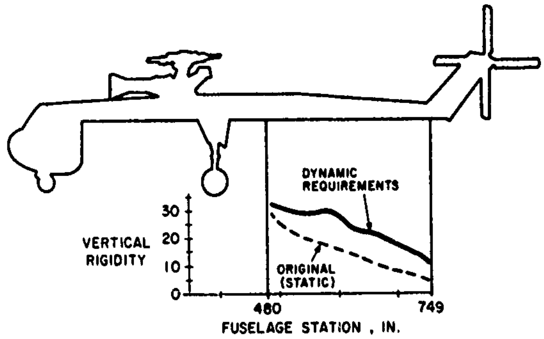


Figure 14. Crane Fuselage Stiffness Requirement.

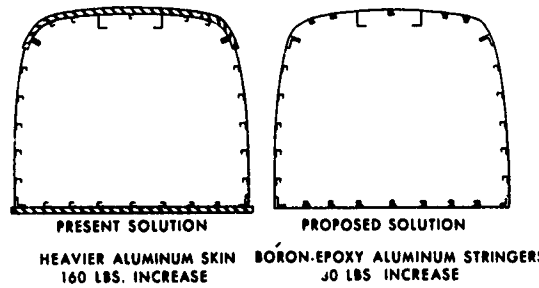


Figure 15. Alternate Stiffening Solutions to CH54B Requirement.

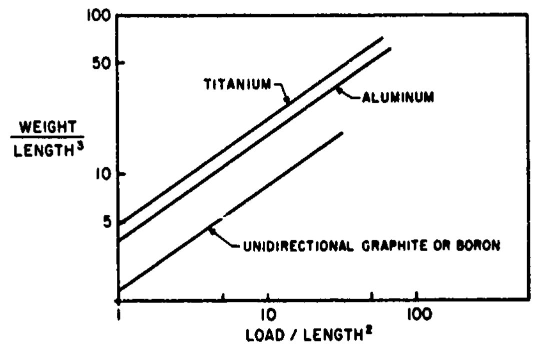


Figure 16. Weight Saving Achievable by Composite for Compression Tubes.

Figure 17. Truss Airframe for CH54B.

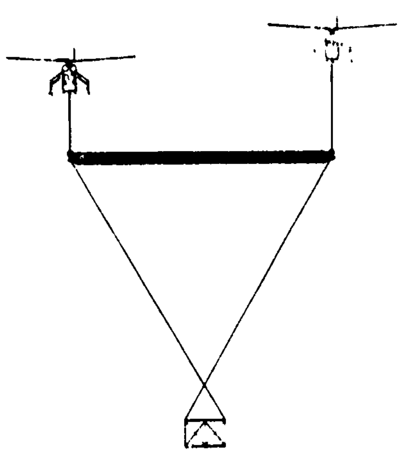
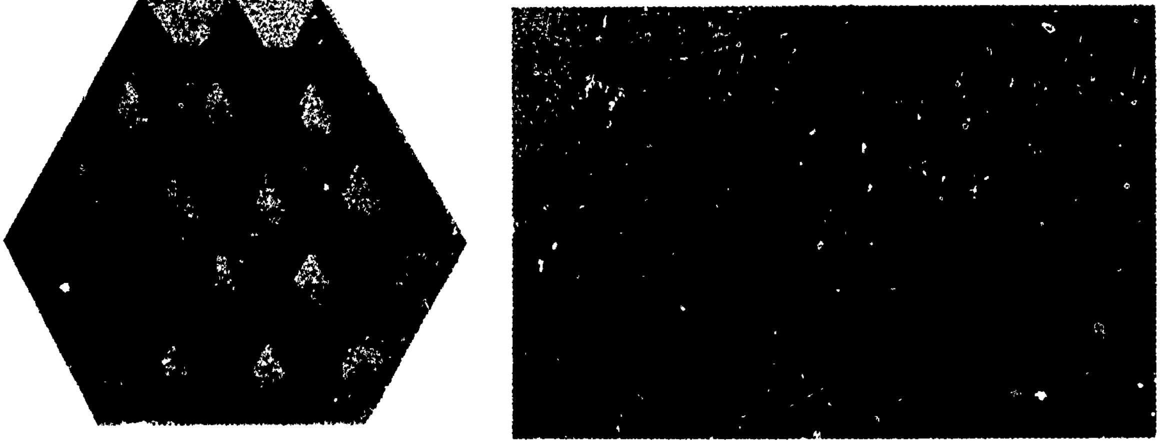


Figure 18. Spreader Bar for Twin Lift.

Figure 19. Composite Cargo Floor.



TETRA-CORE

Figure 20. Novel Structural Concepts Applying Composite Fiber.

Figure 21. Armored Airframe Resistance to Small Arms.

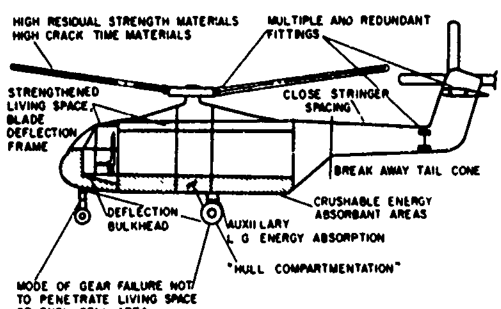
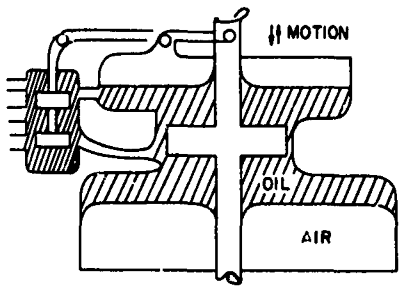
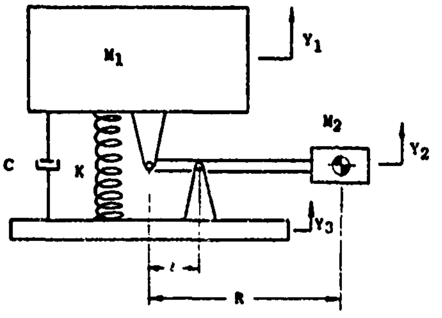


Figure 22. Typical Structural Design Features for Reduced Vulnerability and Increased Crashworthiness.

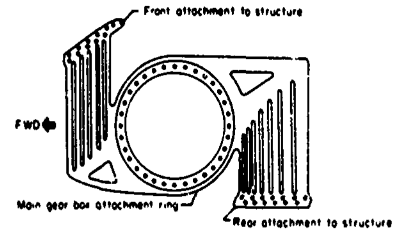


ACTIVE TRANSMISSION ISOLATION

SIKORSKY ROTOR HEAD ABSORBER



KAMAN SEAT ISOLATION



AEROSPATIAL "BARBEQUE GRILL"
FOR TRANSMISSION TUNING

Figure 23. Vibration Suppression Concepts.

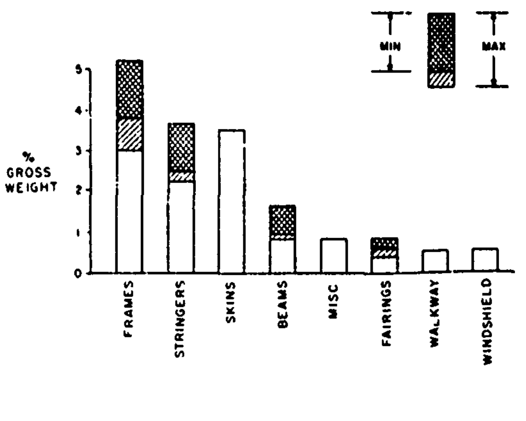


Figure 24. Range of Potential Weight Reduction With Composites (Typical Large Crane Helicopter).

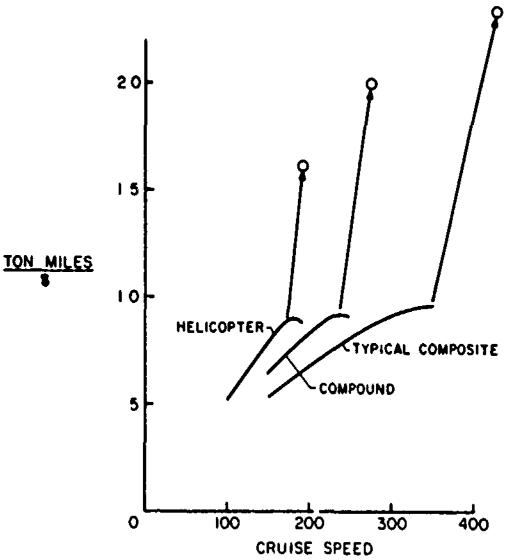


Figure 25. Impact of Advance Technology in Transport Efficiency.

EVALUATION, DEVELOPMENT, AND ADVANTAGES OF THE HELICOPTER TANDEM DUAL CARGO HOOK SYSTEM

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SUMMARY

Helicopter transport of external cargo is recognized by the U.S. Army for its military applications, efficient use of available rotary-wing equipment, and enhancement of aircraft safety. Improvements in this technique could provide the transport of external cargo at the maximum speed of the helicopter, routine operation under instrument flight rules (IFR), precise placement of the load, and could eliminate the problems in hover such as the dust cloud and static electricity.

Feasibility studies have shown the potential of the tandem dual hook concept as a viable base on which to build an improved cargo-handling system. Wind tunnel tests and several full-scale flight tests have confirmed these advantages and have provided design requirements for a new generation of external cargo helicopters.

Production incorporation of a dual cargo hook system is planned for the heavy-lift helicopter (HLH). The system incorporates other features such as variable longitudinal hook positioning, differential winching, load motion feedback, and augmentation of the cargo system operator's vision under conditions of poor light and thick dust.

This paper describes the requirements for an improved helicopter external cargo-handling system, reviews the programs which have established the feasibility of a tandem dual cargo hook system, and explains the system slated for the heavy-lift helicopter.

IMPORTANCE OF EXTERNAL CARGO TRANSPORT

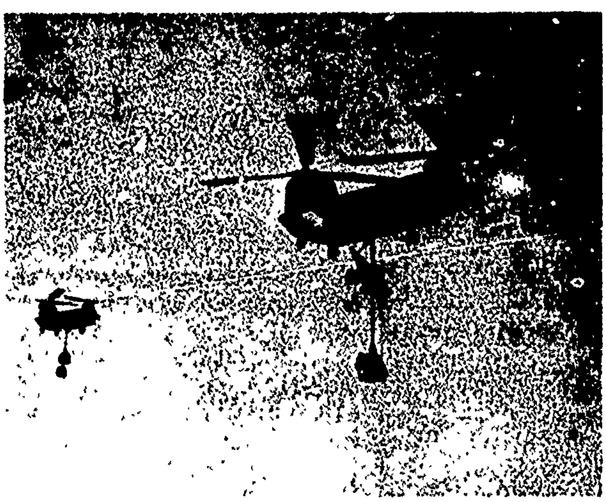

Figure 1. Formation of CH-47 Helicopters With External Loads

The intensive use of the helicopter in the Vietnam war has provided an opportunity to evaluate this aircraft under the most rigorous and extreme environmental and combat situations. Among the many lessons learned is that external transport can account for more than 75 percent of all cargo missions, even when a large cabin and straight-in loading are simultaneously available (1). The key words underlying this preference are productivity and flexibility. By eliminating cargo-loading time and substituting the time for simple hookup and release, ton-miles can be logged in almost direct proportion to operational hours. Given a short-radius, multiload mission, a single helicopter flies many sorties. For a mission involving a series of light loads, the helicopter uses either a piggyback or multiple load technique (See Figure 1).

The short time spent by the externally loaded helicopter near the ground in hover enhances combat safety in terms of reduced exposure to gunfire.

REQUIREMENTS FOR AN IMPROVED EXTERNAL CARGO SYSTEM

The major drawbacks and limitations of the current generation of external load helicopters will be overcome by the successful fulfillment of the following requirements for the soon-to-be-developed heavy-lift helicopter:

- Payload - A design payload of 22.5 tons will for the first time put the helicopter on an economical basis in the areas of unloading of containerized cargo ships and forwarding of the cargo to the users. The military significance of this requirement cannot be overemphasized in light of the deep-water ports that had to be built in Vietnam to overcome waiting by ships for as long as 104 days before offloading (2).

- Multipoint Load Suspension System - In-flight stabilization of external loads by multiple attachments to the helicopter will remove the primary restraint to high forward-flight speeds encountered with single-point cargo suspensions. It will also permit instrument flight with sling loads, lowering of requirements for pilot skill, reduction in pilot fatigue, and positive control of the load in azimuth without ground assistance during load placement.

- Precise Position and Altitude Stabilization in Hover - Stringent limits on position stabilization, coupled with an in-flight load-winching feature, will make this helicopter a true portable crane. For example, compatibility is assured with all types of containerships, including those not possessing winches for below-decks extraction of containers.

- Visual Augmentation System - The helicopter crew will be able to see during covert night operations for hookup and placement of loads without floodlights.

TANDEM DUAL CARGO HOOK SYSTEM

Vertol began to develop dual cargo hook concepts in 1966 with analyses and layouts of longitudinally (tandem) and laterally displaced systems (3). The studies indicated that substantial increases could be gained in the yaw moment reaction which the helicopter could impart to the cargo with a dual hook system over a single, nonswiveling hook. The superiority of the dual hook scheme to other multipoint systems was revealed by its inherent simplicity, low weight, and absence of redundant elements.

Wind tunnel tests were performed in 1967 to confirm the analytical results (See Figure 2). This method of investigation permitted the use of higher airspeeds than current helicopters could achieve and enabled the probing of instabilities without the safety-of-flight considerations of full-scale testing.

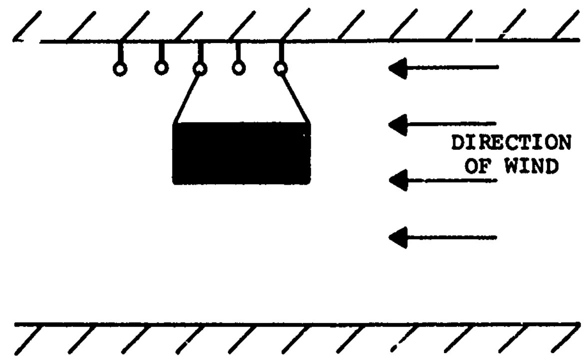

Figure 2. Sling Model in Wind Tunnel

One-eleventh-scale models of a U.S. Army XM102 105mm howitzer and a Conex container were used to determine the ability of the wind tunnel to duplicate full-scale aerodynamic instabilities. Both of these loads had previously shown well-defined, repeatable instability modes in full-scale flight with a single-point hook. The wind tunnel tests of the models showed excellent agreement with flight tests by duplicating correctly the instability modes of both loads. The scaled airspeeds at which the instabilities commenced were also reasonably close to the real thing, as shown in Table I.

TABLE I
COMPARISON OF FULL-SCALE AND WIND TUNNEL TEST RESULTS

Load	Instability Mode	Instability Speed (knots) Full-Scale	Wind Tunnel
105 howitzer 2,900 lb	Coupled yaw-circular pendulum oscillation	80-100	60-80
Conex Box 1,500 lb	Pure rotation	65	40-50
Conex Box 7,500 lb	Pure rotation	80 90	60-80

Following validation of the approach, a group of loads was first tested with single-point and then with dual hook systems at two lateral separations (20 and 40 inches full-scale) and at four tandem separations (20, 40, 80, and 100 inches full-scale). The summarized results of these tests are shown in Figure 3.

These tests confirmed that a dual cargo hook system would stabilize a sling load to speeds which were then beyond the capabilities of current helicopters. They also demonstrated certain advantages of the tandem layout over the lateral, particularly for loads of rectangular planform. For streamlining, a rectangular load must be flown with its long axis parallel to the helicopter at pitch attitudes which avoid aerodynamic load oscillations and provide minimum drag. With the tandem layout, the separation between the hooks controls the change in pitch attitude with airspeed. Furthermore, if the lengths of the tandem cables can be changed in flight, pitch attitude becomes readily adjustable.

More wind tunnel tests were performed in 1968 to evaluate the stability characteristics of the sling loads to be carried by an advanced heavy helicopter (See Figure 4). The objectives were to investigate heavy loads; a tandem hook separation of 24 feet; the effect of helicopter sideslip angle on load stability; and the effect of a cockpit and load-facing operator's cabin ahead of the load.

The maximum stable airspeeds obtained under different loads with the single-point and 24-foot tandem dual hook systems are shown in Table II. The table also shows some of the results obtained with the helicopter in yawed flight.

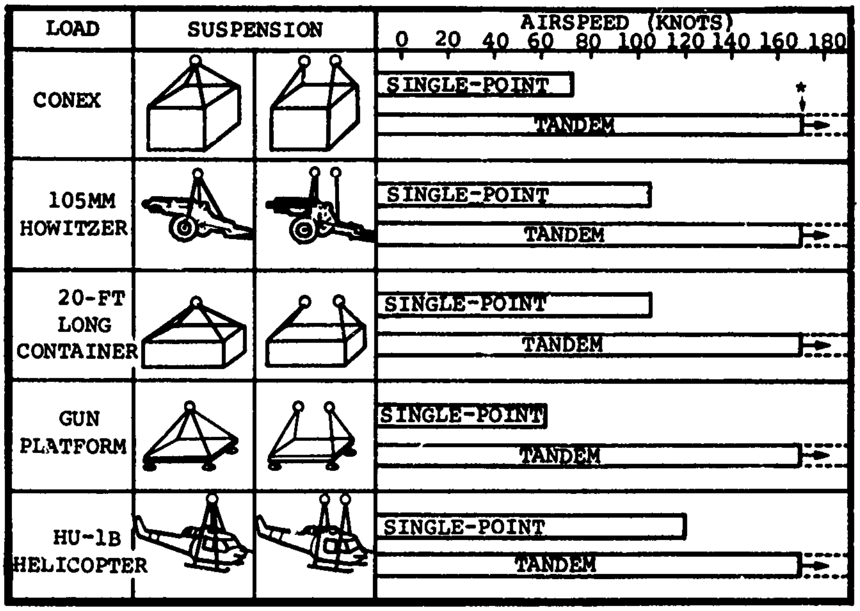

Figure 3. Stability Comparison of Single-Point and Tandem Suspensions

These tests corroborated the earlier trends and supported the conclusion that the next logical step should be a full-scale flight demonstration.

FLIGHT TESTS WITH THE CH-47 HELICOPTER

In 1969 Vertol conducted flight tests to evaluate tandem dual hook systems against a conventional single-point hook (4). The objectives of this testing were to verify the improvements of the dual hook system in load stabilization and airspeed with aerodynamically unstable loads; establish the effects of normal air turbulence; explore the dynamic interaction between the helicopter and the load; and determine the tolerance of the dual hook systems to normal maneuvering of the helicopter.

The test loads were selected to represent three classes of loads: a bluff body (an empty 8 x 8 x 20-foot shipping container weighing 5,200 pounds); an aerodynamic shape (a stripped CH-47 fuselage, 3,650 pounds); and a high-density load (105mm howitzer, 4,980 pounds). The container and fuselage represented high-drag, low-density loads that are usually difficult to transport on a single-point hook.

Figure 4. Wind Tunnel Model of Advanced Heavy Helicopter

The test aircraft was a CH-47B helicopter which was in standard configuration for the single-point tests.

For the two-point tests, the aircraft was outfitted with a steel beam as shown in Figure 5 to support the two suspension cables. The beam was hung from the standard cargo hook and was restrained from movement in the horizontal plane by end-boxes attached to the bottom of the fuselage. The entire rig could be jettisoned in an emergency by opening the standard cargo hook. Fixed-length cables of 7.5, 50, and 100 feet were used to simulate a helicopter winching system; the cables could be attached to the beam at a separation of 12 or 24 feet.

TABLE II
WIND TUNNEL COMPARISON OF ADVANCED HEAVY HELICOPTER SINGLE-POINT AND TANDEM HOOK SYSTEMS

Sling Load	Weight (lb)	Model Scale	Helicopter Yaw Angle	Stable Airspeed (knots) Single-Point	24-Foot Tandem	Limitation
CH-47 helicopter	33,000	1/11	0	120 (load flew broad-side from 40 to 120 knots)	-	Divergent oscillation
			0	-	170	Slight yaw oscillation
			20°	-	170	None
8x8x20-ft container	4,700 (empty)	1/20	0	60	-	Lateral oscillation
			0	-	170	-
			15°	-	130	Against landing gear
M113 armored personnel carrier	20,310	1/20	0	160	-	Slight oscillation
			0	-	170	-
			20°	-	170	-
8x8x40-ft container	5,566 (empty)	1/20	0	100	-	Against landing gear
			0	-	170	
M110 8-in. self-propelled gun	58,500	1/20	0	140 (load flew broad-side from 120 to 140 knots)	-	Large oscillation
			0	-	170	

Figure 5. CH-47B Helicopter With External Load on Tandem Dual Hook System

Each test load was flown with a 24-foot separation on 7.5-foot and 50-foot cables, with and without a spreader bar at the hooks; the shipping container and howitzer were also flown on 100-foot cables. Each load was also flown with a 12-foot separation at a 7.5-foot cable length.

The tests were planned to ensure a safe buildup of airspeed and maneuver severity. Emphasis was placed on reasonably severe pilot-induced upsets for gust simulation and aircraft maneuvering to ensure adequate evaluation of each configuration.

The level-flight evaluation was begun at 50 to 60 knots; speed was then increased to the helicopter power-limited speed in 10- to 20-knot increments, depending on load stability. Load stability was evaluated at each airspeed as follows:

- In straight-and-level flight.
- With control pulse excitation in both directions in pitch, roll, and yaw. The control excitations were 1 to 1.5 inches of control displacement, held for approximately 0.5 second and returned to trim.
- Sideslip maneuvers to the left and right to 15 degrees at speeds below 100 knots and 10 degrees at 100 knots and above.
- Turn maneuvers to the left and right to 30-degree bank angles with operational roll rates on entry and rollout (approximately 10 degree/second roll rates).
- Other normal maneuvers such as accelerations, decelerations, climbs, and descents in the course of the speed buildup.

Flight under simulated instrument conditions was conducted with a hood. Actual instrument experience was gained with many loads flown in conditions of low visibility and in overwater flight where no reference to the horizon was available.

Load-induced linear accelerations in the lifting helicopter pose the most significant problem in instrument flight. These disorienting acceleration forces were at low amplitudes; however, the frequency range was in the band of normal aircraft maneuvering response. The pilot had to sort out his motion cue information and maintain the desired aircraft attitude and flight condition.

With visual reference the pilot has excellent information for monitoring his flight condition and can immediately identify any load-induced cues; he can then relax and ride them out. In instrument conditions the pilot must check out, by instrument reference, all motion cues in maneuvering or gust upsets. False motion cues cause the pilot to become tense and to tire rapidly.

Single-Point Suspension

In hover, all the test loads exhibited a familiar, slow, random yawing under the helicopter. In the climbout and in level flight the container assumed a nominal broadside position and yawed to the left and right through ±45 degrees. This yaw oscillation limited the maximum safe speed for the single-point hook with the container to 40 knots.

The CH-47 fuselage load was stable in the climbout and in the level-flight-cruise buildup. Maximum speed in level flight was 70 knots; this was limited by minimum load-to-fuselage clearance. In partial-power descents at 50 knots, a very abrupt, heavy yaw oscillation developed in the fuselage which excited a very heavy longitudinal swinging and threatened to cause a collision between the load and the aircraft. This instability limited the safe forward speed to approximately 35 knots.

Tandem Dual Hook System

The test loads were flown in the configurations and at the airspeeds shown in Figure 6. The contribution of the spreader bar to stability was inadequate to overcome its disadvantages in load hookup, crew safety, and suspension system dead weight.

In all hover maneuvers the loads were stable in the short pendant (7.5-foot) configuration with excellent load control in azimuth with the two-point suspension. All loads remained stable in the climbout and in the airspeed buildup. The container and fuselage were transported at 120 knots, which was the power-limited maximum speed of the test helicopter. The loads were very stable and well-damped to all pilot-induced upsets and in all normal flight maneuvers. In simulated IFR flight the loads were considered

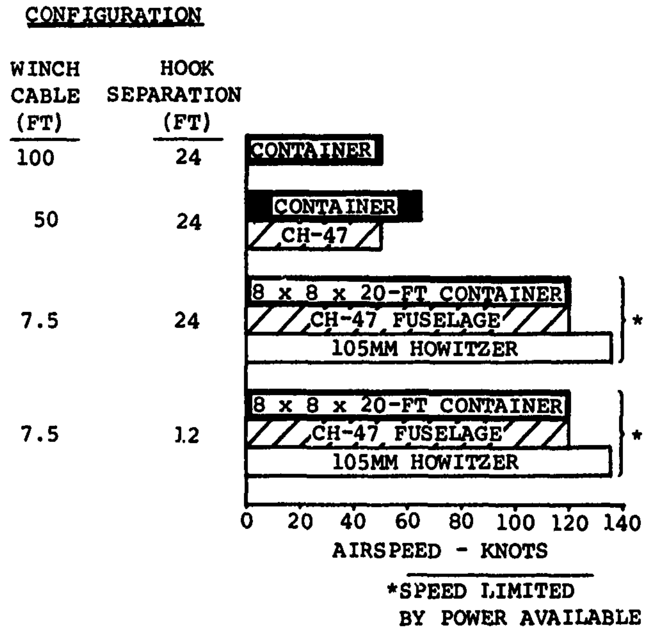

Figure 6. Airspeeds With Tandem Dual Hook

acceptable; however, a lightly damped longitudinal load oscillation followed heavy gust upsets or control excitations. This oscillation produced longitudinal accelerations in the helicopter and made the pilot work harder.

In the long-cable configurations, the container and fuselage loads exhibited a long-period yaw oscillation which produced a coupled lateral and longitudinal swinging due to side lift and drag variations. This tendency toward load oscillation with the long cables limited maximum forward speeds to 50 to 70 knots for the 50-foot length and 50 knots for the 100-foot length. Load stability during hover with the 50- and 100-foot cables was considered acceptable, with light damping to induced-pendulum oscillations. The contribution of the two-point system to heading retention was evident in hover tests, even with the long cables.

All test loads were flown in a short pendant configuration at a 12-foot separation. Load stability with the 12-foot separation permitted carrying the shipping container and the fuselage at the power-limited speed of 115 knots; higher drag from the increased trail angle of the load caused this slight reduction in power-limited speed. There was a slight decrease in stability from the 24-foot separation as evidenced by more load motion following gust or control upsets; however, damping was adequate for acceptable handling in all normal maneuvers.

This flight test program substantiated the capability of a tandem dual cargo hook system to provide load restraint to aerodynamically unstable external loads, thereby enabling safe transport at very high airspeeds. Gusts and normal air turbulence caused transient responses of the loads which made instrument flying difficult, but had no long-term effect. Normal maneuvering of the helicopter was acceptable, with the exception of steep descents at high sink rates, during which some loads could float up and contact the fuselage.

FLIGHT TESTS WITH THE MODEL 347 HELICOPTER

In August 1971 additional flight tests were conducted with the Boeing-Vertol Model 347 advanced-technology helicopter and the tandem dual hook beam shown in Figure 7.

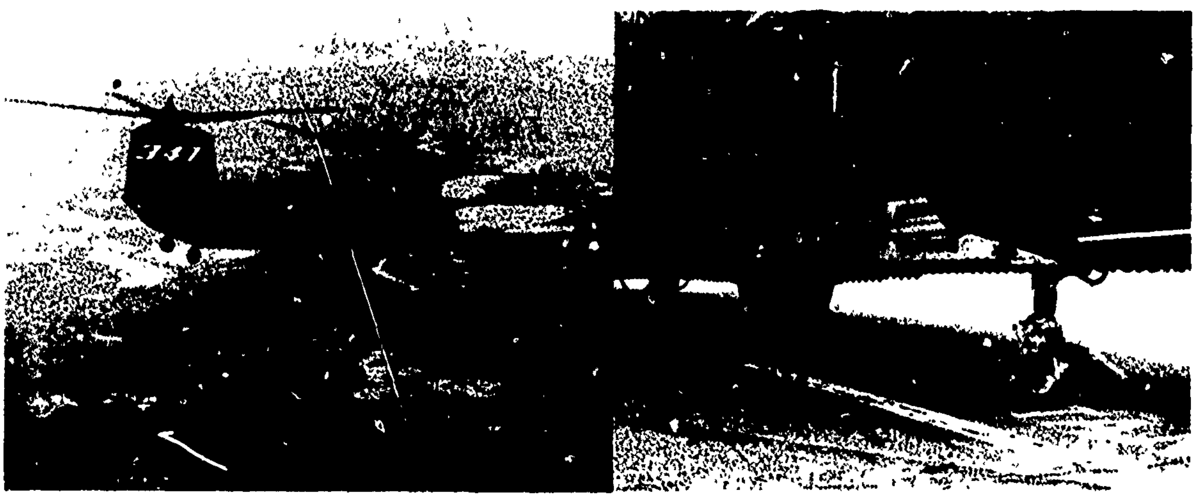

Figure 7. Model 347 Advanced-Technology Helicopter and Details of Tandem Dual Hook Beam

The objectives of these tests were to define the effect of different levels of helicopter stability on the external load and on the capability for precision hover; establish the necessity for active stabilization of the load; and further define the design criteria for the cargo-handling system for the heavy-lift helicopter.

External loads included the 8 x 8 x 20-foot shipping container, both empty and ballasted; an empty 8 x 8 x 8-foot shipping container; and high-density lead blocks. A hook separation of 12 feet was used for the tests.

Aircraft attitudes, rates, and accelerations were measured in each of the three principal axes. Dual hook cable angles were measured in the aircraft longitudinal and lateral axes, along with forward and aft cable tensions and accelerations on the external loads.

Stability of the helicopter was varied by starting with the basic, unaugmented stability level and then incrementally energizing the active stabilization systems. At each stability level, pilot step or pulse inputs were applied individually to each of the pitch, roll, yaw, and thrust controls and the responses of the helicopter and load were recorded. Data were recorded at three airspeeds: hover, 60 knots, and 100 knots; and with two simulated winch cable lengths: 4 feet and 8 feet. Hover data were also obtained with a 50-foot winch cable length.

An example of the type of data obtained during this program is shown in Figure 8. Time histories of helicopter response to a pilot pitch input without a sling load, and of helicopter and load response with a dual-hook-suspended external load are presented. The dual suspension concept provides a dynamically stable load configuration. In comparison to the response characteristics without an external load, two observations can be made. First, the load motion is lightly damped (9 percent critical damping), requiring several cycles before it subsides. The current U.S. military specification covering flying qualities of helicopters (MIL-H-8501) requires a minimum of 11 percent critical damping for instrument flight and 5.5 percent critical damping for visual flight (See Figure 9). Second, the load motion induces both pitch attitude changes and longitudinal acceleration in the helicopter at the pendulum mode frequency. These acceleration forces represent false motion cues to the pilot and can result in pilot-induced oscillations in the control system.

While reduction of data is still in progress on this program, the results confirm the desirability of active load stabilization and also provide a strong data base for control system studies for the heavy-lift helicopter. Items such as the measured maximum cable angles and the effects of maneuvers and external load aerodynamic forces on cable tensions will be used in the design of the cargo-handling system for the HLH.

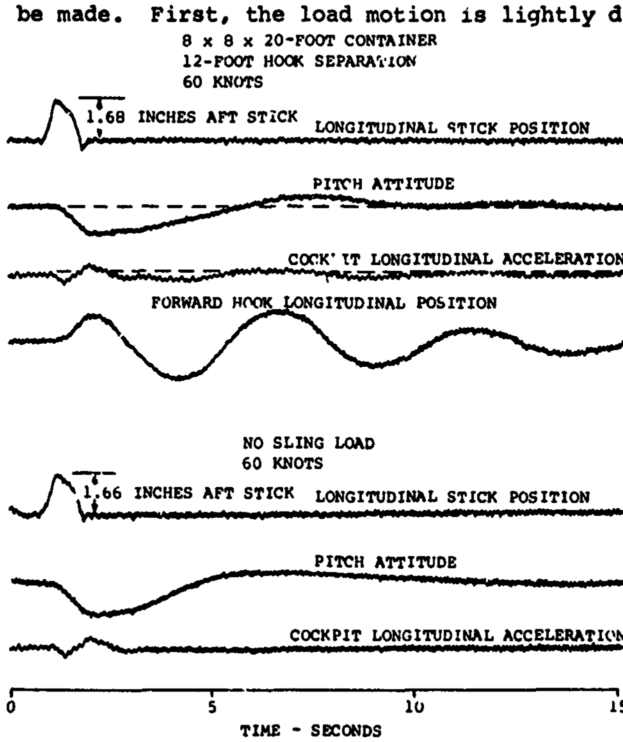

Figure 8. Time Histories of Helicopter Response With and Without Sling Load

HEAVY-LIFT HELICOPTER DUAL CARGO HOOK SYSTEM

The heavy-lift helicopter will incorporate a tandem dual hook system with integral, pneumatically powered hoists as shown in Figure 10. Integral precision hover and display systems will provide safe, routine instrument flight up to the aircraft's power-limited maximum speed with an external load, and accurate placement of the load from an altitude- and attitude-stabilized hover position.

Variable Longitudinal Hook Positioning

The two hoists are mounted on traversing mechanisms which permit adjustments in hook separation from 16 to 26 feet. The 16-foot minimum spacing is based on compatibility with containerships requiring below-decks extraction of the 8 x 8 x 20-foot container. The 26-foot separation was calculated from a hook separation-to-external-load length ratio of 0.6 and a load length of

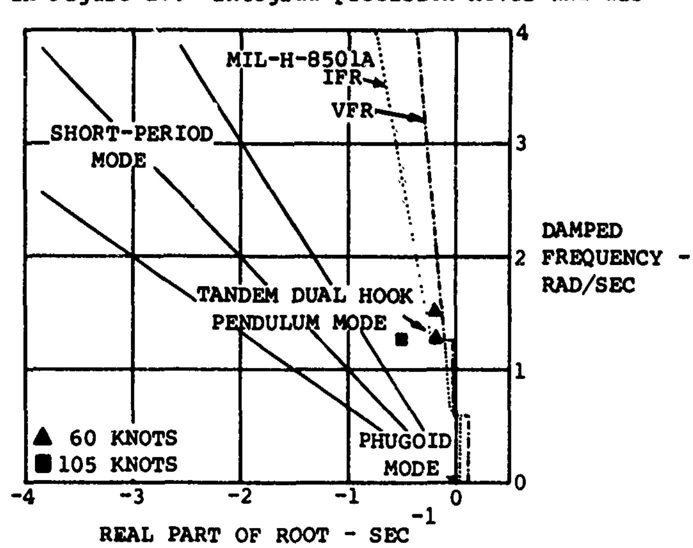

Figure 9. Longitudinal Stability Roots

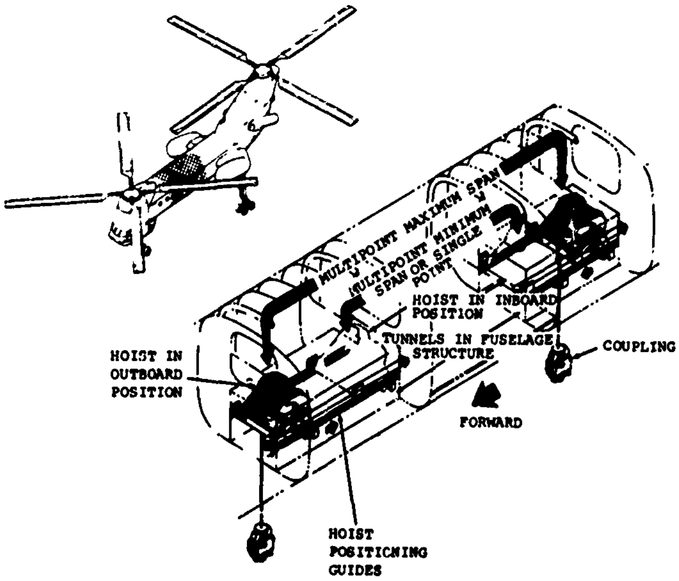

Figure 10. Tandem Dual Hook System for Heavy-Lift Helicopter

of 40 feet. Ratios of at least 0.6 have provided excellent high-speed cargo stability under both visual and instrument flight conditions. Loads over 40 feet in length can be flown, but some degradation of instrument flight characteristics may be encountered.

Load Motion Feedback

The flight control system for the heavy-lift helicopter will include a load motion feedback capability to enable precise control of aircraft and external-load positioning and to ensure stability of the suspended load in gusts. This system will sense load motion and provide stabilizing control as required to effectively damp any random or pendulum load motions. In a typical control situation, the pilot or load-controlling crewman will make the desired control input or displacement and the load motion feedback system will shape the actual control inputs to the rotors to satisfy the pilot's command without exciting any undesirable motion of the load. The effect of wind gusts will be minimized by automatic corrections to maintain a precise hover position. The final configuration of this system will emerge from the recently completed full-scale flight tests and studies of the flight control system now in progress.

Differential Winching

Different cargo attitudes are desirable at various stages of an external-load mission. With the tandem dual hook system load attitude control follows quite naturally from differential adjustment or winching of the two hoist cables.

Figure 11, which depicts the unloading of a containership, demonstrates the requirements for the following load attitudes:

- In hover (velocity, V = 0) over the ship, a level load attitude is required for extraction from the vertical guide rails which support the containers in tiers.
- During climbout at 60 knots, a load attitude providing optimum aerodynamic stability is required.
- In cruise (velocity, V = 125 to 135 knots), the cargo attitude should be adjusted for minimum aerodynamic drag and positive load-stability gradients.

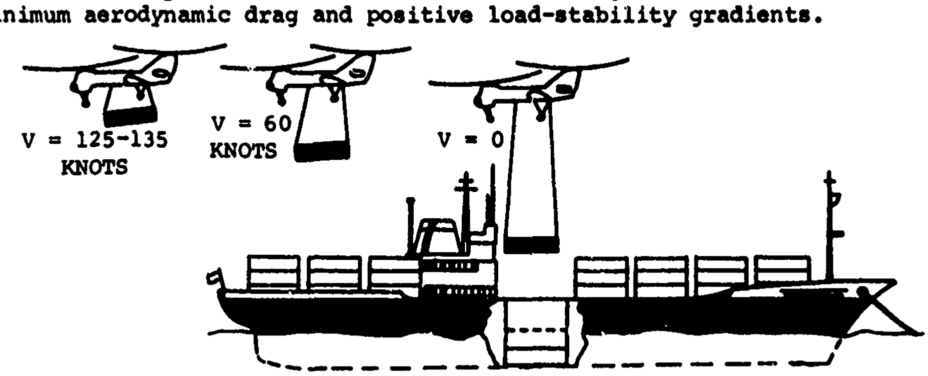

Figure 11. Cargo Load Attitudes at Different Stages of External-Load Mission

Visual Augmentation System

An external-cargo display system incorporating visual augmentation is required to permit precise positioning of the aircraft over a load for pickup and for cargo placement under poor visual conditions caused by darkness, blowing snow, and dust.

A visual enhancement system designed to supplement direct eyesight should have a field of view at least as large as that of the human observer. To provide such a capability, a very wide angle lens, commonly called a fish-eye lens by the optics industry, can be employed. A schematic diagram of a potential heavy-lift helicopter system using a fish-eye lens is shown in Figure 12.

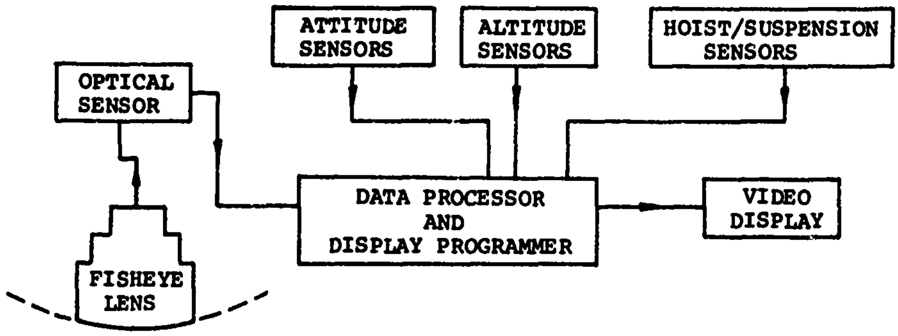

Figure 12. Schematic Diagram of Visual Augmentation System

The pilot display for cargo-handling operations is shown in Figure 13. To hook up the load the pilot need only position the helicopter so that the desired hook touchdown point (represented by the appropriate white cross) is superimposed on the hookup point, and then lower the hook (represented by the appropriate white circle) until the hook engages the cargo.

Figure 13. Pilot's Display From Visual Augmentation System

CONCLUSIONS

Extensive analytical, wind tunnel, and full-scale flight tests have proved the merits of the tandem dual cargo hook concept for stabilizing helicopter external loads. The heavy-lift helicopter will be the first production aircraft to incorporate this system, along with the hover precision, in-flight load winching, and load motion feedback necessary to make it adaptable and versatile. This should open the way to high-speed transport of external loads by helicopters under all weather conditions.

REFERENCES

1. Stuverude, H.N., OPERATIONAL HISTORY AND EXPERIENCE WITH BOEING HELICOPTERS, paper presented at joint American Helicopter Society/U.S. Army Symposium on Operational Characteristics and Tactical Uses of Vertical-Lift Aircraft, Fort Monroe, Virginia, November 1967.

2. McConnell, L.J., Lt. Colonel, U.S. Army, ARMY/NAVY OPERATIONAL EVALUATION OF OFF-SHORE DISCHARGE OF CONTAINERSHIPS, Paper No. 573, 27th Annual National Forum of the American Helicopter Society, Washington, D.C., May 1971.

3. Lancashire, T., et al., INVESTIGATION OF THE MECHANICS OF CARGO HANDLING BY AERIAL CRANE-TYPE AIRCRAFT, Boeing Document D8-0313, The Boeing Company, Vertol Division, Philadelphia, Pennsylvania; USAAVLABS Technical Report 66-63, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, August 1966.

4. Hutto, A.J., QUALITATIVE REPORT ON FLIGHT TEST OF A TWO-POINT EXTERNAL LOAD SUSPENSION SYSTEM, Paper No. 473, 26th Annual National Forum of the American Helicopter Society, Washington, D.C., June 1970.

MATERIALS FOR ADVANCED ROTORCRAFT

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1. INTRODUCTION

I must first apologise for the fact that this paper bears very little resemblance to the title given in the programme - or even to the title on the cover. This is because the whole thing was invented by M. Poisson-Quinton and I may say that it came as a severe shock to see it in print. My initial reaction was to send a rude cable to Paris but after some thought I decided that I would try to conjure new rabbits out of the array of magical materials we now have available to us.

In this I have failed dismally, thus exposing the fact that I am neither inventor nor original thinker but I think I have uncovered something of importance which should be aired and indeed I feel should have been a formal topic on the Agenda. Thus the real substance of this paper is out of order in presentation but in deference to the programme I have taken materials as a background.

2. ADVANCED ROTORCRAFT

The issue which has not been discussed is what we mean by 'Advancement'.

When one who is not a prophet is compelled to prophesy the only way open is to look back into history to some similar time, observe the subsequent pattern of developments, and then argue that our future can be guessed at through parallels and analogies. Provided that this technique is used carefully it can be a valuable guide; unfortunately my analogies all seem to point to what will not happen rather than what will.

Insofar as materials are concerned the only period I have been able to identify is the late 20's when aircraft construction was changing over from wood to metal. If the present time really were to offer comparable opportunities then we could look forward to tremendous developments. Unfortunately the parallel is incomplete for at that time many other features which, in combination, were ultimately to transform the aeroplane, were coming into use. For example the re-tractable undercarriage, variable pitch propellers, new fuels and bigger engines. Also emerging was better understanding of the mechanics of aeroplanes. The true nature of induced drag was clear, an important paper by Melvill Jones[1] brought out the value of reducing drag, it was known how to make skin structures carry loads and there was a growing understanding of airline economics. Just over the horizon were the gas turbine and rocket engines and the probability of supersonic and space flight. Thus the designers of that day had a whole array of new features at their disposal and these were an enormous spur to advancement for almost unlimited improvements in performance and payload seemed to be possible. There was a captive military market and the civil opportunities were most tempting. Perhaps most important for the engineers of that time the technical possibilities generated enthusiasm, they inspired vision and permitted a continuing hope.

In the rotorcraft world no such situation obtains to-day. The limitations on performance are fundamental rather than simply technical and although there is no shortage of schemes for avoiding the limits, those schemes fail commercially because their advantages cannot find a market. The fact is that with the exception of those military-type tasks which require the ability to land and take-off anywhere, all the aeronautical markets appear to have been pre-empted by fixed wing aircraft. Rotorcraft seem to be forty years too late to exploit anything other than the ability to land and take-off vertically. What is worse, the progress of fixed wing aircraft has become the standard of and the golden road to, achievement. Thus when we talk of "Advanced Rotorcraft" we mean machines which will follow the fixed wing path - with a few added virtues.

There have been innumerable attempts to achieve this standard - compound helicopters, tilt wings, ducted fans, tilt rotors, the ABC helicopter and so on. These investigations have taken twenty years or more and they are now so much part of our heritage that we automatically assume that an Advanced Rotorcraft will be some similar new concept.

I believe that this implicit assumption is doing rotorcraft much harm. They cannot possibly have the same standards or follow the same course as fixed wing aeroplanes because the starting points are not the same. Fixed wings and rotary wings are fundamentally different, the former are designed to work at high speed but the latter can fly at zero speed. Nothing we do, from tilt rotors to fan lift, can hide the fact that rotorcraft are meant to fly at low speeds and fixed wing aircraft for high speeds. The courses of their evolution must be totally different and it follows that we must try to evolve our own standards of advancement. Of course one can sympathise with this twenty year search for an aeroplane which combines the virtues of fixed and rotary wings with the defects of neither. But the reason for it is not just the desire to achieve a performance, it is a way and I believe an artificial way, of sustaining the atmosphere which has been one of the mainsprings of aeronautical advance.

It is artificial because this is not the 1920's. Not only is the range of technical possibilities much more limited nowadays but also the attitudes to aeronautics are so different. Fifty years ago everybody wanted speed and pioneering achievements and technological developments. Nowadays they do not; comfort, convenience (most certainly not any form of inconvenience), a distaste for war and an awareness of commercial necessity form the social background for the advancement of rotorcraft.

Against this we must ask what rotorcraft have to offer, or in what sense they are deficient and then decide how the advances should be made. In other words we must first establish what is the market and supply that, rather than just making things because they are interesting and possible. Designers must also be willing to derive encouragement and importance from a new source - for example the fact that they are doing society a service rather than giving a lead in technology.

The feature which rotorcraft have to offer and must be exploited is their ability to get in and out of small landing sites. It confers great flexibility. Speed and economy of operation are obviously important factors but they only become paramount if a competitive form of transport exists. That is why hybrid transport rotorcraft are hard to sell, because they are deliberately attempting to compete with a faster, cheaper method. The form of transport which depends upon flexibility for its appeal is the motor car. From the point of view of speed the advantage lies with the helicopter, particularly over water and congested areas, provided that the distance is between 50 and 250 miles. The disadvantages of the helicopter are that it is not cheap to buy or run and not simple to operate, particularly in poor visibility.

These are the deficiencies to be overcome and it will be obvious at once that there is no place in this scheme for what we generally think of as advanced hybrid Rotorcraft. A 50% increase in speed at the expense of great complexity, some loss of flexibility and an increase in cost does not yet have much appeal. I have no doubt that a market for these devices will emerge but only as a consequence of the wider use of helicopters.

Therefore the improvement of helicopter comfort, ease of flying and maintenance and a little in speed and economy should be our first aim. I would also add that they ought to look more robust and attractive. At the moment they all have what one may charitably call a utilitarian or fundamental appearance but still manage to look remarkably like the prototype of the early 1940's.

If this probably wider civil use is not sufficiently inspiring then one can turn to military applications. Anti-submarine operations scouting and local transport are already recognised as the province of the helicopter but I believe that by the 1980's the fighting helicopter will have come into common use. Indeed I hope so for I believe that this development is necessary to a wider civilian acceptance and appreciation of rotorcraft. I have chosen the phrase "fighting helicopter" deliberately to distinguish it from existing military types which, with one or two exceptions, are transport aeroplanes.

Low level operations, against ground and air targets, will be the sphere of the fighting helicopter. For this purpose they will be heavily armed and armoured, able to fly at 200 knots or so, be very manoeuvreable and have a high rate of climb. Manoeuvreable here means the capacity for high accelerations along, normal to and about the flight path so that the aircraft can rapidly change attitude and position. The speed is necessary to avoid small arms fire as well as for travelling quickly from place to place.

The big basic difference which I see between these helicopters and present day machines will be in the installed power. Something like five times the present level of power is needed to bring helicopters into the same general performance bracket as fixed wing ground attack aircraft. It will be necessary to make rotors and perhaps other lifting and propulsive devices, which can absorb this power. There is great scope here, and in the development of suitable armour, for the use of new materials.

3. SOME NEW MATERIALS

Altogether then my contention is that the conventional helicopter is the machine we should advance. In broad terms we must seek to make it more widely usable. This will undoubtedly mean a change in its appearance and perhaps configuration as the geometric proportions of the various components change. New materials must play a part in this transformation and it is likely that radically new mechanical properties will be wanted.

Obviously the greatest value of new materials will be in the reduction of structure weight but helicopters provide much wider scope. Unlike fixed wing aeroplanes - and this is another reason why a parallel with the 1920's cannot be drawn - rotorcraft make direct use of forces associated with elasticity and inertia as well as with airflow. Thus materials can be employed to improve handling qualities. Examples of this have already been discussed at this meeting. Herr Reichert and Herr Huber have explained how torsionally-flexible blades with a chordwise offset c.g. can be used to improve stability and control. Mr. Balmford has explained how to obtain, in a semi-rigid rotor, the values of flapping and lagging stiffness needed to improve response and maintainability whilst keeping low edgewise bending stresses and avoiding ground resonance.

These two examples, and there are many others, make use of fibre glass and titanium, materials which can be classed as new because they are only just coming into common use. They do not produce radically lighter structures than steels or light alloys but they are now well-known to engineers and we must expect their use to continue and expand. However they are not particularly cheap. Whilst fabrication in glass fibre can be mechanised raw material of aircraft quality is quite expensive, good tooling is essential, a lot of development is necessary and good quality personnel are required. Virtually the same is true of titanium. Fig.1 shows a rotor hub forged in one piece from titanium.

