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NORTH ATLANTIC TREATY ORGANIZATION

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FOREWORD

This AGARD Lecture Series No.108 on the subject of Aircraft Assessment and Acceptance Testing is sponsored by the Flight Mechanics Panel and organised by the Consultant and Exchange Programme.

The Lecture Series reviews the present state of the art of aircraft assessment and acceptance testing of production aircraft. This particular kind of testing is needed to select, from a variety of offers, the best aircraft type for a mission and to check that any aircraft on the production line is identical to the type, with acceptable tolerances.

Emphasis is placed on the practical aspects of this technique in order to help the flight crews and organisations dealing with this activity. Flight test instrumentation methods are set forth that do not require the use of sophisticated ground and airborne instrumentation for data acquisition or large computers for data processing. Flight test techniques described illustrate ways to acquire acceptable results utilizing a minimum of instrumentation or no instrumentation.

J.RENAUDIE
Lecture Series Director

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QUELQUES CONSIDERATIONS SUR LES PROBLEMES QUE POSENT LES ESSAIS D'EVALUATION ET DE RECEPTION

Par J.F. RENAUDIE

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FRANCE

0 - INTRODUCTION

Mon but est d'introduire cette série de conférences consacrées aux

Essais d'Evaluation et d'Admission

J'ai choisi pour ce faire de souligner, et commenter les problèmes qu'il faut résoudre pour accomplir ce genre d'essais.

Mes connaissances concernant ce sujet proviennent principalement de mon expérience des avions français de tous types ; mais je pense que les questions techniques qui se posent sont les mêmes dans tous les pays.

Sur ces sujets je vais présenter des idées personnelles.

Cependant je dois dire, concernant les essais de réception, qu'elles doivent beaucoup aux avis que m'a fourni le Groupe de Réception, Service du CEV chargé de ce type d'essais.

J'ai personnellement participé au programme d'essais en vol d'avions bien connus : les CONCORDE de série, sortant de la chaine de fabrication. Il faut dire que j'ai eu la charge de diriger l'équipe du CEV, pilotes, ingénieurs, mécaniciens qui a participé depuis le début du programme, aux essais officiels d'évaluation des avions prototypes et de présérie, et procédé aux essais en vol nécessaires pour délivrer le certificat de navigabilité.

J'aborderai successivement les sujets suivants :

- les objectifs des essais d'évaluation et de réception
- les critères d'évaluation
- les moyens, l'organisation, les programmes de vol
- les essais de réception considérés comme un cas particulier des essais d'évaluation
- les techniques proprement dites d'essais en vol

1 - LES OBJECTIFS DES ESSAIS D'EVALUATION ET DE RECEPTION

Ces objectifs ne sont pas les mêmes pour le constructeur d'avions et le client qui veut en acheter.

Quels sont les objectifs de la société qui construit et vend un avion ?

- Pour le Directeur des Ventes le but évident des vols d'essais est de présenter l'avion au client de manière à monter en épingle ses meilleures qualités.

De tels vols d'essais sont de préférence entrepris à partir de la Base d'Essais du Vendeur, mais aussi assez souvent à partir de Bases Aériennes du Client. Ce sont ces vols qui sont appelés "vols d'évaluation" dans ce texte.

- Pour le service du contrôle de qualité du constructeur.

Les vols d'essais qui sont importants sont ceux destinés à atteindre les objectifs suivants :

- vérifier que tous les avions qui sortent en série ont les mêmes performances, et les mêmes qualités que l'avion de "type" avec des tolérances acceptables.

- vérifier que les défauts décelés sont mineurs, peuvent être aisément éliminés, et le sont finalement.

Dans ce texte de tels vols sont appelés vols d'admission

- Pour le client les objectifs ne sont pas tout à fait les mêmes.

La première question à aborder est celle de savoir de quel client il s'agit.

Il y a une différence fondamentale entre celui que j'appellerai le "Principal Client" et les autres.

Le "Principal Client" est généralement le Ministère de la Défense de la Nation du Constructeur, qui est potentiellement l'acheteur d'un grand nombre d'avions s'ils satisfont aux obligations des spécifications (performances et missions). Quelquefois le constructeur prend le risque de construire et mettre au point un avion sur ses fonds propres ; plus souvent une aide partielle est fournie à cette "aventure privée" par la Nation ; pour un certain nombre de projets une aide totale est donnée au constructeur depuis le début même de la conception et de la mise au point, ou un peu plus tard, sur des contrats passés avec le Ministère de la Défense.

Dans ce dernier cas le "Principal Client" est le seul qui peut jouer un rôle actif dans chacune des multiples phases du programme, depuis le 1er vol jusqu'à la livraison du 1er avion de série ;

généralement quelque vols d'évaluation sont effectués à chaque phase du programme par les pilotes d'essais du Ministère de la Défense (en France ceux du CEV), de manière à rendre compte des progrès réalisés et du bon emploi des deniers de l'Etat.

De tels vols d'évaluation ne sont pas seulement demandés par le "Client Principal" patronnant et subventionnant un programme, mais aussi par le constructeur lui-même, de manière à vérifier que la mise au point est conduite de manière à parvenir aussi près que possible des exigences du client. Quand plusieurs concurrents sont sur le même programme de tels vols sont essentiels pour eux pour "rester dans la course" jusqu'à l'étape finale, quand le choix du type qui sera commandé est fait.

Les clients autres que le "Client Principal" par exemple les pays étrangers lorsqu'il s'agit d'avions militaires, ne sont pas dans une situation leur permettant d'obtenir des informations complètes sur la mise au point et les progrès réalisés en direction des performances opérationnelles.

Pour ces "autres clients" les besoins sont les suivants :

- Evaluer plusieurs types d'avions de manière à choisir parmi eux celui ou ceux qui satisfont le mieux leurs exigences. Puis négocier un contrat avec le constructeur pour acquérir le nombre souhaité d'appareils.

- Accepter chaque appareil livré au titre du contrat ; ceci veut dire : faire exécuter les vols d'essais nécessaires pour vérifier que chacun d'eux est conforme à l'avion de type, sans défauts.

2 - LES CRITERES D'EVALUATION

Ces critères diffèrent suivant les missions que doit accomplir l'avion ; la liste suivante donne une idée des variations possibles des types d'appareils suivant les missions :

- avion léger pour utilisateur privé
- avion "exécutif"
- avion de transport civil
- avion de transport militaire
- avion de combat, haute altitude
- bombardier
- avion de pénétration, basse altitude
- avion de reconnaissance
- avion embarqué
- appareils à voilures tournantes
- VSTOL

Il n'est pas nécessaire de s'étendre sur les différences entre chacune des missions présentées ci-dessus.

Une fois choisie la catégorie d'avion correspondant à la mission, un certain nombre de critères doivent être pris en compte. Pour l'acheteur dont le budget est limité (et tous les budgets, même les plus élevés sont limités) les premiers points à considérer sont d'ordre financier:

- coût de chaque unité à l'achat
- coût correspondant à la durée de vie de chaque unité
- coût de maintien en état opérationnel.