A material which at the moment appears to have most to offer the helicopter is carbon fibre reinforced plastic and to conclude I propose to outline some of the ways in which it might be used.

3.1. Carbon Fibre Reinforced Plastic (C.F.R.P.)

This is a really unusual material. Like all composites its mechanical properties cannot be simply defined because the fibre orientation, distribution and content and the constitution of the resin, all depend upon the item which is to be made. But a few generalisations are possible.

In its cross-plied form the ratio of the elastic modulus to the density is twice that of metals. Its tensile strength for half the weight is comparable with that of dural and evidence from simple specimens shows that the fatigue stress for infinite life can be as high as 60% of the ultimate stress. It is available in short lengths or as a continuous fibre, as a felt or mat or cloth and in the form of sheet or tape. Bonds with other composite materials are easily made. Broadly speaking it is a material most useful for those structures whose design is governed by stiffness or aero-elastic requirements.

Of course carbon fibre has disadvantages in that it corrodes rapidly if brought into contact with certain metals, the strain to fracture is small so that it is not easy to fabricate, and there are difficulties in learning to design in brittle materials. At the moment the cost is very high.

Nevertheless we can assume that these problems will be overcome in time. Already there have been big advances in methods of fabrication. An example of a tape-winding machine is shown in Fig.2 and recently a very useful machine tool for forming felts has been developed.

Until the proper techniques of fabrication have been learned, the stressing rules are known and a suitable code of airworthiness has been established, carbon fibre will be primarily used to reduce the weight of conventional structure elements. Beyond that time we can think of using it to make possible things which otherwise would not have been.

One item which shows a very big improvement in weight is the drive shaft[2]. Fig.3 shows a 4 ft. long 4 ins. diameter specimen which transmits 700 horse-power. Including the glass fibre end fittings this weighs only 40% of its metal equivalent. A bigger structure now under manufacture is the tail boom of a Wasp helicopter. This is a tapered, truncated cone of irregular cross section about 10 ft. long, designed by stiffness requirements. The weight, even when a plastic super factor of 1·3 is imposed, is only 75% of that of the metal boom. Fig.4 shows a section built to examine the problems of joining the boom to the fuselage proper.

These are examples of fibre used as the main material of construction but there is also great scope for the use of C.F.R.P. as a local stiffening or reinforcing element. Fig.5 shows a glass-fibre tailplane, again for the Wasp helicopter, with a carbon stiffener over part of the upper surface. For a 5% increase in weight the natural frequency of the tailplane was raised by over 20%.

The value of mixing glass fibre and carbon fibre is becoming generally recognised. The selective use of carbon produces structures which are basically cheaper, easier to fabricate and more efficient than elements made entirely in that fibre. Some examples, such as propeller blades with carbon fibre spars and rotor blades with carbon fibre trailing edges are already in use. The torsional stiffness of glass fibre and honeycomb rotor blade is very effectively increased by a skin of carbon fibre.

It is now clear that large overall savings in weight are possible. A study carried out on a helicopter of 8,000 lb. A.U.W. shows that the saving can be as much as 500 lb. with even more if felting techniques can be used to replace castings. There is therefore a good prospect of decreasing the structure weight of conventional helicopters by about 20% (of the structure weight) and this should happen towards the end of the 70's.

As yet there does not seem to be a case for using carbon fibre as the primary spar material in conventional rotor blades. The reasons are that blade stiffness is primarily centrifugal and saving weight in the spar is not helpful if it has to be put back by the mass-balance. But for rotors of high solidity or semi-rigid systems as stiff as the ABC the combination of low weight with high stiffness and good fatigue properties will be invaluable.

4. SUMMING-UP

I have argued that the real advancement of rotorcraft will be the wider use of helicopters - and not in the emulation of fixed wing performance. For civil use the aerodynamic performance will not improve much but new materials should bring about some real gains in economy of operation and in the ease of maintenance. This is especially important because relative to other forms of transport, helicopters have been getting cheaper anyway. If there is now a chance of them becoming cheaper in absolute terms we can look forward to a great expansion.

One exciting prospect is the use of materials to provide better handling qualities; Everyone would like the helicopter to be easier and more positive to fly. I would regard research into this subject as a matter of high priority for there are several issues to be resolved.

Military use of transport and scouting aircraft will continue of course, helped by new materials just like civil aircraft. But the big advance must come in the development of fighting helicopters. One can imagine that these will require rotors of much greater disk loading and solidity, with thinner blades and higher tip speeds. They will have auxiliary lifting surfaces and extra propulsive devices such as dual-purpose propellers. Special materials and new attitudes of mind will be needed to make these aircraft. Sikorsky Helicopters have already shown what is possible by using armour plate as a main structural element. It now only requires some inventive genius to use the attack weapons as part of the structure. Some caution in this respect is necessary though. It was suggested to us that gun barrels, if filament wound in carbon fibre would be much reduced in weight. This was followed up with enthusiasm until it was remembered that the fibres have a negative coefficient of thermal expansion.

Both civil and military helicopters will gain from improvements in their appearance. In order to be more pleasing to the eye their structures should fill more of the volume they occupy. Retractable undercarriages help considerably. The 'Fenestron' is a most welcome innovation for it makes the rear end less untidy and it restores symmetry. In a similar way the ABC should make main rotors more attractive provided that the height requirements do not leave it looking like a parasol byplane. New materials will have a big part to play here and there is a case for taking some of their weight advantage and using it in ways which have aesthetic, rather than economic, appeal.

Many people will no doubt feel that I have dismissed hybrid rotorcraft too lightly and on purely technological grounds they will be right. In proportion to the helicopter such aircraft gain more from the use of new materials but the real question is still whether they will be able to satisfy a market. The proportionately higher gain is due to the fact that at the moment the extra complications and redundant items weigh far too much. That this disadvantage can be eliminated I do not doubt but it is still the case that the hybrid is meant to compete with fixed wing aircraft, which themselves will employ new materials and be even more advanced a decade from now. The market for convertible rotor aircraft, does not yet exist but it will I think be generated when there are more helicopters about and one commercial pressure or another makes increases in performance worth while. Of course when the ultimate in new materials has been developed it is most unlikely, if not inconceivable, that rotorcraft will look as they do now. Their configuration is partly dictated by the materials available and a revolution in their properties will almost certainly lead to new configurations. Some of these will be advanced hybrid rotorcraft. We should therefore be unwise not to continue our basic research into these machines.

Finally a word about the materials themselves. The development of the various composites has led to a better fundamental understanding of the mechanics of materials. In particular it is becoming clear how to make substances which combine the desirable features of ease of fabrication, low density, high resistance to fatigue, ultimate strength, toughness etc. One example of this[3] mixes carbon, boron and epoxy resin. It is therefore likely that real revolution is possible for there is an unlimited passport of new materials for both general and special use. On the other hand it is certain that considerations of airworthiness, economy, durability and the lack of appropriate design experience will make the practical adoption of these inventions a slow process.

References

1. Jones, B. Melville, The Streamline Aeroplane, J.R.Ae.S. January, 1929.

2. Winny, H.F. The Use of Carbon Fibre Composites in Helicopters, Westland Helicopters Limited, Feb.1971.

3. Morley, J.G. The Design of Fibrous Composites having mechanical Properties. Proc.Roy.Land.A.319. 117-126 1970.

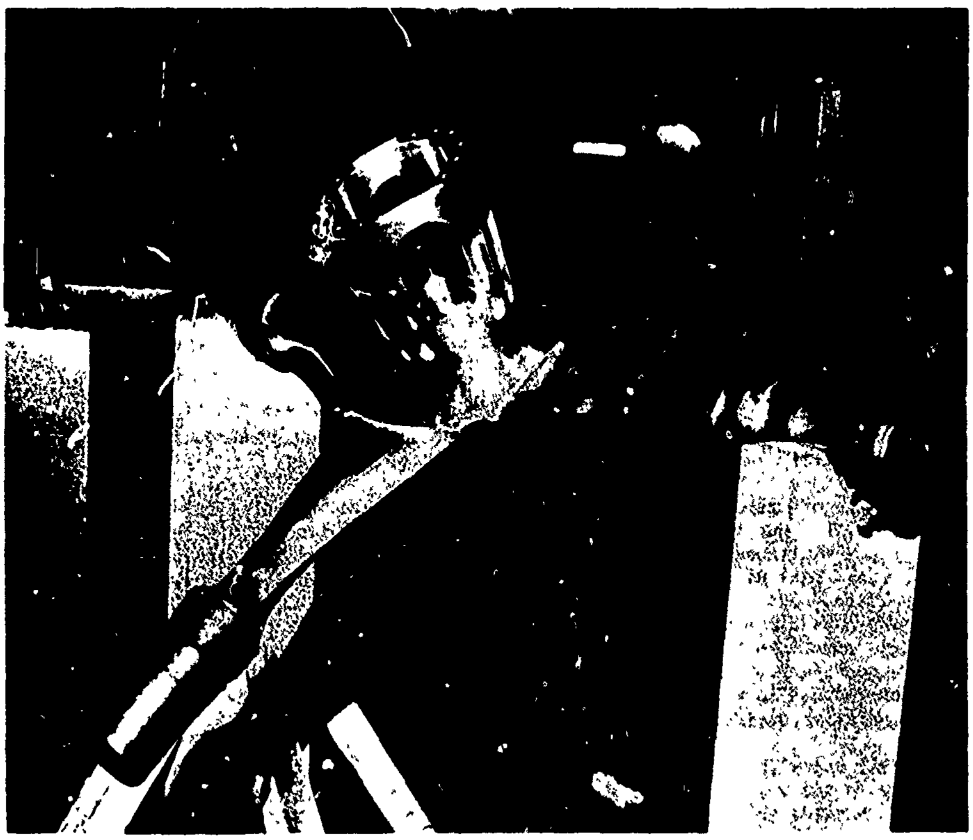

FIG. 1. TITANIUM ROTOR HUB

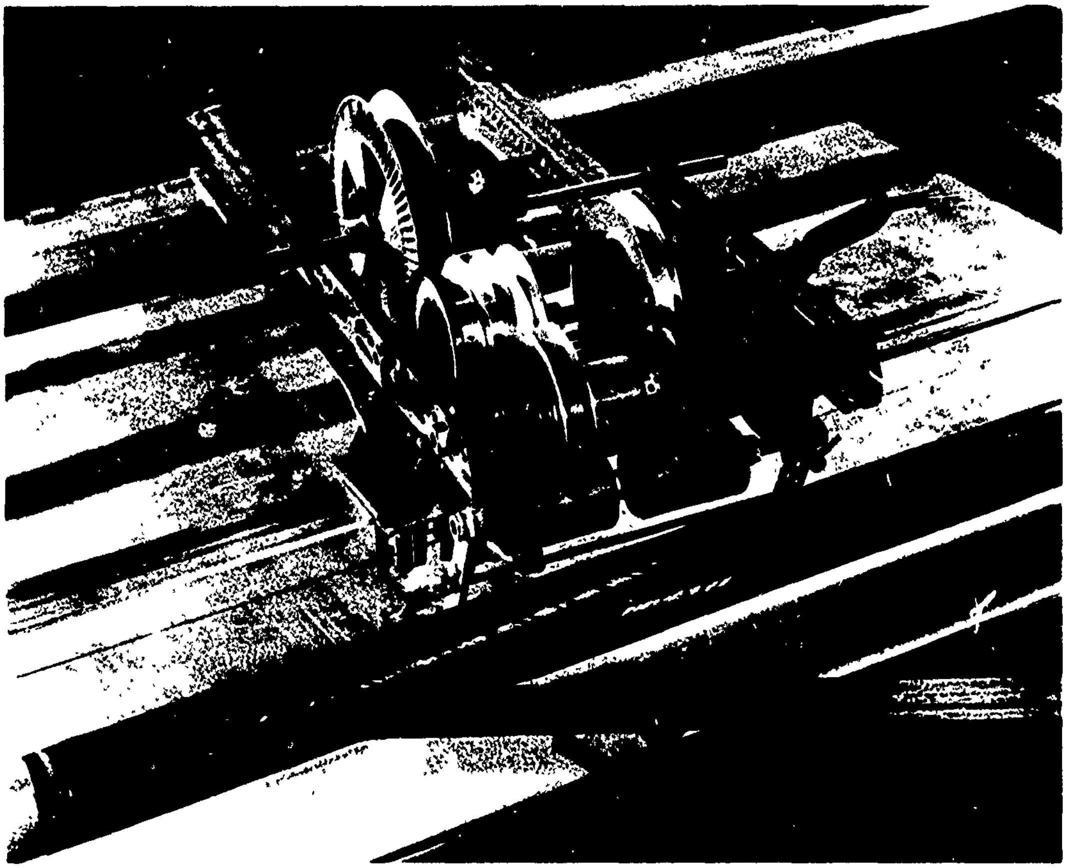

FIG. 2. TAPE WINDING MACHINE

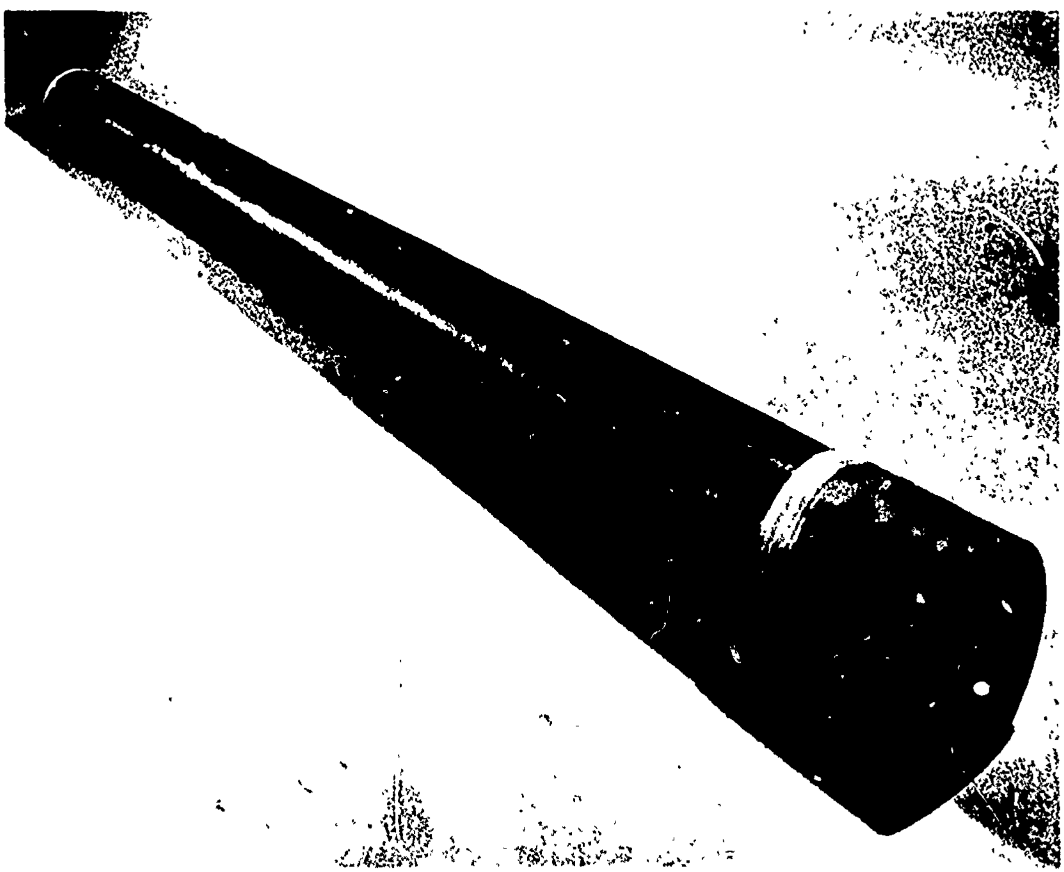

FIG. 3 DRIVE SHAFT IN C.F.R.P.

FIG. 4. SECTION OF TAIL BOOM IN C.F.R.P.

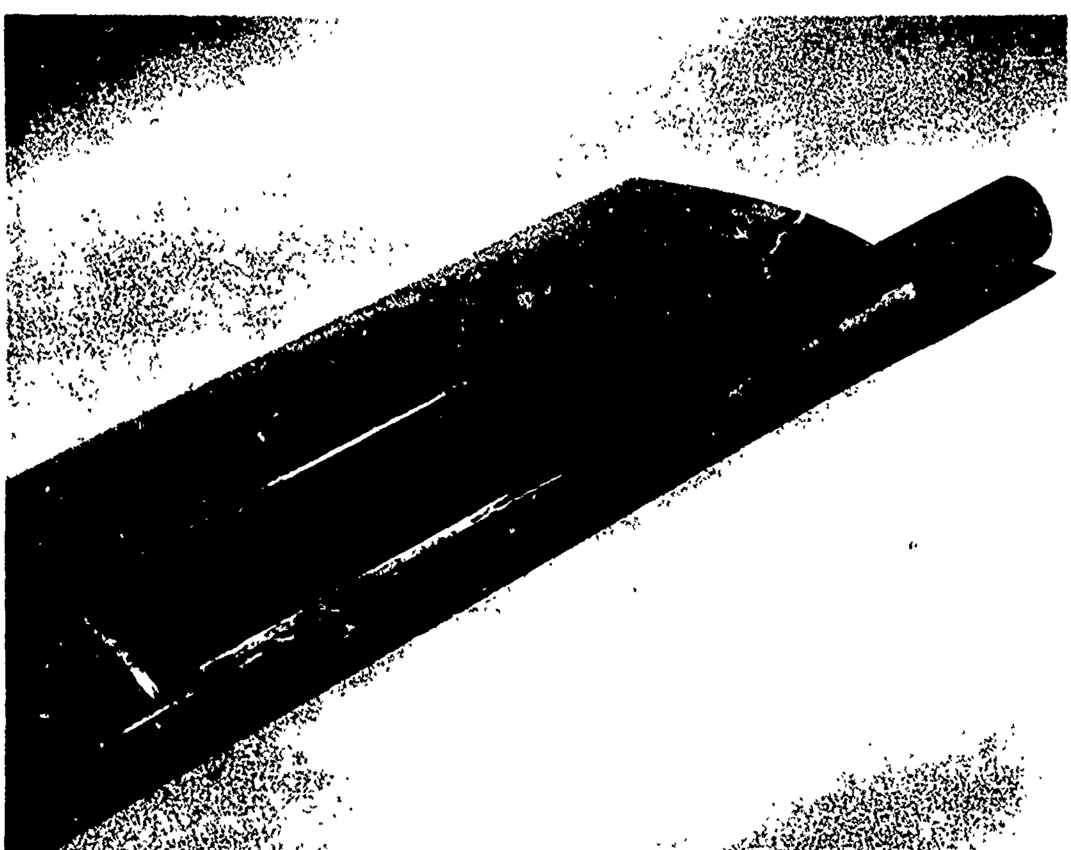

FIG. 5. GLASS-FIBRE TAILPLANE STIFFENED WITH C.F.R.P.

STOPPED ROTOR AIRCRAFT USING CIRCULATION CONTROLLED ROTORS

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SUMMARY

The helicopter is the most successful and best understood type of VTOL aircraft, whilst the subsonic turbofan aircraft is the accepted form of short/medium range commercial transport. The Stopped Rotor Aircraft combines the characteristics of these two types and would seem therefore to offer considerable potential in the VTOL short range transport field. However, closer investigation reveals that the constraints applied by one lifting system on the other are such that a successful compromise is difficult to achieve. This compromise is eased by utilizing the unique properties of the Circulation Controlled Rotor.

The paper examines the fundamental problems of the Stopped Rotor Aircraft and shows why the Circulation Controlled Rotor is ideally suited to such an aircraft. The aerodynamic characteristics of the Circulation Controlled Rotor are discussed and the results of test data presented. Finally, the evolution of a typical stopped rotor aircraft design using circulation controlled rotors is illustrated.

1. INTRODUCTION

The process of evolution in aviation has led to two well established and quite different types of aircraft. On the one hand there is the conventional Subsonic Turbofan Aircraft, and on the other the conventional Helicopter. Between these two extremes there is a whole range of V/STOL types which have been considered and/or developed with varying degrees of success. Figure 1 attempts to summarise the field and indicates where some success has been achieved. The most obvious successes have been Direct Jet Lift, Vectored Jet Lift and Tilt Wing in the VTOL field, and Turboprop in the STOL field. The Stopped Rotor Aircraft, which in principle combines the characteristics of both helicopter and conventional turbofan aircraft, has not appeared prominently in the many and varied V/STOL studies that have been performed by aviation groups during the last decade. It would appear that the problems and penalties associated with the design of this type of VTOL aircraft are such that a feasible solution has either eluded designers completely or the projects that have been evolved are non-competitive. This paper examines the problems of the Stopped Rotor Aircraft and shows why many of them can be solved by using the Circulation Controlled Rotor. The use of the Circulation Controlled Rotor for Stopped Rotor Aircraft was originally proposed by Dr. I. C. Cheeseman of the National Gas Turbine Establishment in reference 1. It has subsequently been studied by Hawker Siddeley Aviation and considerable research work has been completed both by NGTE and HSA.

This paper surveys the work performed by NGTE and HSA and traces the major steps in the HSA project studies.

2. DESIGN PROBLEMS OF STOPPED ROTOR AIRCRAFT

2.1 Disc Loading and Blade Loading

Most conventional helicopters are designed with rotor disc loadings of about 5 lb/sq ft, blade loadings of 80 - 100 lb/sq ft and tip speeds of about 700 ft/sec. Figure 2 shows rotors of these proportions super-imposed on the planform of a typical subsonic aircraft. The wing of this aircraft has an aspect ratio of 5 and a wing loading of 120 lb/sq ft optimised for short range high Mach number cruise and uncompromised by airfield performance considerations. The rotors are assumed to be three bladed since this poses the least mechanical problem associated with folding. Clearly the rotor blades are of such a length that they cannot be accommodated easily when folded. By increasing the disc loading more manageable blade lengths are achieved but only at the expense of wide blade chords which re-complicate the problem of blade folding and lead to wide unwieldy stowage compartments. The tandem layout with a disc loading of 10 lb/sq ft appears to offer a workable arrangement but obviously the problem of folding and stowage would be considerably eased if both disc loading and blade loading were increased. This problem is further aggravated for civil applications where noise is important. It is desirable in these circumstances to limit the tip speed to about 500 ft/sec and this leads to a further reduction in the useable blade loading. Clearly the use of very much higher blade C_L's and hence blade loadings, would help to solve the blade folding and stowing problems.

2.2 Blade Divergence

Conventional rotor blades possess very little bending stiffness in their own right. In normal helicopter flight the bending moments due to lift are reacted almost completely by the stiffness imparted by the centrifugal inertia forces. When the rotor of a Stopped Rotor Aircraft is slowed at the end of transition this centrifugal "stiffness" disappears and lift forces can only be reacted by the real blade stiffness. Although under these conditions the rotor lift is designed to be zero, there will inevitably be lift forces generated due to gusts and other flow disturbances. For any rotor there is a critical flight speed at near zero rotational speed,above which aero-elastic divergence will occur in the blade bending mode. This problem has been investigated in Reference 2. It is likely that divergence can only be avoided with conventional rotors if transition speeds are limited to low levels. This leads to low wing loadings and complicated high lift devices with consequent weight penalties.

2.3 Oscillatory Loads

As the helicopter rotor rotates in forward flight the blades are subjected to a continually varying incidence pattern. This incidence pattern, together with the variations in dynamic head experienced by the blades results in complex oscillatory forces. These forces can be nullified to some extent by the conventional cyclic pitch control. However, this control is normally only capable of first harmonic variation and quite large multi-harmonic forces remain. Under normal helicopter operating conditions the frequency of these oscillatory forces is such that they are largely absorbed by the mechanical/aerodynamic damping and inertia of the rotor itself and only a relatively small proportion is felt by the airframe. During the rotor stopping phase of the Stopped Rotor Aircraft there are several reasons why these oscillatory forces will be more troublesome.

(i) The multi-harmonic content of the forces increases and cannot be counteracted by the cyclic pitch control.

(ii) The frequency of the forces reduces and hence is less readily damped.

(iii) The rotor system needs to be stiffer than that of the conventional rotor in order to avoid the divergence problems of Section 2.2.

For these reasons the Stopped Rotor Aircraft will exhibit very poor ride qualities during the rotor stopping phase unless it is possible to reduce the incidence dependant lift characteristics of the blades.

2.4 Gust Upsets

During the rundown phase on a Stopped Rotor Aircraft the rotor will be very susceptible to gusts. The effect of these gusts will be to produce irregular random forces and moments on the slowly rotating rotors. On some configurations these forces and moments will be unsymmetrical and could lead to serious stability and control problems during the rotor rundown phase. Clearly the high blade loadings and low incidence dependant lift qualities shown to be desirable in Sections 2.1 to 2.3 would also show benefit here.

2.5 Desirable Characteristics for the Rotors of Stopped Rotor Aircraft

From the discussion of section 2.1 it is desirable to use a disc loading of at least 10 lb/sq ft for Stopped Rotor Aircraft in order to achieve blade lengths compatible with stowage. High design blade CL's are then necessary in order to reduce the blade chords and tip speeds to acceptable levels. In order to avoid blade divergence during the rotor stopping phase it is probably necessary to use sections which have low incidence dependant lift characteristics. At the same time the use of very thick sections, particularly at the blade root will help. Combination of high blade loading and low incidence dependant lift will lead to the reduction of the rotor rundown oscillatory forces and the sensitivity of the slowing rotor to gusts.

3. CHARACTERISTICS OF CIRCULAR SECTIONS WITH INDUCED CIRCULATION

3.1 Lift Characteristics

A rotor blade of circular cross section with circulation induced by blowing goes a long way towards fulfilling the requirements of Section 2.5. The lift characteristics of such a section are shown in Figure 3 compared with those of a conventional aerofoil. This data is for a circular cylinder of finite span and aspect ratio 12 and is taken from Reference 3. Other data on circular cylinders are available from NGTE work in References 4, 5, and 6. The total information available on circulation controlled cylinders varies considerably in both quality and performance and much further work is required before a full understanding of the parameters involved is acquired. However, Figure 3 shows a fairly high standard of performance from the available evidence. As can be seen the available lift coefficient for a circulation controlled cylinder is many times greater than that for a conventional aerofoil and this means that the useable blade loadings are at least five times greater than those for conventional aerofoils. Although the incidence dependant lift characteristic at high blowing momentum are much the same as that for a conventional aerofoil the change in dimensional lift for a given incidence change will be much lower for the circulation controlled section due to the high blade loadings that can be used.

3.2 Critical Mach Number

Although the characteristics outlined above suggest that a circular section is ideally suited to high disc loading, low tip speed rotors of low solidity, their use is in fact extremely limited, due to their very low critical Mach Numbers. Figure 4 shows the critical Mach number of a circular cylinder with induced circulation. Under hovering conditions if the high C_L's available are used, tip speeds have to be limited to 300 - 350 ft/sec. This leads to high induced power and high rotor torque which results in significant weight penalties in the rotor drive system. If a reasonably high transition speed is used to avoid low wing loadings and complex high lift systems the low critical Mach number of the circular section leads to difficulties here also. With a transition speed of say 140 kts the rotor r.p.m. has to be restricted to a level low enough to avoid the advancing tip exceeding M = 0.47. This means that the rotor r.p.m. has to be reduced during transition but more important it results in advance ratios approaching unity towards the end of transition. Clearly it is preferable to strike a compromise between the characteristics of a circular cylinder and those of a conventional aerofoil.

4. CHARACTERISTICS OF ELLIPTIC SECTIONS WITH INDUCED CIRCULATION

4.1 Critical Mach Number

The critical Mach number limitations of circular cylinders described in Section 3.2 can be alleviated by using elliptic cylinders. Figure 5 shows the critical Mach number of elliptic cylinders as a function of Thickness/Chord ratio and induced C_L. The use of a 40% ellipse and a tip speed of about 480 ft/sec leads to advance ratios of less than 0.5 during transition for a transition speed of 140 kts without rotor slowing. It is not suggested that the 40% ellipse is the optimum section for the Circulation Controlled Rotor, lower C_L's and higher tip speeds could be used if noise is less important. However, it must be pointed out that attempts to produce circulation control on sections with small trailing edge radii have not been very successful. It could be that a non-elliptic section with a more rounded trailing edge may offer better characteristics.

4.2 Lift

The lifting characteristics of a 40% elliptic section are shown in figure 6. This data is also taken from Reference 3. Further data on the characteristics of elliptic sections are available in References 7, and 8. Compared with a circular section the 40% ellipse shows marginally less efficiency with regard to lift versus blowing momentum. A 40% non-lifting ellipse also shows an incidence dependant lift characteristic which is about one third of that for a conventional aerofoil. The maximum lift coefficients available for a 40% ellipse are somewhat lower than those for a circular section. However, they are more than adequate to achieve the operating conditions suggested in section 4.1.

The data shown in Figure 6 indicates that at positive incidence and high blowing momentum partial stalling of the section is occurring and this is thought to take place between the blowing slots. It is considered that this effect can be improved by repositioning the slots or possibly introducing further slots. Optimisation of the blowing slot geometry is not very advanced and considerably more research is required in this area.

4.3 Drag

The drag of circulation controlled sections has been investigated in references 3 to 8 but at this stage a clear picture does not emerge. In general one would expect that as the blowing momentum is increased wake closure would occur and this has been demonstrated in some of the experimental data. However, in some cases separation bubbles and spanwise discontinuities have occurred and the drag results have been irregular. The foregoing remarks on blowing slot position optimisation apply equally as far as drag is concerned. There is evidence from some of the data that low drag can be achieved at the expense of poor lift/blow efficiency and vice versa.

5. ROTOR PERFORMANCE

5.1 Rotor Geometry

The majority of the work performed by Hawker Siddeley Aviation is concentrated on rotors of parallel chord with 40% elliptic tips and circular sections at 20% radius. The general arrangement of a typical rotor blade is shown in figure 7. This shows three blowing slots which taper linearly from tip to root designed to give a tip C_L of 2 and a root C_L of 8 under hovering conditions and maximum normal acceleration. For reasons which will become apparent later, work has been concentrated on three bladed rotors.

5.2 Induced Power

Many attempts have been made in recent years to refine the process of calculating induced power. These have consisted of making a more accurate assessment of the induced velocity field of a rotor. In the case of Circulation Controlled Rotors the accurate assessment of this velocity field is even more important than in the case of a conventional rotor. The Circulation Controlled Rotor is capable of achieving very high disc loadings with only moderate tip speeds and this results in large helix angles in the rotor flow field. This means that the in-plane induced velocities produce a significant contribution to the induced power. The induced flow field for a rotor of this kind has been calculated by Seed in reference 9.

5.3 Profile Power

It has already been explained in Section 4.3 that the profile drag of circulation controlled sections is not well established. For this reason some doubt exists regarding the level of profile power for these rotors. In the absence of better information the profile drag of circulation controlled sections has been assumed to be equal to that of the same section without blowing.

6. ROTOR MODEL TESTS

6.1 The Models

Model test rotors have been built by both Hawker Siddeley and the National Gas Turbine Establishment. These rotors have embodied both elliptic and circular sections. Figure 8 shows a 4 ft diameter, three bladed rotor built by Hawker Siddeley Aviation. This rotor has tips with 40% elliptic sections tapering to circular sections at the root. Each blade has three blowing slots. A wide variety of tests have been performed with this rotor including free air hover, and airframe interference tests in the Hawker Siddeley 15 ft V/STOL tunnel. Figure 9 shows a 12 ft diameter test rotor at NGTE. This rotor has circular sections throughout and is one of the earliest rotors tested. This rotor has subsequently been modified to give information on the effect of 20% elliptic tips. It has been tested under free air hovering conditions, under forward speed conditions in the RAE 24 ft tunnel, and also under forward speed conditions on a specially constructed mobile test vehicle. Tests on the 4 ft diameter Hawker Siddeley rotor are reported in references 10, 11 and 12 whilst tests on the NGTE rotor are reported in references 13, 14, 15 and 16.

6.2 Experimental Results

It is not intended in this report to quote a comprehensive set of experimental results since these are available in the references. However, in this section a few typical results are quoted.

6.2.1 Isolated Rotor

Figure 10 shows the shaft horse power required together with the adiabatic blowing horse power for the Hawker Siddeley 4 ft diameter rotor under hovering conditions and at 60 and 120 ft/sec forward speed. The measured shaft horse power is in close agreement with estimates. However, the blowing power is in excess of that estimated using the data on blowing requirements from Sections 3 and 4. It is thought that this poor blowing efficiency could be a Reynold's number effect since the blades on the 4 ft diameter rotor are a mere 1.5 inches chord. It is also possible that the blowing power could be reduced at the expense of an increase in the profile shaft power as suggested in Section 4.3. The results shown in figure 10 illustrate the considerable penalties in installed power (about 50%) which one has to pay with Circulation Controlled Rotors compared with conventional rotors.

6.2.2 Rotor Airframe Interference.

The 4 ft diameter Hawker Siddeley rotor has been tested in the Hawker Siddeley 15 ft V/STOL tunnel in conjunction with a half model of a lateral-twin rotor aircraft. The NGTE 12 ft diameter rotor has been tested in the RAE 24 ft tunnel in conjunction with a full span wing forming a central rotor configuration. These two sets of experiments reported in references 11 and 16 provided interesting information on the problem of rotor airframe interference. Figure 11 illustrates the main points resulting from these experiments. Under hovering conditions the lateral-twin configuration shows a wing downforce slightly greater than that for the central rotor configuration. However, as the forward speed increases the wings on the lateral-twin begin to generate lift at a very low forward speed, whereas the wings for the central rotor configuration are unable to generate lift until a much higher forward speed is reached. The implications of this are that a lateral-twin configuration would exhibit much easier transition characteristics than a central rotor configuration, but would suffer more adverse interference effects in the hover and require more installed horse power.

7. CYCLIC CONTROL

In common with more conventional rotors, Circulation Controlled Rotors require cyclic control to the blade lift. This cyclic control is required to provide the necessary trimming moments for CG position, manœuvre and to cope with the effects of forward speed. Figure 12 shows a cyclic control valve as devised by Hawker Siddeley Aviation. This is a sleeve port valve, air being supplied from a plenum chamber through ports into the ducts supplying the rotor blades. The area of the ports is controlled by tapered sleeves surrounding the central supply duct. The two tapered sleeves can be moved apart in order to increase the area of the ports, this being equivalent to collective control on a conventional rotor. Alternatively, the two tapered sleeves can be rotated with respect to each other. In this way the area of the supply ports varied cyclically as the rotor rotates. This form of control is equivalent to cyclic control on a conventional rotor. If the two sleeves are rotated together then the azimuthal position of the cyclic control is varied. Unlike the cyclic control mechanism on a conventional rotor the parts of the cyclic control valve shown in figure 12 do not contain any components with reciprocating motion. This represents a considerable simplification compared with the conventional rotor system.

A model valve embodying the principles shown in figure 12 has been tested on the H.S.A. 4 ft diameter rotor. These tests which are reported in reference 10 demonstrate the complete viability of the valve. Figure 13 shows the effectiveness of the cyclic control valve in producing pitching moment on the H.S.A. 4 ft diameter rotor. The maximum pitching moment produced is equivalent to a shift in lift centre of about one quarter of the rotor radius. In producing control moments of this order it is necessary to apply the cyclic control some 30 - 40° in advance of the required moment vector. Other forms of cyclic control valve for the Circulation Controlled Rotor have been devised by Hawker Siddeley Aviation and the National Gas Turbine Establishment. All these valves are very much simpler than their conventional rotor equivalent and one of them includes a very simple means of achieving high harmonic cyclic control. In addition to the simplicity of the cyclic control valve it is also worth noting that the rotor itself offers further simplifications. The rotor blades have no pitch change mechanism, nor have they any flapping or drag hinges.

8. CONFIGURATION ASSESSMENT

Three possible configurations have been considered for Stopped Rotor Aircraft. These are the Single Central Rotor, the Tandem-Twin Rotor, and the Lateral-Twin Rotor Configurations. The table below compares these configurations.

	CENTRAL ROTOR	TANDEM-ROTOR	LATERAL-TWIN
Advantages	i. Good helicopter background	i. Good helicopter background ii. Balanced torque	i. Good interference ii. Balanced torque iii. Low induced power iv. Symmetrical gust response v. Uncompromised cabin layout
Disadvantages	i. Bad interference ii. Unsymmetrical gust response iii. Rotor torque reaction iv. Cabin layout problems	i. Bad interference ii. Unsymmetrical gust response iii. Cabin layout problems iv. High induced power	i. Poor helicopter background

The lateral-twin appears most promising and was consequently chosen for project studies by Hawker Siddeley Aviation. In choosing the lateral-twin configuration it was realised that there may be rotor stowage problems due to the absence of suitable bodies at the wing tips. However, the characteristics of a Circulation Controlled Rotor with the blowing system turned off are such that the blades can be stowed in line of flight without the need for retracting into stowage bays. Initially it was envisaged that two bladed rotors would be used, simply parked in the direction of flight after transition. However, it soon became apparent that under rotor rundown conditions the level of vibration resulting from the aerodynamic forces on two bladed rotors would be such that the quality of ride in the aircraft would be extremely poor. It was therefore decided that the rotors should be three bladed, parked with one blade facing forward and two blades folded to trail rearwards.

9. POWER PLANT ARRANGEMENTS

9.1 Integral Lift and Propulsion Engines

During the early part of the Hawker Siddeley study it was considered that a common power plant should be used for both vertical and cruising flight. Figure 14 shows the arrangement of the power plant which was evolved at this time. The engines are low by-pass ratio turbofan engines with separate flow ducts. During vertical flight the hot stream is ducted to a free power turbine which is used to drive the rotor via a reduction gearbox. The cold stream is used to supply blowing air to the rotor via the cyclic control valve. The power plants in the wing tip nacelles are connected together by cross shafting and ducting from the cold blowing system. This is necessary to avoid power asymmetries due to engine failure. During transition as the shaft power and blowing air requirements of the rotor diminish, the hot and cold gases are progressively transferred to variable area plug nozzles in the base of the nacelles. At the end of transition all the engine gases leave the propulsion nozzles and the rotor blowing system and drive system is shut down.

As the study proceeded it became obvious that this integral power plant system had several drawbacks. These are as follows:-

(i) At an intermediate transition condition, before the wing begins to generate large amounts of lift, there is a shortage of power to be shared between the propulsion and lifting systems. This results in very sluggish transition performance.

(ii) After transition, since the gases are made to leave the nacelle from two separate nozzles, the hot nozzle produces considerable noise which is disadvantageous for civil applications.

(iii) The free power turbine has to be a variable acceptance design and this produces poor efficiency and high noise.

(iv) The complication of the plumbing associated with the cross-ducting and the various valves controlling it raise doubts on airworthiness.

9.2 Separate Lift and Propulsion Engines

As a result of the disadvantages listed above,it was decided to evaluate a power plant system in which completely separate engines are used for lift and propulsion. Such a system was not considered initially due to the added expense of two power plant systems. However, it was realised that the expense of the complicated control system of the integral lift and propulsion engine system in Section 9.1 would outweigh the cost of the basic engines. Figure 15 shows the arrangement of a lift engine pod for the separate lift and propulsion engine system. In this case the lift engine consists of a lightweight lift jet with the exhaust gases ducted to a power turbine. The power turbine drives the rotor via a reduction gearbox, and also an auxiliary compressor to supply the necessary blowing air. The propulsion engines for this scheme are separate high by-pass ratio turbofan engines and these may be mounted either on the wing or on the rear fuselage. The simplicity of this scheme compared with the one described above is obvious and all the disadvantages outlined in Section 9.1 are removed. Subsequent project studies showed that there was no weight penalty due to the use of the extra power plants.

10. HS 803 SHORT HAUL VTOL TRANSPORT

The HS 803 is one of several Circulation Controlled Rotor projects which have been investigated by Hawker Siddeley. It is aimed at exploiting what are probably the two main benefits to be derived from Circulation Controlled Rotors. These are low noise and the ability to use existing power plants with a minimum of special-to-type development. Figure 16 shows the general arrangement of the Hawker Siddeley 803. The lifting power plants are of the type described in Section 9.2 and consist of Rolls Royce RB 162, lightweight lift jets driving power turbines. The propulsion power plants are Rolls Royce Trent turbofans mounted on the rear fuselage. Of particular interest is the extremely low take-off noise level of 83PNdB at 1,500 ft radius from the take-off point. This noise level is markedly lower than estimated noise levels for any proposed VTOL type. The need for a VTOL aircraft capable of operating into city centres has not been firmly established but it could be that the Circulation Controlled Rotor offers the only means of producing an acceptable noise level for such an aircraft.

Comparative studies performed by Hawker Siddeley indicate that the Circulation Controlled Rotor Aircraft carries an all up weight penalty of some 10 - 15% compared with a fan lift VTOL aircraft. However, it does not follow that this necessarily reflects a similar penalty in first costs, particularly in view of the modest engine development required.

11. OTHER APPLICATIONS

The Circulation Controlled Rotor is seen as a system which allows the development of Stopped Rotor Aircraft, and its application to a Civil VTOL Transport has been discussed. The need for immediate development of this class of aircraft seems doubtful and it is of interest to consider what other applications of the Circulation Controlled Rotor seem possible.

The Stopped Rotor Aircraft possesses characteristics which combine reasonably good hovering efficiency and moderately high cruising speed. These characteristics would lend themselves to missions such as Search and Rescue where it is desirable to avoid delays in transit to the rescue area and then be able to hover whilst effecting the rescue operation. It would seem, therefore, that such an application would be an ideal one for the Circulation Controlled Rotor.

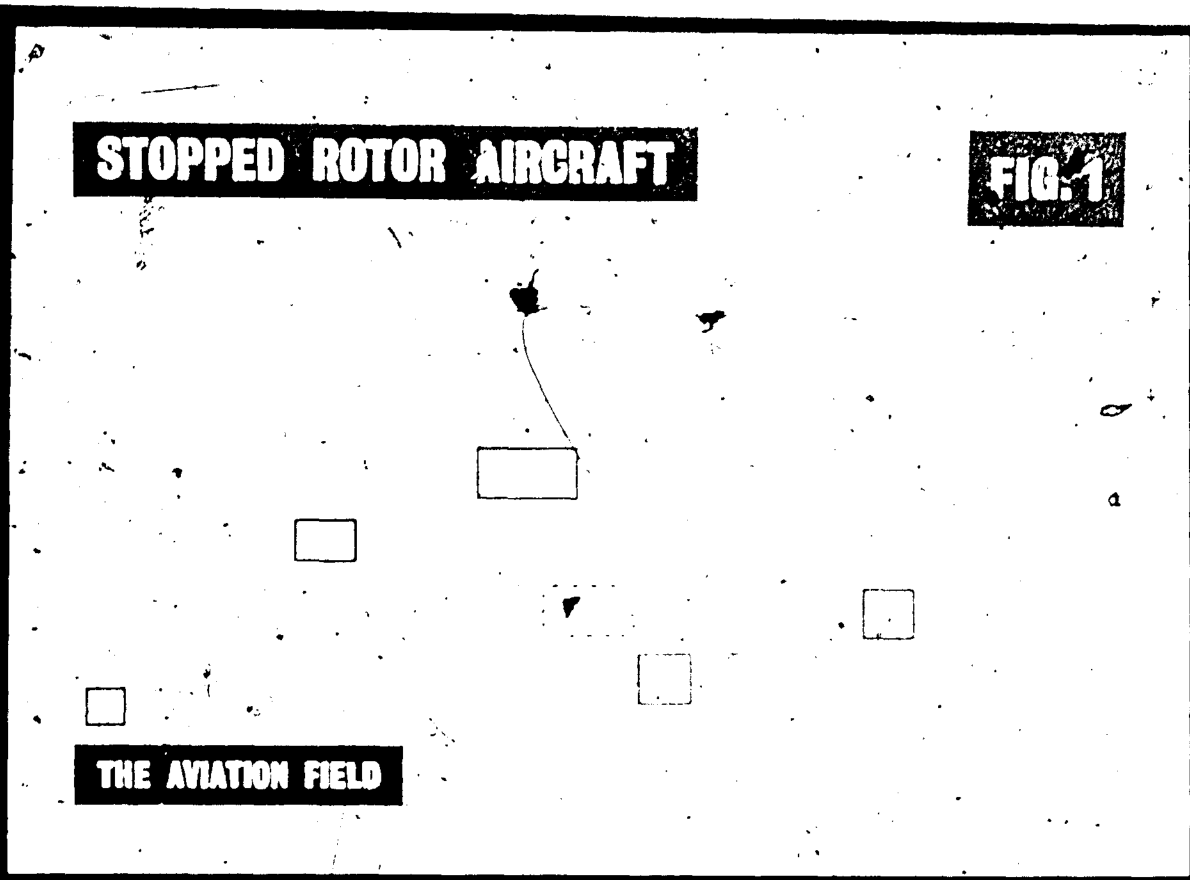
As outlined in Section 7, the Circulation Controlled Rotor is mechanically simpler than the conventional helicopter rotor and should therefore require far less maintenance. It should be well suited to rough industrial and military use. Such applications include a heavy lift crane, and a simple utility military helicopter. Although these applications exploit the rugged simplicity of the Circulation Controlled Rotor they would unfortunately demand the development of a similarly rugged power plant.

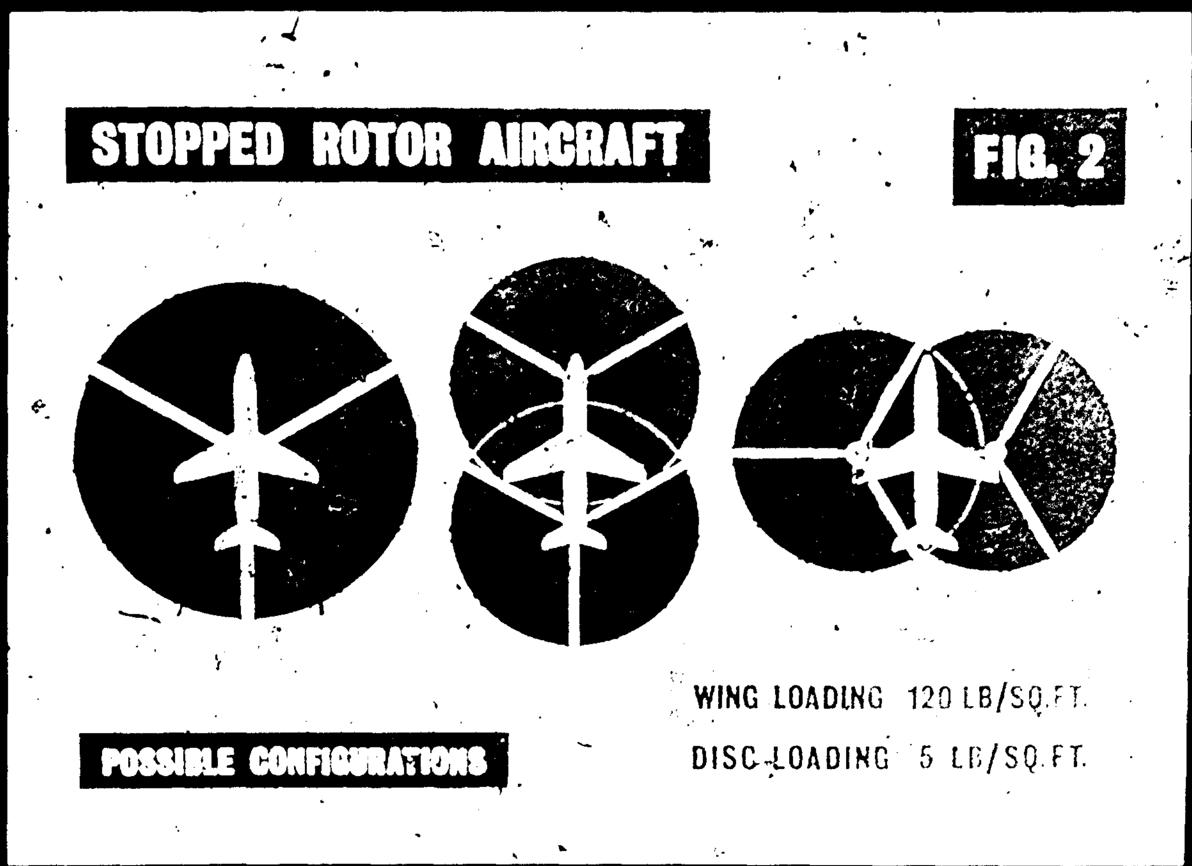
12. CONCLUDING REMARKS

Investigations into Stopped Rotor Air aft have in general not been fruitful due to mechanical problems involved in their design. These problems can be largely overcome by the use of the Circulation Controlled Rotor. Although a considerable amount of research into the Circulation Controlled Rotor is required, it does promise to lead to a civil VTOL Transport Aircraft offering extremely low noise levels and requiring a minimum of engine development. The Circulation Controlled Rotor would also appear to be ideally suited to applications which require a combination of hovering efficiency and high transit speed, and to applications where a rugged, simple, low maintenance helicopter is required.

13. REFERENCES

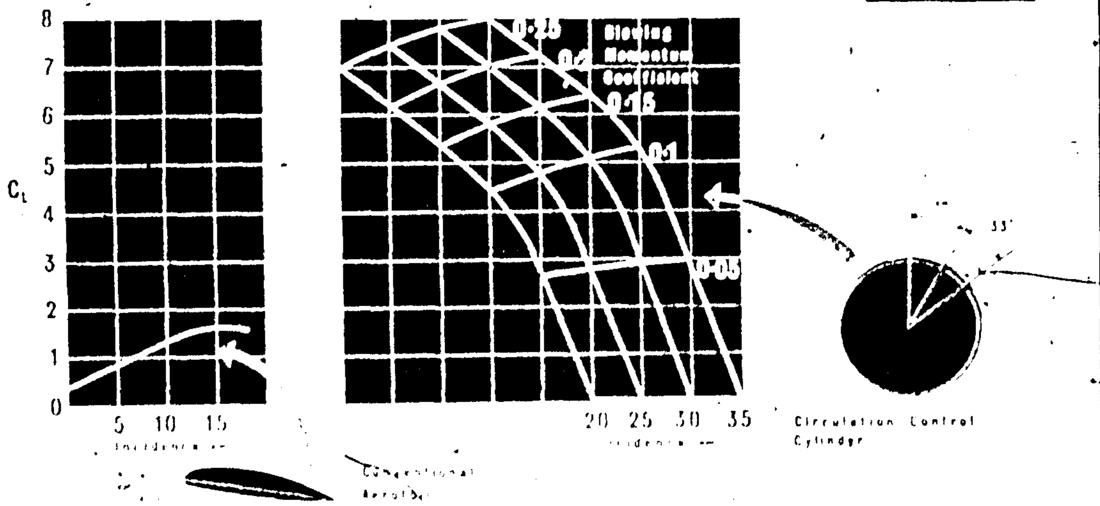
1	Dr. I. C. Cheeseman	A possible design for a vertical take-off aircraft. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. March, 1961.
2	R. E. Donham	Analysis of Stowed Rotor Aerodynamic/Aeroelastic Characteristics. Lockheed California Company, Burbank, 1961.
3	-	Lift - Dependent Drag for Circulation Control Aerofoils. VS/M/MC/803/11.18. Hawker Siddeley, Woodford. June, 1970.
4	J. Dunham	Circulation Control Applied to a Circular Cylinder. N.G.T.E. R.287. National Gas Turbine Establishment, Pyestock. July, 1967.
5	A. R. Osborn	An Interim Review and Appraisal of Two-Dimensional Tests on Circulation Controlled Sections. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. February, 1968.
6	J. Dunham	The Pressure Distribution Round a Lifting Circular Cylinder. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. March, 1964.
7	A. R. Osborn & J. M. Kettley	Wind Tunnel Tests on a 20% Ellipse with Tangential Slot Blowing. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. March, 1966.
8	A. R. Osborn & C. N. Sherbourne	Two-Dimensional Wind Tunnel Tests on 10% and 40% Elliptic-Section Models with Tangential Blowing Slots. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. June, 1968.
9	A. R. Seed	An Investigation of the Induced Velocity on a Two-Bladed Rotor Operating at High Thrust Coefficients in Hover. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. November, 1964.
10	M. B. Howard	Tests in the Hatfield V/STOL Wind Tunnel to Evaluate the Effects of Cyclic Modulation of the B.L.C. Air on the Rotors of the HS 803. ARD/WT/MISC/224. Hawker Siddeley, Woodford. December, 1970.
11	-	Rotor-Wing Interference Study. Proj. 803/23. Hawker Siddeley, Woodford. January, 1971.
12	D. J. Butter	Hover Tests on a Model Circulation Control Rotor. Proj. 803/22 Hawker Siddeley, Woodford. December, 1970.
13	J. Dunham & C. J. Hart	The Profile Drag of a Circulation Controlled Lifting Rotor. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. December, 1964.
14	Dr. I. C. Cheeseman	Circulation Control and its Application to Stopped Rotor Aircraft. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. August, 1967.
15	G. W. Dean, J. M. Kettley & Jenifer N. Roberts	Some Tests on a Circulation Controlled Lifting Rotor Using a Mobile Test Vehicle. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. September, 1968.
16	M. C. G. Smith & G. W. Dean	The Performance of a Circulation Controlled Rotor in Forward Flight Including the Effect of Wing Interference. M.O.D. P.E. Unpublished. National Gas Turbine Establishment, Pyestock. October, 1968.


STOPPED ROTOR AIRCRAFT
FIG. 1
THE AVIATION FIELD


STOPPED ROTOR AIRCRAFT
FIG. 2
WING LOADING 120 LB/SQ.FT.
DISC LOADING 5 LB/SQ.FT.
POSSIBLE CONFIGURATIONS

STOPPED ROTOR AIRCRAFT

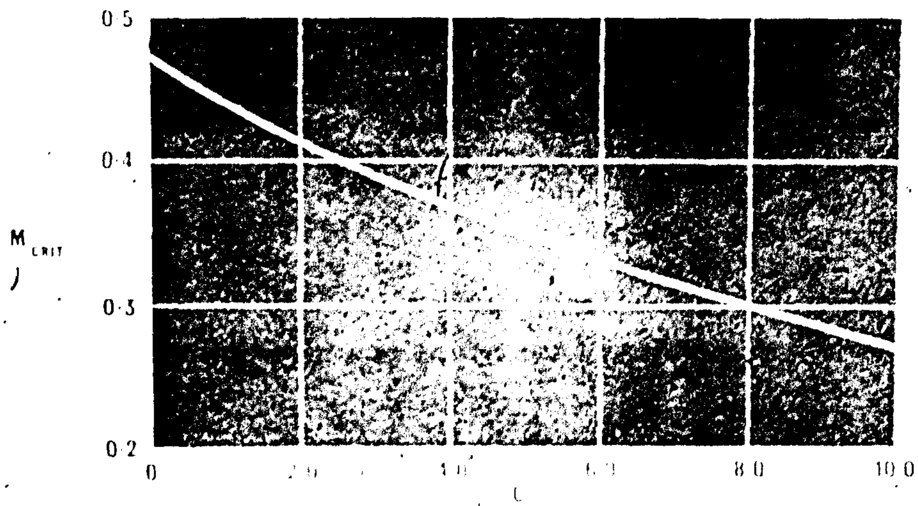
FIG. 3



SECTION CHARACTERISTICS

STOPPED ROTOR AIRCRAFT

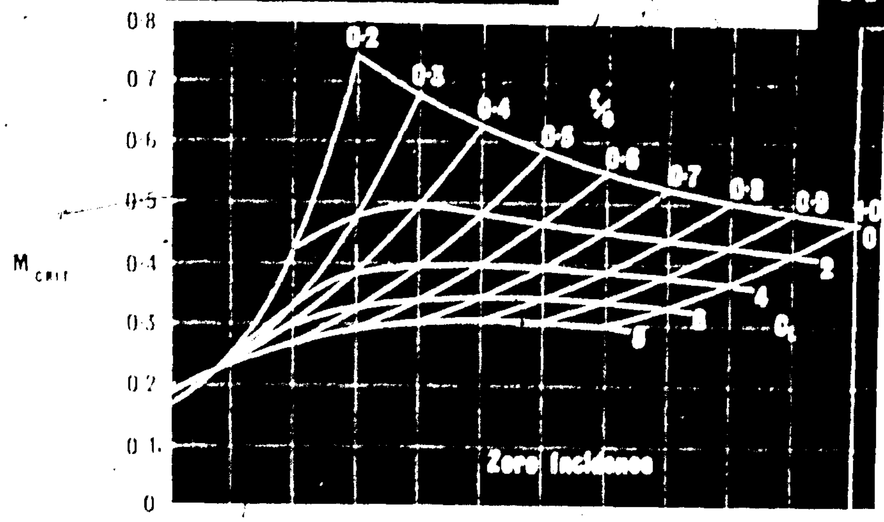
FIG. 4



CRITICAL MACH NUMBER OF CIRCULAR SECTIONS

STOPPED ROTOR AIRCRAFT

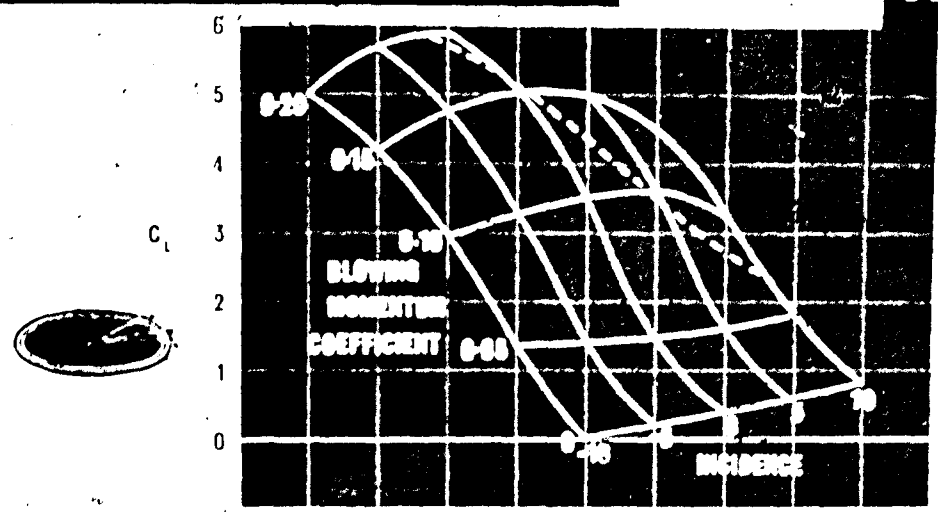
FIG. 5



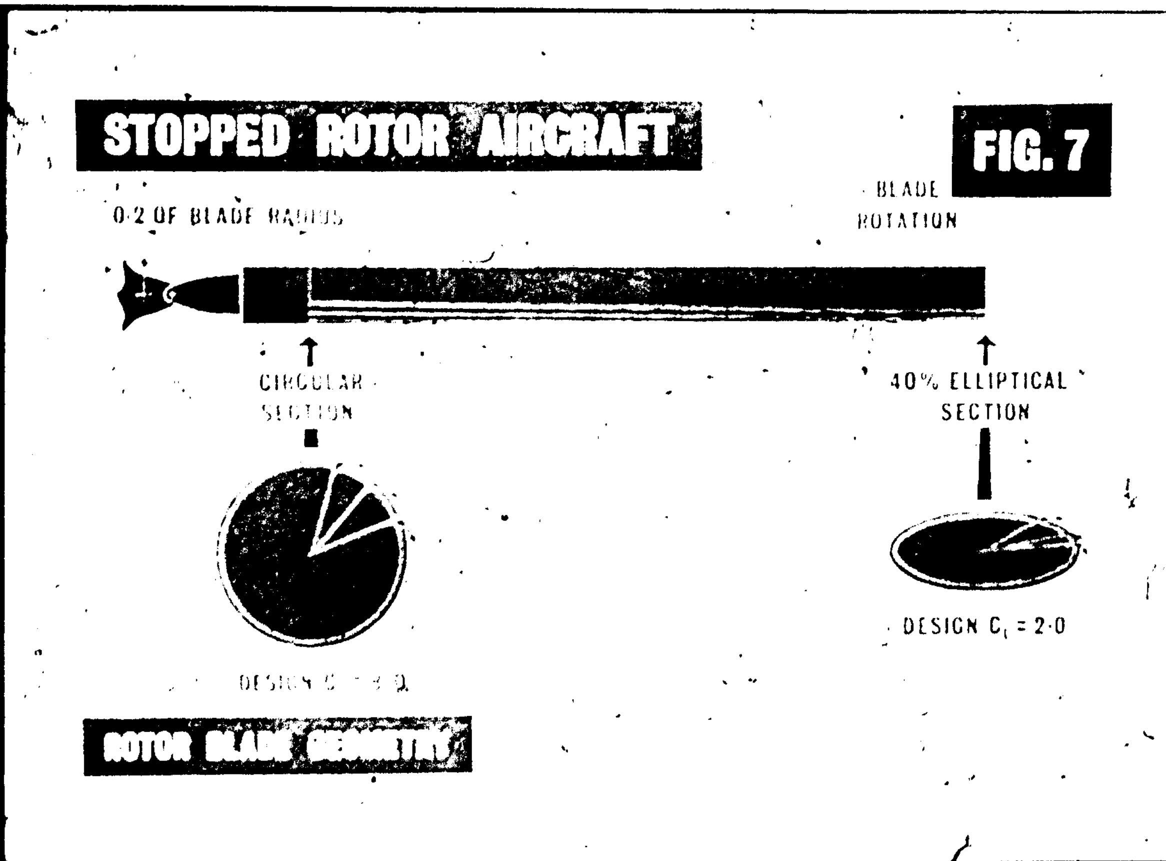
CRITICAL MACH NUMBER OF ELLIPTIC SECTIONS

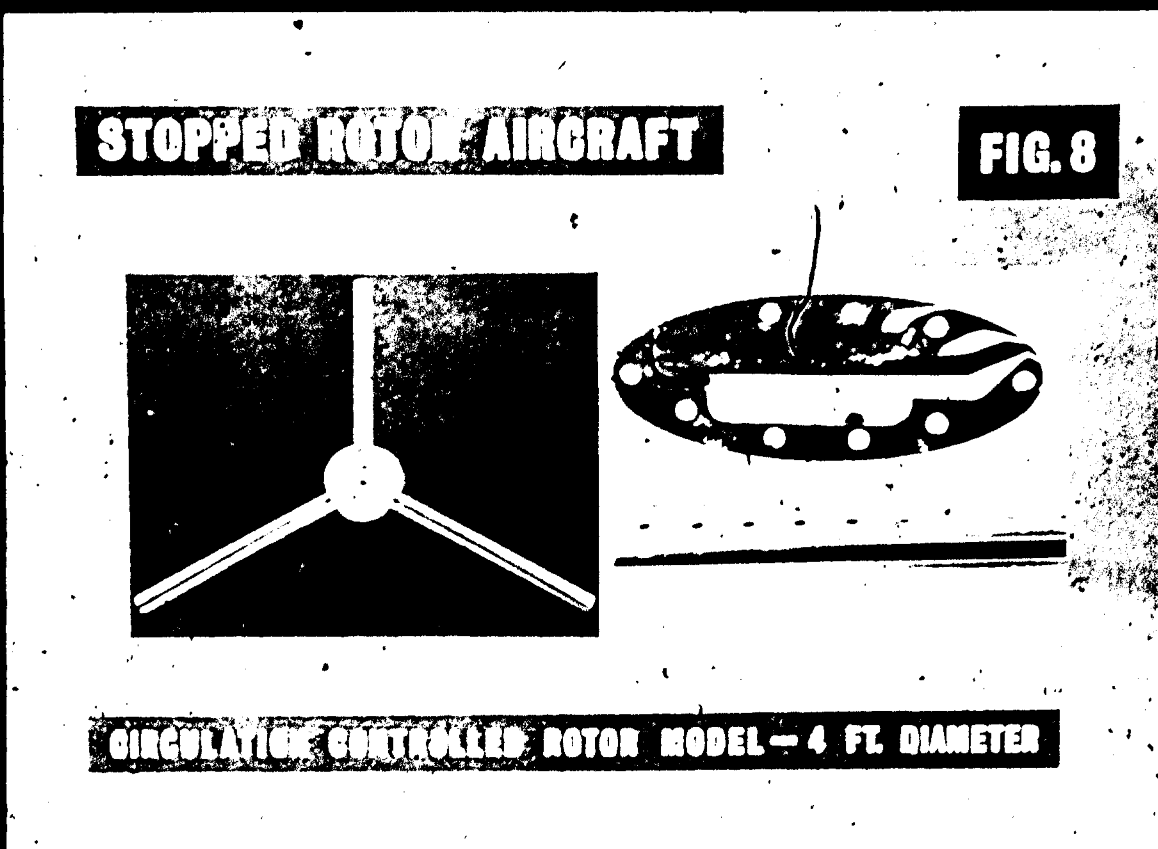
STOPPED ROTOR AIRCRAFT

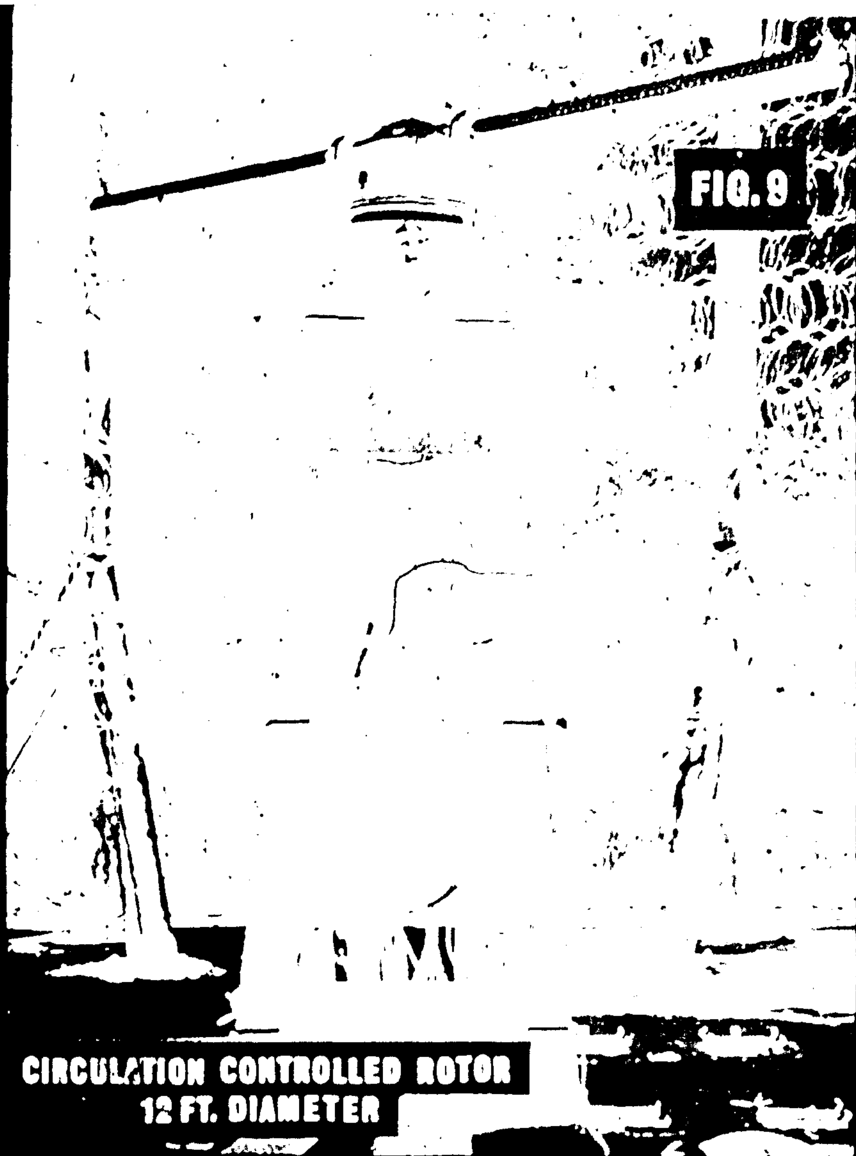
FIG. 6



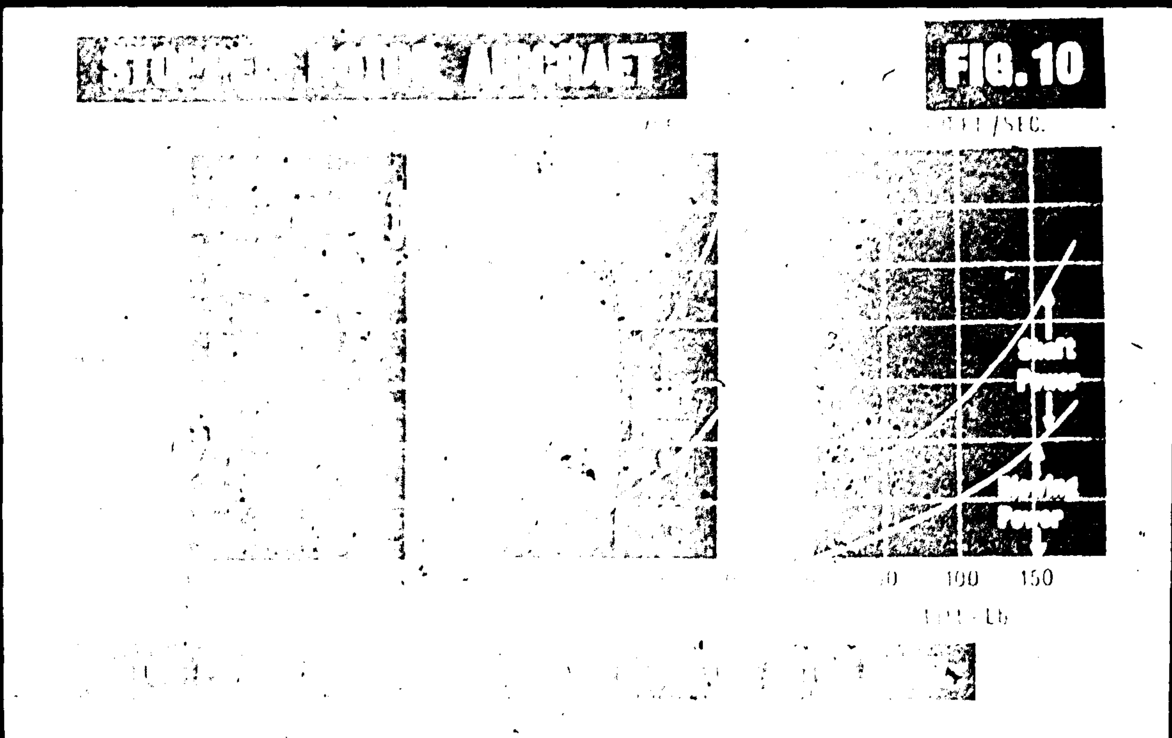
40% ELLIPTIC SECTION CHARACTERISTICS


STOPPED ROTOR AIRCRAFT
FIG. 7
0·2 OF BLADE RADIUS
BLADE ROTATION
CIRCULAR SECTION
40% ELLIPTICAL SECTION
DESIGN C_l = 2·0
ROTOR BLADE GEOMETRY

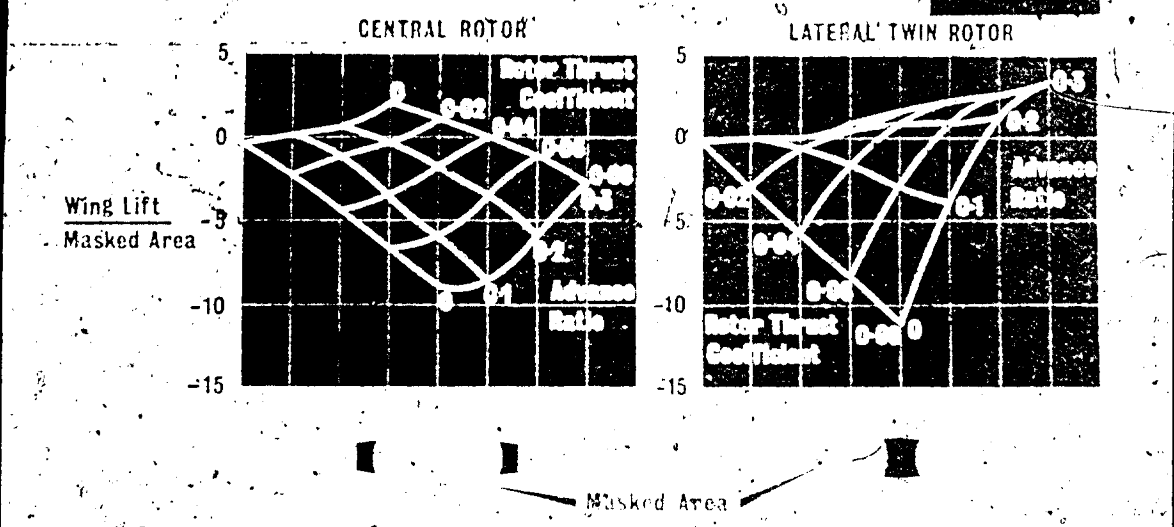

STOPPED ROTOR AIRCRAFT
FIG. 8
CIRCULATION CONTROLLED ROTOR MODEL — 4 FT. DIAMETER



CIRCULATION CONTROLLED ROTOR
12 FT. DIAMETER

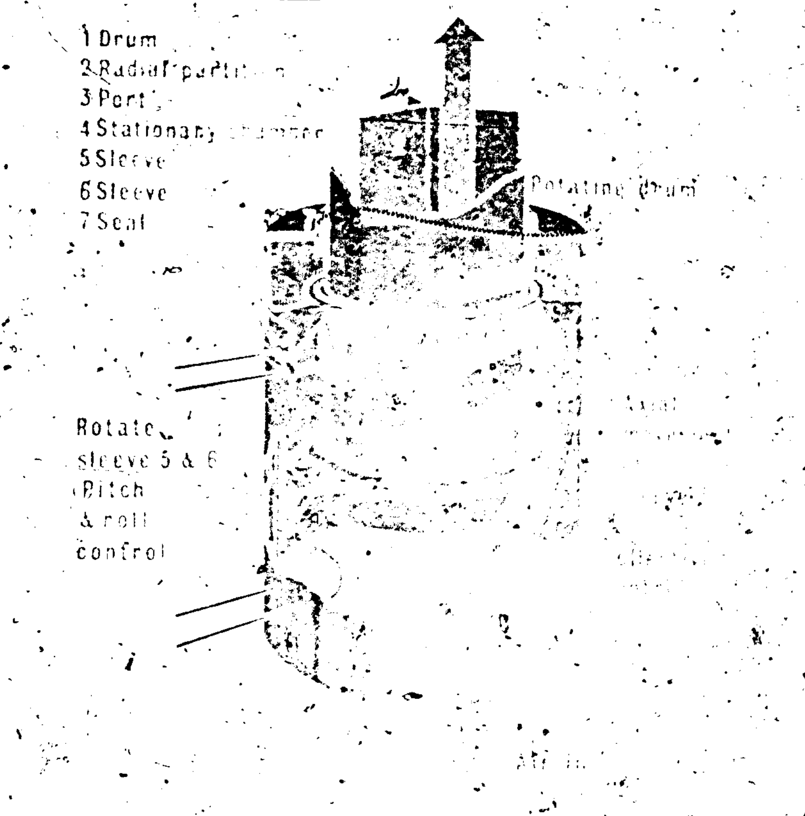


STOPPED ROTOR AIRCRAFT — **FIG.11**



ROTOR AIRFRAME INTERFERENCE

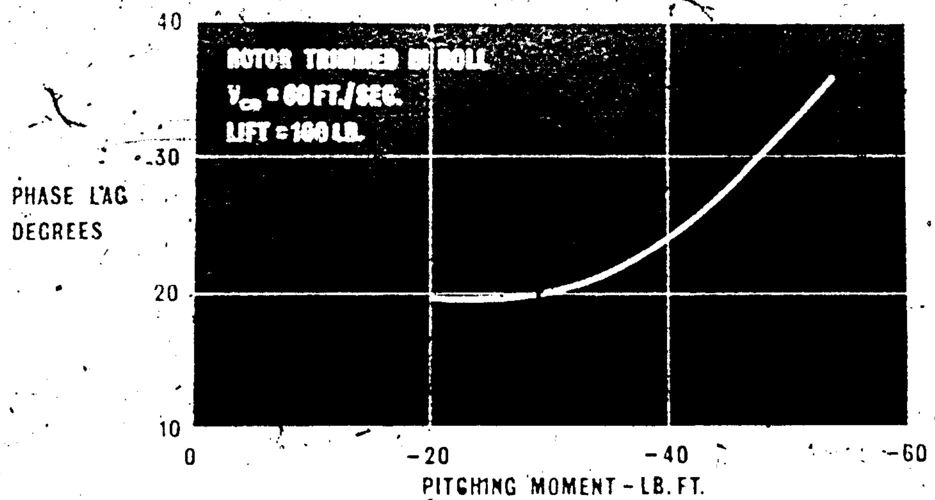
STOPPED ROTOR AIRCRAFT — **FIG.12**



CYLINDRICAL CYCLIC CONTROL VALVE

STOPPED ROTOR AIRCRAFT

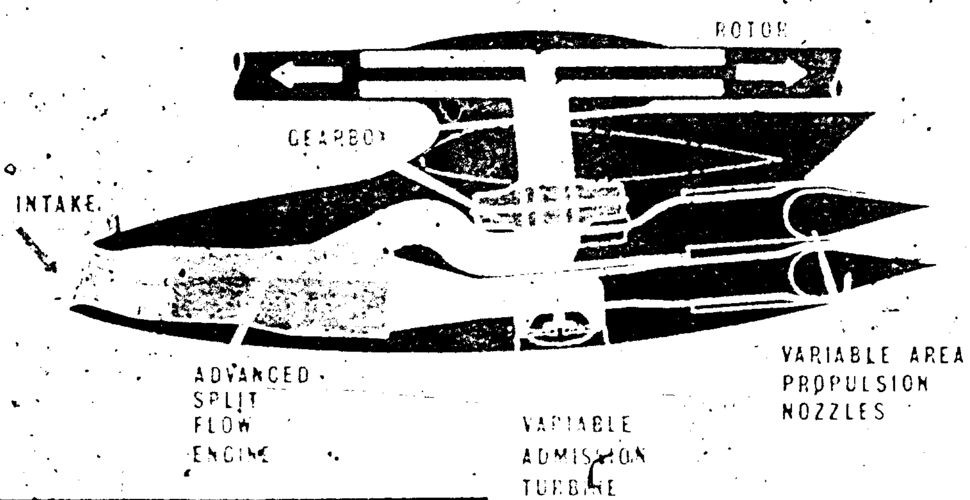
FIG. 13



CYCLIC CONTROL – 4 FT. DIAMETER ROTOR MODEL

STOPPED ROTOR AIRCRAFT

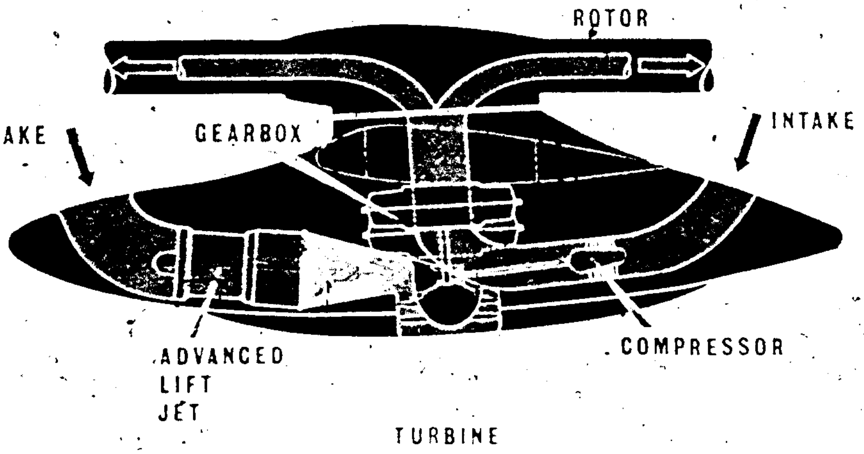
FIG. 14



INTEGRATED POWER PLANT

STOPPED ROTOR AIRCRAFT

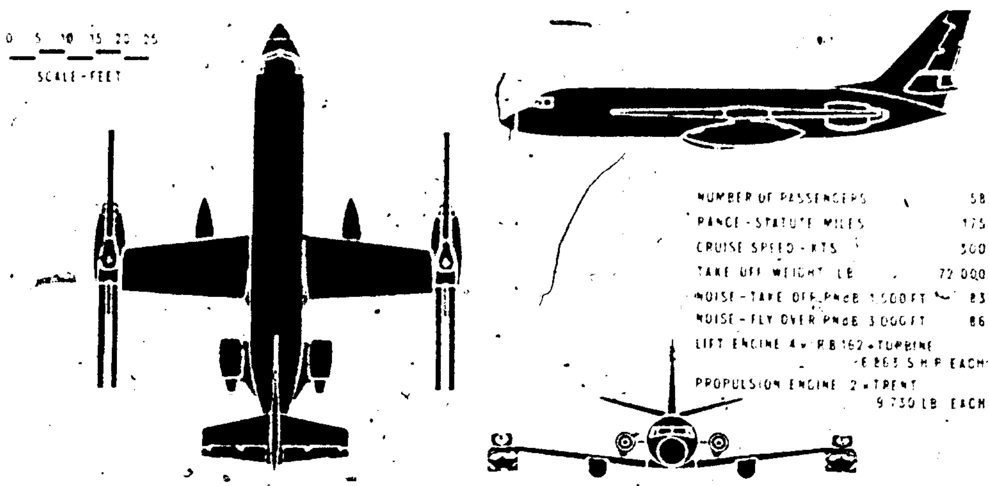
FIG.15



SEPARATED POWER PLANT

STOPPED ROTOR AIRCRAFT

FIG. 16



NUMBER OF PASSENGERS	58
RANGE - STATUTE MILES	175
CRUISE SPEED - KTS	300
TAKE OFF WEIGHT LB	72 000
NOISE - TAKE OFF PNdB 1 500 FT	83
NOISE - FLY OVER PNdB 3 000 FT	86
LIFT ENGINE 4 × RB 162 + TURBINE (6 265 S.H.P. EACH)	
PROPULSION ENGINE 2 × TRENT 9 730 LB EACH	

GENERAL ARRANGEMENT – H.S. 803

FIELDS OF APPLICATION OF JET FLAPPED ROTORS

by

M.Kretz
Chief Engineer
Giravions Dorand
Suresnes, France

SUMMARY

The jet-flap rotor has the advantage of reducing weight and complexity by integrating rotor lift, rotor drive and rotor control at the source of aerodynamic lift on the outboard parts of the blade. Under these conditions, two-blade fixed-pitch rotors may be used for forward airspeeds in excess of 250 knots. Lift control, which in our case includes multi-cyclic components, modulates the aerodynamic force, thereby considerably reducing blade alternating stresses and rotor vibration. Experimental confirmation of theoretical studies was obtained from the tests effected in the 40 x 80 foot wind tunnel at the Ames Research Center on a jet-flap rotor of 40 feet in diameter.

Analysis of the field of application of the jet-flap rotor shows that the cost-effectiveness of this technique is extremely good when applied to heavy helicopter and stoppable rotor designs. Comparison with equivalent mechanically driven heavy-lift rotorcraft shows empty-weight gains of 30 to 40%. Initial cost gains for these vehicles is even higher, approaching 50%. The feasibility of an aircraft having a 0.85 Mach number capability and possessing a stoppable and stowable non-folding two-bladed rotor has recently been established.

The weight analysis in this case also demonstrates the attraction of the jet-flap concept, which combines the features of both low weight and low cost, with a long duration hovering capability. The jet-flap rotor thus makes it possible for the same aircraft to have the high airspeed characteristics of a modern airplane coupled with the low-speed advantages of a helicopter.

When approaching the subject of advanced rotors in general and jet-flap rotors in particular, it is seen that an analysis of the situation has both scientific and technical aspects, often mingled with elements of a philosophical and psychological nature. When replying to the question why have jet-flap rotors not yet taken their place amongst the many rotary wing applications for which they are particularly well suited, one feels that technical reasons alone do not suffice. Thus it would appear useful at this AGARD meeting to try to draw a lesson from the past in order to approach the future better. Our Company first obtained contracts on blown rotor applications twenty years ago. During the years 1954 and 1956, two scaled-down jet-flap rotors were tested in the ONERA wind tunnels and very encouraging results were recorded. These results justified the construction of a rotor of practical dimensions, having a diameter of 12 meters, designed for effecting tests over a wide experimental range (Fig.1). This and subsequent works were effected under French and American contracts, to which the following organizations contributed: the Service Technique Aéronautique (STAé), the Centre de Prospection et d'Evaluation (CPE), the US Army, TRECOM, AVLABS and NASA through its Ames Research Center (References 1, 2 and 3).

The 12-meter rotor, designated DH 2011, was projected in 1959. It underwent its first whirl trials in 1964, and in 1965 it was tested in NASA's 40 x 80 foot wind tunnel at Ames. The results obtained have fully demonstrated the effectiveness of this formula. Work has continued, and a new program executed in the Ames 40 x 80 foot wind tunnel in 1971 has proven the soundness of our forecasts.

In the mean time, between the earliest work and the last test program, twenty years have gone by, a period which, if surprisingly long, is explained by the modesty of the appropriations for developing this technique. Indeed, if we take a look at the hours of work expended in this field, we are surprised by the importance of the results obtained: 200,000 hours, which corresponds to an average of 5 people. This effort has allowed the construction of three rotors and the execution of four wind tunnel test programs. It covers, of course, all theoretical and practical studies effected to date and has enabled us to accumulate valuable experience, which is most promising for the future.

If we now look at evolution in the field of helicopters, we can see that the techniques applied over the last twenty years are reaching their limits. The diversity of new projects prove this, as do recorded performances: world records have not jumped since Sud-Aviation's Super-Frelon flew at 190 knots in 1962. The latest progress has been the introduction of the gas turbine as a power plant. In the present state of affairs, any evolving technical progress demands and will demand efforts in work and cost out of proportion with gain. This evolution can be summed up by the graph shown in Figure 2, where it is seen that the performance obtained for a given technique of new concept increases rapidly at the start and then levels off asymptotically, tending towards limiting values. These curves resemble the learning-curves for a man when learning to execute a given task. In their general form, these curves regulate all human progress and inflexibly remind us of the difficulties we must overcome to establish new working principles and the difficulties by no means neglibible, even if the new concept is of great value, we have in putting to one side well established techniques and breaking with old habits. The psychological aspects hidden behind these curves are far-reaching and we feel them each time a new problem is put to us. We are ever divided between the desire to do better and the necessity to make a halt, a period of rest and exploit what has been achieved. As much as VTOL needs are great, we feel that the present helicopters, with their gas turbines, mechanical transmissions, mechanically controlled blade pitch and multi-blade rotors, hinder progress considerably. The weight-to-price ratio compared with that of fixed-wing aircraft is still very high and it is here that exists the brake to all technic ' progress concerning rotors. Reducing this ratio is both the target and criterion of present and future VTOL development (Fig.3).

The reasons which lead us to believe that the use of a system comprising a jet-engine as a source of compressed gases, a pneumatic power transmission and a two-blade rotor controlled by jet flaps, is likely to revolutionize rotary wing techniques, are many. They are discussed further, but let it be said straight away that an overall advantage appears when analyzing this technique: the empty weight and consequently the acquisition cost of jet-flap rotorcraft are notably less than these employing mechanical systems. Let us quote a few figures. If we look at the weight-to-thrust ratio of the lifting power plant, comprising the power source, power transmission and the rotor, it is seen that it is of the order of 7 in our case, a very high value of jet engines. This ratio is much lower for mechanical systems, where it is between 4 and 5. It should be noted that the lift power plant, which differentiates mechanical systems from jet-flap systems, represents a major part of equipment cost. Comparisons made with mechanical systems give a first indication that for a variety of missions the price ratio between mechanical and jet-flap systems is of the order of 2. The high value of this ratio shows the considerable opportunity for reducing equipment cost. This result is, from another point of view, not surprising when it is recalled that for a given power output, a simple jet engine costs half as much as a turboprop engine.

Overall advantages of light weight, simplicity and low cost produce a large number of basic advantages characterizing jet-flap craft (Fig.4). The jet-flap rotor is fixed pitch, which is a structural advantage dominating this type of rotor and is unique amongst rotary wings in general.

Together with the possibility due to blowing of using two-blade rotors for obtaining high forward velocity, this rotor avoids the present hub and blade-root complications. The difference with conventional systems recalls that between fixed and variable pitch propellers. From the aerodynamic point of view, the jet-flap possesses the property of being able to vary lift very rapidly up to high harmonic frequencies (Fig.5). Trailing edge blowing, which we use, has been the subject of many studies in France (Ref.3) and the United States (References 4 and 5) and is now well known, In forward flight, the aximuthal variation in jet-flap deflection is obtained by a multicyclic control system. This ability to vary lift forces according to a desired law enables the level of vibration transmitted to the fuselage and blade stresses to be considerably reduced. It is simply because of this ability that two-bladed helicopters can be envisaged for forward speeds of 250 knots. The possibilities of stress and vibration reduction using multicyclic control have been demonstrated experimentally this year, and to our knowledge, for the first time, by the DH 2011 rotor, and a large volume of analysis work is presently under way. These latest results will be published by NASA in the near future.

Moreover, rapid lift variation may be usefully employed in the case of transient conditions such, for example, as during the stopping of a rotor, when the feedback channels enable forces operating on the blades to be measured precisely and continuously.

In addition to its structural simplicity, the fixed-pitch, jet-flap rotor has other advantages: it eliminates the danger of aeroelastic vibration and enables rotors of low solidity to be used. In most cases, the average lift coefficient is close to 1, thereby enabling blade surface to be decreased by at least half compared with unblown rotors. Furthermore, the fact that the jet exhausting from the trailing edge induces a region of very low pressure, similar to that at the leading edge, displaces the aerodynamic center from the quarter-chord point towards the middle of the chord, enabling the blade to be balanced much nearer the trailing edge, whence considerable saving in blade weight. It should be noted that the DH 2011 rotor blade has its center of gravity at the 35% point of the mean chord and has never shown any signs of aeroelastic instability.

It is further obvious that jet-flap rotorcraft do not need a tail rotor. Since the rotor is free in rotation, its rpm may be optimized for each flight configuration. Aircraft of this type can easily perform jump take-offs in cases of overload or high ground altitude.

From the point of view of performance, the jet-flap rotor, compared with unblown rotors, enables helicopters to reach high vertical acceleration at high forward speeds. This advantage comes from the fact that the high-lift capability increases as the power supplied to the rotor increases. Let us take the example of a 4-ton tactical support helicopter, designed to fly at 250 knots (Fig.6). This helicopter is capable of executing a 60° – bank turn at 200 knots, producing an acceleration of 2g. Figure 7 shows the general tendancy of load factor variation in forward flight. The shape of this curve is similar to that for power in forward flight. This characteristic differentiates the jet-flap rotor from conventional rotors, whose performance falls off with forward speed.

The ability of modulating blade lift whilst moving round the rotor plane, associated with its high-lift characteristics, enables rotor speed limits to be pushed back and to envisage helicopters flying at 250 knots. This characteristic is of additional advantage in carrying heavy loads, whatever their weight. We can quote here the example of an application which shows a difference of 2 to 1 in cost effectiveness between conventional means and a jet-flap rotor configuration (Ref.6). It concerns an automatic radar observation platform, which carries a 1700 kg load at an altitude of 7000 meters for 5 hours in the hover (Fig.8) From the aspect of layout, the generator is placed vertically in the axis of symmetry of the vehicle. The payload is distributed around the engine in the spherical nacelle. A spherical shape was chosen to reduce aerodynamic forces and perturbations due to rotation.

The project leads to a vehicle total all-up weight of 5400 kg and an empty weight of 1530 kg, only 28.4% of the all-up weight (Fig.9). The ratio between the maximum lift at ground level and the combined weight of the rotor, power transmission and power plant is 7.3. This ratio, as already emphasized, is remarkably high for a helicopter rotor. The rotor is two-bladed, with a loading of 24 kg/sq. m. The platform is powered by a bypass engine, the SNECMA/TURBOMECA M 49 "Larzac", operating as a gas generator with a maximum pressure ratio of just under 1.9.

The cost-effectiveness analysis of this project shows that a platform with mechanical transmission would have cost 90 to 100% more than the jet-flap rotor platform. This cost saving aspect is not limited to this particular case. General application studies (Ref.7) show that light weight is the predominant characteristic of helicopters, and especially of crane-helicopters, fitted with jet-flap rotors (Fig 10).

The crane-helicopter has been the subject of many studies in Europe and the United States and abundant literature exists. Without quoting a case of application, it is desirable to dissipate a number of misunderstandings and a certain mistrust surrounding tip driven rotors in general and jet-flap rotors in particular. This mistrust concerns power transmission and the resulting fuel consumption. In order to do the job properly, one should judge aircraft in a comparative manner for well defined missions, which until now has never been done with sufficient objectivity as long as the starting hypotheses upset the results and final conclusions (see Reference 8 for the difficulties encountered in making comparison studies). Here, we shall limit the discussion to the power losses between the isentropic power at the gas generator outlet and the equivalent mechanical power obtained at the rotor. This loss is essentially due to the total pressure loss (neglecting the loss due to ejection of kinetic energy, the aspects of which are generally well understood) and may be defined by a single parameter, K_T , equal to the difference between the total pressures at the gas generator and ejection nozzle, measured on a stationary rotor divided by the dynamic pressure in the blade. Coefficient K_T has the advantage of depending essentially on duct geometry, such as, for example, the coefficient of aerodynamic friction drag. Experience has shown that considerable effort must be made to reduce this as far as possible by exhaustive test programs. The importance of internal aerodynamics is similar to that for all jet engines. Naturally, in a case of real application, the internal losses would to some extent exceed those given by coefficient K_T , there being added principally the losses due to cooling and the perturbations created by the rotor. However, coefficient K_T is very useful when analyzing losses and constitutes a quality criterion for all tip driven rotors. Appreciation of the sensitivity of the transmission system to pressure ratio variation is much easier with this coefficient. There is indeed close correlation between K_T and the power available for lifting the craft. In Figure 11, it is seen that for constant isentropic power produced by the gas generator, in this case by a fan engine, the power available to be absorbed by the induced velocity, therefore power proportional to lift for a given rotor diameter, has a marked maximum for a given K_T . A difference in pressure ratio, even very small, severely affects power transmission efficiency. We see here the first problem of correctly matching a jet driven rotor. The second difficulty in obtaining high performance is due to the sensitivity of the internal geometry to the local value of coefficient K_{Te} (Figure 12, Reference 9). This figure shows to what extent the local coefficient due to a duct bend can change the value of the losses and consequently rotor matching.

Our experience with cascade in the outboard part of the blade (Fig.13) shows that a coefficient K_{Te} = 12% of the dynamic pressure can be obtained in this case, with very carefully defined cascade shapes derived from a series of tests requiring considerable time. In general, 90° bends with a total pressure loss of 12 to 15% can be made, but the configuration of the whole of the circuit comprising several internal bends requires detailed experimentation. The third difficulty arises from the generator itself, whose pressure ratio and gas temperature must be chosen with precision to obtain the required performance. There is no doubt that these three difficulties closely condition jet driven rotor performance. If, however, sufficient research and development effort is deployed, real power transmission efficiencies can attain the theoretical values mentioned above. Thus overall transmission coefficients of 71% in the case of the radar platform and 62% in the case of a stoppable-rotor helicopter, dealt below, can be obtained. It should further be emphasized that writers not in favor of tip driven rotors often speak of pressure loss "compensation" by the centrifugal effect of the rotor. This hypothesis, which has no physical grounds, would, if true, fundamentally destroy any project, since it is equivalent to limiting the transmission efficiency to 50%.

The VTOL stoppable-rotor aircraft is the third example illustrating the field of application of jet-flap rotors (Fig.14). The project presented here is a particular case of projects effected under contract for the Centre de Prospective et d'Evaluation (CPE). corresponding to a VTOL craft with a stoppable and stowable rotor (Ref.10). This example is characteristic of most craft of this type. In this case, the rotor is used for short periods during flight for take-off, approach and landing. Maximum speed at ground level is Mach 0.85 without reheat. For a tactical mission, the aircraft has an operational range of 1 000 km with full internal tanks and in a ferry configuration, can cover a distance of 4500 km. In its role, the rotor is very different from that of a helicopter. Although lift in the hover state and at low forward speeds is provided by the rotor, the aircraft is controlled in pitch and yaw by means of compressed air jets through swivelling nozzles. Control in roll, however, can be effected by the rotor. The rotor is placed very close to the fuselage and is servo-controlled so as to remain in a fixed plane. Flapping angle, for gusts of up to ± 10 m/s, does not exceed ± 1° during the rotor stopping phase, which occurs at 70 m/s. This stopping phase is very short, taking 7 seconds. The rotor is stopped by reversing the jet ejected at the end of the blade. In this particular case, the jet flaps occupy 50% of the blade radius and use 30% of the flow transmitted to the rotor.

The power source is the same for the aircraft and the rotor, the gas flow being directed either to the rotor or to the rear nozzle for aircraft propulsion by means of a diverter valve. In the project presented, the aircraft is essentially subsonic, but supersonic projects are being studied. In general, the stowable rotor technique seems particularly well adapted for configurations requiring an engine thrust to total weight ratio of 0.3 to 0.7. It should be noted that the noise level of jet-flap rotor aircraft is reduced by slotted nozzles and that the induced velocity of 20 to 25 m/s, corresponding to a rotor loading of 100 to 150 kg/sq. m, allows all-terrain operation without danger of erosion.

From the point of view of its configuration, the aircraft is designed for tactical support, with an added lift rotor. It is to be noted that the rotor part corresponds to a comparatively small fraction of aircraft volume and weight. Thus the elements corresponding to rotor lift occupy 10% of the aircraft volume and constitute 9% of the total weight. In the version studied, the all-up weight is 12000 kg, divided as follows:

A	Airframe	4155 kg
B	Power plant	790 kg
C	All-mission installation	1730 kg
D	Particular mission installation	none
E	Crew	85 kg
F	Fuel	2280 kg
G	Variable loads	1860 kg
H	Rotor	1100 kg
	Total weight	12000 kg

It should be noted that the lift system, comprising the power plant B and rotor H, has a thrust-to-weight ratio of 6.35, a ratio rarely attained by jet engines. According to the weight break-down, the empty weight of the aircraft is 6775 kg and the payload 5225 kg, giving a payload-to-total weight ratio of 43.5%.

The three examples quoted show that the field of application of jet-flap rotors is very wide, covering the whole of the helicopter domain. The cases mentioned correspond to those where the use of a jet-flap rotor is most appropriate from the point of view of its cost-effectiveness advantage over existing projects or projects being studied.

By combining lift, propulsion and lift control in a fixed-pitch two bladed rotor, the jet-flap rotor, associated with a jet engine as power source, constitutes a particularly light weight, simple and low-cost assembly, which enables helicopters to fly at higher forward speeds and to lift heavier loads, thereby exceeding the present speed and weight limits. Its light weight and smaller volume, together with the flexibility of aerodynamic force control by means of jet-flaps, enables it to be stopped in flight and stowed. Thus in the extreme case, its characteristics associated with those of fixed-wing aircraft can produce a new generation of stoppable and stowable rotor VTOL aircraft, where the helicopter and fixed-wing aircraft coexist efficiently in the same vehicle.

REFERENCES

1. – Giravions Dorand. Documents concernant les rotors DH4, DH5, DH2011, 1954-1971.

2. Dorand, R., Tararine, S. *Determination through Wind-Tunnel Tests and Analytical Methods of the Optimum Deflection Devices Suitable for Use on Jet-Flap Helicopter Rotor Blades.* TRECOM No.DA-91-591-EUC-1203, December 1960.

3. Poisson-Quinton, Ph. et al. *Recherches théorique et expérimentales sur le côntrole de circulation par soufflage appliqué aux ailes d'avions.* ONEAR TN No.37, 1956.