L'incidence sur ces critères de la simplicité technique est évidente ; parmi les 3 critères ci-dessus le dernier ne peut être évalué sans quelques essais en vol, mais tous sont reliés aux considérations suivantes qu'il faut garder à l'esprit pour bâtir un programme d'essais en vol d'évaluation :

- type d'opération envisagée : vol à vue, vol toutes conditions météorologiques, vol de jour ou de nuit.
- qualification des équipages ; équipage minimal.
- confort de l'équipage compte tenu de la durée et des caractéristiques de la mission.
- confort des passagers (éventuellement) et bruit.
- niveau de sécurité à assurer en relation avec les règlements de navigabilité applicables.
- limites du domaine de vol (à haute et basse altitude).
- qualités de vol.
- maniabilité.
- complexité de l'avion, de ses systèmes fonctionnels et opérationnels.

et, dernière considération, mais non la moindre :

- les performances correspondant à la mission.

3 - COMPLEXITE CROISSANTE DES AERONEFS

C'est un des problèmes majeurs à surmonter pour établir un programme d'essais en vol d'évaluation et de réception tenant compte de tous les critères, sans pour autant gaspiller les heures de vol, qui doivent être strictement limitées pour des raisons économiques.

Tout en permettant le nombre maximal d'observations ou mesures chaque vol doit être construit de manière à utiliser le moins de temps possible. Une charge de travail équipage très élevée est caractéristique de ce type de vol d'essai.

Une bonne image de la complexité croissante des avions d'aujourd'hui est donnée par la figure 1 qui donne le nombre de paramètres enregistrés pendant la phase de mise au point de divers avions prototypes.

L'emploi de tels systèmes d'enregistrement à capacité élevée n'est pas envisageable pour les essais d'évaluation ou de réception ; on ne peut utiliser que des systèmes plus modestes ; une des conférences de cette série sera consacrée à ce sujet. De toutes façons on doit réaliser que l'information primaire est celle que peut ramener l'équipage lui-même. Ce système d'enregistrement ne peut être qu'une aide pour l'équipage ; en outre il va sans dire que dans de nombreux cas les évaluations en vol sont faites sans l'aide d'aucun matériel spécialisé d'essai ou d'enregistrement, en utilisant les instruments de bord standards.

4 - MOYENS, ORGANISATION, et PROGRAMME des VOLS d'EVALUATION

Bâtir un programme de vols d'évaluation couvrant tous les points importants aussi efficacement que possible dans le temps de vol minimum n'est pas une tâche facile.

Moins on dispose de temps de vol, plus il en faut pour préparer le programme de ces vols.

Pour accomplir cette tâche de préparation puis d'exécution des vols, il existe de nombreux types d'organisation ; presque toutes reposent sur la désignation d'une équipe d'essai responsable de chaque type particulier d'avion.

Cette équipe de pilotes et ingénieurs est désignée par le client ; il lui correspond une équipe similaire désignée par le constructeur ; les deux équipes travaillent en étroite coopération.

Aux U.S.A. cette équipe est appelée "Joint Test Team".

Chacune des deux équipes comprend :

- des pilotes et équipages d'essais ; il est avantageux d'associer à un pilote d'essai spécialiste de la navigabilité un pilote ayant une expérience opérationnelle des missions dévolues à l'avion. Il existe parfois des pilotes expérimentés compétents dans ces deux domaines.

- des ingénieurs navigants d'essais et spécialistes des essais.

- des experts de la conception, de la maintenance, des opérations ; si de tels experts n'existent pas dans l'équipe du client, il faut assurer une liaison étroite avec les divers spécialistes du constructeur, par l'intermédiaire d'un représentant du bureau d'études.

Pour établir le programme des vols, deux aspects doivent être pris en considération :

- ceux relatifs à l'avion
- ceux relatifs à l'environnement opérationnel de la mission et aux moyens d'essai à employer.

(a) points relatifs à l'avion lui même

Les tâches suivantes doivent être accomplies :

- une analyse des exigences de la mission afin d'identifier les plus difficiles à satisfaire.
- une analyse de toutes les tâches opérationnelles dévolues à l'avion afin de choisir les plus utiles et les plus difficiles.

- une comparaison des points critiques trouvés avec ceux identifiés sur d'autres types d'avions ayant la même mission, ou des missions similaires.

- un examen critique des rapports d'essais en vol qui ont pu être obtenus du constructeur et/ou du "Client Principal". C'est une pratique courante de transmettre certains de ces rapports au client lorsqu'ils ont trait à une caractéristique particulière ou originale de l'avion de type.

- un examen critique de toutes les autres sources d'information, telles que rapports d'autres clients s'ils sont d'accord pour les fournir.

(b) points relatifs à l'environnement opérationnel de la mission et aux moyens d'essais associés.

Ce sont les suivants :

- les conditions climatiques spécifiées dictent les lieux géographiques où les vols prendront place ; quand les conditions extrêmes doivent pouvoir être supportées par l'avion, (par exemple climat tropical sec ou climat polaire) il sera généralement nécessaire de répartir les vols entre plusieurs aérodrômes et espaces aériens, ceux où existent à la fois les conditions météorologiques recherchées et les moyens d'essai adéquats (le besoin de moyens de trajectographie est un exemple des difficultés de ce genre à surmonter.)

Les vols peuvent devoir être répartis entre l'aérodrôme du constructeur, une base aérienne représentative des conditions météorologiques, et quelquefois une base d'essai importante dotée de moyens puissants et sophistiqués qui feront gagner du temps.

- la nécessité d'essayer le système d'armes dicte l'emploi d'un champ de tirs où les armes pourront être mises à feu, avec tous les moyens associés de trajectographie ou de sécurité. Par exemple, en France, tous ces essais sont exécutés au Centre d'Essais en Vol de CAZAUX, où plusieurs champs de tir jouxtent l'aérodrôme ; en outre CAZAUX se trouve à côté de la grande base d'essais de missiles air-air nécessitant un grand espace aérien s'exécutent avec les moyens conjugués du CEV de CAZAUX et du CEL de BISCAROSSE.

Il est utile de faciliter le travail de l'équipage pour chaque vol par un moyen d'enregistrement par exemple un enregistreur vocal.

- dans certains cas il est nécessaire de disposer de matériel d'enregistrement de prise de vue cinématographique à grande vitesse, pour permettre l'observation différée de phénomènes trop rapides pour l'observation visuelle directe.

- il faut découper le programme en un nombre raisonnable de vols compte tenu de la durée totale de l'évaluation.

5 - EPREUVES D'ADMISSION

Par épreuves d'admission on désigne une série de vols d'essais ayant un but particulier : contrôler les qualités, défauts, déficiences d'un avion particulier de série comparé à l'avion de référence, appelé "avion de type".

Pour exécuter ces essais, il est possible d'utiliser les mêmes moyens et la même organisation que ceux décrits ci-dessus.

Dans ce cas particulier, la tâche n'est pas aussi vaste et les critères sont moins nombreux puisqu'il s'agit seulement d'une comparaison entre :

- l'avion de type et
- un avion particulier de la chaine de fabrication.

La connaissance de l'avion de type s'aquiert progressivement à partir du moment où l'on a commencé à faire l'évaluation préliminaire au choix de l'avion ; ici encore le "Client Principal" a une meilleure connaissance de l'avion.