4. Mc Cloud, J.L., Evans, W.T. *Analytical Investigation of a Helicopter Rotor Driven and Controlled by a Jet-Flap,* NASA TN D-3028, Septemter 1965.

5. Mc Cloud, J.L., Evans, W.T. *Performance Characteristics of a Jet-Flap Rotor.* Conf. on V/STOL Aircraft, NASA Ames Res. Center, April 1966.

6. Kretz, M. *Plate-forme de surveillance radar.* Contrat CPE, DE.07-37 E.1, December 1968.

7. – Jet-Flap Rotor. Preliminary Application Study LTV Studies Rep. No.2-55400/8R-50466, April 1968.

8. Deckert, W.H., Mc Cloud, J.L., III *Considerations of the Stopped Rotor V/STOL Concept.* J. Am. Hel. Soc. January 1968.

9. Idel'cik *Memento des pertes de charge.* Eyrolles, Paris, 1960.

10. Kretz, M. *Avions d'appui tactique à voilure tournante escamotable.* Contrat CPE, DE.70-46, July 1971.

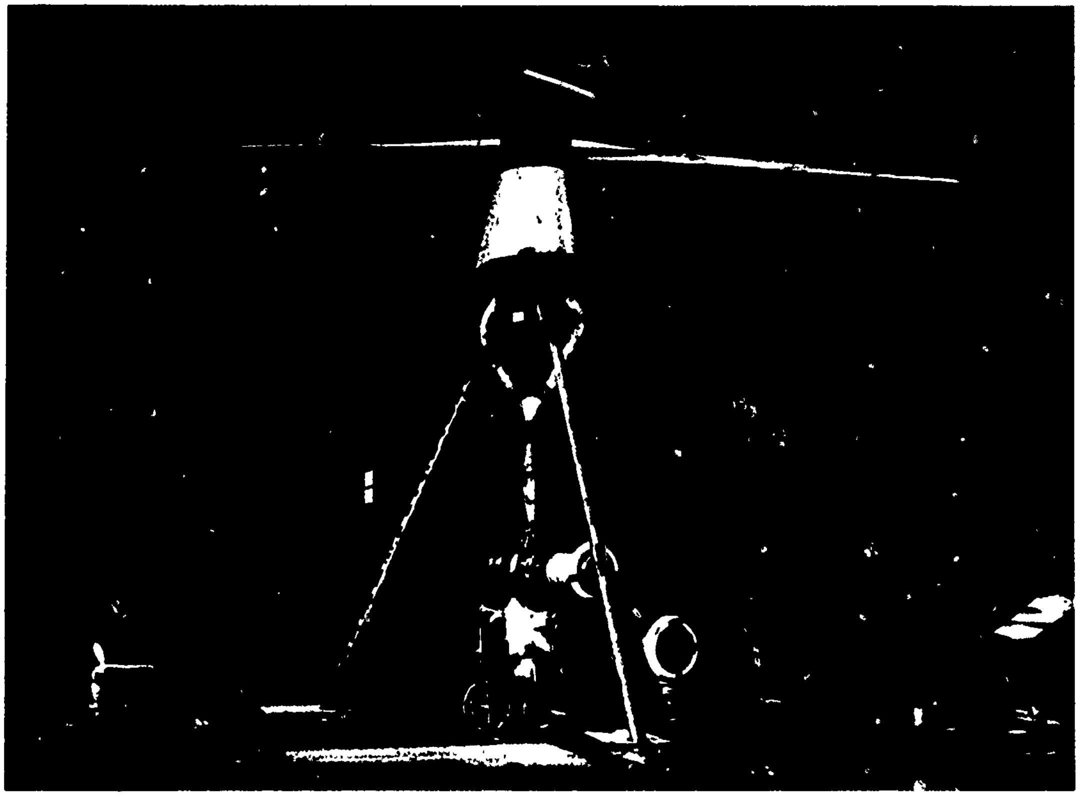


Fig.1 DH 2011 rotor in the NASA's Ames 40 x 80 wind tunnel
Rotor DH 2011 dans la soufflerie 40 x 80 d'Ames, NASA

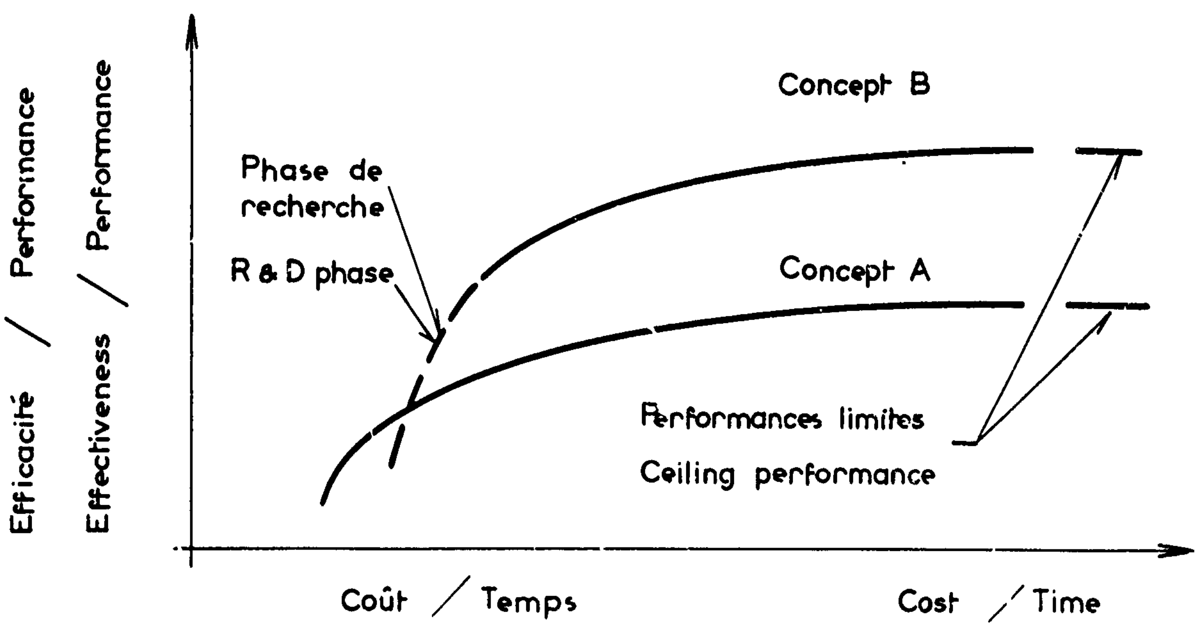


Fig.2 General trends of technical progress
Allure générale du progrès technologique

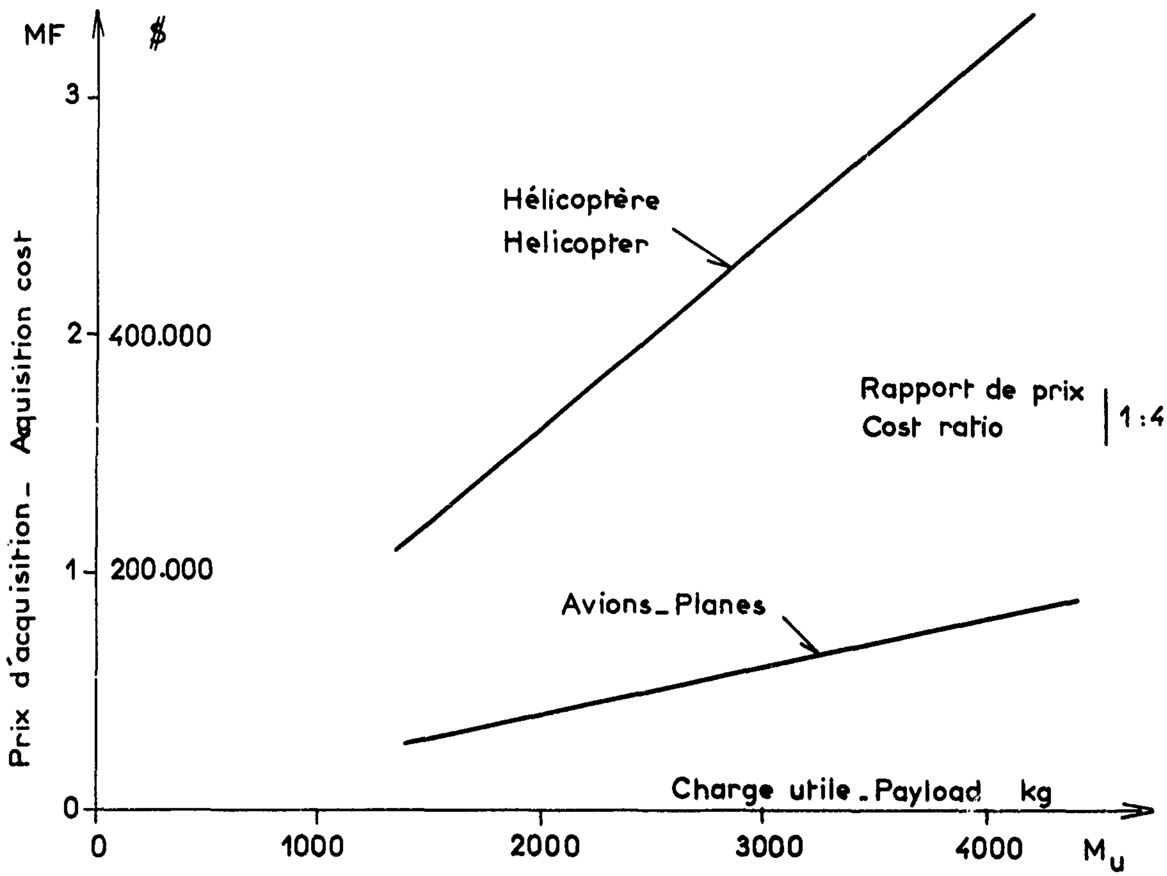


Fig.3 Cost comparison
Comparaison de prix

● ROTOR A PAS FIXE	● FIXED PITCH ROTOR.
● ROTOR BIPALE DE GRANDE VITESSE	● TWO BLADED HIGH SPEED ROTOR.
● RÉDUCTION DES CONTRAINTES ALTERNÉES ET DES VIBRATIONS PAR COMMANDE MULTICYCLIQUE.	● ALLEVIATION OF FATIGUE AND VIBRATION PROBLEMS BY MULTI-CYCLIC CONTROL.
● APPLICATIONS AU ROTOR STOPPABLE GRÂCE AU CONTRÔLE RAPIDE DES FORCES AÉRODYNAMIQUES.	● STOPPED ROTOR CAPABILITY BY CONTINUOUS CONTROL OF FORCES ON THE BLADE.
● ÉLIMINATION DES PROBLÈMES AÉROÉLASTIQUES. PAR RIGIDITÉ ÉLEVÉE EN TORSION.	● ELIMINATION OF AEROELASTIC PROBLEMS BY HIGH RIGIDITY IN TORSION.
● FAIBLE PLÉNITUDE.	● LOW CHORD BLADE
● ÉQUILIBRAGE AISÉ DE LA PALE.	● ALLEVIATION OF WEIGHT BALANCE PROBLEMS OF THE BLADE.
● FAIBLES CHARGES DANS LES COMMANDES.	● LOW CONTROL FORCES.
● RÉGIME ROTOR VARIABLE.	● VARIABLE R.P.M.
● PAS D'ANTICOUPLE	● NO ANTITORQUE DEVICE.

Fig.4 Advantages of jet-flap rotor
Avantages d'un rotor a volets fluides

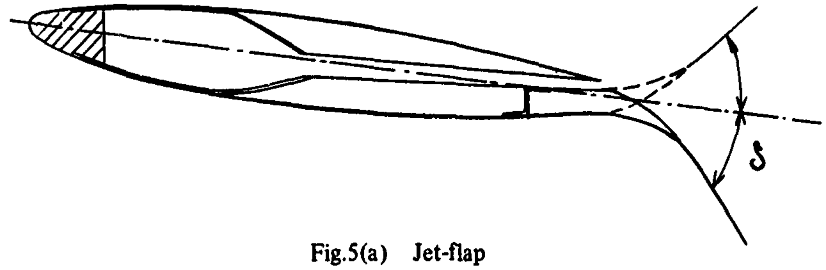


Fig.5(a) Jet-flap
Volet fluide

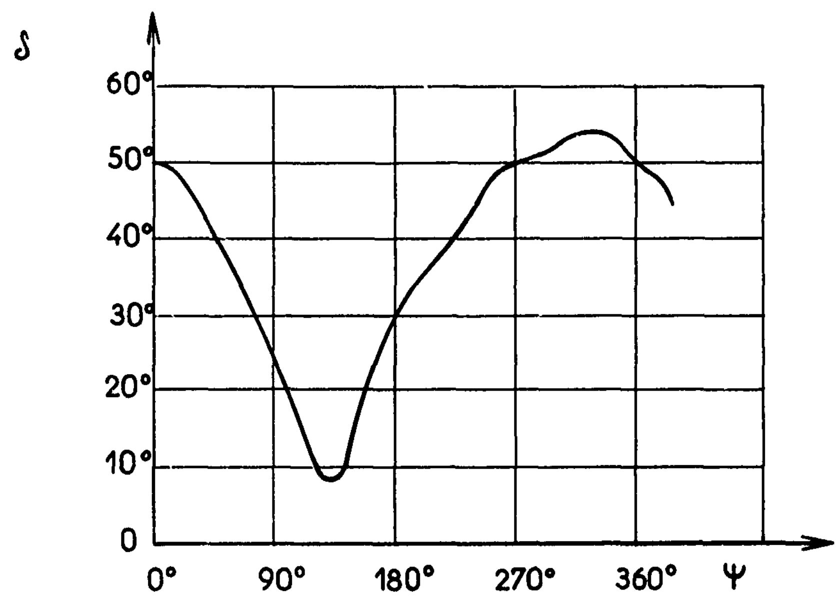


Fig.5(b) Azimuth variation
Loi azimutale

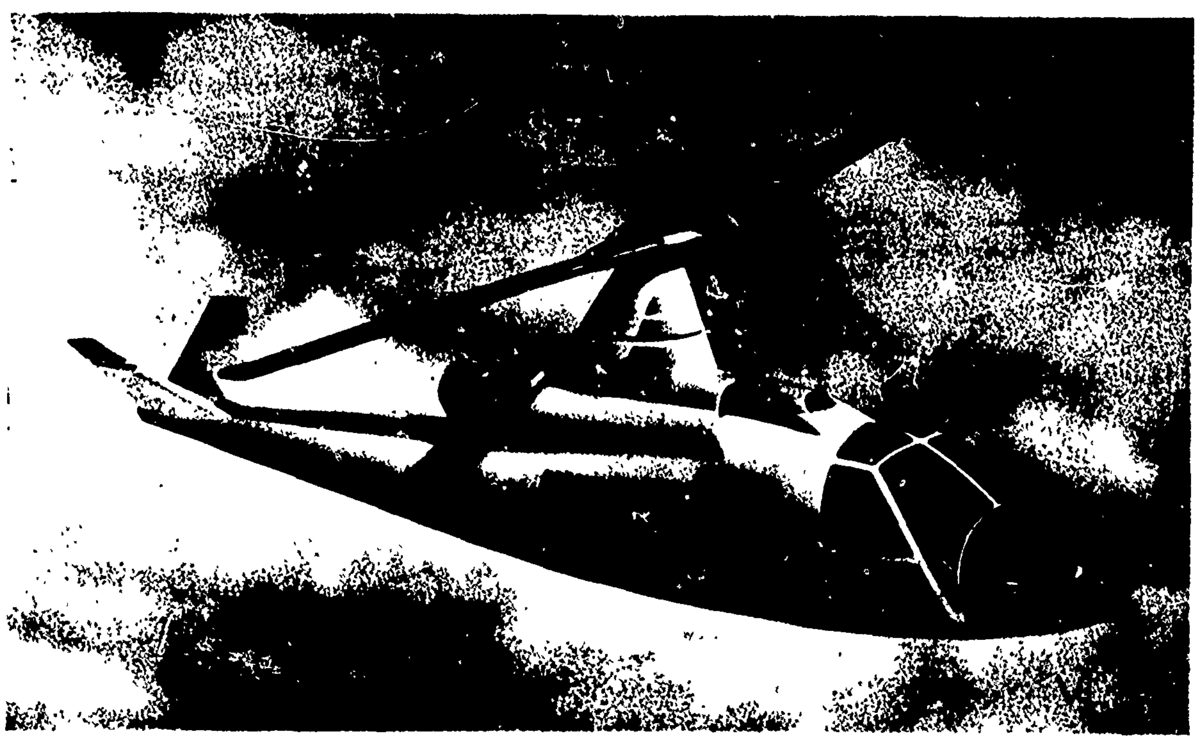


Fig.6 High speed jet-flap helicopter
Hélicoptère rapide à volets fluides

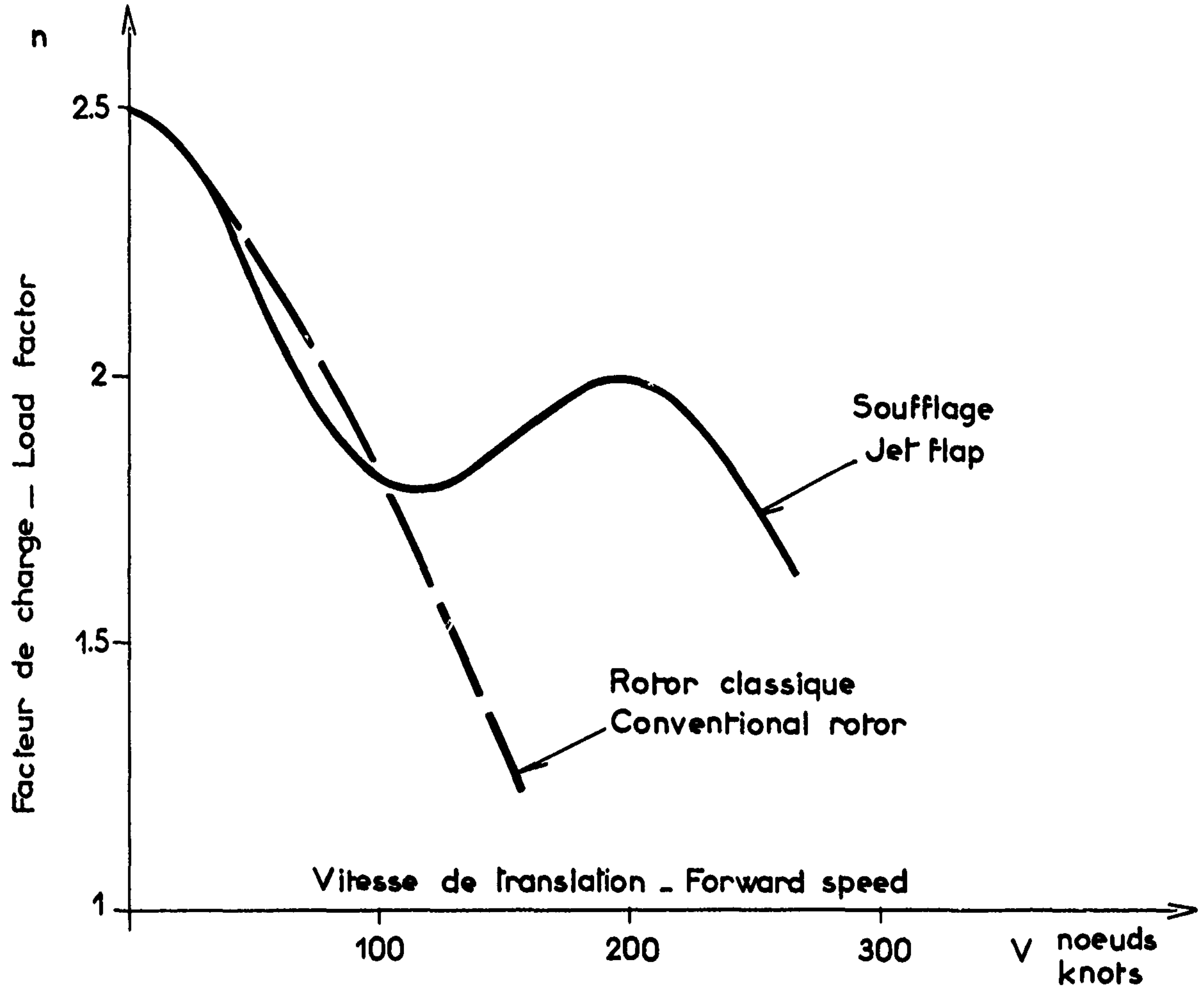


Fig.7 n-V curves
Variation du facteur de charge

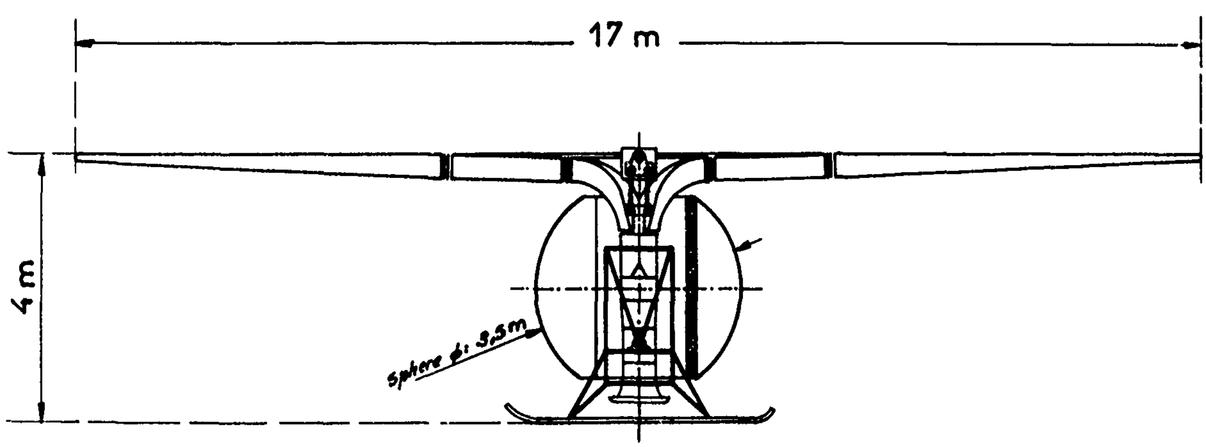


Fig.8 Radar surveillance platform
Plate-forme de surveillance radar

CHARGE UTILE	PAYLOAD	1700 kg
MASSE TOTALE	GROSS WEIGHT	5400 kg
MASSE A VIDE	EMPTY WEIGHT	1530 kg
NOMBRE DE PALES	N° OF BLADES	2
DIAMÈTRE DU ROTOR	ROTOR DIAMETER	17 m
CORDE	CHORD	0.9
PLÉNITUDE	SOLIDITY	6.75 %
PROPULSEUR	ENGINE	SNECMA / TURBOMÉCA : M 49
ALTITUDE DE FONCTIONNEMENT	ALTITUDE	5000 m
DURÉE DE LA MISSION	MISSION TIME	5 h

Fig.9 Radar surveillance platform
Plate-forme de surveillance radar

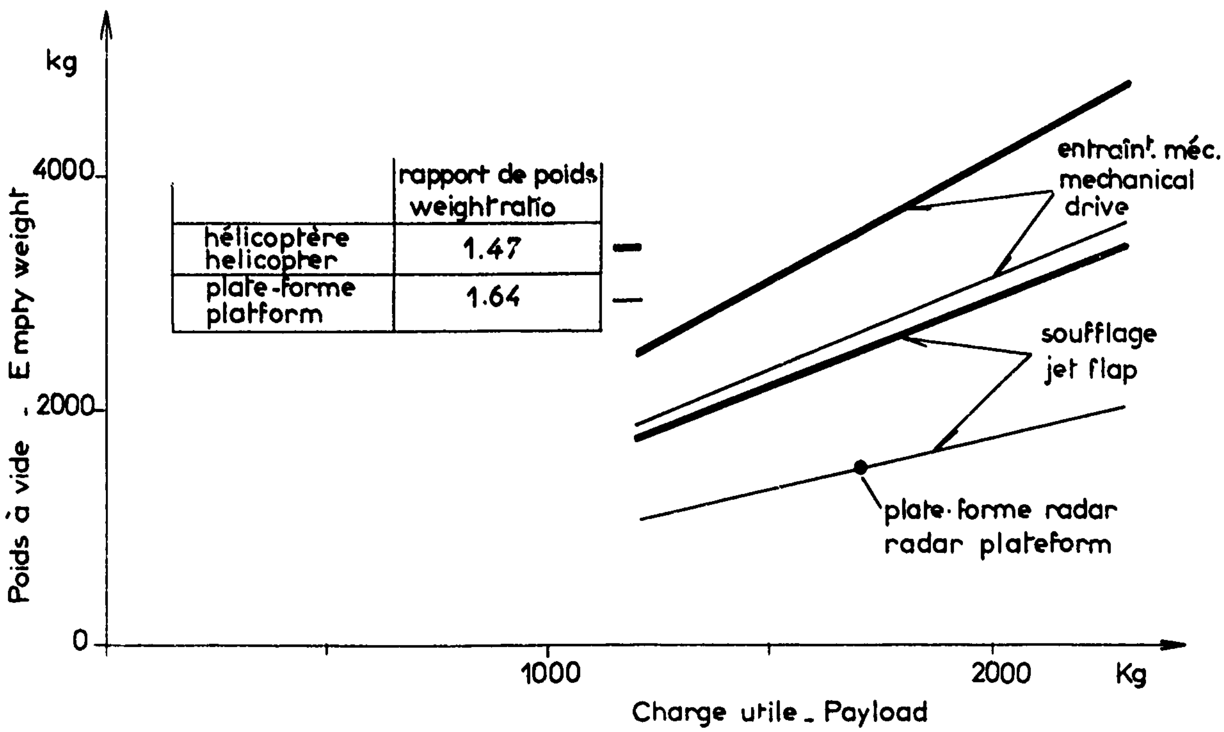


Fig.10 Comparison between shaft driven and jet-flap helicopters
Comparaison entre les hélicoptères mécaniques et a volets fluides

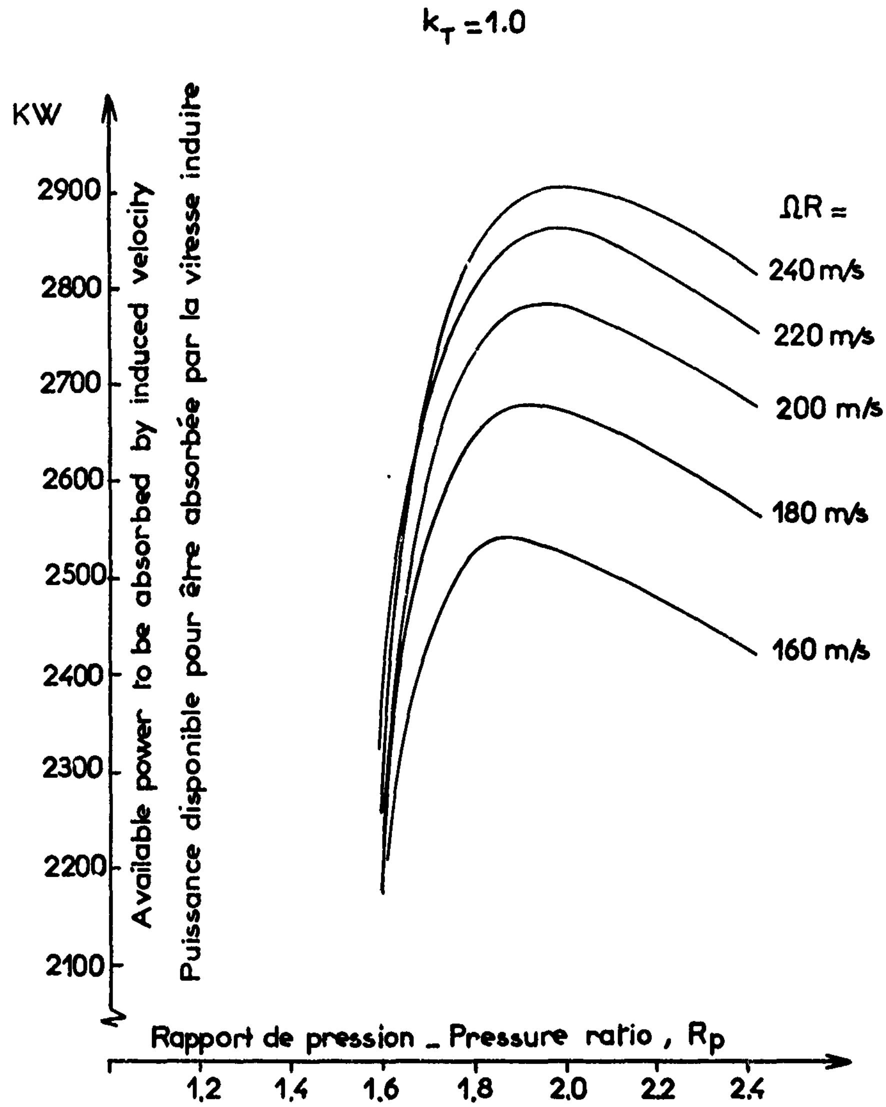


Fig.11 Pressure ratio optimization
Optimisation de rapport de pression

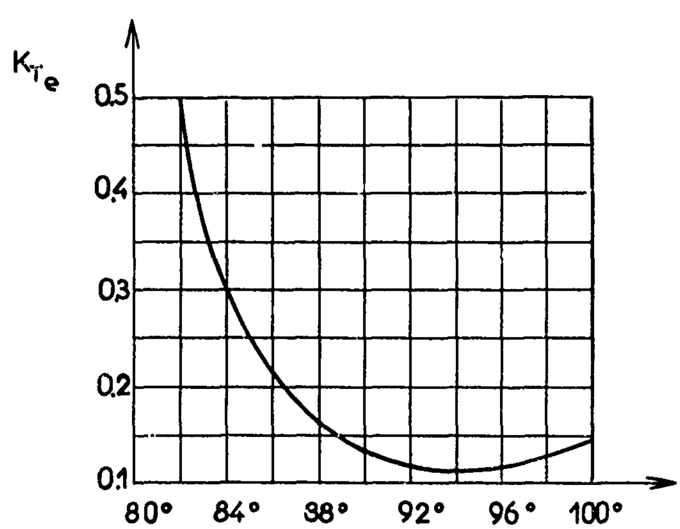


Fig.12 Sensitivity of K_{Te}
Sensibilité de K_{Te}

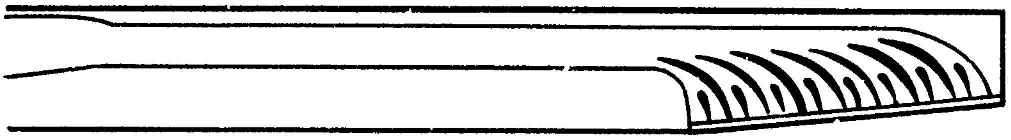
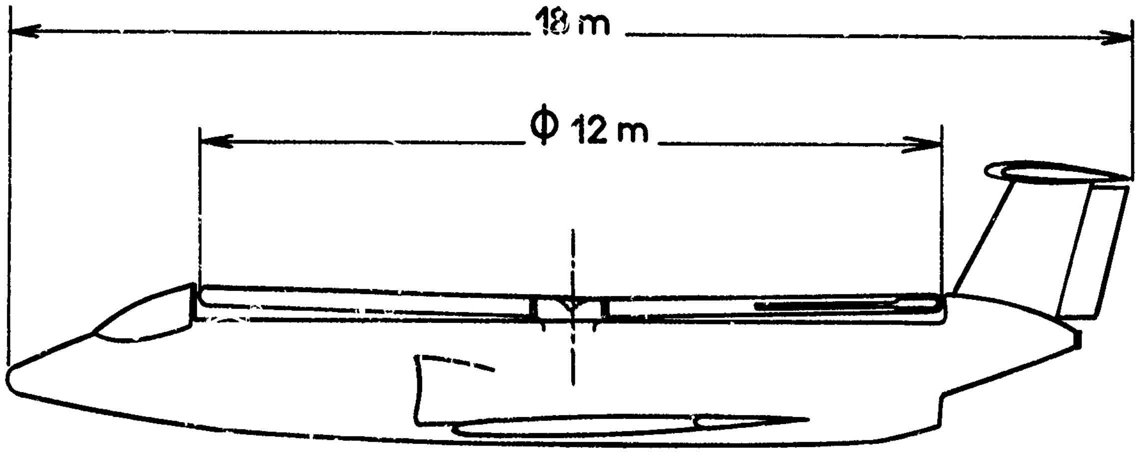


Fig.13 Cascade
Grille d'aubes



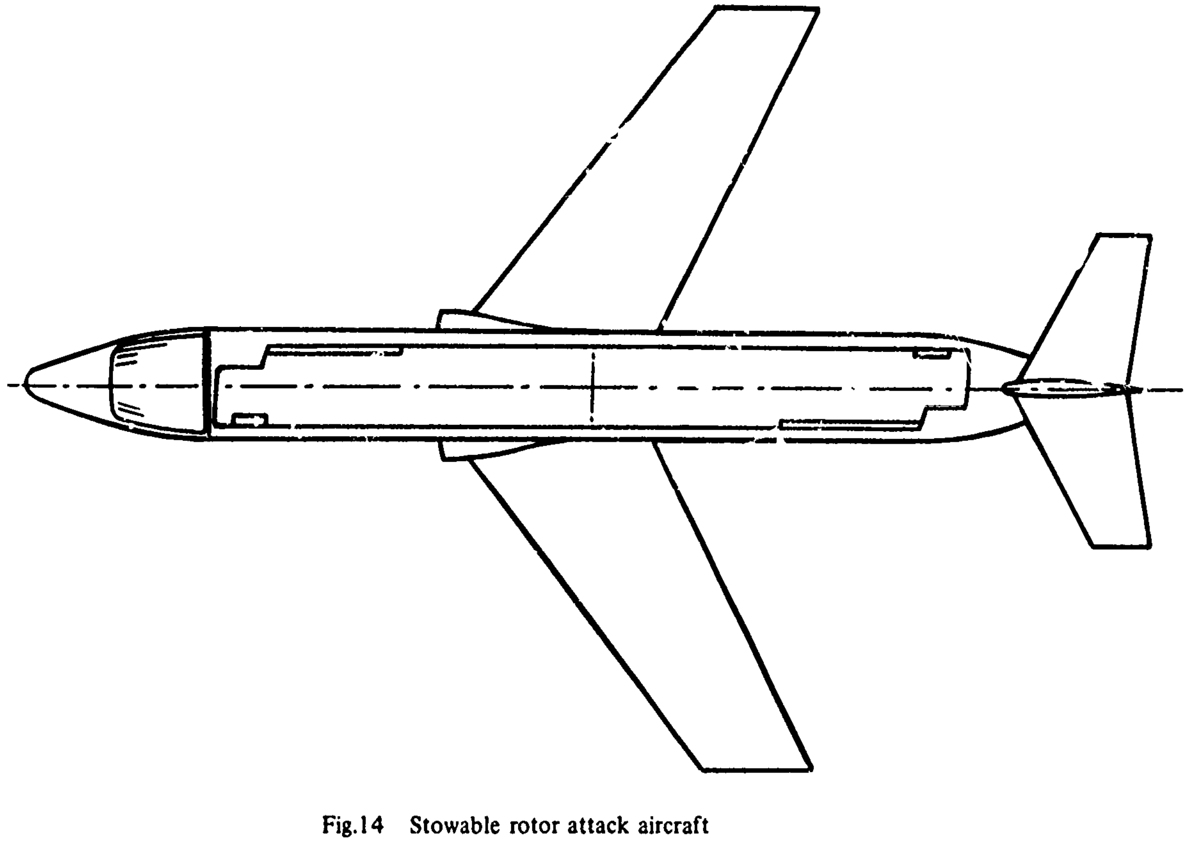


Fig.14 Stowable rotor attack aircraft
Avion d'appui tactique a rotor escamotable

RESEARCH AND DEVELOPMENT ON ROTORS WITH TIP REACTION DRIVE IN GERMANY

by

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SUMMARY

This paper reports upon the German activities on cold, hot and mixed cycle tip jet propulsion for rotors. Research and programs at Messerschmidt-Bölkow-Blohm on cold and large mixed cycle systems are described very briefly in order not to repeat what has already been said in former publications (see references). At Dornier studies and project work were directed to realization of a new rotor technology ducting hot gases of 700° centigrade through the rotor head to the blade tips with optimum efficiency. At VFW-Fokker the H3 conducts preliminary flight tests. This compound project shows a configuration, where the engine produces compressed air to drive the rotor in hovering and low speed flight. In cruise mechanically driven shrouded propellers provide the necessary thrust while the rotor autorotates.

For both projects some results of component testing respectively flight tests are discussed. Aspects of the flight mechanics as decoupling of movements in hovering and advantages of wide rpm-range are shown.

Concluding remarks on the operational applicability and new missions favoring torquefree rotor drive systems are added.

1. INTRODUCTION

Out of the total of today's flying helicopters more than 90% are of the single main rotor/tail rotor configuration.

One should appreciate the usefulness of the advanced types in operation and the ingenuity of mechanical and electronic engineering to their continous improvement. However, the small helicopter in particular is still extremely expensive compared to fixed wing aircraft of corresponding size. The large number of rotating parts cause high development cost and maintenance effort. The direct operating cost of a four seat helicopter amounts to approximately three times that of a four seat fixed wing aircraft. The rotor suffers from inherent instability in hovering flight and coupling of movements around all axes with the vertical in hover. Customers and engineers agree upon the desireability of improving vibration levels, noise levels and reliability of the complex machinery and many other features of present day helicopters.

As old as the helicopter is also the idea of reducing these problems by avoiding the need for anti-torque devices or at least their use to produce lift, thus providing a better ratio of useful load to gross weight, that is more economic use of the power installed. Mechanically complicated designs such as tandem twin rotor, intermeshing or coaxial types are still flying and proving themselves. However, the blade tip propelled rotor had only temporary success through the production of almost 200 Sud-Aviation Djinn two-seaters 15 years ago. Many other projects have been prototype tested and interrupted or abandoned.

In our opinion the main deficiency causing tip reaction projects to disappear was lack of available technology. This, for example, refers to the nonavailability of proper engines, suitable bonding materials and procedures, lack of experience in handling high temperature gases and ducting them through the rotor head to the blade tips.

2. BASIC REMARKS

One of the most discussed inherent handicaps of the blade tip propelled rotor is the poor propulsion efficiency. This is true also for the cold, mixed or hot cycle systems where the working gas is produced by an engine inside the fuselage and expanded at the blade tips thus using the rotor as the power turbine. Of tip jet drives these configurations seem to be the simplest ones and closest to become operationally realized. These systems are the subject of the following discussion.

We distinguish between the so called cold, mixed or hot cycle pneumatic rotor drive systems. For the cold cycle an engine driven compressor delivers pressurized air up to about 200° centigrade. In a mixed cycle low pressure cool air from the fan will be mixed with the hot exhaust gases from the turbine not yet fully expanded, temperatures ranging from 200° to 450° centigrade. Hot cycle drives use up to 750° hot gases coming directly from the pure gas generator part of the engine.

Looking at the properties of such propulsion systems compared to the conventional geared drive it can be said:

- Poor propulsion efficiency, high power installed required.
- Heavy weight rotorblades and rotor hub; large, thick blades need servo control.
- High rate of descent in autorotative flight and forward speed restricted due to aerodynamic drag of rotor and rotor head.
- Yaw control problems.
- High noise level of tip jets.

However:

- No gear boxes, shafting, clutches or anti-torque rotor required; minimum lubrication and oil cooling systems.
- Extremely wide rpm-range, favorable for storing kinetic energy and better adjustment to forward velocity, atmospheric and altitude conditions.
- Inherent antiicing of the rotor.
- Good hovering stability with complete decoupling of yaw and vertical control from pitch and roll motions.
- Good handling qualities in general.
- High ratio of useful load to gross weight.
- Reduced maintenance; minimum of rotating parts result in higher availability and reduced cost.

Of course this fragmentary list does not cover all the numerous aspects. It is necessary to carefully investigate how far disadvantages could be minimized, or suppressed, advantages optimized and contradictions eliminated. During the past decade several steps in this direction have been made in Germany by MBB (Messerschmidt-Bölkow-Blohm), VFW-Fokker and Dornier AG trying to use the benefits of modern technology and advanced turbine engines for tip reaction rotors.

3. PROGRAMS

All three Companies started off with cold cycle jet rotors. MBB tested a 4-m-diameter twobladed rotor, then went to mixed cycle systems for very large units. Fig. 1 shows a 31-m-diameter rotor producing 35 metric tons of thrust, driven by a General Electric GE CJ 805-23 which was an aft fan version of the CJ 805. This engine produced 160 to 180 m^3 of gas per second at a pressure ratio of 1,6 to 1,7 and a temperature of approximately 220° centigrade. Since no military requirements existed for such big cranes no funds were available for a follow-on program. There is sufficient literature available on these research projects.

Dornier in the early sixties developped a single seat helicopter based upon the cold cycle system. Compressed air of 125° centigrade was produced by a turbine driven compressor equipped with a controllable variable inlet vane. A number of fundamental studies directed at improvement of propulsion efficiency and technological possibilities led to an order to build three experimental Do 132 helicopters 2 1/2 years ago. This is a hot cycle five seat test vehicle which, dependent upon the results of the test program, would have allowed to derive a production version. A mock up is shown in fig. 2. First flight of the aircraft was planned for the spring of next year. However, the present financial squeeze interrupted the program at the beginning of this year. Do 132 data are given in Table 1.

Dornier concentrated on substantiation of a new technology to duct 700° centigrade hot gases through the rotor head to blade tips where they are expanded through cascade nozzles. To back up theoretical predictions a test stand rotor has been built very early in the program. The two year test program is almost finished. It's results show that the problems related to ducting hot gases have been mastered and that a significantly higher thrust is developped for a given blade angle and power than anticipated. This will be discussed later. Completed design work and analysis of parts show higher weight of the dynamic system than projected, but [illegible]rience and redesign will bring them back to target weights.

There is a developping market for unmanned tethered rotary wing drones where a derivative of the cold cycle Do 32 fits the requirements. Since the fuel can be fed to the drone via the cable from a ground station, the low propulsion efficiency of the system is no longer critical. This system brings the advantages of good stability in hover and particularly those of a torquefree rotordrive system. As a first step at Dornier a prototype Do 32 has been modified and equipped with remote control. The finalized system, the Dornier "Kiebitz" is illustrated in fig. 3. This project is now entering the development phase for an operational unit based upon military requirements with first deliveries planned for 1974/1975.

VFW-Fokker decided to investigate advanced configurations to avoid development of new technologies. Appreciating the undoubted trend towards higher speeds the system analysis resulted in a compound configuration based upon well known components. It incorporates a cold cycle rotor for hovering and low speed flight and a high efficiency mechanical power transmission to separate forward thrust producers for cruise and high speed flight, while the rotor autorotates. Basic philosophy is: A typical small helicopter needs about .2 hp per kg weight for hovering. This power permits a forward speed in the order of 130 kn. To raise the speed to 180 kn requires compounding and power installed of at least .4 hp/kg, see fig. 4. With such power installed a cold cycle drive can easily match today's standard of helicopter hovering performance. The benefits, simplicity, rpm-variability over a wide range and favorable handling qualities, can thus effectively be used. The high fuel consumption is unsignificant, because, according to statistics, only 5% of the total flight time of helicopters are flown in this regime. The main problem, hence, was to design a proper blade for high forward speed, autorotation and at the same time for tip jet propelled low speed flight.

A laminated 15% thick blade was built with duct area to airfoil section area ratio of .7. Initial whirl test stand trials were performed with a 6-m-diameter rotor and the same rotor was then tested on a single seat test bed H2. Positive results encouraged VFW-Fokker to plan their H-series compound family. First step was to realize the H3, fig. 5, a small, comparatively slow compound. Its basic data are listed in Table 1. A three bladed fully articulated rotor is used for producing lift, two shrouded propellers aside the fuselage center section provide forward thrust.

First flight was made in summer 1970, three years after commencing the program. At this time the selected engine was not yet available. Thus it flew with less than 70% design power. Maximum weight lifted vertically in a dynamic take-off was 800 kg. Correlation of rotor thrust predicted and measured was adequate with flight handling qualities as good as flown by the same pilot on the Sikorsky research flight simulator. The project also ended up with a higher empty weight than anticipated due to the aerodynamically refined airframe and the diverter gearbox between engine, compressor and fans. At the present the originally intended Allison 400 hp 250-C20 is installed. It is now hoped to soon complete the first test phase in a pure helicopter configuration with fans removed, fig. 6. Planned as production model is the H4, a five seat helicopter. Studies have also been made for a twin engine five seat compound, the H5, a mock up of which is shown in fig. 7. These projects suffer from the same financial problems as the Do 132 project.

4. SOME DETAIL RESULTS

Studies and experience at the three Companies have shown that optimisation of the numerous parameters influencing the overall propulsion efficiency of tip reaction drives lead to a best pressure ratio of 2,2 to 2,7, fig. 8. One of the main variables effecting this efficiency is blade chord and ratio of duct area to airfoil section area by a specific design.

A real surprise was the substantially higher thrust measured on the Do 132 rotor as compared to the prediction. After repeated recalibration of the test rig equipment the values were confirmed. Correction for ground effect and recirculation influence were again comprehensively investigated with refined theories and accompanying measurements in the surroundings of the rotor. This also could not explain the difference. A separate test program of this phenomenon was supported by the Federal Ministry of Defense. This program is not yet completed but some results can already be discussed.

Fig. 9 shows the thrust coefficient C_T versus rotor angular velocity omega, ω, see dashed lines, as computed by the usual means. The measured proportionality of C_T-increase and rotational speed matches the increasing nozzle efflux speed and flow. This favores the theory of some kind of tip vortex and therefore circulation control around the blade. It should be interesting to know if a similar effect was found with the XV-9 rotor by Hughes.

Anticipated efficiency was 32% for the H3 rotor, 36% to 38% for the Do 132 dividing net rotor power available by gas horsepower behind the gas generator. Conventional rotors show a comparable efficiency of 72% to 75%. With the aforementioned thrust increase the propulsion efficiency of the Dornier rotor climbs up to a good 50%. This means the rotor could easily lift a gross weight of 1 900 kg at an empty weight of 900 kg based on the actual weights of the Do 132.

Analysing the weight breakdown of the Do 132, fig. 10, it can be seen that the engine weight is 87 kg. The same PT6 with power turbine and shaft gearbox has a weight of 131 kg, that is 50% more. Also the gas generator alone is 20% less in price. The entire dynamic system weighs 20% of the present (strcturally limited) gross weight which compares to an average of 25% for the geared helicopter. This together with the lack of gearboxes, tail rotor and shafts and the corresponding decrease in maintenance and cost should more than compensate for the more complicated rotor and rotorhead construction.

Characteristic internal stresses, gas ducts, blade tip nozzles and corresponding blade nose balance weights in all the pneumatic rotors resulted in a blade heavier than usual, with lock numbers 2/5 to 3/5 of conventional shaft driven blades. This almost suppresses higher harmonics in forward flight thus reducing flapping and lagging motion. Even small hinge offset as in the H3 gives excellent control response and c.g. range. Hovering stability is positively affected not only by the rotor mass but also by the high rpm possible.

A time history of pitch movement is shown in fig. 11 for both the Do 132 with a teetering but heavier rotor and the H3 with a 2% flapping hinge offset after a step input. Flight test confirmed the predicted favorable behavior.

Simulator results indicate the decoupling of longitudinal, yaw and vertical motions. However, yaw control in hovering is still a problem as long as Mil-H-8501 requirements have to be met. Both projects work with yaw nozzles at the tail steadily consuming bleed gas from the main rotor supply, thrust being vectored according to the pedal deflection. The H3 adds additional torque by also vectoring the residual thrust of the engine which reduces power required for yaw control to 3%. For the Do 132 the 5% power drain is increased to 10% for higher yaw rates when one of the pedals is put to the stop. Here the simulator runs show only minimum coupling to the vertical movement for a short full control step input due to the high inertia of the rotor. However, this slight coupling occurs only when the engine is running at maximum power. Otherwise the rpm governor will automatically adjust the engine power to the additional bleed. Both types meet AGARD 577 requirements for yaw control of VTOL aircraft.

In forward flight the fixed wing aircraft type tail surfaces provide good directional and longitudinal stability and more than sufficient yaw control.

Now a few words regarding the available rpm-range. As already experienced with the Do 32 helicopter the high rotor inertia and wide rpm-range reduce the critical area in the height-velocity diagram - the so called dead man's zone - considerably. This property was intentionally developped in the later designs of VFW-Fokker and Dornier AG.

For the geared rotor drive engine torque is an essential parameter. The rotor should be governed at a definite rotational speed as accurately as possible in order to draw the maximum power out of the engine and transmit it to the rotor by mechanical means. Even with the flexible turbine engine of today the shaft torque has to be kept below a certain limit. Thus a speed drop of the rotor also means corresponding loss of power available from the engine at lower turbine speed. For the pneumatic drive, however, even at zero rotor speed the engine can be put to full power, producing a substantial thrust at the blade tips and thus accelerating the rotor very fast to operating speed. Fig. 12 illustrates this for the H3. Power required and effective power available at the rotor match for a distinct engine power setting and given weight. For example at 750 kg, a tip speed of 130 m/s and 13° blade pitch as well as at 205 m/s and 6,3° blade pitch. Of course the power required curves - solid line - remain the same if a shaft drive would be applied. However the power available drop, due to shaft torque limitation and increasing power losses in the free turbine, would follow the dotted line.

For safety considerations the pilot will start for example hovering at a high rotorspeed with low pitch setting. By rapidely pulling the collective to a 13° position and leaving it there he can perform a dynamic take-off coming to a hover again at a higher altitude. Thrust for a short time almost doubles. Based upon the restricted preliminary flight testing of the H3 a dynamic altitude difference in the order of 50 m should be possible. Moving the collective slowly down again the rotor will accelerate without loosing thrust as seen from the small available excess power - shaded area.