Cette connaissance permet d'établir une liste complète des points critiques à vérifier ; une sélection doit être faite dans cette liste pour optimiser l'efficacité des vols de réception, ce qui signifie :

- effectuer le plus grand nombre possible des vérifications les plus importantes,
- réduire le temps de vol autant que possible pour minimiser le coût.

Lorsqu'il y a un grand nombre d'avions commandés la réponse à ce problème difficile consiste à utiliser deux types de programme de vol.

- un programme très complet qui sera appliqué seulement à certains avions choisis sur la chaine de fabrication ; par certains exemples, si 100 avions ont été commandés, ces essais seront faits sur les avions 10, 20, 30 etc... et demanderont 5 à 6 vols par avion (en FRANCE ces avions s'appellent "avions de lot".)

- un programme réduit, d'un ou deux vols, pour chacun des autres avions.

Pour établir ces deux programmes une connaissance aussi approfondie que possible de l'avion de type est nécessaire. C'est pourquoi les services d'Essais en Vol du "Client Principal" sont là encore dans une situation beaucoup plus confortable pour faire ce travail, puisqu'ils ont eu pour tâche d'observer et souvent surveiller le déroulement du programme d'essais en vol depuis le prototype jusqu'au premier avion de présérie ou série ; ils sont bien informés des défauts particuliers à l'avion, des remèdes

à ces défauts, et des caractéristiques les plus "marginales" de l'avion de type.

C'est en particulier le cas du "Groupe Réception" français dont le travail particulier au Centre d'Essais en Vol est d'exécuter les vols inscrits au programme des épreuves d'admission.

Un certain nombre de clients étrangers des avions construits en France, conscients de leurs possibilités limitées dans le domaine des essais en vol ont décidé d'exiger que la Réception des avions qu'ils ont achetés soit faite par le "Groupe Réception" ce qui apporte une sorte de label de qualité, aux avions acceptés par cette équipe très expérimentée.

Pour un type nouveau d'avion, cette expérience s'accumule au fur et à mesure du déroulement du programme de mise au point du prototype. L'équipe participe aux principales phases du programme, schématisées comme suit :

Phase 1 : Préparation du 1er vol du prototype :
Mise au point et évaluation de chaque système fonctionnel et opérationnel système de navigation nouveaux moteurs, armement ; tous ces essais sont généralement effectués sur des avions banc d'essais.

Phase 2 : Essais en vol de mise au point du prototype et des avions de présérie "déverminage" initial, extension du domaine de vol, essais des systèmes ; à la fin de cette phase l'avion de type est bien défini et a subi toutes les modifications nécessaires pour subir l'examen final de la phase suivante.

Phase 3 : Essais en vol de qualification; pendant cette phase, il n'y a, en principe, aucune modification à la définition du type, sauf exception mineure, ou sans influence (comme la couleur des sièg s) sur les essais de qualification; les sujets de ces essais sont :

- la navigabilité : vérification du niveau de sécurité de l'avion compte tenu des règlements applicables.

- les performances de l'avion et de ses systèmes fonctionnels : vérifier qu'ils satisfont les exigences de la mission.

- les performances et les limitations des systèmes opérationnels en particulier les systèmes d'armes, et leur compatibilité avec les exigences de sécurité.

Phase 4 : Vols expérimentaux : d'habitude ces vols sont exécutés avec le premier avion de série, afin de permettre de relever toute déficience pouvant être révélée dans l'environnement réel de la mission qui peut différer fortement de celui des vols précédents : pour les avions de combat, des escadrilles expérimentales sont parfois utilisées dans ce but et aussi pour préparer les consignes d'emploi opérationnel par les Armées de l'Air (ce fut en particulier le cas du HARRIER Britannique) Ces vols expérimentaux peuvent avoir plusieurs buts différents ; ce peuvent être :

- des vols d'endurance, pour déceler en volant à un rythme accéléré les problèmes de maintenance ou d'opérations que les vols d'essais en pouvaient révéler.

- des vols opérationnels, pour déceler les défauts pouvant compromettre l'accomplissement de quelques unes des missions prévues, et limiter ainsi les possibilités d'emploi.

- des essais climatiques pour vérifier que l'avion peut être utilisé partout dans le monde, ou au moins, dans les conditions prévues.

Cette phase est donc consacrée à l'exécution d'une série de vols d'évaluation particuliers qu'on appelle parfois "vols d'évaluation initiale".

D'après la description ci-dessus on peut comprendre qu'une équipe de réception qui a participé à ces quatre phases, ou au moins a été constamment informée des résultats obtenus, à tous les éléments pour définir le programme d'essais de réception le plus efficace.

On voit là encore qu'un équipage de réception étranger est dans une position plus difficile pour établir ce programme, puisqu'il le construit d'après des informations indirectes, qu'il ne peut vérifier, et qui sont parfois sélectives. Pour surmonter ce problème, il lui faut au moins deux sources d'information :

- des informations fournies par le constructeur
- des informations fournies par un autre utilisateur du même avion, et de préférence par le "client principal", si celui-ci accepte de les donner.

6 - PROGRAMMES TECHNIQUES D'EVALUATION ET DE RECEPTION

Comme on vient de l'expliquer ci-dessus, les programmes techniques doivent inclure tous les essais qui peuvent révéler les faiblesses de l'avion. Une revue générale des principales caractéristiques

doit aussi être faite. C'est pourquoi un certain nombre d'essais "Standard" sont inclus dans de tels programmes, les conférences de cette série parleront de certains de ces essais. D'une manière générale, on peut les diviser en plusieurs catégories.

A1 - La vérification des limites du domaine de vol, décrochage ou vitesse minimale, résistance à la vrille, facteur de charge maximal et nombre de mach maximal en virage.

A2 - la vérification des performances, avec ou sans charges externes (s'il y a lieu) ; il faut souligner que le but de tels essais n'est pas de répéter le processus long et difficile qui permet au constructeur d'établir le "manuel de performances" de l'avion.

Le but est seulement de vérifier que les quelques mesures particulières faites en petit nombre ne sont pas en contradiction avec le "manuel de performances". Parmi ces essais souvent faits les exemples suivants peuvent être cités :

- vitesse horizontale mesurée pour quelques masses, altitudes, régime moteur de référence.
- vitesses verticales dans des conditions bien définies, ou temps pour atteindre une altitude de référence, à partir du décollage.
- vitesses verticales avec un moteur en panne
- débit de carburant et distances parcourues par unité de masse de carburant.
- toutes ces mesures peuvent être comparées avec les données du manuel de vol dans la mesure où l'on tient compte des données météorologiques, surtout la température.

A3 - La vérification des qualités de vol dans les conditions les plus critiques de vitesse, nombre de mach, altitude ; les vitesses transsoniques sont généralement parmi les vitesses critiques.

A4 - La vérification des systèmes fonctionnels tels que le pilote automatique, et l'automanette, les systèmes de stabilité artificielle etc...

A5 - La vérification des dispositifs de sécurité

B - Essais relatifs au systèmes opérationnels.

Une liste complète ne peut en être donnée compte tenu de la grande variété des missions ; cependant les exemples suivants peuvent être donnés :

- systèmes de Navigation
- systèmes "d'avionique" comme les radars de suivi de terrain
- radar de poursuite
- systèmes d'armes.

7 - TECHNIQUES D'ESSAIS EN VOL

Cette revue des essais d'évaluation et de Réception ne serait pas complète sans quelques mots à propos de l'entraînement des équipages à ces vols très particuliers et quelques remarques concernant la différence entre ces vols d'essai et les autres sortes de vols.

La densité des informations qui doivent être rapportées par les pilotes et équipages durant les vols d'Evaluation est très forte ; ils doivent le faire par des moyens très simples, usuellement les instruments de vol disponibles à bord. Ils ne peuvent se faire aider par un système de recueil de données sophistiqué et complexe leur permettant de se concentrer seulement sur le pilotage proprement dit. Ils doivent en même temps piloter avec précision et faire les observations et mesures demandées.

Ce n'est pas une tâche de débutants. C'est pourquoi les pilotes d'essais de Réception sont généralement choisis parmi les pilotes les plus expérimentés.

Quand il y a à bord un système de mesure, c'est lorsqu'il est impossible à l'homme de recueillir les mesures nécessaires, par exemple parce que le paramètre à observer varie trop rapidement.

Il va sans dire qu'une qualification élevée, acquise dans une Ecole d'Essais en Vol est essentielle pour qu'un pilote puisse faire ce travail. Dans le monde occidental quatre de ces Ecoles sont bien connues ; elles reçoivent des étudiants de toutes nationalités. Ce sont :

- l'US AIR FORCE Test Pilot School d'EDWARDS AFB -CALIFORNIE, aux USA.
- l'US NAVAL TEST PILOT SCHOOL, au NAVAL AIR TEST CENTER (PATUXENT River)
- l'Empire Test Pilot School Britannique à BOSCOMBE DOWNS

- l'Ecole Française, EPNER (Ecole du Personnel Navigant d'Essais et de Réception) à ISTRES.

Toutes ces écoles enseignent à des pilotes expérimentés à devenir pilotes d'essais ; les Ecoles Françaises et des Etats Unis forment des équipages d'essais complets incluant l'ingénieur navigant d'essais.

Les conférenciers de cette "série" décriront en détail les règles de l'art des vols, d'essais pour chaque type d'essai concerné. Cependant il existe des règles générales applicables à tous ces vols. Je ne les décrirai pas toutes, ce serait trop long. Cependant je veux donner quelques exemples.

Précision des stabilisations de vitesse : il n'est généralement pas possible de faire de bonnes mesures de performances sans garder la vitesse constante ; la précision nécessaire est très souvent meilleure que celle fournie par le pilote automatique. Ceci est pourtant faisable manuellement en air calme en utilisant l'indicateur d'assiette comme instrument de base plutôt que l'indicateur anénométrique lui-même. Pour obtenir un bon résultat il faut un indicateur d'assiette de haute sensibilité. La vitesse est mieux stabilisée en conservant l'altitude constante car cela impose un angle d'incidence ; à incidence constante la vitesse ne peut changer pendant une stabilisation de quelques minutes en air calme. Quand l'indicateur d'assiette n'est pas assez précis, la simple observation de l'horizon peut suffir, pourvu qu'il existe sur le parebrise une référence quelconque, par exemple une marque au crayon gras, qui permet de maintenir le nez de l'avion à une "hauteur" (angulaire) constante au dessus de l'horizon. En France, nous utilisions autrefois un "gadget" appelé collimateur pour obtenir une telle référence réglable ; cet appareil était l'ancêtre des systèmes de "pilotage tête haute" d'aujourd'hui.

Echanges d'énergie cinétique et potentielle gravifique sur les avions de combat à réaction.

Sur tous les avions de combat à réaction une légère réduction de vitesse à régime moteur constant produit un changement d'altitude. Au "deuxième régime de vol" une réduction de vitesse fait perdre de l'altitude. Lorsqu'on mesure un taux de montée un petit écart de vitesse pendant la stabilisation peut engendrer une grosse erreur dans la mesure, lorsque celle-ci est obtenue en divisant la diférence d'altitude par le temps nécessaire pour l'obtenir.

Une méthode de correction sera donnée dans l'une des conférences ; elle repose sur la notion de "hauteur totale".

Nombre de paramètres physiques influents sur la mesure

Quand une mesure telle que celle de la stabilité longitudinale d'un avion doit être faite, le résultat est fonction d'un certain nombre de paramètres, certains peuvent être maitrisés, tels :

La masse, la position du centre de gravité, la position des volets hypersustentateurs, le réglage des moteurs ; d'autres ne sont que partiellement contrôlés, comme la densité de l'air qui varie en premier lieu avec l'altitude. Mais puisque personne, sauf Dieu, ne peut régler la température qui règne à une certaine altitude, il est impossible de contrôler complètement ce paramètre.

On peut citer un certain nombre d'exemples semblables. Pour tenir compte de ce problème, il il est généralement nécessaire de stabiliser la vitesse et l'altitude correctement avant toute manoeuvre d'essai, de manière à disposer d'un moment pour mesurer les valeurs des paramètres exerçant une influence sur le résultat, et définissant aussi des conditions à l'origine servant de référence.

Quand un certain nombre d'essais doivent être entrepris dans les mêmes conditions de vol, afin par exemple de faire des comparaisons, cette pratique est très utile parce qu'il est ainsi plus facile de vérifier, entre les manoeuvres, tout changement involontaire des conditions de vol, par exemple le réglage des compensateurs, qui peut induire une erreur.

Précision des mesures différentielles

Faire des mesures différentielles, cela consiste à mesurer la différence entre le paramètre inconnu et une référence bien définie, plutôt que de tenter une mesure directe de l'inconnu.

Un bon exemple est la méthode d'étalonnage de la prise de pression statique qu'alimente l'altimètre (et l'indicateur de vitesse) : Cette méthode est ancienne, mais si simple qu'elle est encore employée, avec quelques raffinements. On l'appelle la méthode des "Passages à la Tour".

On peut la décrire ainsi :

L'avion survole l'aéroport en ligne droite à très basse altitude au dessus du sol, assez près de la Tour de Contrôle pour pouvoir stabiliser cette altitude au niveau du toit de la Tour de Contrôle. Plusieurs passages sont faits pour couvrir la gamme de vitesses souhaitée, et à chaque passage près de la tour, l'altitude lue est comparée à l'altitude vraie du sommet de la tour.

De manière à s'assurer que l'effet de sol n'altère pas les résultats, l'avion doit voler au moins à une hauteur au-dessus du sol égale à son envergure ; la hauteur optimale est de 1,5 fois l'envergure.

Le caractère différentiel de ce type de mesures apparaît dans la façon de comparer indirectement la hauteur de la tour et la lecture de l'altimètre.

Dans le but de faire cette comparaison une "référence" est utilisée ; avant et après la série de passages de l'avion, celui-ci est immobilisé au sol, à proximité de la Tour de Contrôle, et l'altitude pression est lue, en affichant 1013 mb dans la fenêtre de l'altimètre ; ce réglage est conservé pendant tout le vol.

Exprimée en termes d'altitude pression, l'erreur de statique est donnée par la différence entre la hauteur de la tour et la hauteur obtenue par la différence entre l'altitude lue à chaque passage et celle lue au sol sur l'altimètre.