In summary the storage of kinetic energy not only reduces the critical height-velocitiy area, but also allows the unloading of the pilot from constant rpm monitoring. Safe excess power is available to climb out or accelerate to forward speed very fast under adverse conditions. Similarly for autorotational landings safety is added and pilot skill requirement lowered.

The slope of the power available curve versus tip speed is approximately proportional to the propulsion efficiency of the blade tip nozzle somewhat flattened by the lower secondary losses at lower speed. For the Do 132 the slope of the power available curve is slightly steeper resulting in a smaller rpm-range. However, this is compensated for by the higher mass inertia of it's rotor. In the Do 132 an additional speed governor has been developped to make it still more comfortable for the pilot. Inputs for this governor are rotor speed and exhaust temperature of the gas generator. A PID amplifier adjusts the fuel control according to the pilots rpm setting over an electro-pneumatic servo. The pilot needs only two power settings and has the choice of 405 rpm for dynamic take-off or 325 rpm for optimum hover and forward flight. Similarly the H3 pilot at the present can use an rpm range from 300 to 425, and an optimum hovering rpm of about 360 with 280 to 480 rpm available with the 400 hp engine.

As far as future compound configurations are concerned the hot cycle rotor drive offers a very easy and highly effective method for smooth power distribution. Power can be put either directly to the rotor or partially or completely to a turbine driving fans, shrouded or regular propellers etc. Advanced design studies directed towards a rotary wing recovery device for drone aircraft have shown that for a hot cycle rotor a blade with 15% thickness is feasible. Together with the results of the diverter valve studies it paves the way for a very uncomplicated, safe, lightweight and effective compound helicopter propulsion system. VFW-Fokker built a diverter gearbox for this purpose and also had the transition phase investigated by Sikorsky on their research flight simulator. A smooth and easy transition procedure was obtained. Fig. 13 shows one of the computer plotted simulator flights. With less forward acceleration and unchanged stick position the 30 ft altitude loss is avoidable.

5. MISSION CAPABILITY

Evaluating the missions for which the pneumatic rotor drive is best suited, work done in the recent years permits to state that it's applications are not limited to heavy crane helicopters. For small helicopters the hot cycle, and if a 10% to 20% efficiency improvement is possible also the cold cycle, could supplement not replace the geared helicopters. Furthermore in commercial application a 15% to 20% cost reduction favores the concept, since the higher fuel consumption has a minimum effect on direct operating cost, fig. 14. A 50% higher fuel consumption would raise the D.O.C. by only about 3% or less. A 10% reduction in initial cost reduces direct operating cost by at least 20%. And this amount of cost reduction is possible today, with our present experience and available technology.

For larger helicopters for example designed for airline commuter service, the annual utilisation has to be so high that the higher fuel consumption forms a much bigger part of D.O.C. Thus in this case it becomes a matter of weighing the secondary advantages against this increase in D.O.C. Military use may also be somewhat hampered by the fact that usually the airframe plus engine is only part of the system which proportionately cuts the price advantage. However, the flight handling qualities and maintenance aspects as well as a probable higher availability and easier training might favor a tip reaction system. Finally it has to be taken into account that new applications of rotary wing aircraft in general arise with new systems. Therefore this brief survey should end with a glance to the Dornier drone helicopters.

6. NEW MISSIONS

As mentioned previously the present development of an unmanned remotely controlled tethered rotor platform was derivated from the Do 32 helicopter, fig. 15. In 1964 the Do 32 U, fig. 16, flew for the first time. Here the pilot was replaced by a couple of black boxes. Successful tests resulted in the final concept "Kiebitz" as shown in fig. 3 before. The basic idea consists in stabilizing highly sensitive sensors above ground or sea for a longer period of time. Horizontal air mobility which would have permitted to fly over enemy territory was deliberately abandoned in order to keep the rotor platform outside the range of fire of hostile ground troops or ships. A mobile ground station allows the rotor platform to be quickly displaced.

Airborne platform and ground station are connected by a cable. An expensive flight governing and guidance system is therefore not required. The tether, fig. 17, contains the fuel line as well as the wires for control and sensor signals to be transmitted from the ground station to the platform or vice versa.

Fig. 18 demonstrates the simplicity of the operational "Kiebitz" due to the torquefree cold cycle rotor drive system. A welded framework is devided into two compartments by a fire wall, faired with an easily removable nonstructural housing. The upper compartment contains controls, governor and systems as well as internal components of sensors. Below the horizontally arranged fire wall the engine with accessories and the yaw nozzle are located. Air intake is arranged around the rotor hub to separate it as far as possible from the exhaust gases, at the same time cooling the electronic compartment. Sufficient cooling of engine oil is obtained by a tube cooler wound around the engine air inlet. Sensors usually will be attached to brackets at the outside.

Table 2 shows a summary of missions and sensors for an elevated platform. The five principal missions are surveillance and reconnaissance, communication, fire control, ECM and navigation. The corresponding sensors are based on optical, electromagnetic or acoustical means. Fig. 19 indicates that most of the mission requirements can be met with a small rotor platform now under development. The performance is summarized in table 3 according to specifications of the customer. For the heavier payloads a similar device is under consideration. It is called Argus and is equipped with the dynamic system of the Do 132. It carries up to 500 kg payload or will obtain - with reduced payload - an altitude of 1 000 m above ground station.

Concluding this brief survey it might be mentioned that the development work including the hardware and test programs as well as the application of the pneumatic rotor drive to tethered platforms was done with a total budget of about 42 million German Marks. To us it seems to be well invested money not only looking at today's achievements, but also into future applications: For high speed rotary wing designs - even stowed rotors - the pneumatic rotor drive offers good cost effectiveness and variable rotor speed without the usual mechanical problems. It offers less empty weight at least for big cranes and compounds and eventually a simpler retraction mechanism due to the lack of a mechanical transmission.

References:

1. U. Hütter : " The Problems of Rotor Drive and the Feasibility of Using Very Large Rotors with Tip Jet Drives ". AGARD CP no. 7, Jan. 1966.

2. G. Reichert and E. Weiland: " Low Pressure Reaction Drive System for Large Helicopter Rotors". AGARD CP no. 31, June 1968.

3. G. Kannamüller: " Efficiency Improvement of Helicopter Tip-drive Systems". AGARD CP no. 31, June 1968.

4. K. Pfleiderer: " Review of Bölkow Activities in the Rotorcraft Field" The Aeronautical Journal, Volume 74, no. 716, August 1970.

5. Chr. Fischer: " VFW's H3E ' Sprinter ' ". Vertiflite, Vol.16, no. 7, July 1970.

	DO 132	VFW H3
Number of seats	6	3
Power installed	710 gas HP	400 shaft HP
Rotor drive	pneumatic hot cycle	pneumatic cold cyc.
Fan drive		mechanical
<u>Dimensions</u>		
Rotor diameter	35,2 ft	28,6 ft
Overall length	24,0 ft	30,5 ft
<u>Main rotor</u>		
Number of blades	2	3
Blade airfoil section	NACA 63_4021	NACA 23015
Blade twist	$-5,9^o$	0^o
Permissable rpm range	310 to 420	280 to 480
Rotor disc loading	3,26 lb/sq.ft.	3,26 lb/sq.ft.
<u>Weight</u>		
Empty weight	1 490 lbs	1 075 lb
Takeoff weight	3 150 lbs	2 125 lb.
Payload	1 020 lbs	590 lb.
<u>Flight performances</u> (ISA, sea level, max.gross weight)		
Normal cruising speed	116 kn.	130 kn.
Rate of climb, vertical	680 ft./min.	390 ft./min.
Best rate of climb	1 440 ft./min.	1 280 ft./min
Range (10 min reserve)	220 nm	270 nm

Table 1 Technical datas of DO 132 / VFW H3

MISSION / SENSOR	SURVEILLANCE RECONNAISSANCE	COMMUNICATION	FIRE CONTROL	ECM	NAVIGATION
OPTICAL	• TV CAMERA • LLTV		• BEAM RIDER (LASER) • IR TRACKER		
ELECTRO-MAGNETIC	• RADAR (GROUND, SEA, AIR) • DIRECTION FINDER	• OMNIDIRECTIONAL ANTENNA • DIRECTIONAL ANTENNA • LOW FREQUENCY ANTENNA	• RADAR TRACKER	• ECM RADAR • RF JAMMER	• BEACON • TRANSPONDER
ACCOUSTICAL	• DIRECTION FINDER				

___ UNDER DEVELOPMENT - - - IN PREPARATION

Table 2 Summary of missions and sensors for an elevated platform

- ▲ Flight altitude of up to 300 m above ground
- ▲ Continuous operation for 24 hours
- ▲ Readiness for operation at 300 m above ground within 8 minutes
- ▲ Recovery and readiness for transport within 5 minutes
- ▲ Capability of operation at wind velocities of up to 14 m/sec ± 8 m/sec gust
- ▲ Ground-based equipment compiled to a set on a cross-country vehicle

Table 3 Requirements for a tethered rotor platform

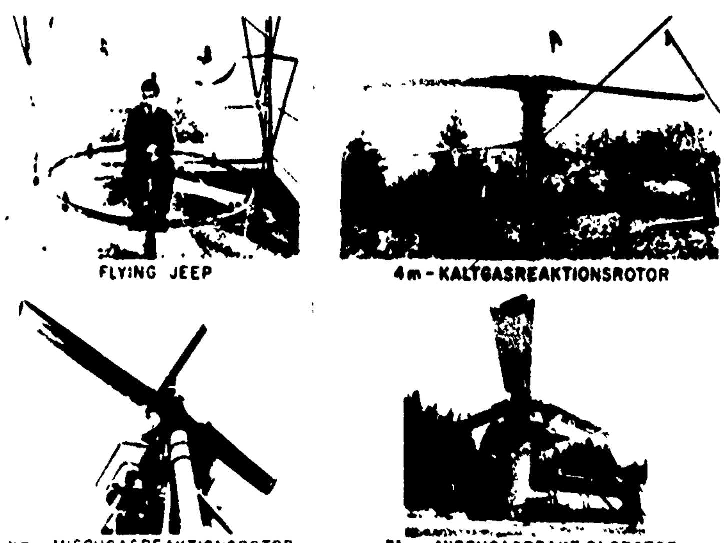

Fig. 1 Pneumatic rotors build and tested by MBB

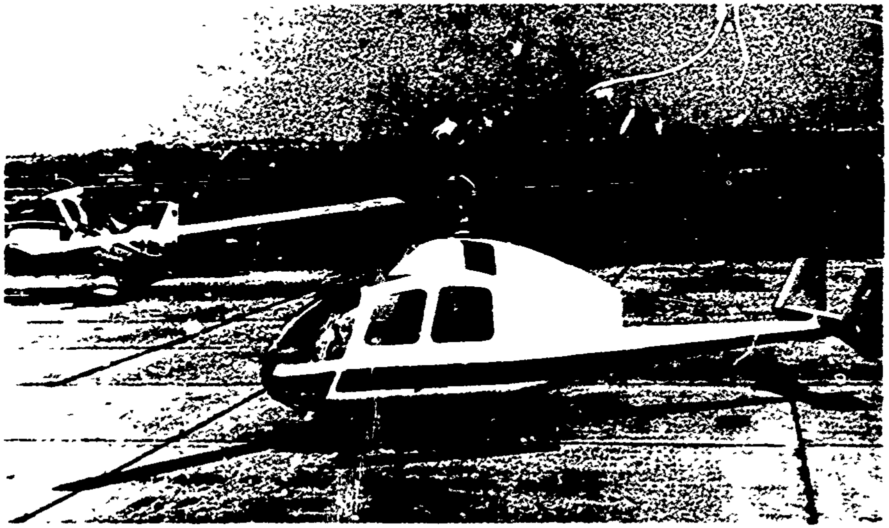

Fig. 2 DO 132 mock - up

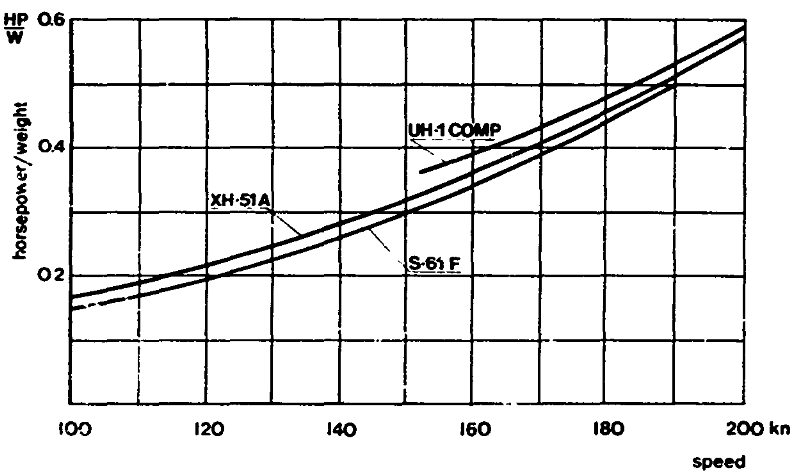

Fig. 4 Shaft horsepower to weight ratio of compounds

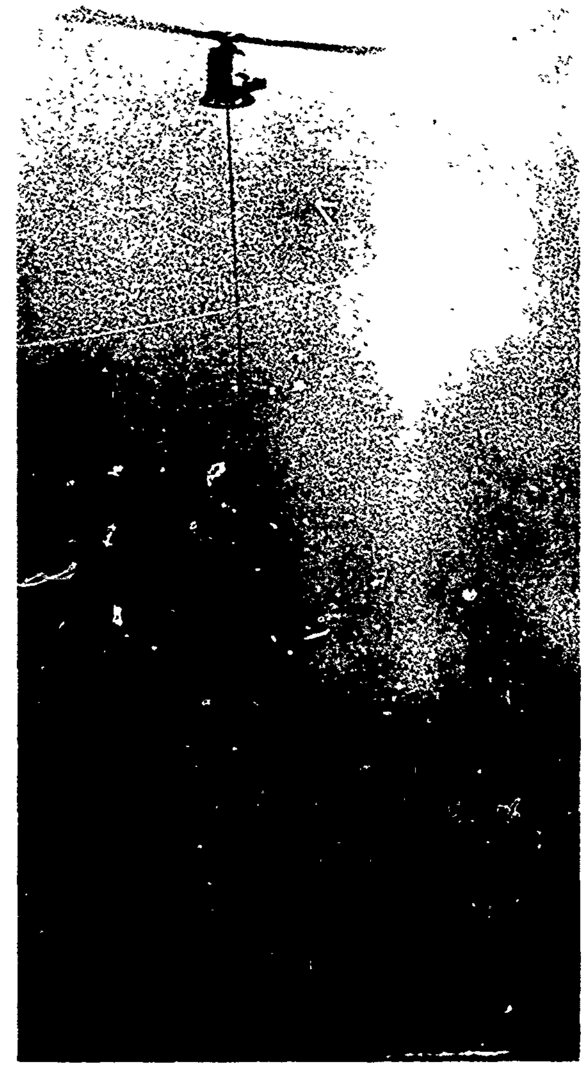

Fig. 3 Tethered rotor platform DORNIER „Kiebitz"

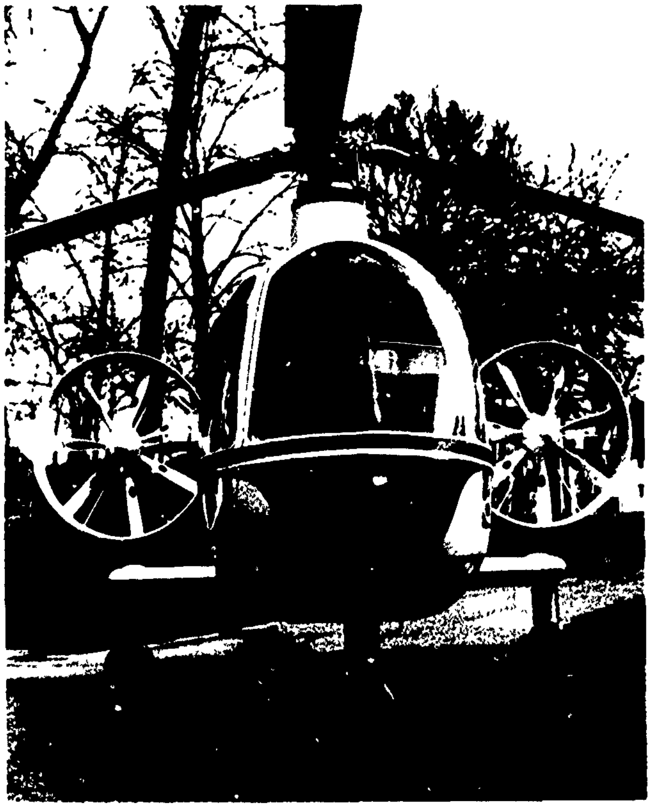

Fig. 5 VFW H3 small compound prototype

Fig. 6 VFW H3 during initial test flight

Fig. 7 VFW H5 mock-up

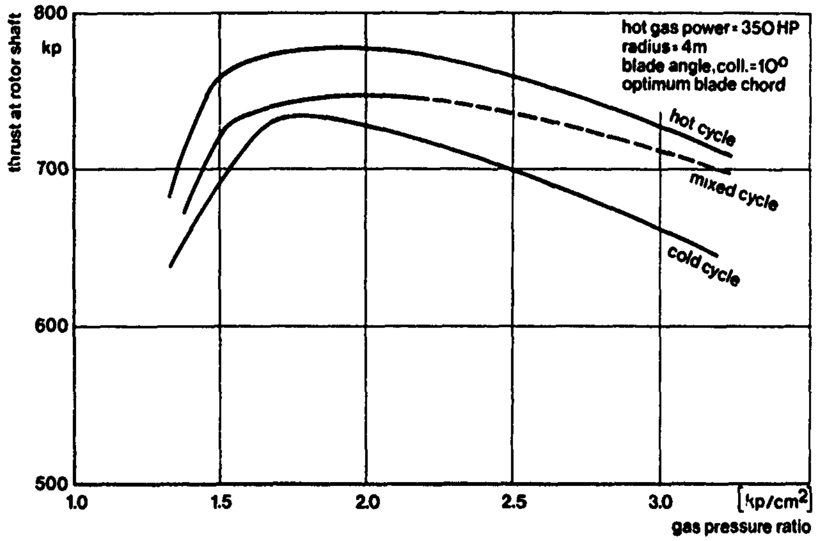

Fig. 8 Comparison of cold, mixed, hot cycle tip jet drives

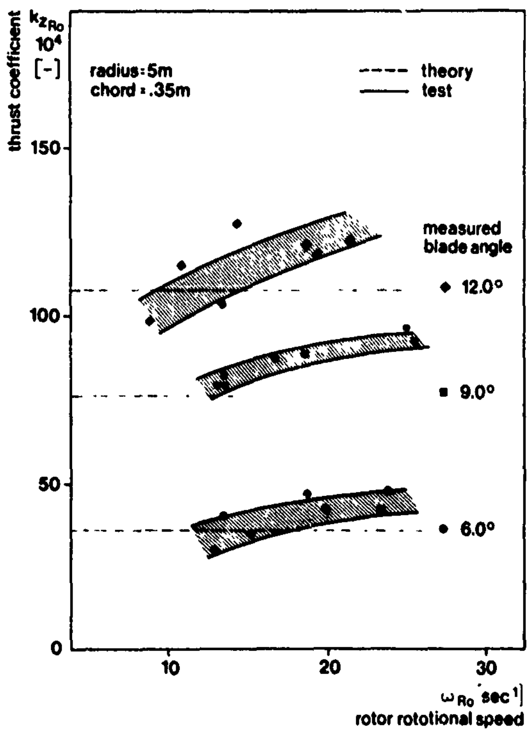

Fig. 9 DO 132 hot gas rotor thrust measurements

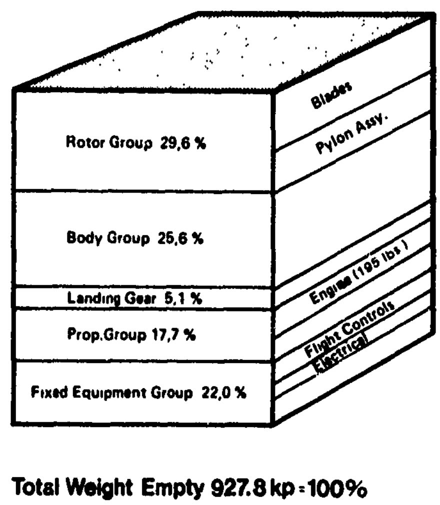

Fig. 10 Weight breakdown of DO 132

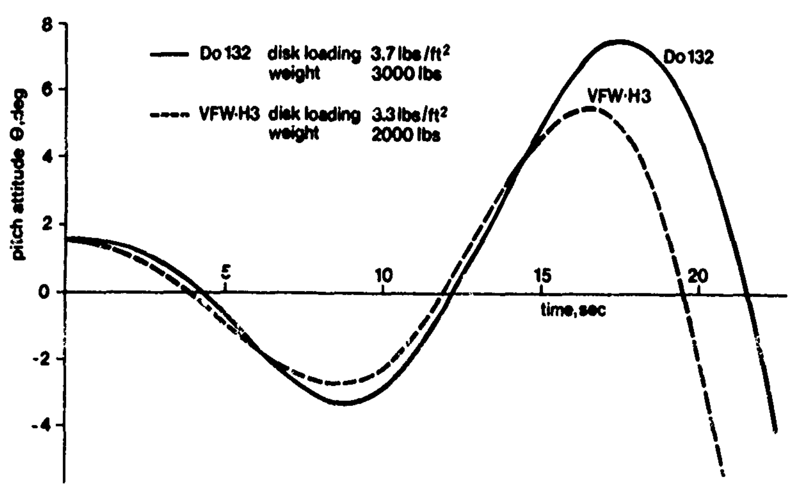

Fig. 11 Time history of longitudinal attitude

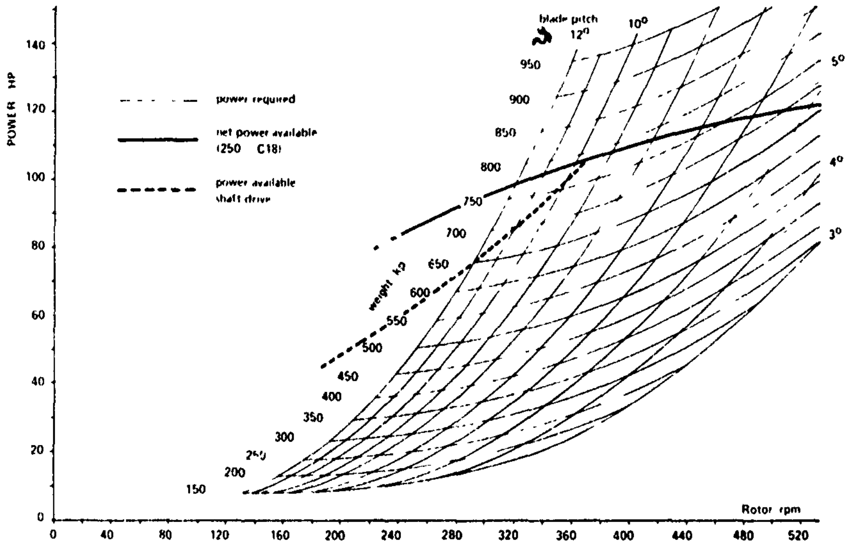

Fig. 12 VFW H3 power required/power available versus rotor rpm

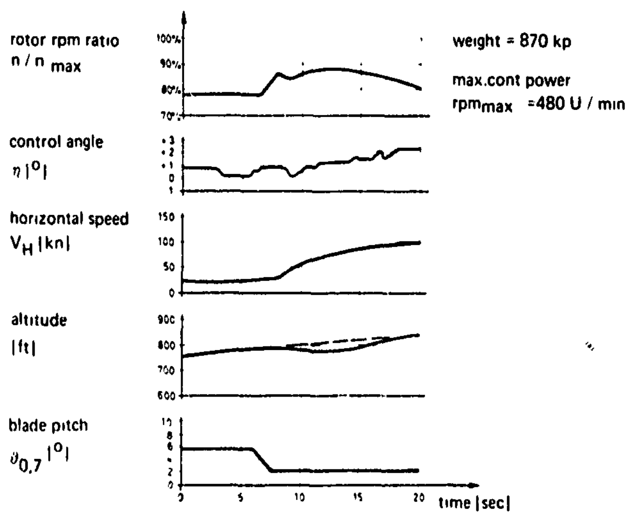

Fig. 13 Printed simulator flight, forward transition for H3

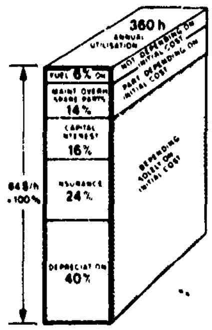

Fig. 14 Break-down of direct operating cost / hour for a conventional produktion helicopter

Fig. 15 DO 32 single seat cold cycle helicopter

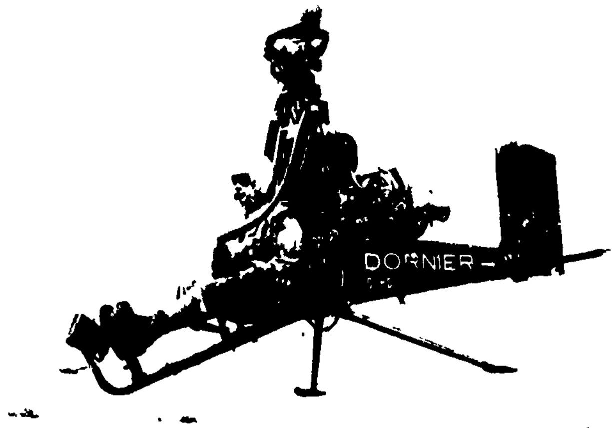

Fig. 16 DO 32 - U tethered experimental drone

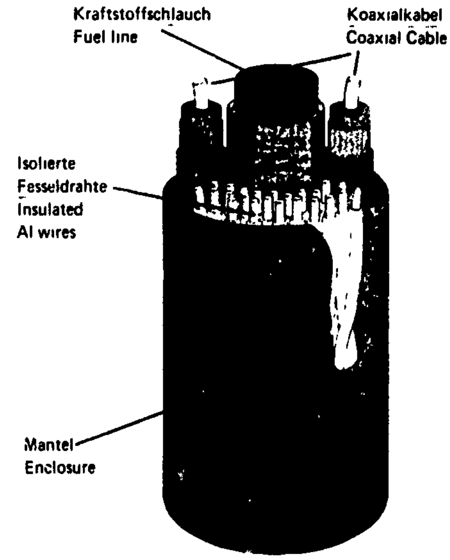

Fig. 17 Tether cable

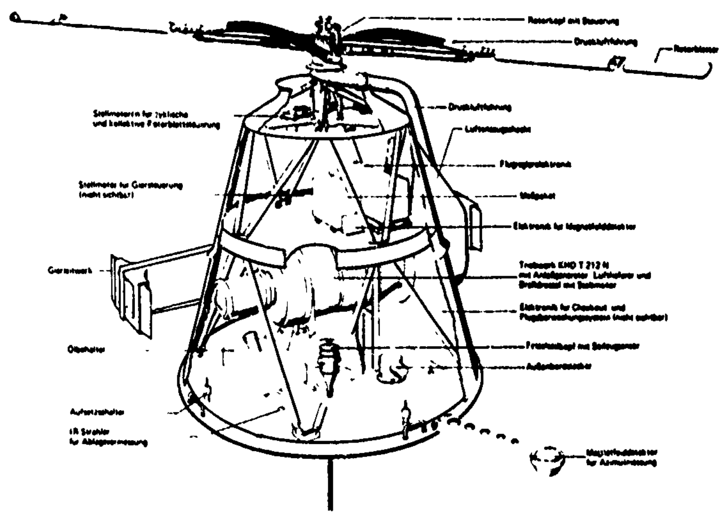

Fig. 18 Operational DORNIER „Kiebitz" design

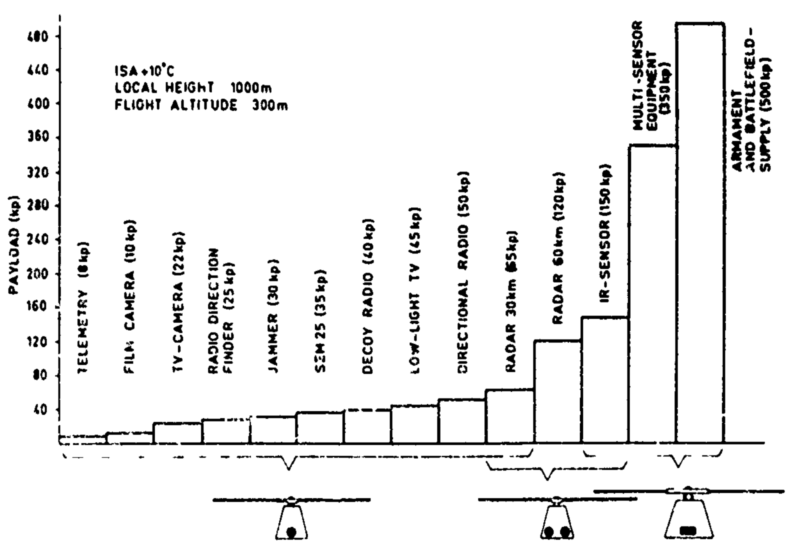

Fig. 19 Payload spectrum

SURVEY OF TILT ROTOR TECHNOLOGY DEVELOPMENT

by

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SUMMARY

A review is made of the development of tilt rotor technology since the XV-3 program in the late 1950's.

A brief comparison of the capabilities of the tilt rotor with other rotary wing configurations for a transport mission is shown. Tilt rotor performance and dynamic model tests are described. Analytical methodology development is reviewed and predictions are shown to compare well with model test data in the areas of performance, aeroelastic stability and flying qualities. It is concluded that the technology is now in hand to develop a prototype vehicle.

INTRODUCTION

This paper presents a review of the state-of-the-art of tilt rotor technology. While the data presented has been developed primarily from Boeing programs in this field, it is presented against a background of other work, particularly by NASA and Bell, in order to provide an overall perspective.

The feasibility of the tilt rotor concept was demonstrated in the late 1950's by the Bell XV-3 shown in Figure 1. This was basically a test bed aircraft and while it successfully performed its purpose of substantiating feasibility, it also pointed up some fundamental technical problems of the tilt rotor configuration, especially in the dynamics area (References 1, 2).

Boeing started major tilt rotor development in 1966 as a result of a study which examined many potential low disc loading configurations in applications for next generation military transport missions. One output of this study is shown in Figure 2. This figure shows the effect of design cruise speed on design gross weight for various configurations, all having equal payload, radius and hover capabilities. The helicopter, with or without wings, runs out of propulsive force around 200 knots. This can be extended to 250-300 knots by compounding but at a weight penalty of about 20%. The study also showed that the power required for a 250 knot compound would be from 50 to 100% greater than that of a 180-200 knot helicopter. The tilt rotor offers speeds of 300-350 knots with a weight penalty less than that of the compound and with the same power as the helicopter. At the expense of some additional weight and power, the tilt rotor has speed potential in excess of 400 knots. Many subsequent tilt rotor application studies have confirmed the benefits of the configuration for many missions, both military and civil.

Technology Development

This attractive performance potential caused Boeing to initiate an extensive program of tilt rotor technology development. New analytical tools had to be developed. A good background of analytical methodology was available from helicopters and tilt wings, particularly in the areas of rotor performance and loads. However, this still needed modification to deal with the special cases of high twisted flexible rotors used on the tilt rotor configuration. The biggest task in the analytical development was in the dynamics area where the large flexible rotors mounted at the tips of relatively flexible wings presented more potential problems in aeroelastic stability both of individual blades and of coupled propeller/nacelle/wing systems than are experienced either by a helicopter or a propeller-driven airplane.

Model Testing

All of this new methodology had to be substantiated by model tests. In the last five years Boeing has completed over 3,500 hours of tilt rotor model testing. Because so much of the work was in the dynamics area, this also required development of the technology of building and testing dynamically-scaled models, both powered and unpowered.

5-1/2 foot diameter semi-span model which Boeing has recently used for examination of the aeroelastic characteristics of the coupled wing/rotor system. This model uses a hingeless rotor with inplane frequency of about .75/rev. After demonstrating freedom from aeroelastic instabilities when mounted on the nominal stiffness wing, a special soft spar was built for this model to examine stability boundaries. During these tests the data shown in Figure 8 was developed indicating good agreement between predicted and measured instability boundaries.

A tilt rotor, like a helicopter, is susceptible to mechanical instability or ground resonance. Mounting a rotor on a flexible wing also results in the possibility of mechanical instability in the air or air resonance. Ground resonance has been extensively investigated for helicopters and extension of the analytical methodology to include airframe flexibility gives excellent prediction of the boundaries of air resonance in the cruise mode as shown in Figure 9, which shows the decay of modal damping with increase in rpm for a semi-span dynamically-scaled wing and rotor.

Design studies have indicated no problem in selecting wing and rotor characteristics which will keep instability boundaries well removed from operating conditions without paying any substantial weight penalty. This is shown in Figure 10 which depicts the predicted aeroelastic stability boundaries for a 12,000 lb. gross weight tilt rotor aircraft as compared to the operating flight envelope. The wing used in this study was designed from strength considerations only and did not have to be modified to provide the stability characteristics shown.

Flying Qualities and Gust Sensitivity

The blades of a tilt rotor aircraft in the cruise mode are lightly loaded and operate at very low section angles of attack. Changes in rotor attitude and velocity can therefore make large changes to local section angles of attack and, therefore, to total rotor loads. The rotor static derivatives therefore become a major component of the total aircraft derivatives and can produce large destabilizing moments. The variation of the rotor static derivatives with blade flapping frequency is shown in Figure 11. This effect presents no surprise and follows directly from the increased flapping motion of the more flexible blades. Less well known is the effect of lag frequency on the rotor derivatives shown in Figure 12.

As the lag frequency is reduced, the pitching moment derivative reduces sharply and actually changes sign and becomes a stabilizing moment on the airplane. This effect is large because of the fact that in the cruise mode at high collective pitch settings blade motion in the disc plane produces large changes in local blade section angle of attack, unlike a helicopter where inplane motion of the blades produces only small velocity changes and substantially no change in airfoil angle of attack. As shown in the figure, a soft inplane rotor having a natural frequency of about .75/rev produces lower forces and moments than either a rigid propeller or gimballed rotor (inplane frequencies 1.3 to 2.5 per rev) or an articulated rotor (inplane frequencies of .25 to .3 per rev). Substantiation of this predicted variation was obtained from the semi-span dynamic model discussed earlier in this paper. By varying the rpm the inplane frequency could be varied from about .75/rev up to more than 1/rev and the variation in pitching moment derivative predicted and measured as shown in Figure 13. Without lag accountability the pitching moment predicted would increase with decreasing rpm due to the increase in advance ratio. Thus, the rotor dynamics can be used within limits permitted by blade load and aeroelastic stability considerations to improve the flying qualities of the airplane.

However, it still remains clear that because of the large lightly loaded rotors a tilt rotor will tend to have a higher gust sensitivity than a conventional propeller-driven airplane with the same wing characteristics. Since, however, the propellers incorporate cyclic pitch control, the capability is available to use this cyclic pitch as a feedback control system in order to alleviate gust effects. Analyses and wind tunnel tests have confirmed the effectiveness of such a system.

The gust sensitivity of the tilt rotor applies also to longitudinal or axial gusts. The reason, again, is the low mean blade section angle of attack in the cruise mode, resulting in large percentage changes in thrust for a given change in axial velocity. This effect was noted on the XV-3 where pilots observed both a galloping and a yawing tendency in rough air at high forward speeds. Here, again, as in the case of inplane gusts, a major improvement in airplane response can be obtained by use of a feedback control system into collective pitch. The extent to which response to a gust can be reduced is shown in Figure 14. It can be seen that the response to a longitudinal gust can be almost eliminated by suitable feedback.

Summary of Technology Development

In summary, we can say that tilt rotor performance is predictable by available analytical methods which substantiate the 100-150 knot speed improvement over a helicopter without increase in power. Developments in aeroelastic analysis over the past five years now permit prediction of stability boundaries and are well substantiated by dynamic model tests. Freedom from aeroelastic instability can be readily achieved throughout the flight envelope with little or no penalty in weight or complexity. Flying qualities are predictable and by taking advantage of potential rotor dynamics and by use of feedback in the rotor collective and cyclic pitch control system, can be made as good or better than those of a fixed wing airplane in the cruise mode or a helicopter in the hover and transition modes.

Other technology areas, which it has not been possible to cover in this brief review, are also well in hand.

Noise, a major obstacle to introduction of many V/STOL and STOL concepts, is not a problem for the tilt rotor. In hover, the high blade twist results in a blade load distribution which is higher inboard and lower outboard than a helicopter, resulting in lower noise levels. A tilt rotor has no difficulty in meeting proposed noise standards such as 95 PNDB at 500 feet distance in hover. The use of low cruise rpm (typically about 70% of hover rpm) to improve propulsive efficiency also results in very low noise levels in cruise.

Vibration, a major helicopter problem ever since the first helicopters flew, is greatly reduced in a tilt rotor. The rotor never operates in the high tangential flow field associated with high speeds in the helicopter mode and, in all regimes of flight, the wing of the tilt rotor is an effective rotor isolation system, providing large attenuation of any rotor excitations before they reach the fuselage.

Overall it can be concluded that tilt rotor technology has been developed to the state where industry could go ahead with an operational tilt rotor aircraft with high confidence. This same conclusion was drawn by Bell in Reference 6.

So now that we have this technology, what are we going to do with it?

Need for Flight Demonstration

Both commercial and military operations can benefit substantially from the realization of the unique characteristics of this configuration, combining the hovering efficiency and flying qualities of the helicopter with the cruise efficiency and flying qualities of the fixed wing airplane.

The risk involved in introducing any new aircraft system, even of a conventional concept, is now so high in terms of cost that the U.S. Department of Defense has instituted a "Fly before you buy" philosophy. Clearly, the risk of starting a program aimed at the introduction of a production aircraft with a new configuration incorporating new technology represents a still higher risk even when the technology has been as well substantiated through analyses and model tests as has the tilt rotor. The probability, therefore, that the tilt rotor could be developed in one shot for a specific mission starting from today's state of technology is very poor. If the technology we have developed is ever to be applied to an operational aircraft, an intermediate flight step of a demonstrator aircraft or proof-of-concept vehicle which could be developed for a far more modest expenditure than a true operational prototype appears absolutely necessary.

This aircraft must demonstrate far more than the feasibility aimed at in the XV-3 and other similar test beds. It must provide an honest flight demonstration that all the technology is in fact in hand and it must substantiate the predicted benefits of the configuration by exploring the full flight envelope and demonstrate the aerodynamic, dynamic, structural and operational capability of the tilt rotor concept. NASA has plans for the development of such a vehicle which would be in the 10-12,000 lb. size class and might be configured as shown in Figure 15. I would appeal to all who are interested in the development of operational V/STOL aircraft to provide active support for this program to ensure that a "proof-of-concept" vehicle is, in fact, built and flown so that the large investment which government and industry have made in tilt rotor technology development may be put to use in operational aircraft and not just filed as an interesting academic exercise.

References

1. Pilot Evaluation of the Bell Model XV-3 Vertical Takeoff and Landing Aircraft, Report ATO-TR-62-1, U.S. Army Transportation Materiel Command, February 1962.

2. Deckert, W. H. and Ferry, R. G., Limited Flight Evaluation of the XV-3 Aircraft, Air Force Flight Test Center Report TR-60-4, May 1960.

3. D. vanWagensveld, et al, Wind Tunnel Tests of the Aerodynamics and Dynamics of Rotor Spin Up, Stopping and Folding on a Semi-Span Folding Tilt Rotor Model, USAF Report AFFDL-TR-71-6-Volume VII, September 1971.

4. Investigation of the Performance of Low Disc Loading Tilt Rotors in Hovering and Cruise Flights. (NASA Contract NAS2-5025) Boeing-Vertol Division Report D160-10013-1, March 1971.

5. Loening, D. G., Grief, R. K., and Kelly, M. W., Full-Scale Wind-Tunnel Investigation of the Longitudinal Characteristics of a Tilting-Rotor Convertiplane, NASA Technical Note D-35, December 1959.

6. K. G. Wernicke and H. Kipling Edenborough, Full-Scale Proprotor Development, Presented at the 27th Annual National V/STOL Forum of the American Helicopter Society, May 1971.

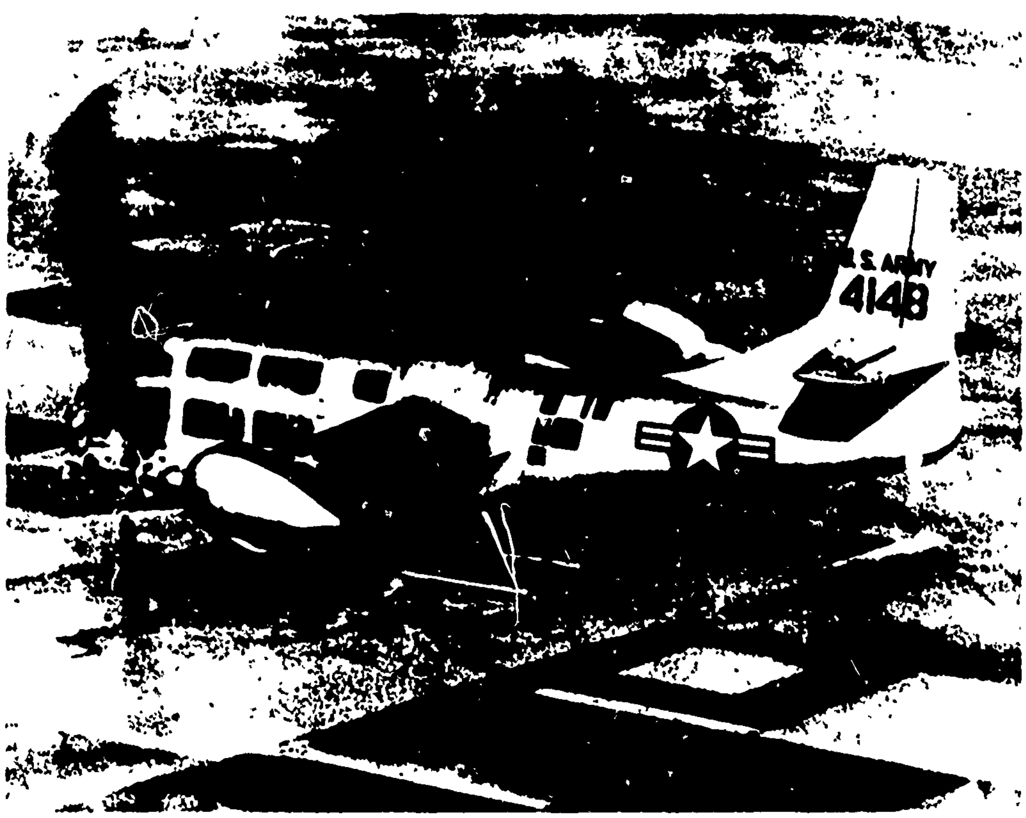
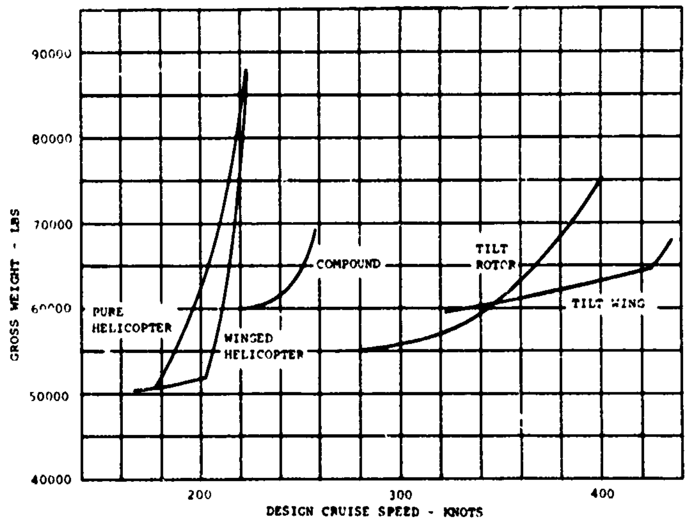


FIGURE 1. BELL XV-3 TILT-PROP ROTOR VTOL



Configuration Comparison - LTTAS Mission

Figure 2

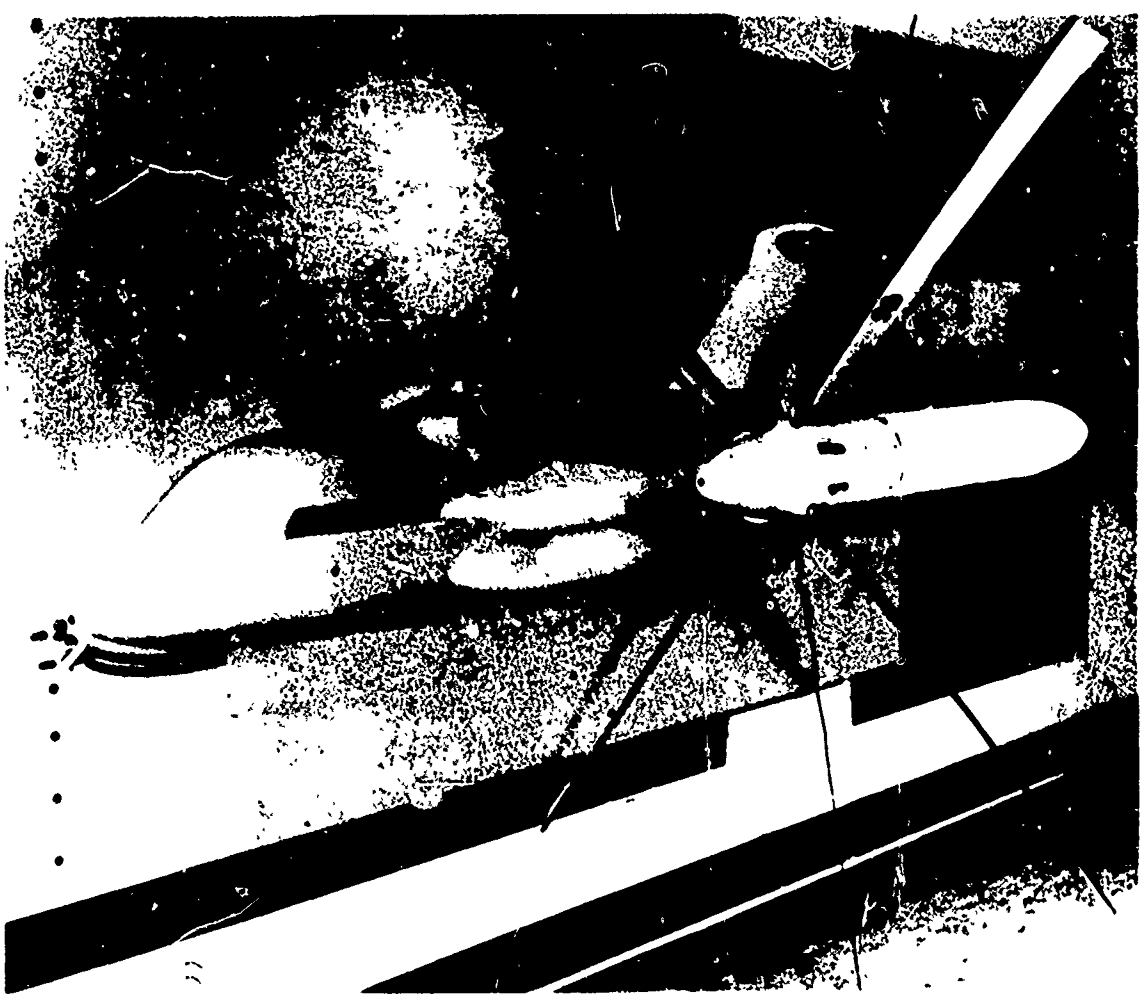


Figure 3

Semi-Span Dynamic Model

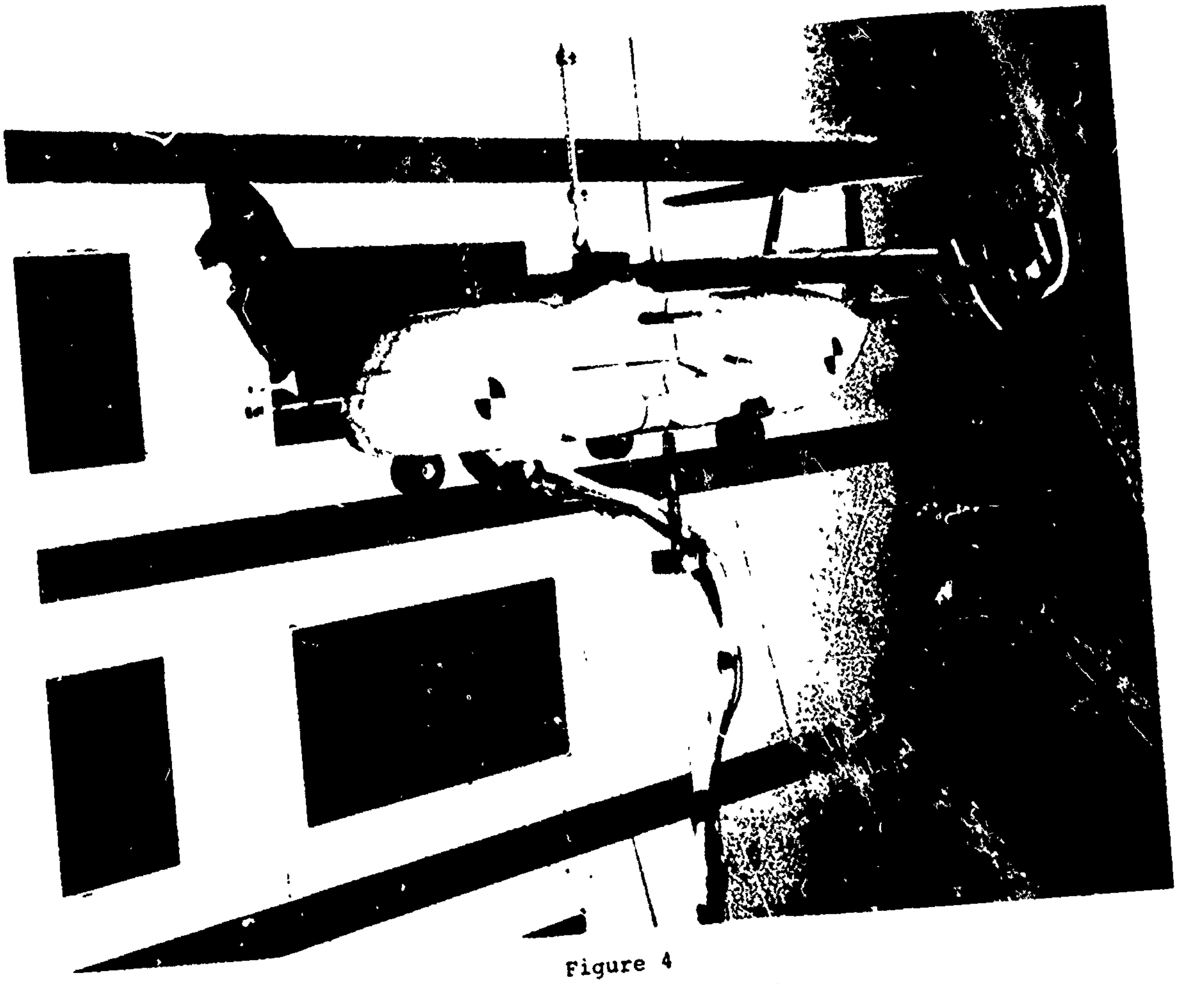
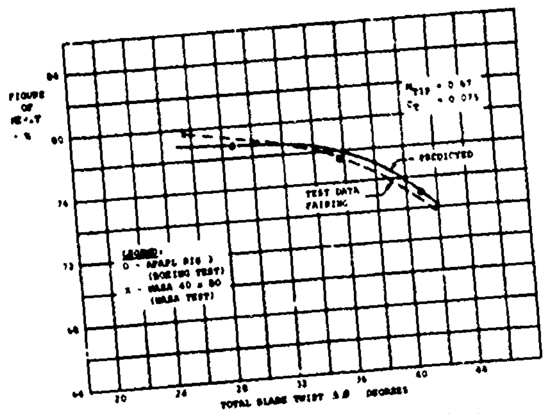