Un des "raffinements" appliqués aujourd'hui à cette vieille méthode consiste à disposer un appareil photo rigidement fixé sur le toit de la Tour de Contrôle.A chaque passage un opérateur prend une photo de l'avion passant devant la tour ; ceci fournit un moyen de corriger toute différence pouvant exister entre le niveau de chaque passage et celui du toit de la Tour.

8 - DEUX EXEMPLES

J'ai choisi deux exemples de programmes d'essais de réception.

CONCORDE

Les vols de réception de l'avion n° 13 sortant de la chaine de fabrication ont été entrepris en septembre 1978 en suivant le même programme que pour tous les autres avions de ce type construits en Grande Bretagne et en France.

Pour couvrir ce programme, il a fallu 6 vols, et un nombre total d'heures de vol de 17 heures incluant 7 heures à vitesses supersonniques, et 10 atterrissages.

Ce programme de vol comportait 78 essais individuels détaillés ci-dessous :

Qualités de vol	5 essais
Entrée d'air	8 essais
Système anénométrique	4 essais
Propulsion	3 essais
Système de carburant	4 essais
Génération électrique	5 essais
Système hydraulique	11 essais
Conditionnement d'air	11 essais
Dégivrage	2 essais
Pilote automatique	10 essais
Système de navigation	6 essais
Système de radiocommunications	3 essais
Radars météorologiques	1 essai
Systèmes commerciaux	4 essais
Système enregistreur d'accident.	1 essai

MIRAGE III E

Le programme de réception comporte 4 vols des types A, B, C et D - le profil de ces vols est donné sur les figures 13 à 16.

On peut voir d'après ces figures, qu'à l'exception du vol D, consacré au système de navigation, aucun des trois autres vols n'est spécialisé dans un domaine spécifique. Ces trois vols sont organisés de manière à couvrir autant de points à vérifier qu'il est possible.

9 - CONCLUSION

L'exécution d'un vol d'Evaluation ou d'Admission demande beaucoup de travail à l'équipage, puisque la grande densité des observations à faire est la caractéristique générale des programmes de vol.

Ces essais ne sont pas des essais en vol "de deuxième ordre". Ils demandent l'habitude d'équipages expérimentés.

Pour faciliter leur préparation et leur programmation une connaissance en profondeur de l'avion et de son programme de développement est utile.

J'espère que la série de conférences qu'ouvre cet exposé sera utile à tous ceux qui auront à accomplir ce travail.

LES PLANCHES SE TROUVE SUR PAGE 1-20.

SOME COMMENTS ON THE PROBLEMS INVOLVED IN AIRCRAFT ASSESSMENT AND ACCEPTANCE TESTING

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0 - INTRODUCTION

My purpose is to introduce this lecture series devoted to

Assessment and Acceptance Testing.

I have chosen to point out, and comment on some of the many problems which have to be solved when such testing has to be made.

My personal knowledge of the subject comes mainly from my experience on French aircraft of any kind ; but I think that the technical questions raised are the same in any Country.

The views that I will present are personal, but they were strongly influenced by the advice which I was given by our French official team dealing with acceptance testing, named the "Groupe Reception", which is a service of the French Flight Test Center (C.E.V.).

I have been personally involved in the flight testing programme of a well known aircraft : CONCORDE, since I am the leader, on the official side, of the team which has dealt with the entire programme, from the preparation of the first flight, in 1965 (completed in 1969) to the first landing at KENNEDY Airport USA (19 oct. 1977).

My presentation will adress the following subjects

- objectives of Assessment and Acceptance Tests
- criteria for assessment
- means, organisation and programmes of flying
- acceptance testing as a particular case of assessment
- the technique of flight testing.

1 - OBJECTIVES OF ASSESSMENT AND ACCEPTANCE TESTS

These objectives are not the same for the aircraft manufacturer and the customer who wants to purchase it.

What are such objectives for the company manufacturing and selling the aircraft ?

+ For the sales manager the obvious purpose of flight testing is to present the aircraft to the customer emphasizing its best qualities.

Such flight testing will preferably be carried out at the vendor's facilities but sometimes also at the customer's Air Force base. In this paper such flight tests are called "assessment tests" or in a shorter way "ASST".

+ For the manufacturer's quality control services, the flight testing which is important is that needed to meet the following objectives :

- check that any aircraft from the production line has the same performance and qualities as the "type aircraft", with acceptable tolerance
- check that the defects are minor and can be easily cured by proper adjustments.

In this paper such flight tests will be called "acceptance tests" or in a shorter way "ACCT".

+ For the customer the purposes are not exactly the same.

The first question to raise is : what customer ?

There is a major difference between the "Main customer" and the other ones.

The "Main customer" is generally the Ministry of Defence of the Nation of the Manufacturer, which is potentially a buyer of a large number of aircraft if they meet the specifications adequately (performance and missions). Sometimes the manufacturer takes the risk of building and developing an aircraft on his own funds ; more often some assistance to this private venture is given by the Nation ; for a number of projects full assistance is given to the manufacturer from the very beginning of the development, sometimes later on, with contracts funded from the budget of the Ministry of Defence.

In the latter case the "Main customer" is the only one who can play on active role in the completion of the various phases of the programme from the first flights to the delivery of the first operational aircraft ; in particular some flight tests are carried out at each step of the programme by the flight test pilots committed by the Ministry of Defence in order to report on the progress made, and to check the expenses incurred.

Such flights are not only requested by the "Main customer" sponsoring and financing a programme, but also by the contractor himself, in order to check that the development is directed toward a target as close as possible to the needs of the customer. When several competitors are on the same programme, such flights are essential for each of them to stay in the competition up to the final stage, when the choice of the type ordered is made.

Customers other than the "Main customer" as described above, for example the foreign countries, when the aircraft requested is military, are unable to get all the information about the development of the aircraft and the progress made toward the operational performance.

For these "other customers" the needs are the following :

- assess various types of aircraft in order to choose those which are closest to their requests, and negotiate a contract with the manufacturer to get the required number of aircraft

- accept each individual aircraft delivered on the basis of the contract ; this means ; undertake all the flight testing needed to check that each of them is identical to the type aircraft, with no defects.

2 - CRITERIA FOR ASSESSMENT

These criteria differ with the mission to be fulfilled by the aircraft.

- light aircraft for private users
- executive aircraft
- airline transport
- Air Force transport
- Fighter , high altitude attack
- bomber
- low level penetration aircraft
- observation aircraft
- carrier aircraft
- rotorcraft
- VSTOL.

The differences between each of the above missions do not need to be expanded.

Once the category of aircraft is selected and the mission requirement defined, a number of criteria have to be taken into consideration. For the buyer with a limited budget (all the budgets, even the highest ones are limited) the primary consideration is a financial one :

- cost of each unit
- life cycle cost
- operating cost.