Figure 4
Full-Span Dynamic Model



Hover Performance Correlation
Figure 5

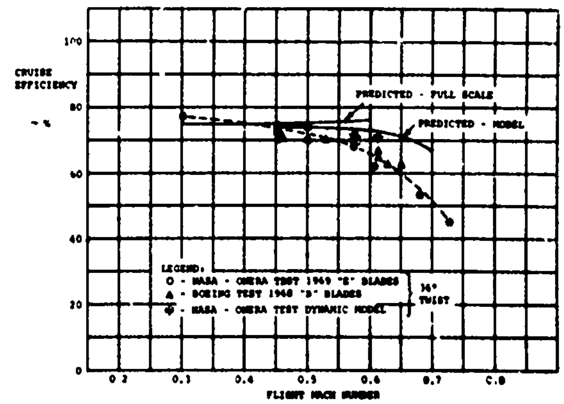


Figure 6

Cruise Performance Correlation

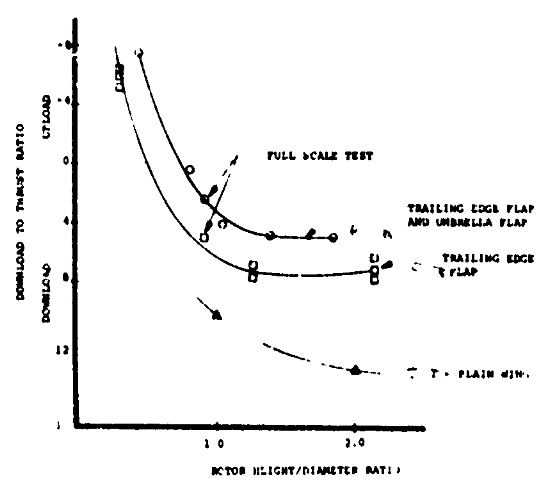


Figure 7

Download Reduction

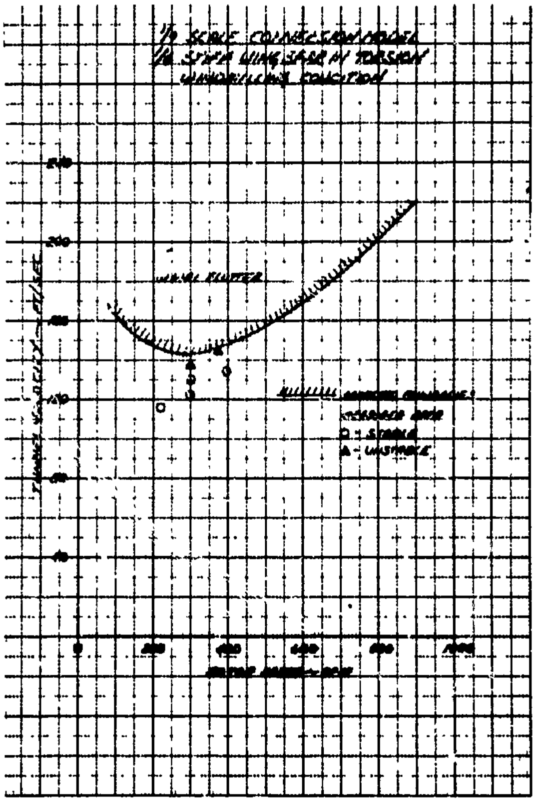


Figure 8
Whirl Flutter Stability Correlation

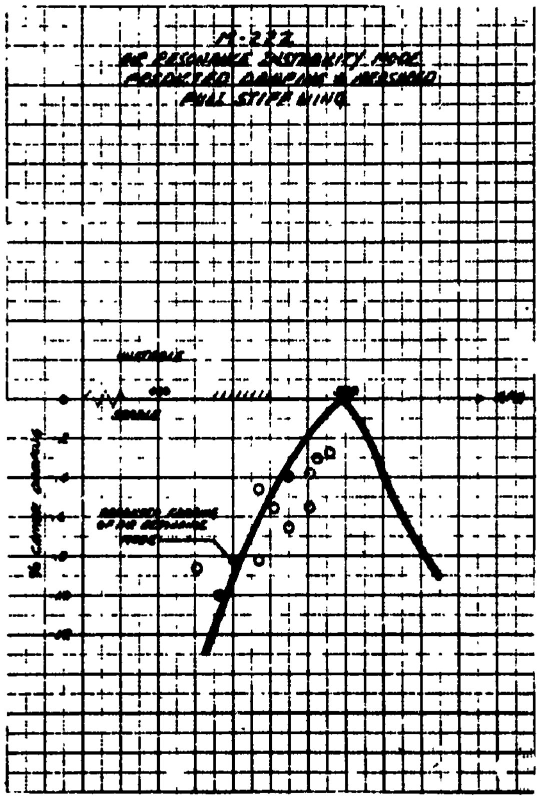


Figure 9
Air Resonance Stability Correlation

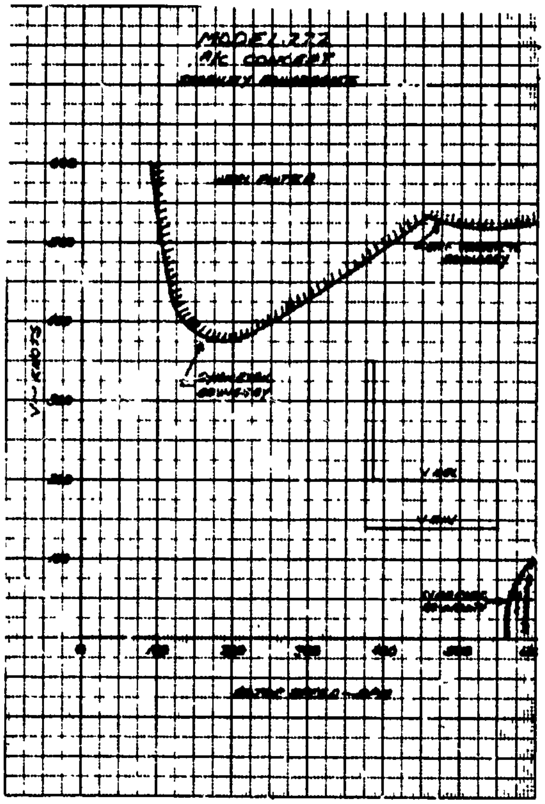
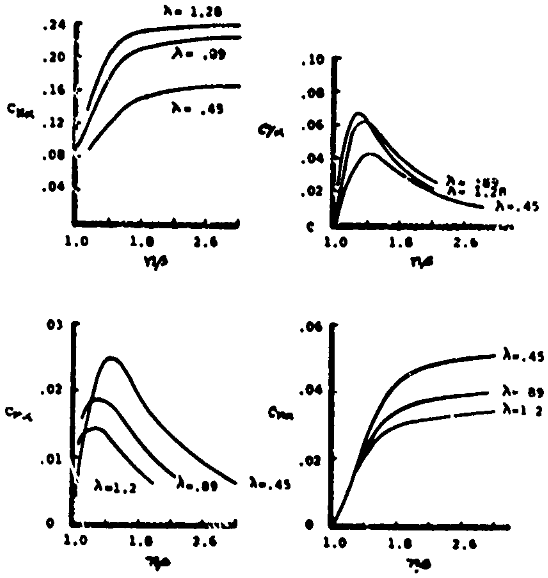


Figure 10
Aeroelastic Stability Boundaries



VARIATION OF DERIVATIVES WITH FLAP FREQUENCY, LAG FREQUENCY CONSTANT (=10.0)

Figure 11

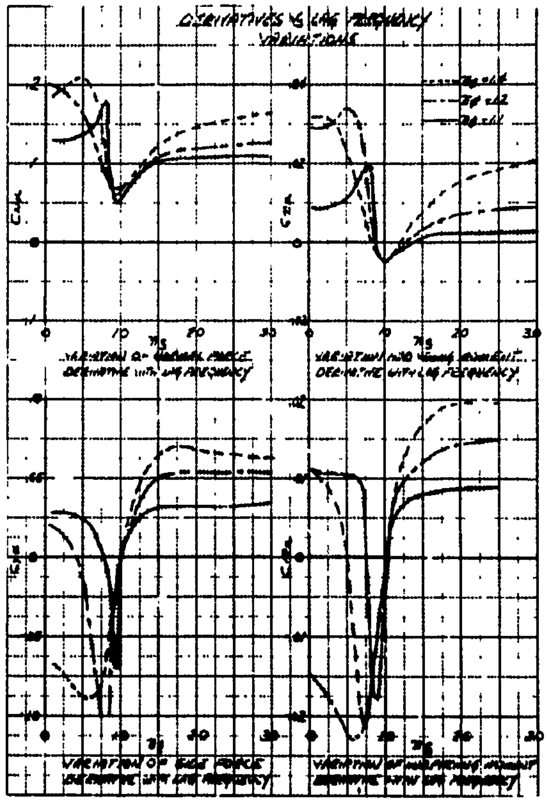


Figure 12

Effect of Lag Frequencies
on Rotor Derivatives

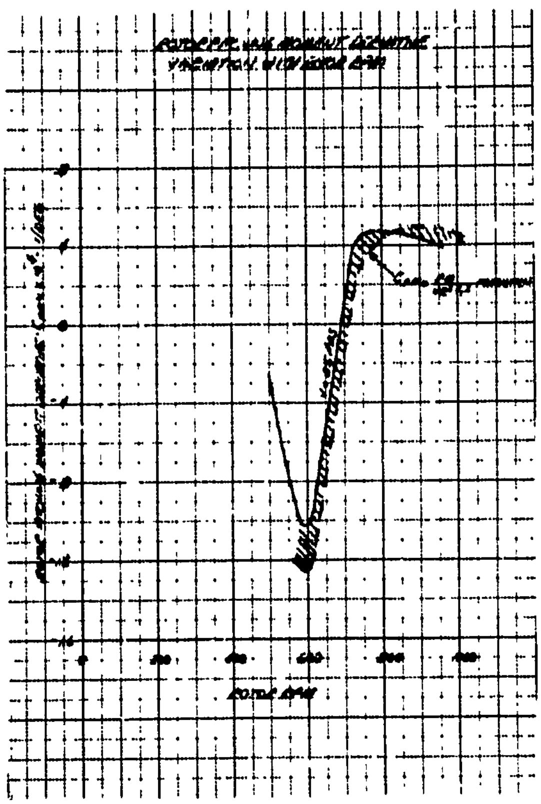
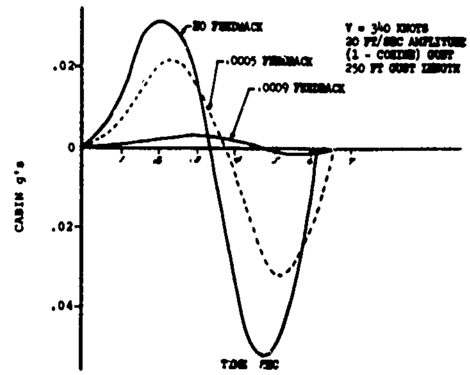


Figure 13

Correlation of Derivative
Tests With Prediction



CABIN ACCELERATION WITH EFFECT OF FEEDBACK

Figure 14

Gust Alleviation by Feedback Control

Figure 15

Tilt Rotor Demonstrator Aircraft

LE FENESTRON, SOLUTION NOUVELLE DE ROTOR DE QUEUE

par J. GALLOT, Chef du SERVICE AERODYNAMIQUE

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RESUME

Le rotor de queue caréné développé par l'Aérospatiale, pour le SA 341 "Gazelle" représente une solution nouvelle pour résoudre le problème du contrôle en lacet des appareils de type A.D.A.V. En effet, cette formule n'est pas soumise aux problèmes inhérents au rotor arrière classique : instabilité dynamique ou vibrations importantes à grande vitesse. Ses performances intrinsèques, légèrement moins brillantes en stationnaire, deviennent meilleures en vol d'avancement, et indépendamment de l'économie de puissance réalisée, la solution rotor caréné permettra d'atteindre des vitesses supérieures, la limitation due au décrochage étant fortement atténuée. Pour un convertible très rapide, on peut même envisager sans difficulté technique particulière, son escamotage complet si besoin est.
Dans l'état actuel, le rotor de queue caréné apparait comme une solution satisfaisante du point de vue qualités de vol, qui représente un progrès dans le domaine de la sécurité au sol et en vol, et qui, moyennant quelques améliorations simples pourrait constituer également un progrès sensible du point de vue bruit.

Notations

$$Cy = \frac{Fy}{\frac{1}{2}\rho S u^2}$$

Fy = poussée rotor caréné, positive à droite de l'appareil

ρ = densité de l'air

S = surface du disque balayé par les pales

u = vitesse périphérique

μ = paramètre d'avancement

V = vitesse d'avancement

INTRODUCTION :

Malgré l'expérience que les hélicoptéristes ont des rotors de queue à faible charge au disque, il est indéniable que cette solution constitue une servitude gênante sur le plan technique et opérationnel.

Sur le plan technique, la solution rotor de queue classique nécessite un transfert de puissance à grande distance en permanence, ce qui multiplie les organes mécaniques. Il opère dans des conditions vibratoires et aérodynamiques difficiles à cause de sa position en extrémité de queue et des interactions très sévères avec le souffle rotor principal, le sillage du fuselage et la dérive. Du fait de la sévérité de ces conditions de fonctionnement, le rotor arrière classique est soumis à des contraintes importantes qui limitent fortement la durée de vie de ses éléments. De plus il est généralement critique au point de vue stabilité de fonctionnement, par suite de la souplesse relative de la structure qui le supporte.

Sur le plan opérationnel, le rotor de queue est un ensemble fragile et dangereux. Au voisinage du sol, le contact est toujours possible avec des pierres ou des branches, ce qui conduit dans la majorité des cas à sa destruction.

Dans les opérations au sol où le rotor tourne, il représente un danger permanent pour le personnel qui se trouve à proximité immédiate de l'hélicoptère.

En vol, tout incident concernant la transmission arrière ou le rotor de queue est catastrophique puisqu'il nécessite un atterrissage immédiat en autorotation, avec une très faible liberté de manoeuvre.

La solution du rotor de queue caréné, ou "Fénestron", développée par l'Aérospatiale pour le SA 341 Gazelle (figure 1), permet de s'affranchir des servitudes inhérentes au rotor arrière classique.

De plus compte tenu de la technologie retenue, cette solution présente un progrès certain dans le domaine de la fiabilité et de la maintenance.

DESCRIPTION DU FENESTRON (figure 2)

Le rotor de queue "Fénestron" est constitué d'un rotor à pales multiples de faibles dimensions, articulées en pas seulement, qui tourne à l'intérieur d'un tunnel aménagé dans la dérive verticale en extrémité de la poutre de queue.

Le tunnel possède un bord d'attaque arrondi et un léger divergent pour améliorer le rendement lorsque le Fénestron travaille pour compenser le couple dû au rotor principal.

Les pales en alliage d'aluminium sont obtenues directement par matriçage. L'articulation en pas s'effectuant sur paliers auto-lubrifiants, elles comportent, au pied, une partie cylindrique en face du palier d'incidence, le levier de pas est constitué par une excroissance de la pale elle-même.

Le rotor est supporté directement par une petite boîte de transmission comportant un couple de pignons coniques, qui transmet le mouvement de rotation en provenance de la boîte de transmission principale. Sur ce boîtier une simple servo-commande permet de modifier, sans effort important au pied.

La dérive, elle-même, calée à une certaine incidence par rapport à l'axe appareil, est cambrée de façon à fournir une poussée latérale anticouple en translation. L'ensemble est complété à la partie inférieure par un sabot caréné capable d'absorber une certaine énergie en cas d'atterrissage très cabré.

AERODYNAMIQUE DU FENESTRON

- Vol stationnaire :

Au point de vue performance au point fixe, il est bien évident qu'une solution rotor caréné pour un anticouple d'hélicoptère est attrayante puisqu'elle permet théoriquement à iso diamètre un gain de puissance de l'ordre de 30 % pour une même poussée. En effet le tunnel améliore le rendement de l'anticouple pour deux raisons :

a) il y a réduction des pertes en bout de pale que subit normalement un rotor libre

b) la différence de pression qui s'établit entre la face amont et la face aval de la dérive, représente théoriquement la moitié de la poussée totale et en pratique environ 30 %.

Il ne faut pas oublier aussi qu'un rotor arrière classique en présence d'une dérive même petite a une efficacité diminuée de 5 à 10 % classiquement, par suite des intéractions défavorables avec la dérive.

Par contre, l'intégration d'un rotor caréné dans la structure de l'hélicoptère nécessite une réduction importante du diamètre pénalisante au point de vue puissance. Le choix définitif de ce diamètre est donc un compromis entre les performances et des considérations d'encombrement et de poids de dérive.

Dans le cas du SA 341, l'adaptation retenue n'est pas trop pénalisante puisqu'elle conduit dans le cas de la masse maximale de 1700 kg en vol stationnaire HES au niveau de la mer, en atmosphère standard, à une perte de 4 % sur la puissance totale par rapport à un hélicoptère équipé d'un rotor arrière conventionnel.

Des essais en soufflerie sur maquette à l'échelle 1/2, et au banc grandeur réelle ont permis de vérifier les prédictions de calcul et d'améliorer le fonctionnement du Fénestron :

- la forme du carénage a été optimisée en ce qui concerne la forme des lèvres d'entrée et la légère diffusion adoptée.

- l'influence du jeu entre les pales et le tunnel a été testée

- les profils de pale retenus ont été choisis d'après les résultats de soufflerie

Il est apparu de plus que le Fénestron avait une efficacité croissant régulièrement jusqu'au décrochage pratiquement, contrairement au rotor arrière classique dont l'efficacité diminue rapidement même lorsque le décrochage est encore assez éloigné (figure 4). Il faut noter que les pales du Fénestron étant de plus extrêmement raides, le comportement vibratoire au voisinage du décrochage est nettement plus sain que pour un rotor arrière classique.

Les essais sur banc appareil en présence du rotor principal en effet de sol, ont montré que le fonctionnement du Fénestron était très légèrement perturbé par le souffle rotor principal sans doute parce que la différence de pression existant sur la dérive est modifiée par l'apparition d'un vent relatif.
Ce phénomène peu gênant a été retrouvé sur appareil en vol stationnaire. Bien qu'il faille moins de puissance à l'hélicoptère dans l'effet de sol que hors effet de sol, il faut légèrement plus de pied dans l'effet de sol comme le montre la courbe (figure 5) donnant l'évolution du pied en fonction de la hauteur de l'appareil au dessus du sol.

- vol d'avancement :

Le fonctionnement du Fénestron en vol d'avancement a priori pouvait poser des problèmes. Notamment on pouvait craindre un décrochage de la lèvre d'entrée en attaque oblique, ce qui aurait eu pour effet d'introduire des variations brutales de moment de lacet et des contraintes alternées importantes dans les pales.

Dès le stade conception, pour réduire l'importance éventuelle de ce phénomène et surtout pour réduire la puissance passant dans la boite de transmission arrière en translation à grande vitesse, la dérive a été vrillée et cambrée de façon à ce que la presque totalité de l'effort anticouple nécessaire dans ce cas soit fournie par la dérive et non par le rotor. Le contrôle de lacet restait néanmoins assuré par la variation de pas rotor.

Les essais en soufflerie, effectués sur la maquette motorisée (échelle 1/2) pour des rapports d'avancement allant jusqu'à 0,35 et dérapage variable jusqu'à ± 8 degrés, ont montré que l'ensemble dérive-fénestron avait une efficacité régulière en fonction du dérapage (figure 6) quelque soit la vitesse simulée et le pas affiché, ce qui éliminait la possibilité d'un décrochage de la lèvre du carénage en vol d'avancement. Par ailleurs, l'efficacité de la commande de pas ne présentait aucune anomalie dans le domaine exploré (figure 7). Les essais ont montré aussi que la stabilité statique, en lacet, d'un appareil équipé de cette solution, est pratiquement déterminée par sa dérive verticale, ce qui nécessite une surface de dérive importante et efficace.

Les poussées maximum réalisées en soufflerie en vol d'avancement sont nettement supérieures aux poussées nécessaires à la fonction anticouple, même dans le cas d'une dérive non portante (figure 8) et surtout dépassent largement les possibilités d'un rotor arrière classique ayant même poussée maximum en vol stationnaire. De ce fait dans le domaine exploré, les possibilités de manoeuvre avec un fénestron sont importantes, alors que le même appareil équipé avec un rotor classique serait limité en vitesse et en dérapage.

Du point de vue performances en vol d'avancement, il est évident qu'un Fénestron pratiquement déchargé nécessitera beaucoup moins de puissance qu'un rotor arrière classique surtout à grande vitesse où celui-ci fonctionnerait près du décrochage.

Dans le cas du SA 341, les mesures effectuées en vol ont permis d'estimer à 70 % la diminution de la puissance passant dans la mécanique arrière par rapport à la solution classique pour une vitesse de 250 km/h. Cette économie ne se retrouve pas intégralement dans le bilan complet puisqu'il faut tenir compte des trainées induites par les 2 solutions et de la trainée induite par la portance de la dérive. Néanmoins, il reste un bénéfice global chiffré aux environs de 2 %.

STABILITE - MANOEUVRABILITE - MANIABILITE EN LACET :

Les essais en vol sur SA 341 ont permis d'expérimenter cette solution à plus de 310 km/h en palier et 350 km/h en descente, ce qui a permis de confirmer en général les essais de soufflerie. On a pu noter les remarques suivantes :

1°) La courbe de déplacement pédale en fonction du dérapage à la vitesse de croisière (figure 9) présente une zône où la pente, généralement très satisfaisante pour la stabilité, accuse une diminution très nette. Il s'agit de la position du pied qui correspond à la poussée nulle du Fénestron et par conséquent il semble qu'il y ait une légère plage de moindre efficacité lorsque la circulation d'air dans le tunnel s'inverse.

2°) En autorotation à grande vitesse, la dérive fournit un effort important qui n'est plus nécessaire et qui doit être compensé par une poussée Fénestron en sens inverse.

C'est pour ces deux raisons que les modifications suivantes ont été appliquées au cours de la mise au point :

1°) Suppression du vrillage et diminution du calage de la dérive pour limiter la poussée négative en autorotation et pour décaler vers des valeurs de dérapage moins usuelles, la zône de moindre stabilité

2°) Accroissement de la surface de la dérive et adjonction d'oreilles latérales profilées pour donner un peu plus de stabilité propre à l'appareil dans la zône de poussée Fénestron nulle qui reste traversée lors du passage en autorotation.

Dans la configuration adoptée, le rotor caréné fournit ainsi en translation une légère poussée anticouple sans que la puissance absorbée ait augmenté notoirement. Cette poussée ne s'inverse jamais sauf en cas de dérapage important ou en descente en autorotation.

Dans la configuration série, l'appareil présente une bonne stabilité dynamique.

Pour une puissance donnée et une vitesse déterminée, la position des pédales peut demeurer constante indéfiniment. En virage l'appareil se pilote simplement par action au manche en latéral sans réaction spéciale désagréable.

La manoeuvrabilité avec le Fénestron est déterminée par les marges au palonnier qui sont suffisantes dans tout le domaine de vol. En vol stationnaire H.E.S ou D.E.S, l'évolution de la position des pédales avec la masse réduite représentée sur la figure 10 laisse une marge suffisante en altitude compte tenu de la bonne efficacité du Fénestron quel que soit l'effort développé.

La stabilisation des différents caps au vent se fait avec une précision correcte jusqu'à des vents de 30 kts tout en laissant une latitude de manoeuvre suffisante (figure 11). Enfin en vol de translation l'évolution de la position du palonnier représentée sur la figure 12 en fonction de la vitesse dépend du niveau de puissance sur le rotor principal et du délestage par la dérive avec la vitesse. Dans tous les cas la garde au pied a été jugée satisfaisante, même à grande vitesse en autorotation.

La maniabilité en vol stationnaire, bien symétrique, peut être caractérisée par la réponse de l'appareil en lacet à un échelon de 10 % de la course totale des pédales. Dans le cas du SA 341 on obtient une vitesse angulaire de lacet d'environ 40 degrés/seconde.

En vol d'avancement, la réponse appareil devient dissymétrique. Par exemple à 100 kts, sur un échelon de 5 % on obtient à gauche une réponse en vitesse angulaire de lacet plus importante, à gauche 13°/s contre 7°/s à droite.

Cette dissymétrie d'efficacité déjà relevée sur les courbes de stabilité statique (figure 9) n'est cependant pas gênante, compte tenu de la bonne stabilité de l'hélicoptère et du mode de pilotage, et passe inaperçue du pilote, s'il ne recherche pas vraiment ce problème.

VIBRATIONS ET EFFORTS DANS LES COMMANDES :

Le fait que le Fénestron soit presque totalement déchargé en vol d'avancement et par conséquent soit alimenté par un flux très faible, laissait supposer que la dissymétrie d'efforts aérodynamiques à laquelle est soumise un rotor arrière classique, serait fortement atténuée dans la solution Fénestron. Ceci a été pleinement confirmé par les mesures de contraintes en vol effectuées par straingages collées sur les pales.

Un large domaine de vol a été balayé pour ces mesures :

- vol stationnaire et manoeuvres rapides en lacet

- vol de montée et de descente en autorotation à toutes les vitesses d'avancement possibles.

- vol de translation jusqu'à 350 km/h en descente à la puissance maximale.

Les contraintes les plus élevées rencontrées dans le rayon de raccordement au pied de pale correspondent au vol stationnaire avec manœuvre rapide de rotation à droite, elles sont de l'ordre 1,5kg/mm2. En vol horizontal à la vitesse de 260 km/h et dérapage nul, elles ne dépassent pas 1 kg/mm2. Par ailleurs le dérapage a une influence très faible sur les contraintes enregistrées (voir figure 13) et dans tous les cas de vol rencontrés le niveau de contraintes est nettement inférieur à la limitation correspondant à la durée de vie infinie.

De ce fait on peut éliminer les essais de fatigue effectués sur les pales prélevées en cours de production Série, tout en garantissant une durée de vie théorique infinie, contrairement au cas des pales d'un rotor arrière classique.

Les efforts de commande dépendent de la définition géométrique et massique de la pale et de sa position en corde par rapport au moyeu, ainsi que des moments de frottement engendrés par le palier autolubrifiant dans lequel la pale est encastrée.

Au cours de l'étude, l'adaptation a été faite de façon à ce que les efforts de commande soient faibles en vol de croisière. Par contre en vol stationnaire il s'ensuit que les efforts de commande sont relativement élevés et c'est la raison pour laquelle une petite servo-commande hydraulique simple corps a été montée sur la boîte arrière. En cas de panne hydraulique, les efforts sont donc nuls en croisière et ne deviennent plus élevés qu'en vol stationnaire (environ 30 kg au pied) ce qui est admissible compte tenu de la faible durée de vol nécessaire dans cette configuration pour se poser.

COMPARAISON FENESTRON - ROTOR ARRIERE AU POINT DE VUE BRUIT :

Des essais comparatifs ont été effectués sur des appareils au sol, entre un rotor arrière classique et un Fénestron. Dans les deux cas, les pales du rotor principal avaient été enlevées. Le microphone était déplacé sur le sol à hauteur du centre du rotor arrière et les mesures ont été effectuées pour les points indiqués sur la Figure 14.

Le Fénestron se caractérise essentiellement par deux raies à 1300 Herz et 2500 Herz correspondant aux harmoniques 1 et 2 du bruit rotationnel. La directivité du bruit est maximale au voisinage de l'axe à 45° par rapport à l'axe rotor.

Il se différencie donc nettement du rotor arrière classique qui tournant plus lentement, émet un spectre de raies plus riche en harmoniques. Pour le rotor arrière classique dont la fréquence fondamentale du bruit rotationnel est 68 Herz, on a mis nettement en évidence les 7 premières raies, ainsi qu'un bruit à large bande qui apporte une contribution notable à très basse fréquence (160 Hz).

L'atténuation du bruit avec la distance est plus importante dans le cas du Fénestron (12 PNdB entre 25 m et 50 m) contre 6 PNdB pour le rotor arrière classique. Donc bien que le niveau perçu dans le cas du Fénestron à faible distance soit plus important, compte tenu de l'atténuation plus grande avec la distance, les niveaux perçus sont du même ordre de grandeur à 50 m, et à grande distance le Fénestron possède un niveau sonore inférieur.

Par ailleurs, en vol de croisière, bien qu'aucune mesure n'ait été effectuée, on peut estimer que le Fénestron étant pratiquement déchargé par la dérive, ne sera pas soumis à des phénomènes de compressibilité comme un rotor arrière classique et aura par conséquent un niveau de bruit plus faible à grande vitesse.

SECURITE - VULNERABILITE :

La solution rotor de queue caréné présente un progrès très net dans le domaine de la sécurité au sol et en vol.

Le fait que le rotor arrière soit plus petit, et inclus dans un tunnel de protection, avec une bonne matérialisation en rotation, exclut pratiquement l'accident pour le personnel au sol. En cas d'atterrissage brutal, trop cabré, le Fénestron est à l'abri de détériorations consécutives au contact avec le sol. En vol en cas d'incident concernant la transmission arrière ou le rotor de queue lui-même il est possible de continuer le vol avec un léger dérapage à la vitesse de croisière jusqu'à la destination prévue puisque la plus grosse partie de l'anticouple est fournie par la dérive et de se poser alors bien sûr en autorotation. Dans des conditions équivalentes un appareil à rotor de queue classique serait obligé de se poser immédiatement en autorotation et l'on conçoit l'avantage au point de vue sécurité qu'offre la solution Fénestron pour l'utilisateur civil ou militaire.

La vulnérabilité du rotor caréné peut s'analyser en considérant :

- les possibilités de contact avec des objets étrangers
- les probabilités d'impact de balles ou d'éclats d'obus tirés par l'ennemi

En ce qui concerne les possibilités de contact avec les corps étrangers, le Fénestron est bien protégé par son carénage contre les obstacles fixes que ce soit branche d'arbre, buissons ou pierres émergeant du sol.

De plus en vol stationnaire, il parait très improbable que les cailloux ou graviers soulevés par le souffle rotor principal, et même ces morceaux de glace se détachant du rotor principal en atmosphère givrante puissent pénétrer dans le tunnel après un changement de direction à 90° que leur interdit pratiquement leur seule inertie. En vol d'avancement le risque d'ingestion d'objets étrangers est encore plus minime puisque le flux d'air traversant le Fénestron est pratiquement nul.

En ce qui concerne la vulnérabilité aux balles tirées dans l'axe du rotor caréné, on démontre facilement que les risques d'impact sont proportionnels à la surface vulnérable définie comme la somme de la surface réelle des pales et de la surface balayée par les pales pendant le temps mis par le projectile pour traverser le disque rotor.

La comparaison avec le rotor arrière classique montre que le risque dans le cas de la balle de fusil, est deux fois plus faible dans le cas du Fénestron.

Par ailleurs même en cas d'endommagement, les pales rotor caréné travaillant à un niveau de contraintes très faible étant de plus courtes et pleines, sont moins susceptibles de se rompre en fatigue que des pales de rotor arrière classique. Des essais en vol ont montré qu'on pouvait tolérer un balourd important sur l'ensemble sans que le niveau vibratoire ne se détériore trop et que l'ensemble arrière ne soit endommagé. Donc une détérioration du rotor quoique improbable, avec des conséquences moins graves que dans le cas du rotor arrière classique.

MAINTENANCE :

La maintenance de l'ensemble arrière d'un appareil équipé de la solution Fénestron est nettement plus faible que celle d'un appareil à rotor de queue classique, de par la simplification technologique réalisée :

1°) La réduction à des valeurs très faibles de la puissance absorbée par le rotor en translation, décharge presque complètement la boîte et l'arbre de transmission arrière, ce qui permet d'augmenter fortement leur T.B.O

2°) La faible poussée fournie en translation par le rotor lui-même et ses conditions de fonctionnement sont telles que les pales et les éléments de commande ne subissent que des contraintes de fatigue très faibles, ce qui leur confert des durées de vie infinies.

3°) La suppression des articulations de battement pour les pales et l'utilisation de paliers autolubrifiants pour le pas, éliminent les travaux d'entretien rotor

4°) La vulnérabilité réduite du rotor dans un environnement opérationnel diminue la nécessité des interventions pour échanges ou vérifications

5°) Les opérations de démontage et remontage elles-mêmes sont grandement simplifiées et ne nécessitent pratiquement aucun réglage de la part de l'utilisateur.

CONCLUSION :

La solution rotor anticouple caréné type "Fénestron" présente donc un progrès important dans le domaine de la sécurité et de la maintenance. Sa mise au point n'a pas posé de problèmes particuliers. Cette solution a donné satisfaction au point de vue performances et qualités de vol dans le domaine de vol exploré par le SA 341 Gazelle. Il reste maintenant à en déterminer les limites d'utilisation pour les vitesses plus importantes qu'atteindront dans un proche avenir les combinés ou convertibles à voilures tournantes.

Fig.1

Fig.2.ROTOR-FIN GENERAL ARRANGEMENT

DISPOSITION ROTOR DERIVE

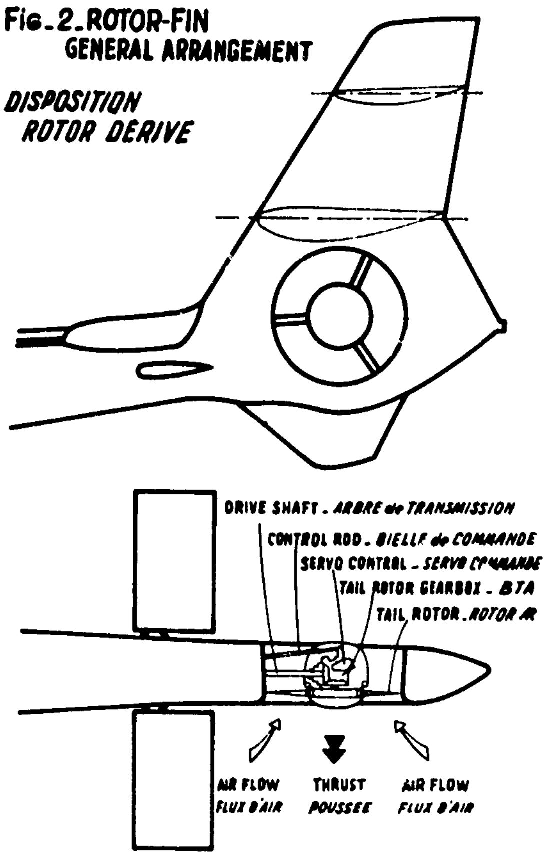

Fig.2

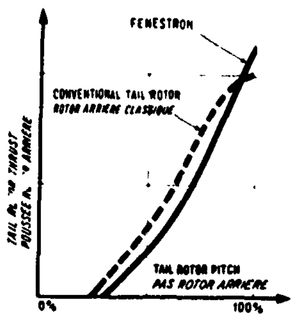

Fig 4. TAIL ROTOR THRUST VS PITCH IN HOVER
POUSSÉE ROTOR ANTICOUPLE EN STATIONNAIRE

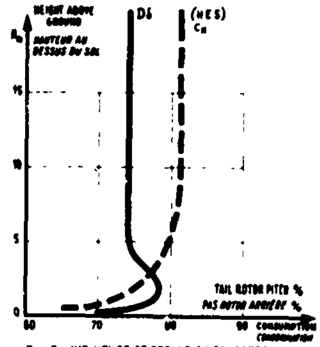

Fig 5. INFLUENCE OF GROUND ON FENESTRON
INFLUENCE DU SOL SUR LE FENESTRON

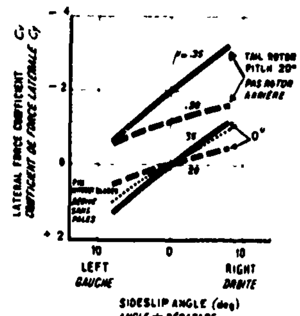

Fig 6. FENESTRON THRUST VS SIDESLIP
(WIND TUNNEL TESTS)
POUSSÉE FENESTRON EN FONCTION DU DÉRAPAGE
(RÉSULTATS SOUFFLERIE)

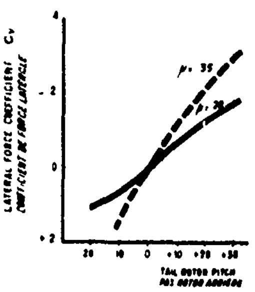

Fig 7. FENESTRON THRUST VS PITCH
(WIND TUNNEL TESTS)
POUSSÉE FENESTRON EN FONCTION DU PAS
(RÉSULTATS SOUFFLERIE)

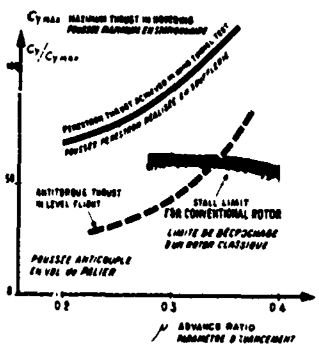

Fig. 8. COMPARISON OF THRUST POSSIBILITIES OF CONVENTIONAL ROTOR AND FENESTRON
COMPARAISON DES POSSIBILITÉS EN POUSSÉE DU ROTOR CLASSIQUE ET DU FENESTRON

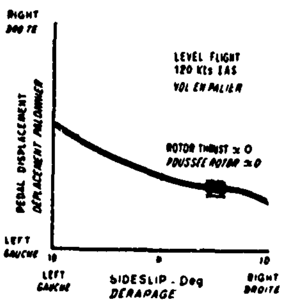

Fig 9. PEDAL DISPLACEMENT VS SIDESLIP
DÉPLACEMENT PALONNIER EN FONCTION DU DÉRAPAGE

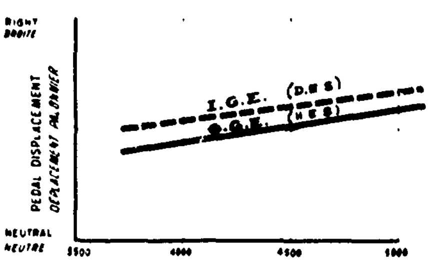

Fig 10. PEDAL DISPLACEMENT VS WEIGHT TO DENSITY RATIO IN HOVER
DÉPLACEMENT PALONNIER EN FONCTION DE LA MASSE RÉDUITE EN STATIONNAIRE

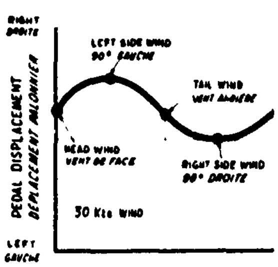

Fig. 11. PEDAL DISPLACEMENT VS WIND HEADING IN HOVER
DÉPLACEMENT PALONNIER EN FONCTION DES CAPS AU VENT

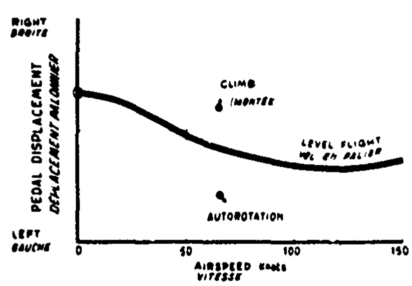

Fig. 12 PEDAL DISPLACEMENT VS FORWARD SPEED
DÉPLACEMENT PALONNIER EN FONCTION DE LA VITESSE

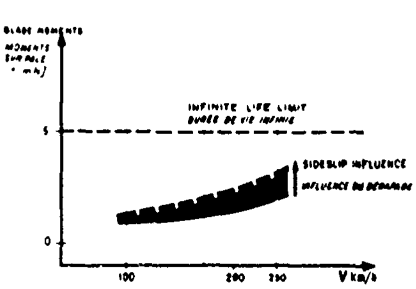

Fig 13. INFLUENCE OF SPEED AND SIDESLIP ON STRESSES
ÉVOLUTION DES CONTRAINTES AVEC LA VITESSE

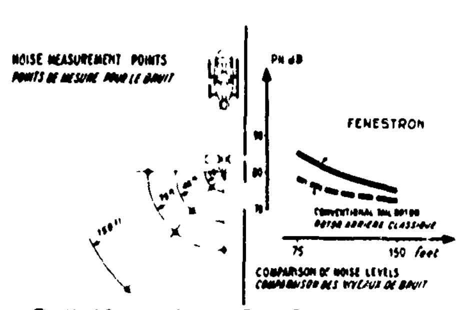

Fig. 14. NOISE MEASUREMENTS ON FENESTRON
MESURE DE BRUIT SUR FENESTRON

DEVELOPMENT OF THE ABC ROTOR*

by

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SUMMARY

The development of the ABC rotor is traced from conception through small scale model wind tunnel testing, full scale analysis, design, fabrication and ultimate wind tunnel testing of a 40 ft diameter rotor in the NASA-AMES 40 ft x 80 ft wind tunnel.

In particular, the principal design tradeoffs resulting from the early analyses and testing are discussed along with their expected impact on the full scale rotor characteristics. Materials and manufacturing methods employed are covered including the more important difficulties that were surmounted during the nearly five years of development.

Finally, the major test programs are outlined including blade balancing, turbine test bed operation and full scale wind tunnel testing in the 40 ft x 80 ft NASA-AMES facility up to speeds of 180 knots and advance ratios of .91. Significant results of these tests are presented, and applications to aircraft systems discussed.

Some 35 years ago Glauert proposed a helicopter rotor system that precluded retreating blade stall without increasing rotor speed and thereby the advancing blade tip Mach number and producing higher compressibility power losses. The system operates by allowing the advancing blade to produce more lift than the retreating blade. This is accomplished in a natural fashion since the advancing blade always "feels" a higher dynamic pressure than the retreating blade because of its higher airspeed (See Figure 1). As embodied in Sikorsky's Advancing Blade Concept (ABC), the unbalanced aerodynamic moment thus produced can be reacted by another rotor turning in the opposite direction or by an asymmetrical, fixed aerodynamic surface. The coaxial ABC rotor configuration was chosen to avoid the load sharing problems associated with combining fixed and rotating lifting surfaces while simultaneously minimizing the high load path generated by the large lift moments of each rotor. Control of such a rotor is achieved as in other conventional coaxial systems by simultaneous cyclic pitch variation of both rotors about orthogonal axes. In addition, optimization of performance, rotor moments, stresses and vibration is easily effected by use of differential cyclic and collective pitch in the form of either separate trim controls or integrated coupling with the standard flight controls. The basic aerodynamic and control principles are described in detail in References 1 and 2.

At the outset it was recognized that the design process is an iterative one involving, amongst other things, elements of geometry, mathematical analysis, manufacturing constraints and test results, all based on specifications or design criteria that in themselves frequently derive from subjective marketing goals as well as considerations of technical feasibility. As a design goal which appeared achievable, a rotor system for a hypothetical aircraft of 14,500 lb gross weight with a maximum level flight speed of 230 knots was chosen. Such a machine (Figure 2) would have a wide range of roles and missions in both military and commercial service. Moreover full scale testing of a rotor system of this size would eliminate the doubts and risks often associated with theoretical predictions and extrapolation of small scale test data. Based on OGE hovering considerations alone, a rotor diameter of 40 ft was established. It is worthwhile to note that ABC rotor diameters are not dictated by forward flight considerations as with conventional rotors where retreating blade stall enters the picture. Other factors which would constrain ABC rotor diameters are disc loading and autorotative landing characteristics in the case of single engine machines.

Initial structural investigations showed that a relatively thick root end would be required even with the use of such materials as titanium and advanced fiberglass. A closed solution with an estimated blade weight of 260 lb was reached after 18 iterations (the actual blade weight turned out to be 256 lb). During the course of this early design effort a set of model blades 4 feet in diameter (1/10 scale) was constructed for subscale tests in the UARL 4 ft x 6 ft subsonic pilot wind tunnel. The blades were geometrically similar except for twist and were dynamically similar in that the ratio of the first flatwise cantilever bending frequency to rotational frequency was approximately equal to that of the full scale design. Coaxial hover performance tests were also conducted with this model rotor (Figure 3). The results, presented in detail in Reference 2, were very encouraging. At this point a Corporate decision was made to construct the 40 ft diameter design for testing in the 40 ft x 80 ft NASA-AMES wind tunnel up to speeds of 190 knots and advance ratios of 1.25. Management placed a second major constraint on the design, viz. that the blades, and preferably also the rotor hub/shaft, would have to be man-carrying, i.e. they should have a reasonable fatigue life (1000 hr) and incorporate such standard

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Sikorsky safety and convenience items as the pressurized Blade Spar Monitoring System (BIM®) and pre-tracked blades. Additionally, it was required that the rotor system weight fraction be at least as good as current state-of-the-art helicopters and compound aircraft. Use of tooling and manufacturing methods consistent with quantity production was also to be considered wherever possible. To achieve these ends required some 4-1/2 years and $7.5M of corporate funding. A contract was negotiated with USAAVLABS for $250,000 and the use of the NASA-AMES facility.

ROTOR BLADE DESIGN

Loads for the ABC rotor were determined by utilizing a digital computer program to determine the aeroelastic response of flexible rotor system components based on the normal modes technique.

The equations of motion were arrived at by consideration of equilibrium of aerodynamic, dynamic and elastic moments and forces acting at discrete radial blade stations. The resulting moment and force equilibrium equations were differentiated to express them in terms of local blade loadings, and the blade displacements were expanded in terms of the appropriate uncoupled blade vibratory modes. Structural and centrifugal stiffness terms were replaced by equivalent uncoupled modal natural frequency expressions. Blade deflections were then expressed as finite series summations of uncoupled normal vibratory modes, and a closed loop iteration of modal amplitude was performed by the digital computer to obtain a convergent aeroelastic response steady state solution.

Blade section aerodynamic forces and moments were based on quasi-steady-state aerodynamic theory. To account for effects of stall, compressibility, Reynolds Number, reverse flow and airfoil section, no small angle assumptions were made regarding the magnitude of either the blade section angle of attack or the inflow angle, and the computer program was provided with the appropriate two-dimensional airfoil data (C_L, C_D, C_M versus angle of attack for various Mach and Reynolds Numbers) necessary to evaluate the aerodynamic forcing functions.

The final design was the result of over 21 iterations, beginning with a complete fiberglass blade and ending with a blade having a monolithic titanium spar, fiberglass skin and a sealed honecomb trailing edge.

At the outset a blade made completely of fiberglass was investigated; however, the transition from fiberglass to metal at the root end attachment proved to be both heavy and not too amenable to analysis. In addition the blade deflections resulting from the very low elastic modulus required what were considered excessive amounts of interrotor separation. To quantify the critical derivatives, it was necessary to perform some design studies on the hub/shaft.

HUB/SHAFT WEIGHT AND DRAG DERIVATIVES

Results of these studies indicated the weight derivative for this item to be about 70 lb/ft for titanium. Drag tests on a 1/2 scale model at super critical Reynolds Numbers in the UARL pilot wind tunnel indicated an equivalent parasite drag area of .5 ft^2/ft of shaft length.

INTERROTOR SPACING OPTIMIZATION

To optimize rotor spacing, several other factors were considered. It was impossible, for example, to have a spacing of less than 22" at the rotor shaft centerline because of the kinematic constraints of the blade pitch controls; therefore this automatically became the minimum boundary. Based on the aircraft design criteria of 14,500 lb gross weight and a Vmax of 230 knots, it was determined that blade tip contact would occur with an interrotor tip spacing of 42". The calculations involved considered some vertical load factor capability and angular rate generation which produces gyroscopic blade deflections. These latter effects were minimized by mechanically phasing the cyclic pitch control inputs in accordance with relationships discussed in References 2 and 3. Proper phasing of these inputs results in cancellation of approximately 70% of the blade deflection due to gyroscopic coupling. This is graphically illustrated in Figure 4.

Based on this information, an arbitrary 10" was added to the 42" to allow for uncertainties in the analyses, thus bringing the interrotor tip spacing to 52". In addition, it was decided to design frangible tips for the upper rotor to ensure that failure would occur on all three blades of one rotor only, thus precluding a mass unbalance in the unlikely event of an unforseen system instability resulting in contact of the two rotors.

Finally, in view of hub/shaft weight and drag considerations, a compromise was made with regard to the precone angle. At the design gross weight of 14,500 lb, a precone angle of +2.5° yields the minimum steady flatwise bending stresses. Stress and flutter calculations showed the upper rotor could be overconed 2.5° to a +5° precone angle, and the lower rotor underconed the same amount for a zero degree precone angle. This resulted in a 30" interrotor spacing at the shaft centerline for the 52" tip separation.

FLUTTER STABILITY

A flutter analysis of the ABC rotor blades indicated that the blades would be free from flutter for all test conditions scheduled for the AMES Wind Tunnel test program. The most critical condition for flutter was the 190 knot condition at 352 RPM which is two RPM above the maximum continuous value This condition represents the maximum advancing blade tip Mach Number. The normal rotor speed is 310 RPM. For the measured control system stiffness, the upper rotor was calculated to be nearer to the stability boundary at the 190 knot, 352 RPM condition. Because of the conservatism of the analysis, however, complete stability was indicated for the wind tunnel test program.

The upper rotor blades are more critical from the standpoint of classical flutter since they are set at a positive precone angle of five degrees in addition to having a mass axis that is aft of the shear center locus which lies approximately at the quarter chord. The mass axis runs close to the 30% chord line at the outer end of the blade. It was this unbalance that could have produced a susceptibility to flutter. The flutter analysis used to evaluate the rotor was developed under U.S. Army funds and is fully described in Reference 4. For the ABC rotor, the mathematical model included only the inertia of the outer 1/3 of the blade. However the mode shapes used were those computed for the entire blade. Initially, the stability of a single blade was examined, the structural and aerodynamic influence of other blades being neglected. The aerodynamic forces acting on the blade were calculated as noted earlier in this paper with the additional assumption that the airflow over the blade at $\psi = 90°$ exists for all time. Thus the stability of the blade was examined at $\psi = 90°$ only, which is considered to be the most critical azimuth position with regard to flutter of the ABC blades. Deflection of the blade was represented by the first three flatwise modes together with the first torsion mode. The inertia of the inner 2/3 of the blade was omitted, as the flutter calculations overestimate the stabilizing effect of the inboard portion of the blade. Examining the stability of the outboard 1/3 of the blade at $\psi = 90°$ was considered to make the method of analysis very conservative. For a significant flutter response, the blade would have to exceed the flutter speed over a substantial portion of its azimuth travel, not just at the $\psi = 90°$ position where the tip Mach No. reaches its maximum value. Therefore the calculated flutter margin for the ABC rotor blade at $\psi = 90°$ adequately ensures complete freedom from this form of instability. Moreover, the high blade torsional stiffness (first mode = 27/rev) precludes torsional divergence. Although advance ratios during the subsequent full scale testing in the NASA-AMES facility did not exceed .91, the analysis was carried through to show stability beyond an advance ratio of 2.8.

SPAR MATERIAL SELECTION

To minimize deflections, a higher modulus material was required. As Figure 5 shows, there is little choice among the three state-of-the-art materials (steel, aluminum, and titanium) from the standpoint of specific stiffness. However titanium does have a strength to weight advantage as illustrated in Figure 6. To maximize this advantage, the next major design iteration employed a nest of adhesively bonded titanium sheets for the spar, since sheet stock has higher allowable stresses than large forged material. The principal problems with this design were the dimensional accuracies required for proper nesting, and lack of good NDT (Non-Destructive Test) methods for determining the bond quality.

Concurrent with the preceding developments, a number of exotic methods of producing large, closed titanium sections were investigated. These included various types of roll forming, both hot and cold, as well as diffusion bonding and electron beam welding of open sections. The large reduction in allowable vibratory stresses in the weld zones, plus rather expensive tooling, precluded the latter approach. The cold forming methods available at the time were also quite expensive; moreover the tendency of titanium to gall severely when worked cold eliminated these techniques also. Recent technical developments, however, make this method deserving of review for future applications. A hot, roll forming process was then selected and the next year-and-a-half spent in attempting to develop this method. Unfortunately, the concept appeared to be too far ahead of the state of the titanium metalworking art, particularly with regard to maintenance of dimensional tolerances.

During the time when the hot forming processes were under development, a parallel investigation of advanced composite application was undertaken with the General Dynamics Corporation of Fort Worth, Texas. As implied in Figures 5 and 6, the payoff in terms of weight alone might significantly offset the higher initial cost of these materials. This is especially true in the case of rigid rotors as compared to articulated rotors where a minimum centrifugal force must be maintained to limit the coning angle. A boron fiber/epoxy resin system was used to fabricate some typical sections, one of which is illustrated in Figure 7. Structural tests of several specimens showed great promise; however, the root end attachment problem did not appear soluble within the established budget and schedule, and satisfactory NDT techniques were lacking. After spending considerable effort on the foregoing processes with relatively little success, it was decided to produce a spar tube by conventional machining methods and impart the sectional shape and twist requirements in a single hot creep forming operation. At this point in time, advanced forging techniques had been developed which produced enough anisotropy in the titanium material to yield relatively high fatigue strength in large monolithic structures. Figure 8 is an exaggerated longitudinal section of the spar tube before creep forming. The reverse taper near the root end was a means of reducing weight. Of special interest is the variation in wall thickness with radius, the maximum being over 1.66" near the root with the minimum of .075" occurring at the tip. From the military viewpoint this provides distinct vulnerability advantages. At the root end where moments and stresses are highest, the section is thickest. Moreover the spar remains circular for the first three feet from the center of rotation, thus presenting obliquity to a large percentage of hits. In the outboard region where stresses are relatively low, the spar may be holed without any great concern. In fact a large number of holes already exist to provide for tuning weight retention. Figure 9 shows a finished spar after creep forming.

BLADE SKIN DESIGN

Having chosen 6Al-4V titanium as the principal bending load carrying member to minimize weight and deflection, the choice of a skin was based primarily on the requirement for maximizing torsional fatigue strength and rigidity, particularly in the outboard region where the classical flutter boundary is sensitive to the local torsional stiffness.

Fiberglass appeared to have the desired characteristics to achieve these ends. In particular, 1002S glass was selected over the more common 1002E material because it has both a 15% higher stiffness and fatigue strength. The cloth was delivered as prepreg and laid up to a thickness of .060" on a male mandrel with all the fibers oriented at 45° to the feathering axis, thus maximizing the torsional strength and rigidity.

FINAL BLADE DESIGN

A complete blade assembly before painting is shown in Figure 10. The principal elements are illustrated in Figure 11 at station 60 (5 ft out from the center of rotation) and station 216 (2 ft in from the blade tip). The titanium spar, S-glass skin, honeycomb aft section and foam filler are clearly seen. The vertical member just aft of the spar transfers the shear loads developed in the aft section to the spar. The moment developed by these loads is reacted as a couple in the skin. The four cavities in the foam spaced around the spar serve as vent corridors to permit rapid escape of the dry nitrogen gas used to pressurize the spar in the BIM® crack detection system. Some concern was expressed with regard to the rate at which gas would escape with the spar surrounded by the foam and a relatively impermeable and extensive skin to spar bond area.

Blade tuning was accomplished by installation of relatively small tip weights. Southwell plots of a representative blade are shown in Figure 12. As mentioned earlier, the chordwise blade cg was located at 30% mac; this required no special counterweights. Spanwise static balance was accomplished on the upper rotor by the addition of lead shot suspended in a copolymer compound and on the lower rotor by removal of small amounts of material from the tuning weights. It is significant to note that there was only a 1-1/2 lb total spread in weight among the eight final blades fabricated with a nominal weight of 256 lb per blade.

Dynamic balance of the rotor blades was performed on the 3,000 HP blade balance stand in Stratford. Figure 13 shows the upper rotor installed on this facility. The slope of the pitching moment vs. angle of attack function was determined during rotation, and the slopes of all three blades on each rotor made coincident by rearranging a number of chordwise counterweights located at the root end inside the fairing cap. The zero moment intercept of each blade was adjusted by bending the outboard trim tab to shift the curve. Later developments in the program gave indication that these operations may not be necessary in very rigid rotors.

The third major operation conducted on the balance stand was blade tracking. At this point it is worthwhile to note that all balancing and tracking operations were performed with an articulated head on the balance stand. This has no significance for the propeller moment and aerodynamic moment balance since these moments act about the feathering bearings. However, it was felt that the high flatwise stiffness of the blade system when installed on the rigid hub/shaft would limit blade deflections to values around the threshold of the electronic blade tracker for unbalanced lift forces that could produce significant one per rev vibratory airframe responses. A constant track variation with blade angle would have required only a simple blade push rod adjustment as is normally done on all Sikorsky pre-tracked production blades, and the blade stenciled accordingly. However, even a linear track variation with blade angle would have posed some serious problems. As it turned out, the total non-constant out-of-track was only 9/16" maximum for a thrust per rotor variation from zero to 11,000 lb. These excellent results were attributed to the very close tolerances and high degree of repeatability achieved with the use of hard tooling throughout the major sub and final assembly operations. As a result of these efforts there was no evidence of any one per rev vibration throughout the subsequent test program.

TEST PROGRAMS

Throughout the 4-1/2 year program a large variety of tests were run to determine such things as materials properties and processing requirements, environmental characteristics, and ballistic resistance. The major test programs and their results are covered in the following sections.

ENVIRONMENTAL TESTS

Rainfall, humidity and altitude cycling tests were performed in an environmental chamber on samples of the aluminum honeycomb-foam-fiberglass sandwich used in the blade construction to ensure that moisture could not become entrained in the blade under severe service conditions and create a one per rev unbalance with subsequent separation of the honeycomb core from the skin because of corrosion of the aluminum.

BALLISTIC RESISTANCE

Figure 14 shows the exit damage caused by several 7.62 mm ball rounds fired at a range of 70 meters. The exit damage is more severe than the entry damage. The three holes showing the least damage were caused by rounds entering at zero obliquity. The fourth hole which was created by a round entering at 30° to the blade surface, exhibits the worst damage. Even so, the damaged areas are relatively restricted and very little spall was created. A simple foam-fiberglass-epoxy resin kit was used to make the field repair shown in Figure 15. No honeycomb is needed for the repair and blade balance is not affected since the replacement material weights are sufficiently close to those of the parent substances. As mentioned earlier, spar holes are difficult to create near the heavy root end, and of smaller consequence outboard where working stresses are low. In any case, although the blade would have to be replaced if the spar were extensively damaged, there is sufficient residual strength in the spar after such damage has been incurred to permit the aircraft to return to base. Any opening created by a projectile or fatigue crack would of course be detected by the BIM® crack detection system described earlier.

CONTROL SYSTEM SPRING CONSTANT DETERMINATION

In addition to the normal control system proof and operational tests, it was necessary to determine the spring constant of each rotor's control system to ensure the absence of classical flutter. The very high torsional stiffness of the blade could have been easily negated by a soft control system spring in series; control system slop was also held to a minimum. It was intended to allow the slop to build up during the subsequent test program to determine its effect on system stability and response. Results of these tests indicated that both rotors would have a satisfactory flutter margin throughout the entire range of air speeds, rotor speeds and advance ratios to be tested in the wind tunnel.

VIBRATION TESTS

A ground shake test was performed on the ABC rotor system module to determine its natural modes of vibration and the proximity of each natural frequency to the 1 p and 3 p rotor frequencies. In addition the transmissibility of the rotor-module system was determined.

For the conduct of the shake tests, the rotor blades were removed and replaced with an equivalent weight which included a unidirectional shaker on the upper rotor head. The module weight and inertial properties were similar to those expected for the AMES wind tunnel test. The tunnel supporting strut stiffness was also simulated. Early analyses had indicated the possible existence of a lateral mode and a coupled longitudinal/vertical mode. A passive spring system was installed to detune the lateral mode and an active hydraulic/pneumatic servo system was designed to isolate the coupled mode.

Both systems were incorporated in a single package at the tail strut. The non-isolated configuration consisted of ball joint restraints at the forward struts and a longitudinal pin restraint at the tail strut; a CH-53A main rotor brake was installed instead of the pin at the conclusion of the test to provide a remotely controlled lock/unlock system for the active isolator during the wind tunnel testing. The results of the vibration testing are presented in Figures 16 and 17.

TURBINE TEST BED FUNCTIONAL TESTS

To demonstrate the structural adequacy of the entire rotor-module system prior to actual full scale testing in the 40 ft x 80 ft AMES wind tunnel, the system was operated throughout the load ranges anticipated in the wind tunnel program on a turbine test bed facility illustrated in Figure 18. Thrusts up to 22,000 lb and interrotor moments of 74,000 ft-lb were achieved, although the high moments were obtained by the fundamental one per rev cyclic control input whereas under actual forward flight conditions both fundamental and second harmonic loads would be generated by the asymmetric airloading. The control console, instrumentation and interrotor tip clearance systems were checked during this portion of the program. A total of 37 hours was accumulated on the turbine test bed of which 27.5 hours were run in the coaxial rotor configuration. Approximately 2.5 hours were run above the rotor red line speed of 350 RPM in the coaxial configuration.

Both rotating and non-rotating rotor stability tests were performed on single and dual rotor systems to confirm analytical natural mode frequencies and the absence of instabilities. Step and sinusoidal (0 to 10 cps) inputs were applied to the rotor through the collective and lateral cyclic control systems to provide an indication of the rotor system damping characteristics. Step inputs up to ± 1 degree of rotor blade pitch angle were used with various initial settings of rotor lift and shaft moment combinations.

The step input excitations resulted in no noticeable transient responses, thereby giving an indication of good rotor system damping characteristics. The sinusoidal excitations were run continuously and slowly from 0 to 10 cps during which time only minor amplitude build-ups were observed as expected when the forcing frequencies crossed the blade natural frequencies. The data from these tests were used to modify a multibladed normal modes computer program that was used to determine the rotor response at the advance ratios anticipated in the full scale tests in the AMES wind tunnel.

At the conclusion of the turbine test bed program, the entire module assembly including main rotor blades was airshipped to the NASA-AMES facility at Moffett Field, California.

FULL SCALE WIND TUNNEL TESTS

Figure 19 shows the complete ABC rotor module and blades assembled in the 40 x 80 ft NASA-AMES wind tunnel. The shiny metal units on the floor house six separate sensing units for the Chicago Aerial blade trackers that were located around the rotor azimuth at the six blade passage points and which were modified to measure rotor separation instead of blade track. Figure 20 illustrates the console used to control and monitor the rotor system. The closed circuit TV receivers were used both as a visual display for the rotor system during tunnel operation and as a back-up rotor spacing monitor. The rotor separation was observed by following the blade tip path traces formed on the TV screen by the rotor blade tip lights. In the illustration some of the crew can be seen working on the rotor head. One of them is actually sitting on a blade. This was a routine method of inspecting and servicing the rotor hub/shaft and instrumentation, the crewman propelling himself around the head by pushing on the other rotor. The center section of the console contains the control switches and monitors. Included in this latter group are the individual rotor and total system yaw, roll and pitch moment measurements. The basic control positions were measured by potentiometers mounted on the feathering hinges. The output signals were resolved along the module axes to provide direct readout of total system longitudinal, lateral and collective control as well as indications of differential longitudinal, lateral and collective pitch. Interrotor blade tip path separation sensed by the modified optical blade trackers was recorded on the six vertical meters located on the right hand console. Two banks of 16 vertical reading meters each were installed on the left hand console to indicate the steady and half amplitude vibratory loads and stresses at the 16 most critical points throughout the rotor dynamic system. These included rotor blade stresses as well as rotating and stationary control loads. These measurements, in addition to approximately 25 others on both rotors, were also permanently recorded on magnetic tape. Several vibration pickups located on the rotating portion of the rotor head and the stationary module were included.

Figure 21 is a map of the basic wind tunnel test conditions exclusive of the rotor control variables. These were varied to provide both trimmed and untrimmed total moments and interrotor moments throughout a range of rotor thrusts up to 23,000 lb at various rotor shaft angles representing flight regimes from forward propulsion through autorotation. Figure 22 is a map of the flight ranges tested. Maximum values of lift and propulsive force were limited by blade stress. The maximum speed of 180 knots and the minimum speed of 80 knots were constrained in the first instance by installed tunnel fan power, and in the second case by tunnel choking. The maximum advance ratio was limited by a combination of blade stress and vibration caused by having to operate the rotor at speeds coincident with the first flatwise and edgewise modes as well as a principal mode of the wind tunnel drag balance. Operation at airspeeds between 140 and 160 knots was precluded because of a roughness condition peculiar to the wind tunnel. It had been planned to investigate higher advancing blade tip Mach numbers and one additional rotor phasing configuration (interrotor blade passage at $\Psi = 0°$ was the only point checked), but wind tunnel schedule limitations prevented further testing.

During the course of testing at the maximum tunnel speed of 180 knots, a major portion of the lower rotor slip ring assembly separated from the hub/shaft. This section contained, in addition to a great deal of wire and cabling, four electrical junction boxes weighing six pounds apiece. At least two of these boxes struck blades in each rotor. Figure 23 shows the damage to the upper rotor blade at 90% radius. The hole is approximately two inches wide and three inches deep. Figure 24 shows the same damage looking directly into the leading edge; the actual junction box that caused the damage is shown underneath but is turned sideways to illustrate the "half-moon" dent caused by the blade strike. Figure 25 is a photograph of the lower rotor blade that sustained the least damage. The junction box is shown on top of the blade for comparison. It is of interest to note that all of the aerodynamic trim tabs were completely bent out of shape. Despite this, and the severe damage to both rotors, when the incident occurred there was no change in vibration level or stresses throughout the 1-1/2 to 2 minutes required to "land" the rotor, i.e. shut down the rotor and tunnel. Subsequent examination of the data revealed only a sharp peak in edgewise stress at the moment of impact followed by a return to the levels that obtained just before impact. This leads to the possible conclusion that rotor blades of sufficient rigidity may not require aerodynamic balancing. Similarly, propeller moment slopes and even blade tracking requirements may also be eliminated. The use of accurate, hard tooling is a prerequisite for elimination of the tracking requirement, since close dimensional repeatability, particularly with regard to angular twist, is paramount in minimizing the lift slope differences between blades.

Finally, despite the extensive damage to both blades, they are completely repairable by replacing the skin, and foam and honeycomb filler. Minor damage on the remaining blades was easily repaired with field kits immediately after the incident.

WIND TUNNEL TEST RESULTS

PERFORMANCE

The basic performance characteristics and the implications of the performance results are covered in Reference 5. For the sake of continuity, however, Figures 26 and 27 serve to illustrate the large improvement in lift capability made possible by the more efficient use of the advancing blade. Figure 26 compares the results of an H-34 rotor test conducted earlier in the AMES wind tunnel and reported in Reference 6. The curves are non-dimensional profile drag polars plotted at 180 knots for optimum shaft angle. In addition, the ABC rotor data represent the optimum differential lateral cyclic pitch, i.e. the data are plotted at values of differential lateral cyclic pitch that yield the maximum lift/drag ratio consistent with acceptable stresses. Some approximate values of total lift have been spotted on each set of curves. Two salient features of the ABC rotor can be deduced from these curves:

1. The lift of the ABC rotor at its best lift/drag ratio is always at least twice that of the H-34 rotor at its best lift/drag ratio. The lift for the best lift/drag ratio is found at the point of tangency of a straight line between this point and the origin.

2. Above the best L/D point the conventional H-34 rotor efficiency drops rapidly because of the onset of retreating blade stall; stresses and vibration increase correspondingly such that power, stress and vibration limits are reached simultaneously at about 13,000 lb of lift. The ABC rotor on the other hand demonstrates a continually increasing lift capability even well above the best L/D point because of the absence of retreating blade stall. The maximum lift of 23,000 lb was limited only by stress. Since the rotor system was designed for an aircraft of 14,500 lb gross weight, this represents a load factor of nearly 1.6 g's.

In addition, the data further show that at 180 knots the ABC rotor can lift nearly 2.5 times as much as the conventional H-34 rotor for reasonable amounts of propulsive force, e.g. for a propulsive force of 1,050 lb representing an equivalent flat plate of 10 ft^2, the H-34 has a maximum lifting capability of 7,070 lb where the ABC rotor can lift 17,100 lb. A portion of this lift capability can be traded off for increased forward propulsion thus permitting higher speeds with the same aircraft, or the same speed with a larger machine.

Figure 27 shows the efficiency of the ABC rotor tested at AMES as a function of lift capability. The wind tunnel test results are compared to results of flight tests of the CH-53 helicopter; these data are plotted at 135 knots which represents the best L/D air speed for both the CH-53 and the particular ABC rotor tested. The best L/D speed for an ABC rotor can be varied over a wide range by judicious selection of airfoils and rotor blade geometry. The lines of constant D_P/L represent parasite drag normalized with respect to lift and are indicative of the propulsive capability of the rotor systems. As an example, for a D_P/L of .072 the lift/drag ratio of each rotor is approximately equal to 6.8; however, the ABC rotor has nearly twice the lifting capacity at this L/D. Moreover, if more propulsive force is required, e.g. if D_P/L is increased to .10, the conventional CH-53 rotor lift capability falls off rapidly whereas the ABC rotor lift capability, being very high at the outset, can sustain a reduction while actually increasing in efficiency. The design gross and empty weight lift coefficients for the current ABC rotor are shown on the graph along with typical mission and empty weight lift coefficients for the CH-53.

Figure 28 shows the theoretical and actual maximum rotor lift coefficient as a function of advance ratio.

Finally, Figure 29 illustrates the beneficial effect of reducing tip speed and the advancing blade tip Mach No. for the actual rotor tested at AMES at an aircraft gross weight of 14,000 lb. Again, the high lift capability of the rotor is used as a "negotiable security", which in this case is used to purchase increased lift/drag ratio. The decreasing L/D with speed is caused principally by the high negative blade twist (-10° nonlinear). In fact the AMES test results indicate that the -10° is too negative even for the original design speed of 230 knots. A blade designed specifically for speeds above 250 knots would in all probability have no twist. For example, the present rotor produces a download on the retreating blade at an advance ratio of .91, equal to 1/6 of the total rotor lift. By eliminating all twist the L/D ratio could be increased by 1/3. The loss in hovering performance incurred thereby could be largely regained by increasing the taper ratio from the present 2:1. Further increases in high speed rotor performance can be achieved by employing stiffer materials such as carbon fiber composites to permit reduction of the root end airfoil thickness which is presently 30%. Full scale wind tunnel tests have shown that the use of double ended, cambered airfoils could also increase the lifting efficiency as much as 50%.

LOADS, MOMENTS AND STRESSES

Loads, moments and their resulting stresses were generally as predicted, i.e. the fundamental blade stresses are a direct function of the lift offset radius on the advancing blade, and the higher harmonics of stress that are generated by the asymmetrical airloads in forward flight are in reasonable agreement with the results of the normal modes analysis. Figures 30 and 31 illustrate, respectively, both measured and predicted values of higher harmonic flatwise and edgewise stresses. In the areas of highest stress the analysis is conservative. Control loads followed a similar pattern with no instabilities observed. In this latter regard it is worthy of note that no structural instabilities of any kind appeared throughout the entire program including the .91 advance ratio condition where it was necessary to operate simultaneously on the first flatwise and edgewise modes.

Tip deflections, as might be expected, are a function of combinations of the principal variables, viz. shaft angle (rotor angle of attack) lift (collective pitch), interrotor rolling moment (differential cyclic pitch) and advance ratio. The -10° non-linear blade twist further complicates the situation. Generally, tip separation between rotors is a minimum at the best lift/drag ratio where the difference in lift between the advancing and retreating blades is a maximum; peak stresses also occur in this region. These conclusions apply for a given lift. Maximum load factors within the stress limits were achieved at off-optimum control settings. Figure 32 tabulates some typical values.

STABILITY AND CONTROL RESULTS

Figures 33 through 35 represent the static stability derivatives of the 40 ft diameter rotor normalized for an aircraft of 8,700 lb gross weight and a pitching inertia of 15,300 slug-ft^2. A typical conventional rotor has been plotted for comparison. Of particular interest is the relative slope of M_α in comparison with M_{A1S}. At 150 knots, for example, M_{A1S} is approximately 3-1/2 times the value of M_α. Since the aerodynamic damping is basically related to M_{A1S} and the gust sensitivity is a function of M_α, it is anticipated that an ABC rotor helicopter would be reasonably insensitive to gusts while exhibiting good pilot response characteristics.

ABC ROTOR APPLICATIONS

The ability of the ABC rotor to maintain its lift over a wide range of speeds makes it ideally suited for high speed applications. The demonstrator aircraft shown in Figure 2 represents a machine of 14,500 lb gross weight with a maximum level flight speed capability of 230 knots at this weight employing the 40 ft diameter rotor tested at AMES.

A logical development of the ABC rotor is illustrated in Figure 36, which shows a 90-passenger transport of about 104,000 lb gross weight capable of reaching speeds up to 400 knots. This is accomplished by slowing and eventually stopping the rotor to keep the advancing blade tip Mach Number relatively constant. A variable speed transmission is not required, since the rotor, which is essentially autorotating above the forward airspeed for minimum required power, can be decoupled from the powerplants; the rotor could be slowed either aerodynamically or mechanically. Extensive sub-scale wind tunnel tests of these flight regimes were conducted at the United Aircraft Research Laboratories and are described in Reference 2. The results support the thesis that the ABC rotor system can be applied to high speed aircraft.

A second interesting application of this rotor system in the lower speed range has been proposed to the U. S. Army by Sikorsky Aircraft. Figure 37 shows the AARV (Aerial Armored Reconnaissance Vehicle) operating in a possible role. For this mission, the vehicle employs the "reconnaissance by fire technique" to determine the enemy location. Survival in this hostile environment required an armored fuselage and an extremely invulnerable rotor/dynamic system. The former is achieved with dual hardness steel armor, and the extremely rugged ABC rotor is expected to demonstrate the latter. Elimination of the tail rotor requirement further enhances the potential invulnerability of the system. The simplicity of the rotor system makes it compatible with forward area operations where environmental conditions are severe and maintenance capabilities are at a minimum.

CONCLUSIONS

Results of over 62 hours of operation of the 40 ft diameter ABC rotor at speeds up to 180 knots, advance ratios up to .91, lifts over 23,000 lb and interrotor moments exceeding 74,000 ft lb indicate the following:

- The system lift capacity is over twice that of articulated rotors up to 180 knots. Moreover, the design lift coefficients were maintained up to .91 advance ratio.
- Measured performance is in good agreement with that predicted by rigid blade theory.
- The possibility of eliminating the conventional rotor anti-torque power requirements is clearly indicated.
- The principal control derivatives measured in the 40 ft x 80 ft NASA-AMES wind tunnel are well approximated by rigid blade theory. Measured longitudinal static stability characteristics are in excellent agreement with flexible blade theory.
- Characteristics of the measured rotor blade stresses, hub stresses and control loads are as expected from earlier model tests and analyses. Minimum blade stresses occur in the rotor lift coefficient design range. Stresses at maximum lift/drag ratios are within continuous operating limits.
- Total fixed system vibration is lower than predicted in the design lift coefficient region.
- The rotor system was found capable of operating over the entire range of wind tunnel conditions without any major mechanical or structural difficulties and without evidence of divergent stresses or other instabilities.
- Sufficient data were obtained to substantiate or modify the theories and analyses necessary to design a demonstrator aircraft.

The four year program developed materials technology and fabrication techniques in sufficient depth to produce viable dynamic system hardware for a production ABC rotor aircraft.

Finally, based on the results of the entire effort, it appears that the capabilities of the ABC rotor system can be optimized and extended directly to full scale, higher speed aircraft.

REFERENCES

1. U.S. Patent No. 3,409,249 issued November 5, 1968 to R. R. Bergquist, et al assignors to United Aircraft Corporation, E. Hartford, Connecticut.

2. Cheney, M.C., Jr., THE ABC HELICOPTER, AIAA Paper No. 69-217 presented at the joint AIAA/AHS VTOL Research, Design and Operations Meeting, February 17-19, 1969.

3. U.S. Patent No. 3,521,971 issued July 28, 1970 to M. C. Cheney, Jr. assignor to United Aircraft Corporation, E.Hartford, Connecticut.

4. USAAVLABS Technical Report 68-18B, PREDICTION OF ROTOR INSTABILITY AT HIGH FORWARD SPEEDS, C. J. Astill and C. F. Niebanck, U. S. Army Aviation Material Laboratories, Fort Eustis, Virginia, February, 1969.

5. Paglino, V. M., FORWARD FLIGHT PERFORMANCE OF A COAXIAL RIGID ROTOR, AHS Paper No. 524 presented at the 27th Annual National Forum of the American Helicopter Society, May 19-21, 1971.

6. USAAVLABS Technical Report 68-3, EXPERIMENTAL STUDY OF THE PERFORMANCE AND STRUCTURAL LOADS OF A FULL-SCALE ROTOR AT EXTREME OPERATING CONDITIONS, V. M. Paglino and A. H. Logan, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, July, 1968.

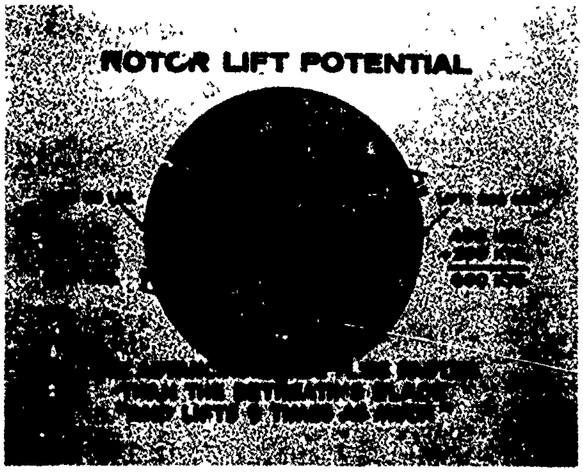


Fig. 1

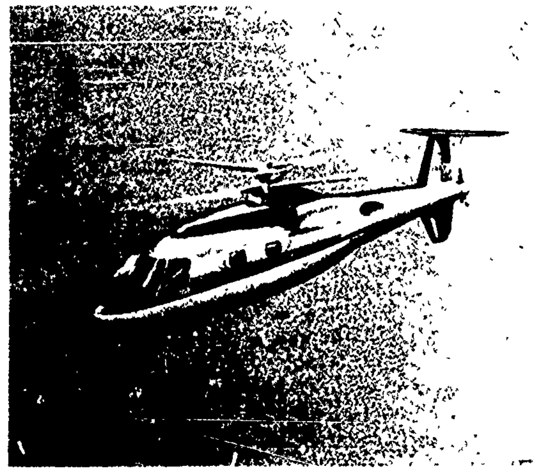


Fig. 2 ABC Rotor Demonstrator Aircraft

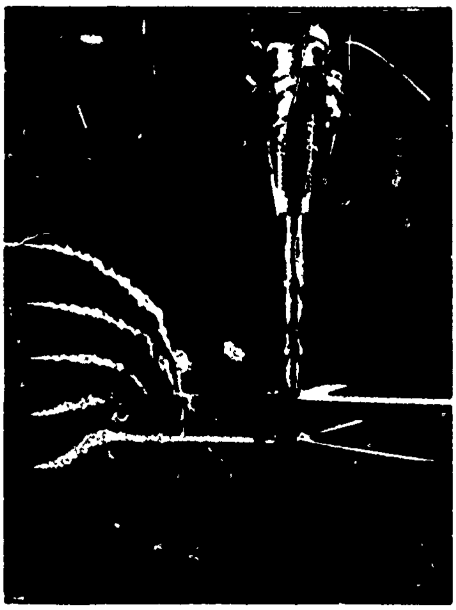


Fig. 3 Hover Model Smoke Tests

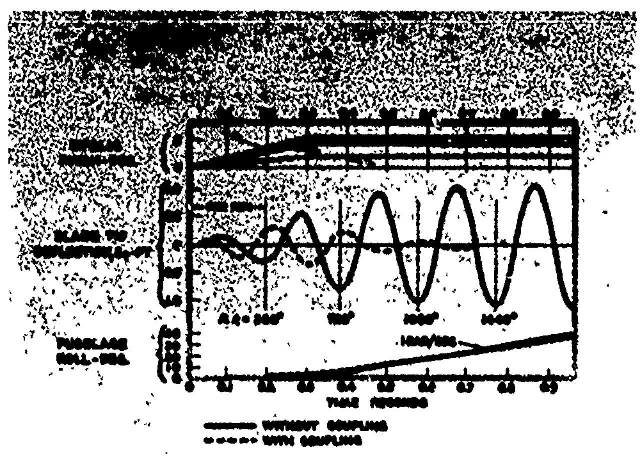


Fig. 4 ABC Rotor Control Coupling

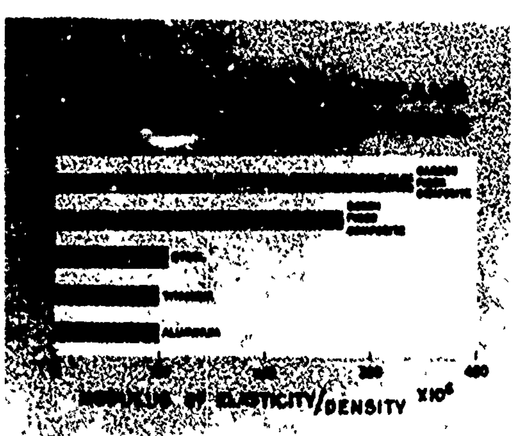


Fig. 5

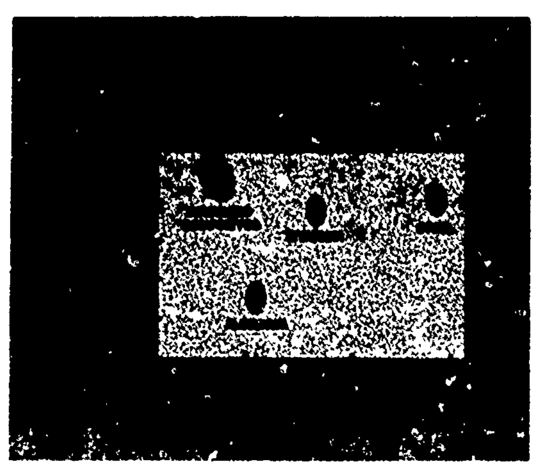


Fig. 6

Fig. 7 Boron/Epoxy Blade Spar Tip

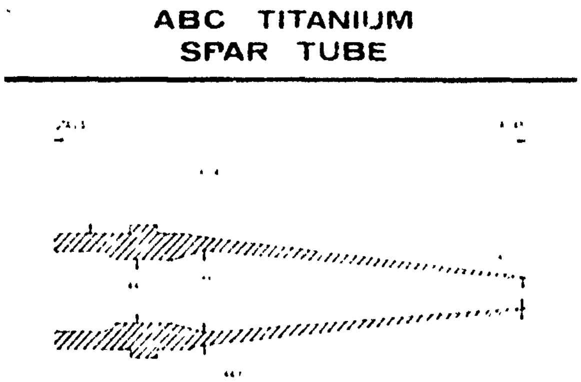


Fig. 8 Blade Spar Cross Section

Fig. 9 Completely Formed Spar

Fig. 10 Final Blade Assembly

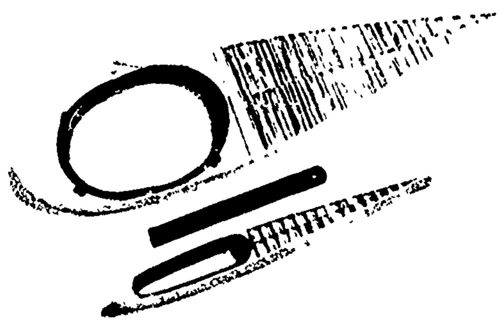


Fig. 11 Blade Sections: Top Sta. 60, Bot. Sta. 216

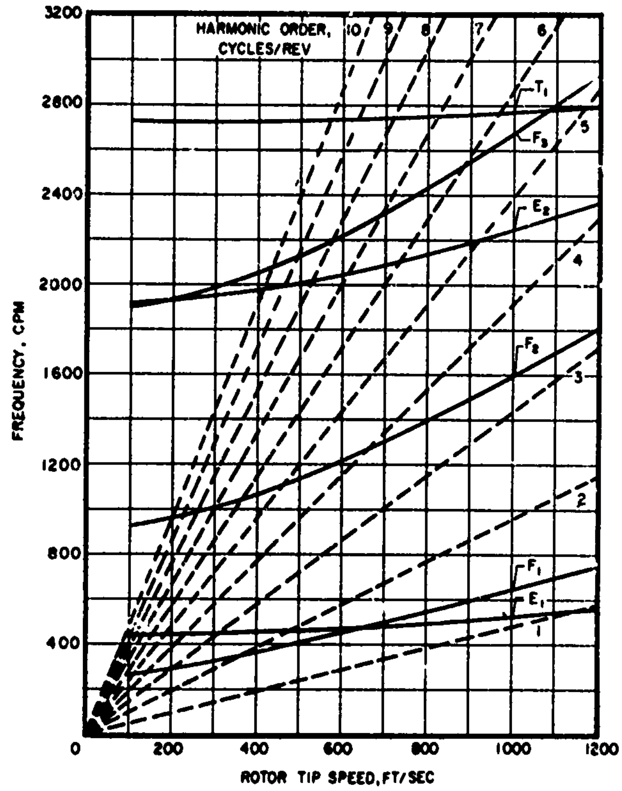


Fig. 12 Principal Blade Modes

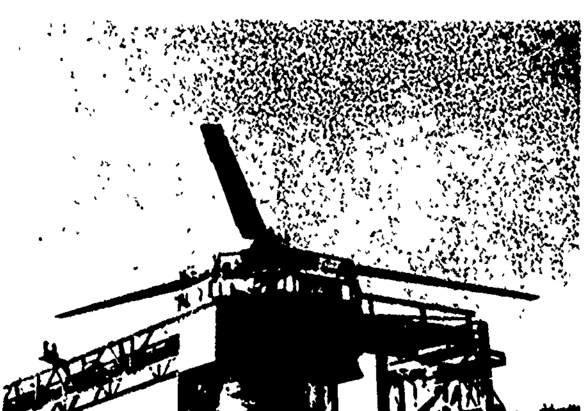


Fig. 13 ABC Rotor on 3000 HP Blade Balance Stand

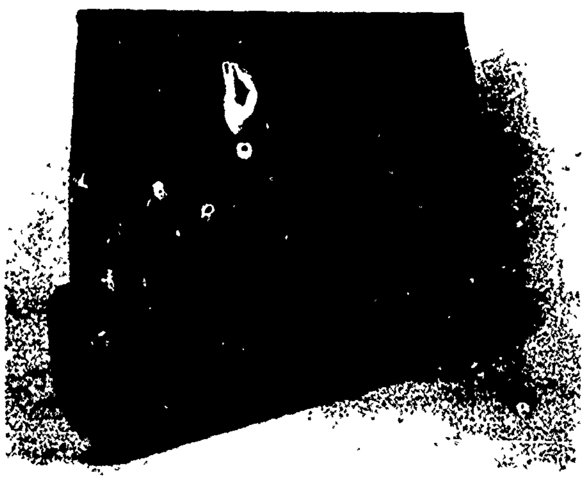
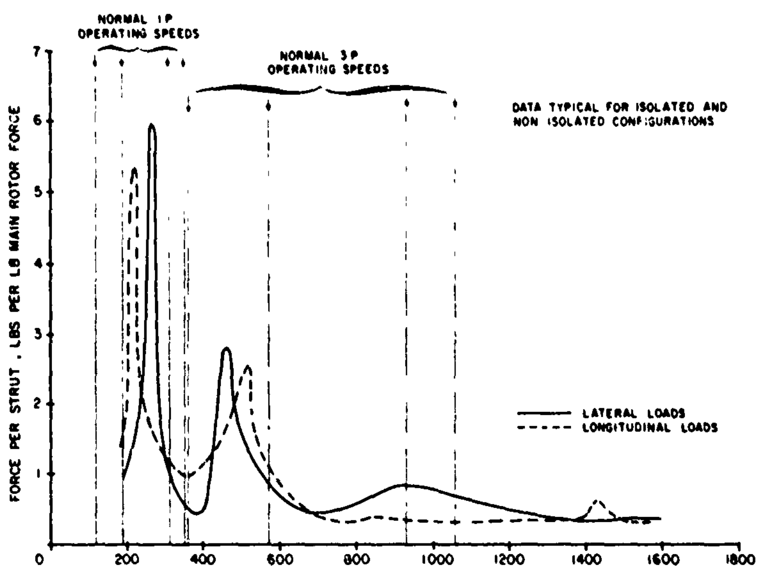


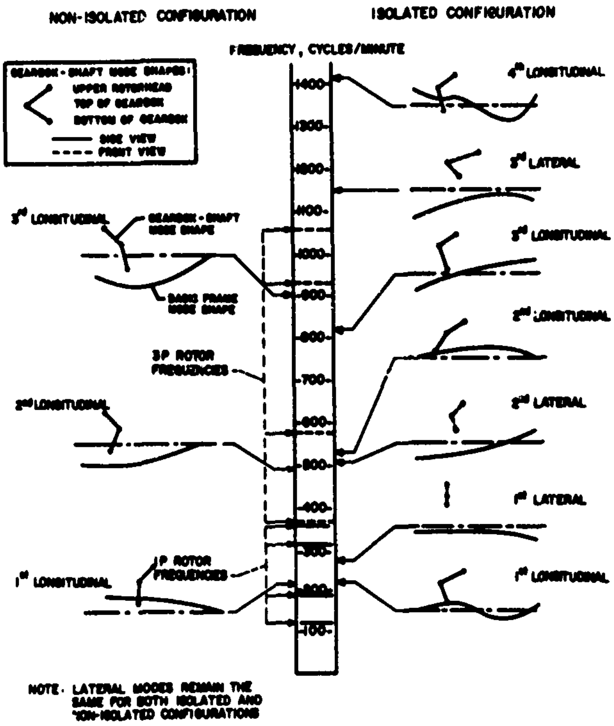
Fig. 14 Exit Damage from 7.62 mm Ball Ammunition

Fig. 15 Field Repair of Ballistic Damage



TRANSMISSIBILITY OF ROTOR INPLANE LOADS TO THE FORWARD MODULE SUPPORTS

Fig. 16



ABC MODULE PRINCIPAL MODE SHAPES AND FREQUENCIES.

Fig. 17

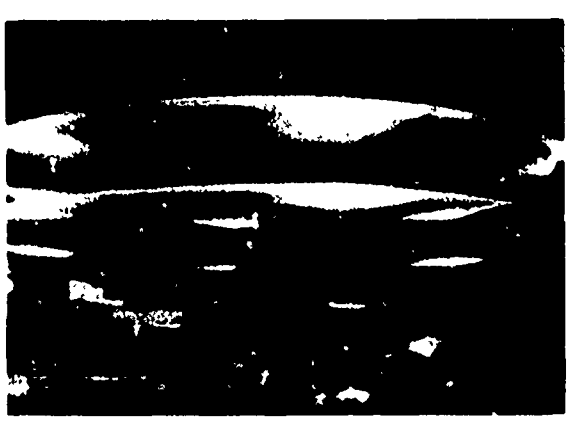


Fig. 18 ABC Rotor on Turbine Test Bed

Fig. 19 ABC Rotor in NASA-Wind Tunnel

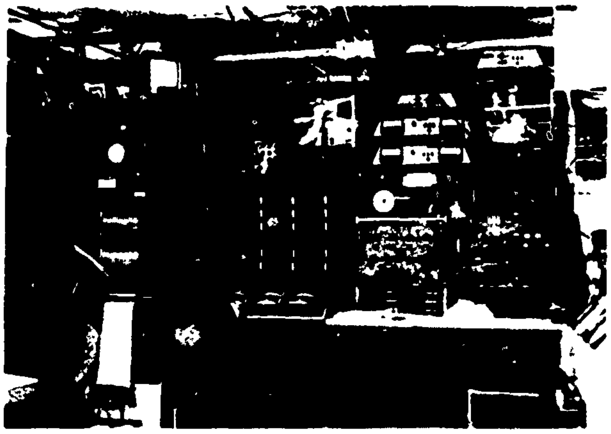


Fig. 20 Control Console and Monitor Panels

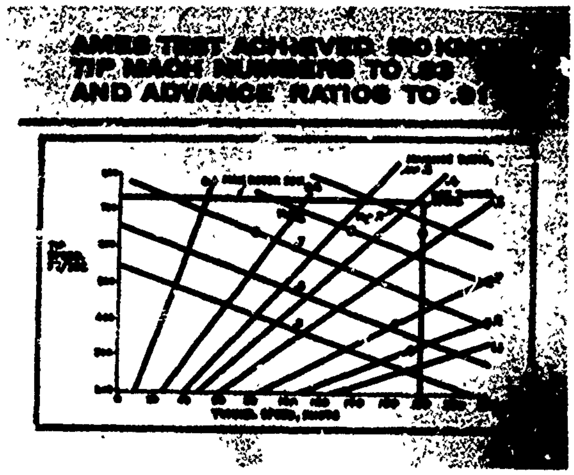


Fig. 21

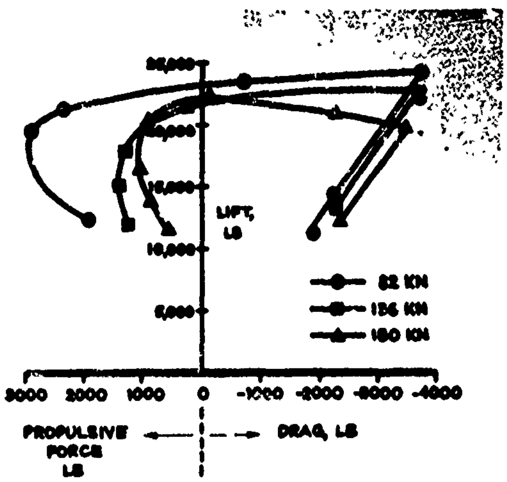


Fig. 22 Flight Range Test Map

Fig. 23 Upper Blade Damage at 90% Radius

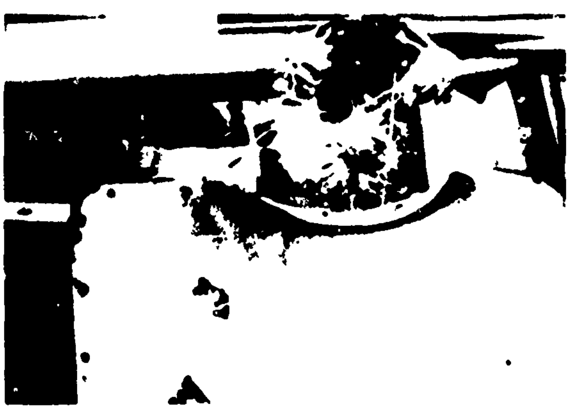


Fig. 24 Upper Blade Damage and Junction Box

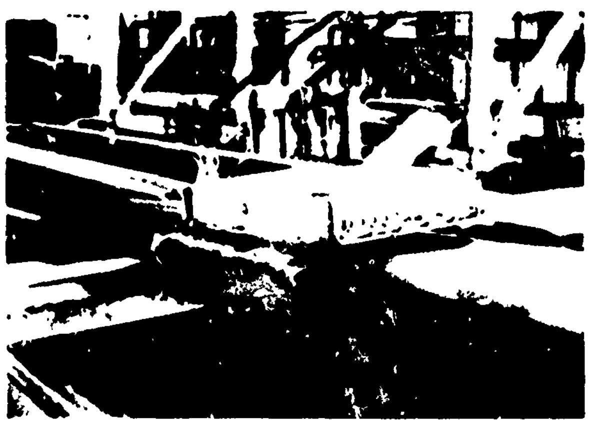


Fig. 25 Lower Blade Damage and Junction Box

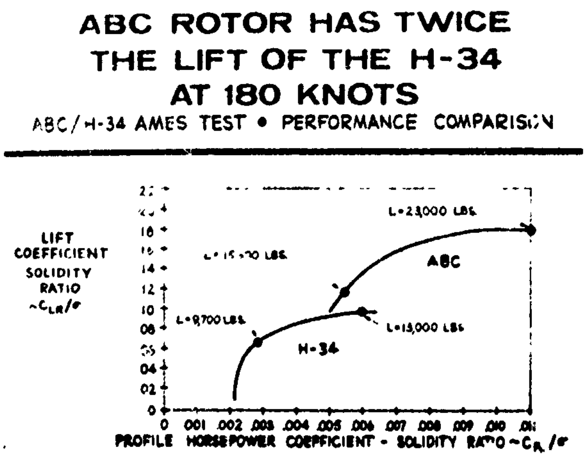


Fig. 26

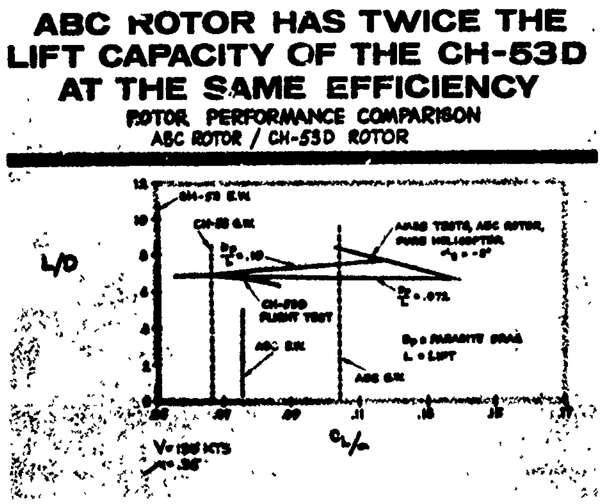


Fig. 27

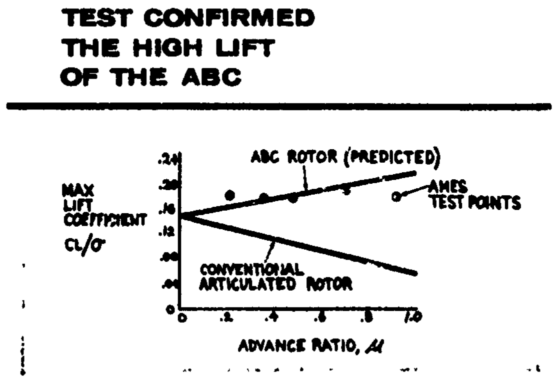


Fig. 28

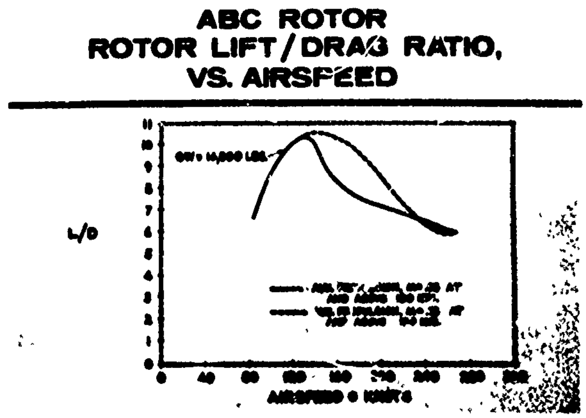


Fig. 29

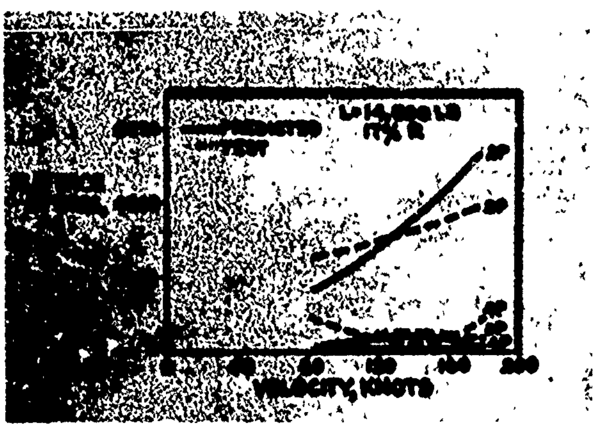


Fig. 30 Higher Harmonic Flatwise Stress

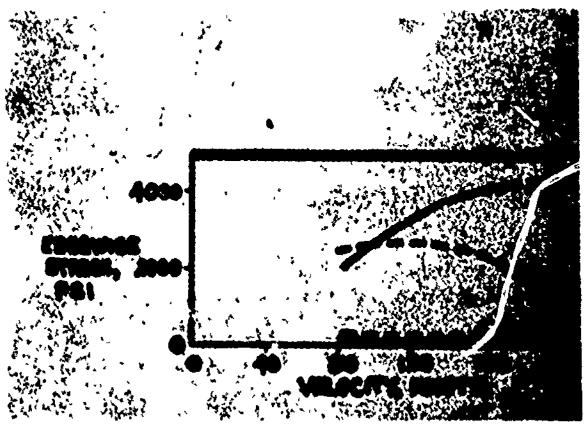


Fig. 31 Higher Harmonic Edgewise Stress

TYPICAL LIFTS, ROLLING MOMENTS & TIP SEPARATIONS

V = 136 KNOTS ● μ = .36 ● ΩR = 660 ft/s

SHAFT ANGLE, DEG.	LIFT, LBS.	ROLLING MOMENT/ROTOR, FT.-LBS.	TIP CLEARANCE, IN. CALCULATED	TIP CLEARANCE, IN. AMES TEST
-8	14,200	43,500	—	27
-8	19,300	63,000	14.3	15
-8	16,700	66,500	10.6	13
-4	13,750	33,000	—	30
-4	16,700	26,500	26.3	30
-4	19,000	53,500	—	24
0	13,100	18,000	28.5	28
0	19,150	47,000	—	22
0	15,700	47,000	18.0	20
0	21,700	72,000	11.6	10

Fig. 32

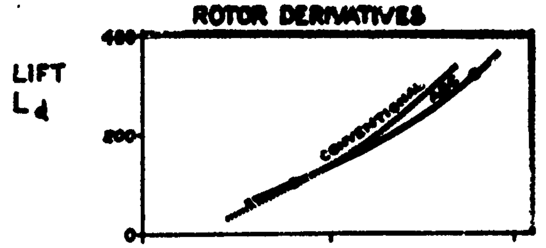


Fig. 33 Lift, Normalized by Mass (ft/sec^2/rad) vs. Speed (knots)

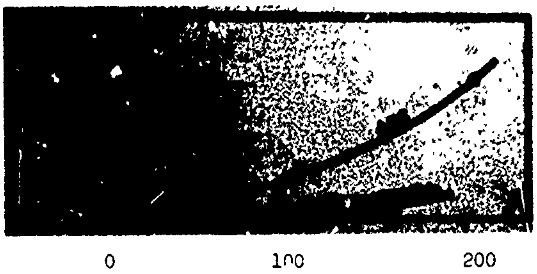


Fig. 34 Static Pitching Moment Normalized by Inertia (rad/sec^2/rad) vs. Speed (knots)

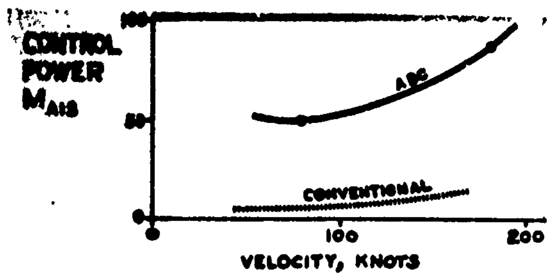


Fig. 35 Pitch Control Moment Normalized by Inertia (rad/sec^2/rad) vs. Speed (knots)

Fig. 36 ABC Rotor 90 Passenger Transport

Fig. 37 ABC Rotor Employed on Aerial Armored Reconnaissance Vehicle (AARV)

R.A.E. EXPERIENCE IN THE USE OF A PILOTED GROUND-BASED SIMULATOR FOR HELICOPTER HANDLING STUDIES

by

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Bedford, England.