The impact of technical simplicity on these criteria is obvious ; among the three above criteria, the last one implies some kind of assessment flight testing ; all of them are clearly related to the following considerations which must be kept in mind for any assessment flying :

- type of operation which is intended : visual flight conditions, or all weather operations, by night or by day
- crew qualification and number ; minimum crew
- crew comfort according to the duration and characteristics of the mission
- passengers comfort (if applicable) and noise
- level of safety to be provided in conformity with the relevant airworthiness regulations
- flight boundaries (at low and high speeds)
- flying qualities
- handling capabilities
- complexity of the aircraft, its functional systems, its operational systems

and, last but not least :

- *performance for the mission*

3 - INCREASING COMPLEXITY OF AIRCRAFT

This is one of the major problems to tackle in order to build up a programme of flight testing for assessment and acceptance purposes taking into consideration all the criteria, without wasting the number of flying hours, which for economical reasons must be strictly limited.

In order to cover the maximum number of observations therefore each flight must be planned within the minimum time. A very high crew workload is the characteristics of such flying.

A good picture of the increasing complexity of aircrafts of to day is given by fig. 1 which shows the number of recorded parameters for the development phase of various prototype aircraft.

Such high capacity data recording systems is not feasible for acceptance and evaluation testing ; only simpler test equipment can be used ; one paper of this series will be devoted to this subject ; one must realize anyway that the primary information is that collected by the crew itself. The data recording system is only a back up of the crew ;it goes without saying that very often assessments are made without the help of any test equipment or recording system, using only the standard cockpit instrumentation.

4 - MEANS, ORGANISATION AND PROGRAMME OF ASSESSMENT FLYING

Drawing up a programme of assessment flying covering all the major questions as efficiently as possible within the minimum flying time is not an easy task.

The less flying is allowed, the more time is needed to prepare it.

To achieve this task of flying preparation, and flying itself, many types of organisation can be used, but one of the most efficient is the following :

A flight test team in charge of each particular type of aircraft to be assessed.

This team and its leader is selected by the customer, who request that similar team of pilots and experts be selected by the manufacturer ; the teams will work in close cooperation with each other.

Sometimes the name of "joint test team" is given to this association.

Each team must include the following members :

- test pilots, and crews ; preferably one of the test pilots should be specialised in the airworthiness problems while the other should have the relevant operational experience of similar missions. Experienced pilot are very often familiar with both fields of expertise.
- flight test observers and specialists.
- design, engineering, maintenance and operation specialists ; if these are not available in the customers team, then close liaison must be arranged with the corresponding manufacturer's specialists preferably through a design office representative.

To draw up the flight programme two aspects have to be taken into consideration :

- those related to the aircraft
- those related to the operational environment of the mission and the testing means to be used.

(a) items related to the aircraft itself

The following tasks have to be performed :

- an analysis of the mission requirements in order to identify the most difficult to meet
- an analysis of all the operational tasks involved in order to select the most useful , and the most difficult
- a comparison of the critical items found with those identified on the other types of aircraft with the same mission or similar missions
- a critical examination of the flight test reports released by the manufacturer and/or the main customer. It is a normal practice to give some of these reports to the customer when they deal with some particular characteristics of the type aircraft (here the word "type" stresses the difference between the aircraft under development and the "product" which is sold)
- a critical examination of all the other sources of information, such as reports from other customers if they agree to release them.

(b) items related to the operational environment of the mission and the associated test means

These are the following :

- the specified climatic conditions dictate the location, where the flying will take place ; when extreme conditions have to be met (for example dry tropical climate, or polar climate) it will generally be necessary to distribute the flights among different airspaces and airfield, these where the right climatic conditions prevail and those where the test means are adequate (for example need of external tracking systems). The flights may have to be distributed among the manufacturer's airfield, a representative air base chosen by the customer and sometimes a major flight test center with sophisticated equipment.
- the need to test the weapon system dictates the location of the testing range, where weapons can be fired, with all the associated tracking and safety means. For example in France all these tests are carried out at the Flight Test center at CAZAUX, where there is a firing range ; additionally CAZAUX is located in the vicinity of the big missile testing range of CENTRE D'ESSAIS des LANDES at BISCAROSSE
- the need to facilitate the crews observation in each flight by a suitable recording system, such as a voice recorder
- in certain cases the need of high frequency recordings or highspeed cinematographic equipment
- the reasonable number of test flights and amount of flying time to be alloted, taking into account the overall duration of the assessment.

5 - ACCEPTANCE TESTING

Acceptance testing is a particular kind of flight testing, directed toward a very specific purpose : controlling the quality and detecting any defect or deficiency of an individual production aircraft as compared with the type aircraft.

To complete this task, the same means and organisation as described above can be applied.

In this particular case, the objectives are not as broad as described above, and there is a smaller number of criteria to apply, since this is only a comparison between :

- the type aircraft, and ...
- an individual aircraft from the production line.

The knowledge of the type aircraft results from the preliminary assessment made prior to the choice which has lead to the purchase contracts ; here again, the "Main customer" has a better knowledge of this aircraft.

On the basis of this knowledge a list of critical checks can be set up; a selection has to be made in order to optimize the efficiency of the acceptance flying, which means

- perform the greatest number of the most important checks
- reduce the flying time as much as possible for cost considerations.

When there is a large number of aircraft ordered the answer to this difficult problem is to have two types of acceptance flying programmes :

- an extensive one, which will be applied only to selected aircraft on the production line for example, assuming that a number of 100 aircraft have been bought, these tests will be made on the aircraft number 10, 20 ,30 etc. which will require for example 5 or 6 flights (in FRANCE these A/C are called"Avion de lot")
- a reduced one consisting in one or two flights for all the other.

To draw up these two programmes, the extensive and the reduced one a knowledge as large as possible of the type aircraft is required. This is why the Flight Test Services of the "Main customer" are here again in a much more comfortable position to do this job, since they were able to observe and very often monitor the development of the programme from the prototype to the preproduction or first production aircraft ;thus they are well informed of all the defects, remedies to these defects, and of the weakest characteristics of the type aircraft.

This is in particular, the case for the French "groupe Reception " whose particular job at the CEV (Centre d'Essais en Vol) is to make the flight testing needed to complete the acceptance trial programme.

A number of foreign customers of the French-made aircraft who are well aware of the limitations of their own flight testing capabilities have decided to request that the acceptance testing be made by the "groupe Réception" since this is a kind of official quality label which is given to an aircraft accepted by this team of experienced people.

For a new type of aircraft this experience is built up progressively with the team designated to accept an aircraft of a new type,participating in all the phases of the programme which are summarized schematically as follows :

Phase 1:preparation of the first flight of the prototype ; development and assessment of individual functional and operational systems equipment, navigation systems, new engines (if applicable), armament ; all these tasks are generally completed on flying test beds.

Phase 2 :flight testing for the development of prototype and preproduction aircrafts ; debugging, extension of flight envelope, system tests ; at the end of this phase the type aircraft is well defined and all the modifications are incorporated for presentation to the final examination of the following phase.

Phase 3 :qualification flight testing ; during this phase, there are in principle, no modifications made to the definition of the type, with the exception of minor ones, or irrelevant ones (such as the colour of the seats :) which means that they do not affect the qualification tests ; the subjects of these tests are :

- airworthiness : verification of the aircraft safety level as requested by the regulations
- performance of aircraft and functional system : check that it meets the specifications of the mission
- performance and limitations of operational systems in particular weapon systems weapon delivery, compatibility with safety.

Phase 4 :experimental flying;usually these flights are made with the first production aircraft, to take account of any minor deficiency in the real environment of the mission, which could be very different from that of the previous phases ; for fighter aircraft experimental squadrons are sometimes used for this purpose and for preparing the operational use by the Air Force (this was the case of the British VTOL Harrier). These experimental flights can deal with several different purposes :

- endurance flights to check with a high rate of flights any maintenance or operation problem that the previous test flights were unable to reveal

- operational flights to check any defect which could prevent the completion of some of the planned missions and limit the use of the aircraft

- climatic tests to check that the aircraft can be used under world-wide conditions, or at least in the planned conditions.

This phase is therefore devoted to a particular, and very important series of assessment flights which are sometimes called "initial evaluation flights".

From the above description it can be understood that an Acceptance Team which has had the opportunity of being involved in, or at least informed, about these four phases has all the needed elements to define the most efficient programme of Acceptance tests.

Again it can be seen that a foreign acceptance team is in a more difficult position to draw up such a programme since it must rely on indirect information , that it cannot check, and that can be sometimes filtered. To overcome this problem this team needs at least two sources of information :

- information from the manufacturer

- information from any other user of the aircraft, and preferably from the main customer "if he agrees to release them.

6 - TECHNICAL PROGRAMMES OF ASSESSMENT AND ACCEPTANCE TESTS

As explained above the technical programmes must include all the tests which could reveal the weaknesses of the aircraft, as well as its most interesting performance. A general overview of all the main characteristics must also be provided. This is why a number of typical tests are generally included in such programmes. The papers of this series will address a selection of these tests. They can be divided into several categories :

A - tests related to the aircraft itself considered as a platform on which the operational systems must work properly. In this category there are

A 1 checks of flight boundaries, stalling or minimum speed, spin resistance, maximum g and Mach boundaries in turning manoeuvers.

A 2 check of performance, with or without external loads (if applicable) ;it must be pointed out that the purpose of these tests is not to repeat the difficult and lengthy process which enables the manufacturer to establish the performance manual of the aircraft.

The aim is only to check that the individual measurements made in a small number are not inconsistent with the "performance manual". Among the tests which are often made the following examples can be quoted :

- horizontal speed measured for a few reference weights, altitudes, thrust regimes

- vertical speeds under well defined conditions, or overall time to reach a reference altitude, from take off

- engine out vertical speeds

- fuel flow and range characteristics.

All these measurements can be compared with the data of the flight manual if due account of the atmospheric conditions,in particular the temperature, is taken.

A 3 - check of flying qualities under the most critical conditions of speed,mach number altitude ; the transonic regime is one of these critical range of speeds.

A 4 - check of functional systems, such as autopilot, autothrottle, stability augmentation systems, etc...

A 5 - check of safety devices

B - tests related to the operational systems.

A complete list of these tests cannot be given ; the following examples can be quoted :

- Navigation systems
- Avionic systems, such as terrain following radars
- Target acquisition radars
- Armament testing.

The papers in this series will cover these subjects.

7 - FLIGHT TEST TECHNIQUES

This overview of assessment and acceptance testing would not oe complete without a few words about the training of crews in this very particular flying and some comments stressing the difference between flight testing and any other kind of flying.

The density of information to be gathered by the pilots and crews during assessment flights is very high ; they must do this with simple means : usually the standard instruments on the dashboard. They cannot be helped by a sophisticated and complex test instrumentation system leaving them free to concentrate only on the flying itself : they must simultaneously control the aircraft accurately and perform the observations and measurements required.

This is not a beginner's task. Therefore acceptance test pilots are generally the most experienced flying personnel.

When some instrumentation is available on board its purpose is to collect the data which a human being would be unable to collect in particular because the reponses are too fast.

It goes without saying that the high qualification reached in a Flight Test School is essential to do this job. In the western world four of these schools are well known ; they teach Students from any foreign countries. These are :

- the US AIR FORCE Test Pilot School at EDWARDS AFB California, USA
- the US Naval Test Pilot School at the Naval Air Test Center , MARYLAND, USA
- the Empire Test Pilot School at BOSCOMBE DOWN , GREAT BRITAIN
- the French Test Pilot School, EPNER at the CENTRE D'ESSAIS EN VOL ISTRES, FRANCE.

All these Schools teach experienced pilots who will be trained to become test pilots ; the French and US Schools provide overall training for complete crews including flight test engineers and observers.

Each Author in this series will describe the state of the art of flight testing for each Subject. But there are general practices applicable to all these flights. I will not describe each of them, it would be too long. I intend only to give a few examples.

Accuracy of speed stabilisation : it is impossible to make proper performance measurements without keeping the speed constant. The accuracy required is very often better than that achieved by the autopilot. This is manually feasible in still air with the attitude indicator as a basic instrument rather than the airspeed indicator itself. For this purpose the attitude indicator must offer high sensitivity. The speed is better stabilized by keeping constant the attitude which governs the angle of attack ; at a constant angle of attack the speed cannot change during a stabilisation of a couple of minutes in still air. When an attitude indicator with high sensitivity is not available the sight of the horizon may suffice, provited that some reference on the windshield such as a pencil mark enable the pilot to keep the nose of the aircraft at a constant elevation above the horizon. In France we used in the past a device named "collimateur" to provide this adjustable reference ; this was the ancestor of the head up displays of today.

Exchanges of potential gravific energy and kinetic energy on jet fighters

On all jet fighters a slight reduction of speed at constant thrust setting produces a change in altitude. On the backside of the power curve, a reduction in speed will result in a lose of altitude. When measuring a rate of climb a small change of speed during a stabilisation can produce a serious error in the measurement of the rate of climb, when this measurement is made directly on board, by dividing the difference in heightby the time necessary to produce it.

A method of correction of this measurement is based on the concept a energy height which will be explained in this series.

Number of physical parameters influencing in the requested measurement

When a measure such as the longitudinal dynamic stability of an aircraft has to be made,the result is a function of a number of parameters. Some of them can be controlled, such as

The weight, CG location, flap setting, engine setting ; some other can be partially controlled such as the air density which varies primarily with altitude. But since nobody except God can control the temperature of the air at a given altitude. It is impossible to control this parameter completely.

A number of similar examples can be quoted. To cope with this problem it is generally necessary to stabilize the speed and altitude correctly prior to any test manoeuver, so that the parameters exerting an influence on the measurement, and defining the reference conditions may be measured.

When a number of tests have to be carried out under the same flight conditions in order to make some comparison this practice is very useful because it is easier to check between each manoeuver any indavertent change in the flight conditions (for example the trim setting) which could produce an error.

Accuracy of Differential measurements

Differential measurements consist in practice in measuring the difference between the unknown parameter and a well defined reference rather than trying to get directly the unknown one.

A good example is a method of calibration of the pressure error of the static pressure port feeding the altimeter (and the Airspeed indicator) : this method is old, but so simple that it is always used with some refinements. It is called the "Control Tower method", in French "Passages à la tour".

It can be described as follows :

The aircraft flies over the airfield along a straight line at a low level above the ground, close enough to the control tower to be able to stabilize its level at the height of the roof of the control tower ; several runs are made to cover the range of speeds needed and the altimeter readings at each run are compared to the true height of the tower.

In order to ensure that ground effects do not influence the results, the aircraft must be flown at least one wing span above the ground - 1,5 wing span is preferred.