SUMMARY

Two studies using a ground-based piloted flight simulator for the assessment of helicopter handling qualities are described. The first simulation, of a Westland Wessex, was performed to establish the simulation techniques required for effective representation of handling behaviour. The second study was of the Westland Lynx, and was conducted, in co-operation with Westland Helicopters Ltd., prior to the first flight of that helicopter in order to provide assistance in the early development programme.

Results of the two simulations are discussed, and the experience gained from these tests is used to suggest some requirements for valid simulation.

1. INTRODUCTION

For over a decade, the flight simulator of the Aerodynamics Flight Division, Royal Aircraft Establishment, Bedford, has been used for investigations into aircraft handling characteristics. The tasks of the Simulator Section have been threefold; basic handling research (stability and control requirements, operational techniques for new classes of aircraft, etc.), assessment of handling qualities of new aircraft during the early development stage (Concorde, A300 Airbus, BAC 221 etc.) and advice on simulation to manufacturers and other users of simulators. Until recently the work has been concerned largely with conventional fixed-wing aircraft (with two excursions into the VTOL field), particularly in the areas of take-off and landing. However, the projected use of the simulator as a tool in helicopter handling studies and the need to give advice on helicopter simulation prompted a simulator-flight comparison of the Westland Wessex, aimed at establishing the techniques required for representation of the handling behaviour of a helicopter. Once some measure of confidence in this simulation had been achieved, the simulator was used for a study of the handling qualities of the Westland Lynx, in preparation for initial flight tests.

This paper discusses both simulations and summarises some requirements for effective simulation of helicopter handling qualities in the light of R.A.E. experience in this and previous fixed-wing simulations.

2. AIMS OF THE SIMULATIONS

2.1 Wessex Exercise

This preliminary exercise was aimed at providing a measure of confidence in the use of the simulator for helicopter handling studies.

The problems to be considered were:-

(a) data required for representation of the helicopter aerodynamics;

(b) mechanization of the aerodynamic data and equations of motion in a suitable form, particularly in view of the somewhat limited computational capacity available;

(c) effective incorporation of the pilot in the control loop (visual and motion cues etc.);

(d) establishment of the areas of validity and of the limitations of the simulation; in particular, the extent of limitations imposed by deficiencies of data or environmental cues.

The Wessex was chosen as subject for this exercise, as such a helicopter was operated by the Aerodynamics Flight Division (Fig. 1) and the pilots performing the major part of the simulation work were familiar with its handling features. It was also hoped to obtain direct comparisons of tasks performed by the same pilot in the simulator and in flight. This helicopter is a modified version of the Sikorsky 5.58, built by Westland Helicopters Ltd.

2.2 Lynx Exercise

The Westland WG 13 Lynx is a twin-engined helicopter with a hingeless rotor (Fig. 2); a 4-channel automatic flight control system (AFCS) is fitted. The hingeless rotor gives higher control powers and damping than the conventional articulated rotor, but also inherently imparts a greater degree of instability to the vehicle, particularly at high speed.

Simulator tests were conducted about five months before the first flight of the Lynx (which was on 21st March of this year). The areas of study included handling with and without AFCS throughout the speed range, the ability of the pilot to cope with AFCS failures, and possible handling improvements to be derived from modification of the AFCS control laws.

3. DESCRIPTION OF THE SIMULATOR

The principal elements of the simulator are shown in block diagram form in Fig. 3. The representation of the helicopter was programmed on the analogue computer (approx. 200 amplifiers, 80 multiplications) and responded to the pilot's control inputs and to disturbances introduced by the simulator operator. Computer outputs fed the sources of motion, visual and aural cues for the pilot, to enable him to complete the simulation loop. Sixteen computed variables were recorded by two 8-channel pen recorders, and records of the pilot's and operators' commentary were made during the tests.

Several of the simulator elements are described below in more detail.

3.1 Cockpit Interior

The simulator cockpit is normally equipped with aircraft controls, but for the helicopter work these were removed; a Wessex cyclic stick, spring and trim units, collective lever with modified friction device, and yaw pedal damper were fitted. For the Lynx simulation, the cyclic forces, trim rates, pedal damping and the length of the collective lever were all altered to values closer to those proposed for the Lynx.

Figures 4 and 5 show the cockpit interior for the two simulations.

Flight instruments were limited to those relevant to the handling studies, and were in general modified from appropriate aircraft instruments. The actual instrument layouts used were not those of the real helicopters but were dictated in part by the restricted panel space in the simulator cockpit. Auto-stabiliser controls and indicators were mounted on the left-hand console for the Lynx exercise.

3.2 Motion System

The motion system, shown in Fig. 6, was originally built with two axes of motion (pitch, with some incidental vertical translation, and roll), but was modified just prior to these simulations to give 4 axes (pitch, roll, yaw, heave). The performance of the various axes is shown in Fig. 7 and table 1; the yaw performance is by far the least satisfactory, and was further aggravated by high friction which caused a sharp jerk on reversal of direction of motion.

TABLE 1

AXIS	TRAVEL	MAX. RATE	MAX. ACCELERATION
Pitch	$\pm 15^\circ$	40°/sec. up 70°/sec. down	100°/sec.^{2} *
Roll	$\pm 15^\circ$	60°/sec.	400°/sec.^{2}
Yaw	$\pm 15^\circ$	35°/sec.	200°/sec.^{2} *
Heave	2.3 ft. up 1.2 ft. down	3 ft./sec.	18 ft./sec.^{2}

* Pitch and yaw maximum acceleration are design figures; actual values not yet available.

Attempts to improve the response and eliminate the breakout jerk were unsuccessful (for reasons not yet fully understood) and the false cue given by the jerk swamped any useful cue that might be derived from this axis; yaw motion was not used for the bulk of the tests.

3.3 Visual System

The visual representation of the outside world was provided by a Redifon moving belt television system. A television camera tracks over a model of an airfield and surrounding countryside in response to the computation of the helicopter's position and attitude; the resulting view is presented on a television monitor in front of the pilot. The field of view of the system is somewhat limited (approx. 45° in azimuth, 35° in pitch) and for the Lynx simulation an extension of this field of view was obtained by projecting a shadow horizon line onto the dome surrounding the cockpit. Viewed through translucent side window panels, this division of the external world into light and dark hemispheres provided an additional source of pitch and roll attitude information.

Two television models were used, one on a scale of 1:2000 giving coverage of an area 12 nm by 4 nm up to an altitude of 1500 ft, and the second of 1:700 scale, with reduced coverage ($5\frac{1}{2}$ nm x 2 nm x 700 ft).

For the Wessex simulation, a small fan, rotating at main rotor speed on top of the cockpit and below a light bulb, was used to simulate rotor flicker. The varying light intensity was detectable through the semi-opaque side windows of the cockpit. This device was not used for the Lynx simulation as the shadow horizon used in that simulation required the same mounting points as the flicker unit.

3.4 Aural Cues Generation

Aural cues were supplied from two loudspeakers behind the pilot's seat. For the Wessex simulation a mixture of filtered white noise and one particular constant frequency was used to represent transmission and engine noise, to which was added a 1/rev. and 4/rev. beating sound to simulate blade noise. The intensity of the 1/rev. increased with vertical acceleration of the helicopter. In the Lynx simulation, the frequency of the single note was modified by computed engine torque, to indicate the rise and fall of engine speed under varying load conditions.

4. WESSEX SIMULATION - PROGRESS AND RESULTS

This section is a summary of the problems, limitations and effectiveness of the Wessex exercise, the account being divided between those factors concerning the mathematical representation of the helicopter, and those affecting the presentation of the vehicle to the pilot. However, there is, of necessity, a substantial link between the two areas.

4.1 Mathematical Model

A comprehensive simulation of a helicopter is far more complex than the fixed-wing studies previously performed on the Aerodynamics Flight Division simulator, and the first problem to be encountered was that of fitting an adequate representation of the helicopter within the limited computer capacity. It is worth quoting here from a paper[1] given to the American Helicopter Society which described a simulation of the CH46 twin rotor helicopter on a 950-amplifier computer: "Considerable manpower and effort .... and a constant struggle to remain within the limits of computer capacity and motion base travel were required to obtain a useful simulation".

For the Wessex simulation, only 200 amplifiers were available, which inevitably led to simplifications in the modelling of the representation of the helicopter, but with, it was hoped, minimum loss of fidelity in those areas relevant to the aims of the exercise. It was decided to place emphasis on simulation of handling in normal flight conditions, paying little attention to extremes of speed or manoeuvre. Perfect engine governing was assumed, giving constant rotor speed in the aerodynamic equations. A simple blade element approach, with uniform downwash distribution, was used for the main rotor; blade stall and compressibility effects were ignored. Quasi-static flapping and coning were assumed, and a much simplified tail rotor representation, synthesised from wind tunnel test results, was used. The form of kinematic equations chosen led to indeterminacies at zero airspeed; however, hovering relative to the ground was achieved by performing tests in a headwind of 10-15 kt.

Semi-empirical body forces and moments were adapted from values previously used in computer studies by Westland Helicopters Ltd. (WHL). Aerodynamic ground effects and the dynamics of the landing gear were not included in the simulation. Ground contact was simulated by inhibiting portions of the computation until thrust exceeded weight.

4.2 Comparison of Mathematical Model with Flight Data

Shortly before the simulation, some flight measurements of trims and control responses were taken in the Wessex helicopter operated by the Aerodynamics Flight Division, for the purpose of comparison with the simulator representation. It was impracticable at the time to fit instrumentation appropriate to the validation tests and the measurements were necessarily limited; in particular, no direct measurements could be taken of blade flapping or feathering or of fuselage sideslip angle. In order to obtain values of the blade angles, the pilot's control positions were recorded, and all tests were performed without autostabiliser (which would have added unmeasured inputs to the pilot's commands). In this unstabilised condition, steady trim values were hard to obtain, and responses were soon terminated by large vehicle attitudes. The amount of information derived from the flight tests was thus less than had been hoped, and much of the assessment of the simulation had to rely on qualitative pilot assessment rather than quantitative data and response comparisons.

Fig. 8 shows trim data for simulator and flight tests. The shapes of the longitudinal cyclic, collective pitch and pitch attitude data points are similar in flight and simulator, though there is a degree or so difference between collective pitch results (so far unexplained, but most likely due to instrumentation errors in the aircraft since the simulator values agree with WHL flight measurements). The lateral cyclic figure shows a difference in shape due to the assumption of uniform downwash in the simulator - a simplification having minor effect on the response characteristics of the helicopter, and employed because of lack of computer capacity.

As initially set up and flown, the simulated Wessex exhibited a marked lack of damping in yaw compared with the actual helicopter, and significant left pedal was required for trim, the amount increasing with speed whereas in real life the pedals remain close to central over a large portion of the speed range. Flight measurements of tail rotor angle and main rotor torque suggested that downwash from the main rotor was acting on the fuselage and fin to provide some of the balancing torque and hence reducing the amount of tail rotor pitch required. In the forward flight condition, the downwash through the tail rotor would also reduce the pedal angle for a given rotor thrust. 'Thumbnail' calculations on the magnitude of such forces showed that the downwash could make a significant contribution both to the shift of the pedal position and to the damping in yaw; the rather tentative values from these calculations were included in the simulator model, giving yaw damping which appeared to the pilots to be closer to real flight, and removed some of the offset pedal displacement. The plot of tail rotor angle in Fig. 8 shows the revised simulator values. The comparison is still not satisfactory, but tail rotor angle is sensitive to sideslip of the helicopter and this was not measured in flight. Further resolution of the discrepancy was not possible with the limited flight instrumentation.

4.3 Motion Drives

The philosophy behind the form of drive laws used for the motion system is one that is in common use, any minor differences between simulators being imposed by the particular limits of the motion system in use. Staples has described the philosophy in his paper[2] to the AGARD FMP Symposium on Simulation; a brief outline of the general principles is given below.

The motion sensors of the inner ear are of two kinds, the semi-circular canals sensing rotational motion, and the utricles which are sensitive to linear acceleration; choice of motion drive laws is concerned with effective stimulation of these two sets of sensors. Consider firstly, roll motion of the simulator; motion about this axis will stimulate both sets of receptors, the semi-circular canals by direct rotation and the utricles by reorientation with respect to the gravitational vector. Thus in the prov. ion of rolling cues by roll of the cockpit, undesirable lateral acceleration cues will result, and conversely, use of the motion to provide a lateral acceleration cue will involve spurious rolling sensations. This conflict is resolved (or more correctly, a compromise is achieved) by considering the initial motion response after a disturbance to be of prime importance. The pilot's first indication of an external disturbance or of the magnitude of response to a control input is from the motion - he detects acceleration. This will be followed by cues of velocity and position from visual sources (outside world, instruments). Then, if the initial motion cue is correct, subsequent false cues imposed by the simulation limits will to some extent be masked by the correct visual cues that the pilot is by then receiving. With this principle in mind, a suitable roll motion law is*:

$$\text{cockpit roll angle} = K_1\left(\frac{\tau_1 s}{1+\tau_1 s}\right) \times \text{aircraft roll angle} + K_2\left(\frac{1}{1+\tau_2 s}\right) \times \text{aircraft lateral acceleration}$$

The first term in this expression gives an initial roll response like that of the real helicopter (but reduced by gain K_1), but then washes out the signal so that for sustained back angle the simulator cockpit returns to wings level and does not expose the pilot to a false lateral acceleration cue. The second term provides roll angle as a source of lateral acceleration cues, but is lagged to minimise the false rotational cues generated in reaching this roll angle. Inevitably, high frequency components of lateral acceleration are lost (but those could be replaced if sway motion were available).

A directly analogous approach is used for the pitch motion and a similar emphasis on initial cues leads to a heave drive law of

$$\text{cockpit motion} = K_3\left(\frac{\tau_3 s}{1+\tau_3 s}\right)^2 \times \text{aircraft vertical displacement}$$

The motion scheme outlined above has been used with success by R.A.E. in simulation of fixed wing aircraft, and was similarly used for the helicopter studies. Due to the limited travels available and to the desire to avoid rapid washout or large false cues, there has to be some compromise between gain K and time constant τ for the various axes; values were optimised by pilot assessment. For these simulations the values were approximately as given in the following table (there were minor differences between the two simulations and variations were tried for particular tasks):

TABLE 2

AXIS	USED FOR ROTATIONAL CUES		USED FOR LINEAR CUES	
	GAIN K	TIME CONSTANT τ SEC	GAIN K	TIME CONSTANT τ SEC
Pitch	0.7	2	1	1.5
Roll	0.35	1.5	0.5	1
Heave	-	-	0.5	0.5

The signal used to provide aural cues of 1/rev. and 4/rev rotor noise was also fed to the pitch motion to give an impression of rotor vibration.

4.4 Pilots' Comments

The following section is a compilation of comments made by pilots during and after simulator trials. Thirty-six pilots flew the simulator; of these, 12 were familiar with the Wessex.

In preparing this simulation, the original intention was to consider only the higher speed end of the flight envelope. The limited motion capability, restricted field of view and poor performance of the moving belt display at very low speeds led to the belief that simulation of the hover would probably be unsatisfactory; however, hovering was included for completeness and to give a full assessment of simulation limits. It is not surprising therefore, that the bulk of criticism was directed at the simulation of flight near the hover. In particular, height judgement from the visual display was poor, and the limited heave motion did not give the pilots sufficient awareness of the magnitude of their collective pitch inputs. There was mechanical slack in the drive to the television belt, masking a change from forward to rearward flight until this slack was taken up, by which time the helicopter could have picked up a significant speed (it should be noted that this TV system was not intended for rearward flight). Pilots commented that, "once off the ground the major problem was height judgement due to the poor visual cues" - "it was impossible to judge the last 15 feet" - "it is difficult to know when you have stopped moving".

Another area of adverse comment was that of yaw control, in spite of the modifications to the aerodynamic representation. The incorrect pedal position with change of speed was criticised and the pedals were described as 'sloppy', 'light' and 'loose'. The yaw response felt underdamped compared with that of the real helicopter.

* In implementation on the computer, the first term is rearranged to give $K_1\left(\frac{\tau_1}{1+\tau_1 s}\right)$ x aircraft roll rate, to avoid problems of axis resolution.

In other than these two areas, criticism was generally minor. It was very pleasing to find the overall acceptance of the simulation by all but one or two of the pilots, particularly in view of their wide range of backgrounds and, for a high proportion of them, the absence of previous simulator experience. All felt that the impression of the size of the helicopter was correctly conveyed; those who were familiar with the Wessex were more particular in both their criticism and their praise. Stabilised and unstabilised flight conditions were evaluated and "the real relationship (between the two conditions) was almost universally reproduced". Although the visual and motion deficiencies made precise hovering harder than in real life, it was still felt that this "didn't invalidate the usefulness of the simulation for this mode of flight"; assessment of cyclic control power and responce was still possible and relevant.

A number of deficiencies of the cockpit layout were criticised; "the instruments are too close, making scanning very difficult" - "the small outside world confuses me" - "I am not sitting in a normal flying position". However "the mainfold deficiencies of the cockpit were considerably more apparent when entering than when flying the simulator. Once involved in a task the fact that you are using 'normal' flying controls mostly overshadows the strangeness of the surroundings".

Most pilots were impressed by the effectiveness of cockpit motion, both for its contribution to the pilot's environment, and as a source of controlling cues. Without motion "it is non-aircraft" - "I feel that I am sitting still with the world moving beneath me" - "control is more difficult", and "only the motion enables me to fly this aircraft in the unstabilised mode". Motion helped the pilot to gauge the size of his control inputs, resulting in reduced, but more efficient, stick activity. The rotor shake added considerably to the environment - the increase in vibration level when pulling 'g' was particularly liked.

Summarising, there was general acceptance that a useful representation of Wessex handling had been achieved, with reservations on the simulation of yaw behaviour and the hover. It was felt that the addition of yaw motion would do much to remove the impressions of underdamped and loose response (removal of pitch or roll motion led to similar impressions of poor control in those axes) and efforts were made - without success - to bring the yaw motion to a satisfactory state for the Lynx exercise.

Overall, therefore, the simulation of the Wessex enabled that of the Lynx to be approached with reasonable confidence, but any problems that might occur in yaw or hover control were to be interpreted with caution.

5. LYNX SIMULATION

This simulation was conducted in close co-operation with the test pilots and aerodynamics staff of Westland Helicopters Ltd., the WHL pilots performing well over half the total tests; pilots from the R.A.E. provided the remaining contribution. Tests were directed at the handling of the basic, unstabilised helicopter, the benefits obtained from the automatic flight control system (AFCS) and the suitability of the AFCS laws, pilot reaction to AFCS failures, and the identification of possible problem areas.

The Lynx has now started flight trials, which allows some assessment of the effectiveness of the simulation. A complete comparison is not possible at the present time, but one or two areas of particular interest will be discussed.

5.1 Mathematical Model

The model used for this exercise was more detailed than that of the Wessex simulation. The simplifications of constant rotor speed, quasi-static coning angle, and the lack of ground effect, landing gear, blade stall and compressibility effects were still present, but the model included flapping as a first order motion, non-uniform downwash, and a detailed tail rotor representation. The equations were generated in a form which avoided indeterminacies at the hover, removing the zero speed limitation of the Wessex simulation. The whole model was derived, with minor simplifications, from that used by WHL for fixed-base simulation and computer studies.

5.2 Handling of the Lynx

The stabilised helicopter, as represented on the simulator, was a pleasant, responsive machine with ample control power in all axes. The AFCS provided attitude stabilisation in pitch and roll (rate stabilisation for large roll angles), rate damping in yaw and a collective pitch channel driven by normal acceleration (the collective pitch acceleration control, CAC). The unstabilised aircraft was easily controlled in roll throughout the speed range; pitch presented no major difficulties until beyond 100 kt., when the basic instability of the rigid rotor gave a divergent but still controllable response. Control could be retained, with a high pilot workload, up to the maximum speed simulated of 160 kt. Addition of the CAC improved the stability, postponing divergence to around 120 kt. and giving easier control throughout the speed range.

Previous fixed-base tests, using the same mathematical model, had suggested a limit of controllability of around 100 kt., so the present tests gave more confidence for the initial flight trials, which were performed without stabilisation.

The Lynx has now flown with and without CAC up to 160 kt., and the pitch and roll behaviour of the helicopter has been much as predicted by the simulator tests. This close similarity between flight and simulator has given confidence in the simulator predictions, thus allowing more rapid progress in early flight tests than would otherwise have been advisable.

It is worth mentioning one or two other results of this exercise. Firstly, the pitch and roll trim rates chosen for the aircraft were assessed in the simulator and found to be far too low for pilot convenience; initial flight tests confirmed this, and trim rates are now being increased. Secondly, the two significant problems encountered in the Wessex simulation, yaw control and hover representation, again were in evidence. Height control near the hover was even harder than with the Wessex simulation because of the high gearing of the collective control and there were slight fears that the control might be oversensitive in the real aircraft. Flight tests have proved the fears to be groundless; although the collective control is undoubtably powerful, there is no tendency to overcontrol.

In the simulator, yaw damping appeared poor at the hover, a feature which again has not been evident in flight.

6. DISCUSSION

6.1 Results of the Two Simulations

The validation simulation of the Wessex received general acceptance by a large number of pilots. A useful simulation was achieved even with a relatively small computing capacity and limited visual capability. There were two major weaknesses; the simulation of yaw behaviour, and of height control near the hover. It is interesting to note that both these problems occurred in attempting close control of axes for which little or no motion cues were present. The pitch and roll motion capability of the simulator is quite good, and no significant problems were encountered in control about these two axes. There was no yaw motion - and yaw control appeared loose and underdamped at all speeds. Heave motion was very limited - and height control became a problem at the hover when fine and continuous use of collective pitch was required, and where control of motion in the vertical plane becomes largely independent of pitching motions. At speed, the collective lever is in general more of a trimming device than a continuous control, and short-term height control is achieved by pitch changes as much as (if not more than) by collective lever inputs. In the simulator it was quite easy to fly at speed at very low altitude, with adequate control of height; the pitching motions gave the pilot information about change of flight path, information which near the hover came from the weak heave cues and inadequate visual display. The major deficiencies of the simulation can thus be attributed in large part to inadequate motion cues. It is worth repeating that the limitations of the visual display were an undoubted contribution to the difficulty in hovering, but it is considered that the lack of motion is more significant. The motion provides acceleration cues which are phase advanced relative to the weak visual cues of rate and enable them to be interpreted more readily; the converse does not apply (it has been shown that powerful visual cues will disorientate pilots in the absence of corresponding motion).

The aerodynamics of the fuselage, including the interaction between the main rotor downwash and the fuselage and tail rotor, can play a significant part in the handling behaviour of a helicopter. 'Thumbnail' calculations on the effects of downwash on fin and fuselage for the Wessex suggested a damping contribution from that source as large as that obtained from the tail rotor. More accurate assessment of these effects is necessary, and the simulation has been useful in indicating a problem area and the need for more supporting data.

Initial feedback from flight tests suggests that the Lynx simulation has been valuable in giving confidence to early flight work. The good read-across from simulator to flight in terms of stability and controllability has enabled a more rapid expansion of the flight envelope than would have been advisable without the simulator tests. Although the Simulator Section is no longer directly involved in the Lynx flight development programme, contact is being maintained with WHL so that comparison of the simulation with flight results can be used to benefit future simulation work.

6.2 Thoughts on Simulation Improvements and Requirements

The above conclusions have indicated where simulation improvements are desirable. In particular, improved yaw and heave motion could prove very valuable. The requirements for angular motion can be quite modest; for example, there is little virtue in having more than $\pm 10^\circ$ or 15° of roll motion - the lateral acceleration cues generated at 15° bank are probably as large as would normally be directly required, and far larger than can be tolerated as a false cue. A similar amount of pitch is adequate unless extreme vehicle attitudes are being simulated and, if similar yaw motion drive laws are used, $\pm 15^\circ$ of yaw should suffice.

The problem axis is, of course, heave. Here one can provide only a very watered-down simulation of the real aircraft motion, except for certain limited tasks such as precision hovering. Provision of linear motions is expensive and space-consuming; as a result, most motion systems are limited to two or three feet of heave travel, which appears to be insufficient to provide the cues necessary in direct control of the vertical motion of the helicopter. However, for tasks in which control is primarily through the rotational axes, i.e. in the bulk of flying away from the hover, the representation of vehicle motions can probably be adequately presented by the angular motions.

Qualitative impressions of simulated flight with and without motion, and the difficulties experienced in those axes with limited or no motion, lead to a belief that motion is essential to produce handling results that can be interpreted with any reasonable degree of confidence. The cost of a motion system with adequate angular motions, and some heave travel, should not represent too large a proportion of the total cost of a simulation facility.

The other major deficiency encountered, an inadequate visual display at the hover, presents greater problems. The television type of display is very expensive, and for many purposes (excluding the hover) a simple presentation of attitude might be sufficient. Figure 9 shows a shadow horizon presentation similar to that used to supplement the television picture in the Lynx simulation - the costs of this device are relatively low.

For hovering flight, position information is also required, and preferably with a greater field of view than that provided by the television display. A number of shadow displays have been used in the past (e.g. by Northrop in their rotational simulator[3]). R.A.E. are at present developing a similar system for use in hovering and low speed flight; this form of display is again costly, with major problems of drive accuracy, light output, and mounting rigidity. It is worth repeating that the simple shadow presentation of attitude could be adequate for quite a wide range of tasks.

7. CONCLUSIONS

A validation simulation of the Westland Wessex has achieved overall success in the representation of handling characteristics, although flight-simulator comparison was based largely on qualitative pilot opinion and not on direct flight measurements. Two areas of simulation difficulty (yaw and hover control) were revealed, and the relevance of motion cues to these problems has been considered. It has been found that effective simulation of the handling behaviour of the Wessex was possible even with a limited computational capacity and hence a simplified representation of the helicopter aerodynamics.

The simulation of the Westland Lynx, performed prior to first flight of the helicopter, has been valuable in giving confidence to early flight tests and has made possible a more rapid expansion of the flight envelope than could have been achieved in the absence of the simulation.

In the future, it is hoped to follow this initial excursion into rotary wing simulation with general studies of helicopter handling to supplement the research work of the helicopter divisions of the R.A.E. and to develop motion and visual systems to achieve greater simulation fidelity.

8. REFERENCES

No.	Authors	Title
1	B. B. Blake N. Albion R. C. Radford	Flight simulation of the CH-46 helicopter. Paper 361, 25th Annual National Forum of the American Helicopter Society. (1969)
2	K. J. Staples	Motion, visual and aural cues in piloted flight simulation. In AGARD Conference Proceedings No. 79 (Simulation). (1971)
3	J. B. Sinacori	Application of the Northrop Rotational Simulator to helicopters and V/STOL aircraft. USAAVLABS Technical Report 70-26. (1970)

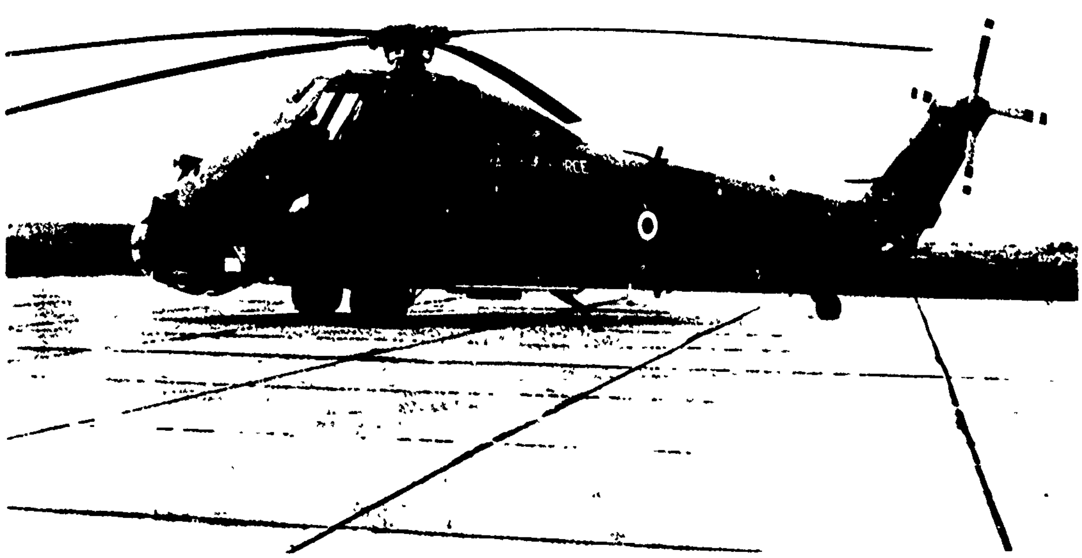


Fig 1 Westland Wessex

Fig 2 Westland Lynx

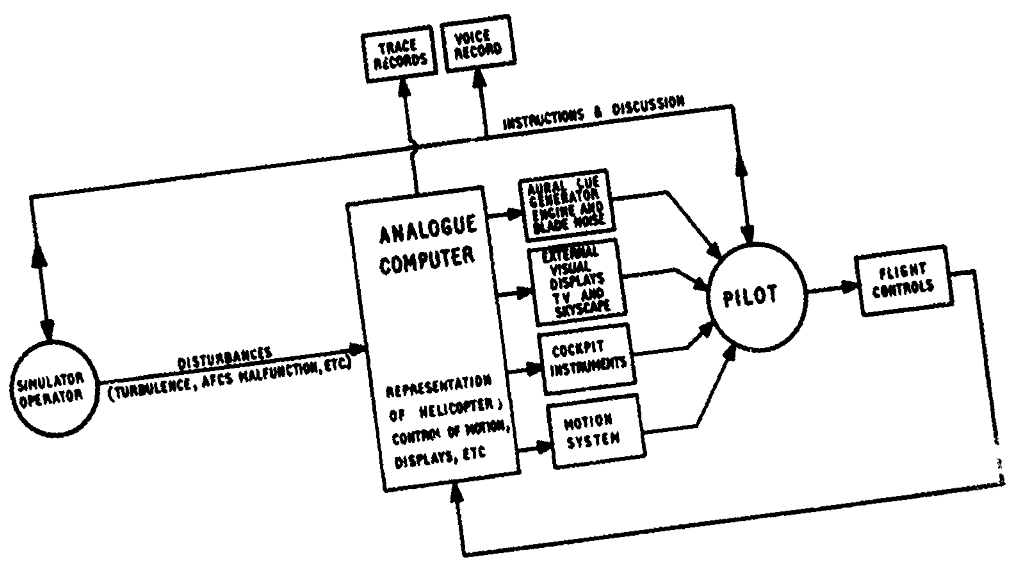


Fig 3 Block Diagram of Simulator

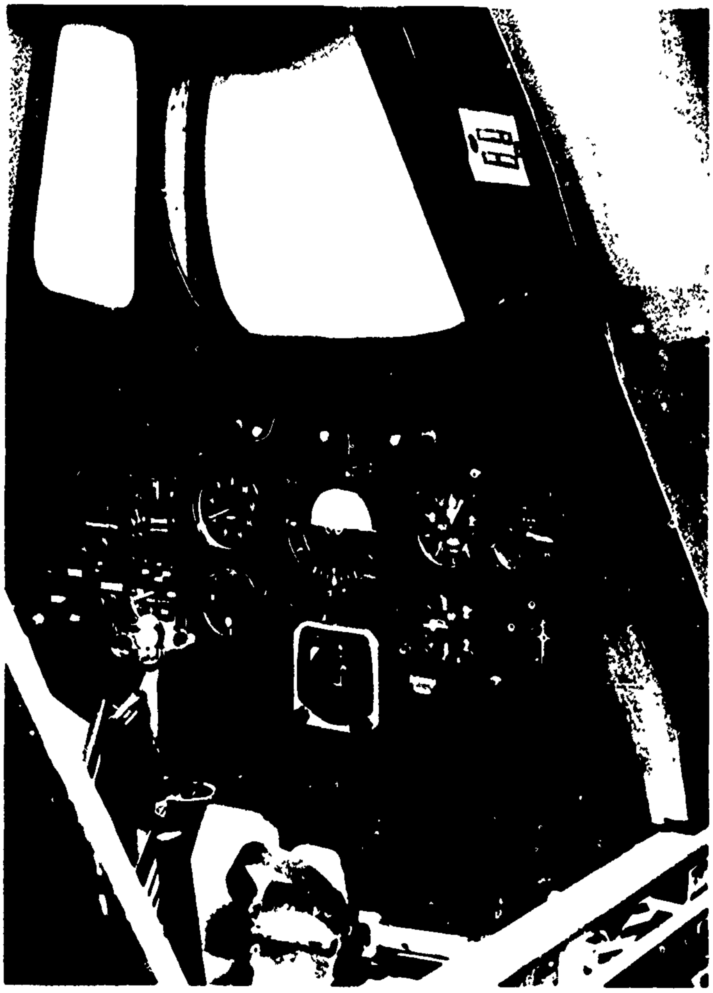
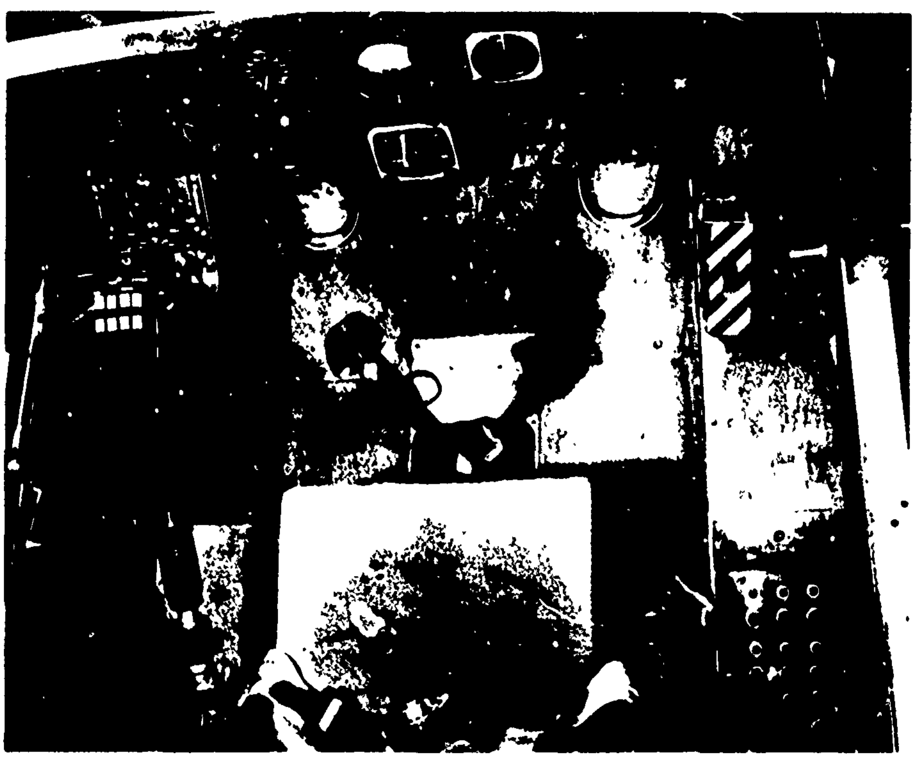


Fig 4 Cockpit Interior, Wessex Simulation



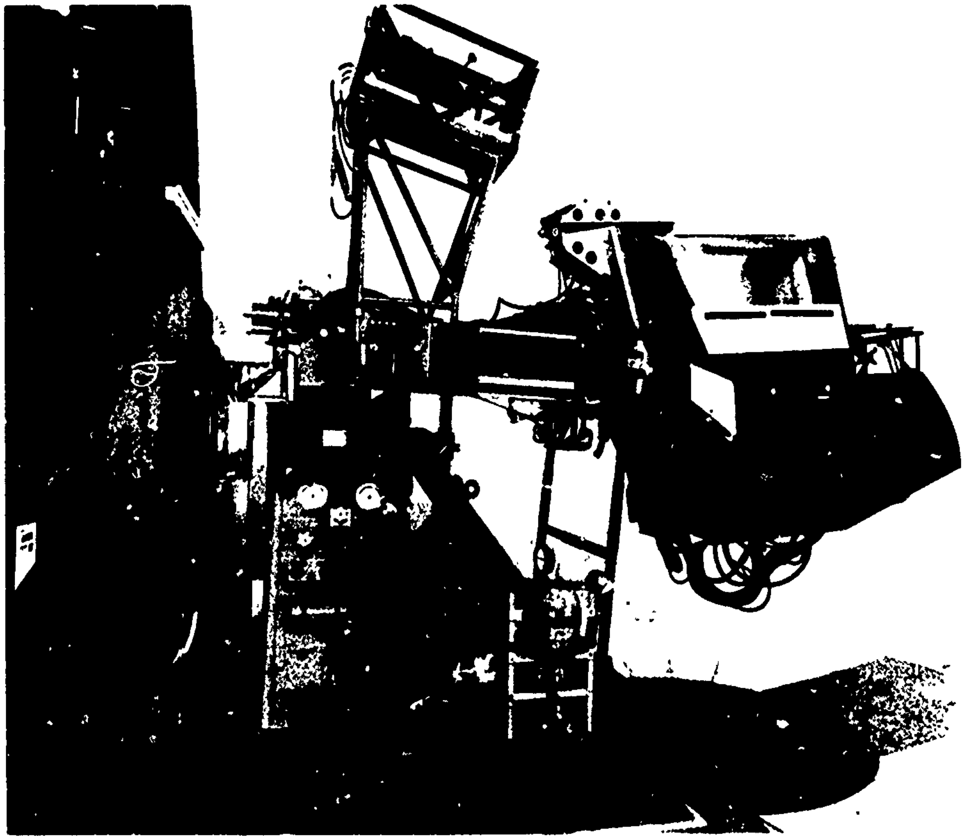


Fig 6 Cockpit and Motion System

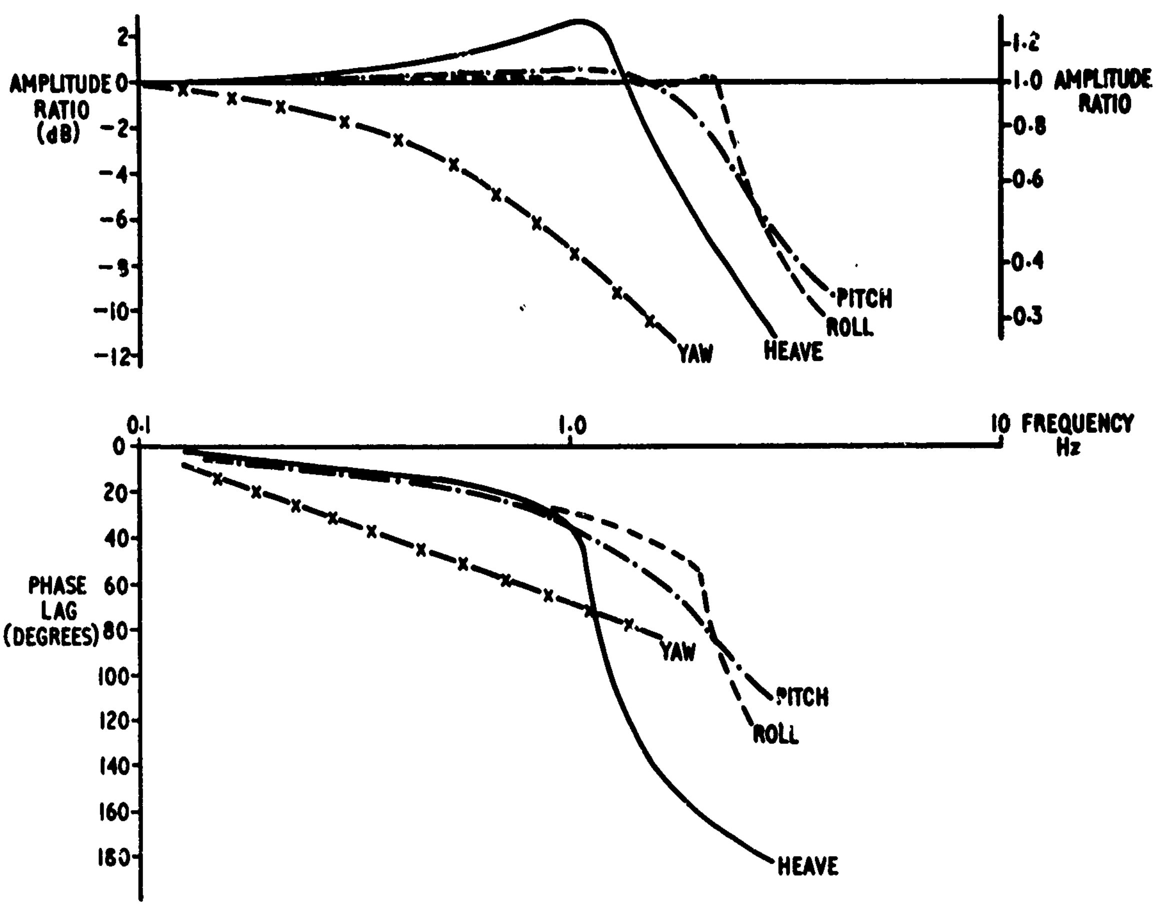


Fig 7 Cockpit Motion - Performance Characteristics

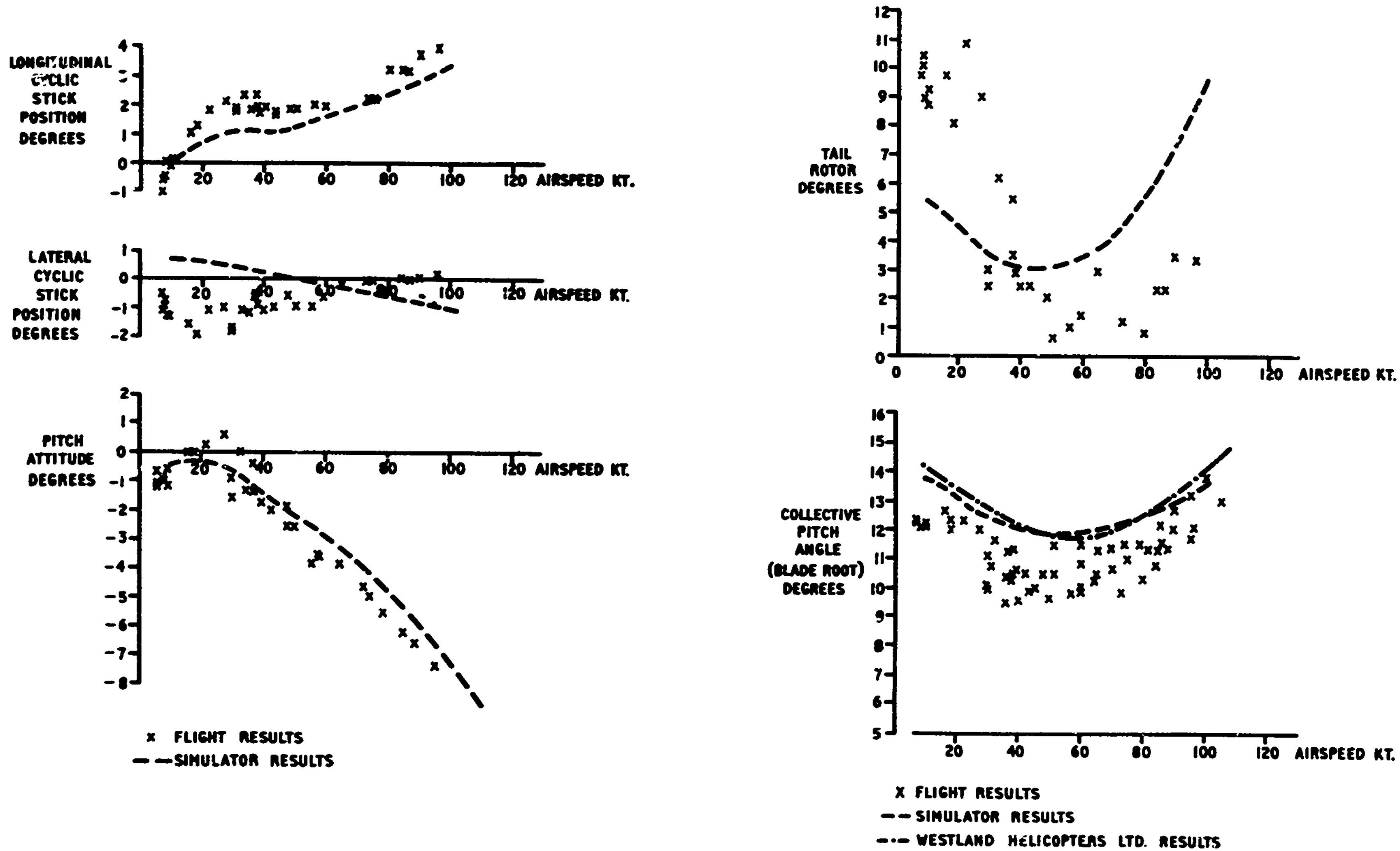


Fig 8 Wessex Trim Data - Flight and Simulator

Fig 9 Shadow Horizon Display