The "differential"character of this type of measurement appears in the usual practice according to which the height of the tower and the altimeter reading are not compared directly.

For this purpose a "reference" is used:before, and after the series of runs the aircraft stands at zero speed, on the ground, in the vicinity of the control tower, and the pressure altitude is read, setting 1013 mb in the window of the instrument ; this setting is kept during the entire flight.

If expressed in term of pressure altitude the static pressure error is given by the difference between the height of the tower and the height measured as the difference between the pressure altitude read during any run and that read at the "reference".

One of the "refinements" applied today to this old method consists in using a photo camera fixed on the roof of the tower. At each run an operator takes a photo of the aircraft passing in front of the tower. This provides a means of correcting any difference between the level of each run, and that of the tower roof.

8 - TWO EXAMPLES

I have choosen two examples of acceptance flight test programme :

CONCORDE

The Acceptance flights of the 13th airplane of the production line of CONCORDE were carried out in september 1978 according to the same programme as all the other British or French aircraft of that type.

The numbers of flights needed to complete this programme was 6 for a total of 17hours including 7 hoursat supersonic speeds, and 10 landings.

The flying programme included 78 individual tests detailed below :

Flying qualities	5	individual tests
Air intake system	8	
Air data systems	4	
Propulsion	3	
Fuel system	4	
Electrical generation	5	
Hydraulic system	11	
Air conditiouing system	11	
Deicing system	2	
Auto Pilot	10	
Navigation system	6	
Communications system	3	
Meteorological radars	1	
Commercial devices	4	
Accident data recording system	1	

MIRAGE III - E

The acceptance programme includes 4 flights of the types A, B, C and D ; the profiles of these flights are given on figures 13 to 17.

It can be seen from these figures that, with the exception of flight D, devoted to the navigation system, none of the other three flights is specialized in a single specific field. The three flights are organised so as to cover as many items to be checked as possible, in order to increase the efficiency.

CONCLUSION

Achieving assessment and acceptance flight testing is very demanding for the crews, since a very high density of items to be checked is the characteristic of the flight programme.

These tests are not "second" rate" flight tests. They require the skill of experienced crews.

To help their preparation and programming a good knowledge of the aircraft and its development flying is usefull.

It is hoped that this lecture series will be of assistance to those who have to perform this work.

A G A R D
LECTURE SERIES 108

A great number of documents have been published concerning flight test techniques the following list is a selection of AGARD documents and other publications.

AGARD DOCUMENTS

M N 1 - AGARD flight test manuel (4 volumes) 1959

CP n° 85 - Flight test techniques 1971

CP n° 160 - Take of and landing 1974

CP n° 187 - Flight / ground testing facilities correlation 1975

CP n° 223 - Flight test techniques 1976

CP n° 260 - Stability and control 1978

CP n° 242 - Performance prediction methods

AG n° 160 - Volume 1 - Basic Principles of flight test instrumentation

AG n° 160 - Volume 2 - In flight temperature measurements

AG n° 160 - Volume 3 - The measurement of fuel flow

AG n° 160 - Volume 4 - The measurement of Engine Rotation Speed

AG n° 160 - Volume 5 - Magnetic Recording of Flight Test Data

AG n° 160 - Volume 6 - Open and closed loop Accelerometers

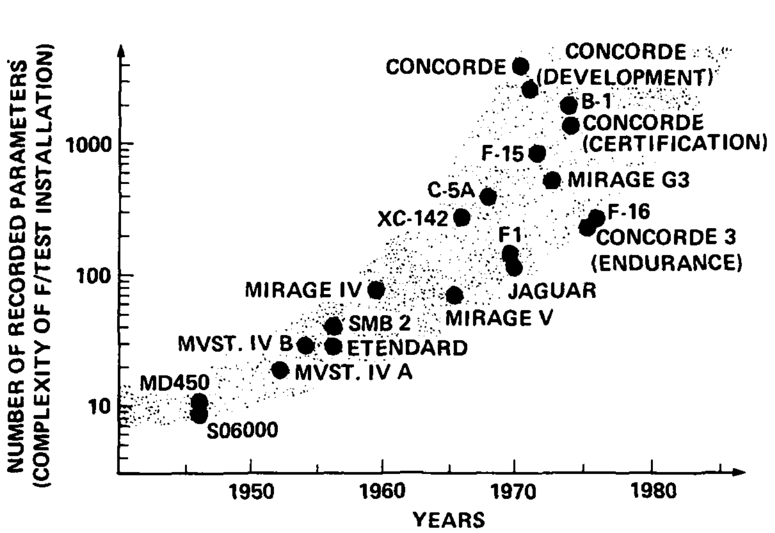
Additional volumes of this series are in progress. A new series on flight test techniques will be produced very soon.

BOOKS

Very few books have been published about flight testing the following one was formerly used by the french flight test school EPNER. In spite of the age of this document it remains widely used in European countries where french can be understood.

ESSAIS EN VOL (2 volumes) by J.F. RENAUDIE published DUNOD FRANCE 1960.

1



Trend in parameters recorded during flight tests

2

OBJECTIVES OF ACC. and ASS. TESTS

MANUFACTURER		CUSTOMERS	
SALES MANAGER	QUALITY CONTROL SERVICE	MAIN CUSTOMER (supporting the manufacturer)	OTHER CUSTOMERS
EMPHASIZE THE QUALITIES OF THE AIRCRAFT TO SELL	CHECK EACH PRODUCTION AIRCRAFT WITH REFERENCE TO THE TYPE AIRCRAFT	CHECK EACH STEP OF DEVELOPMENT PROGRAMME	SELECT THE BEST AIRCRAFT FOR THE MISSION

3

CRITERIA FOR ASSESSMENT

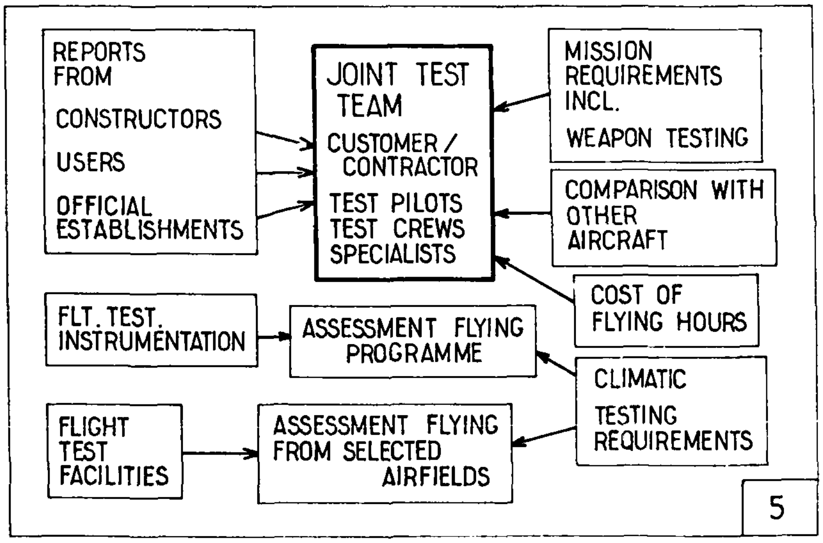
- MISSIONS

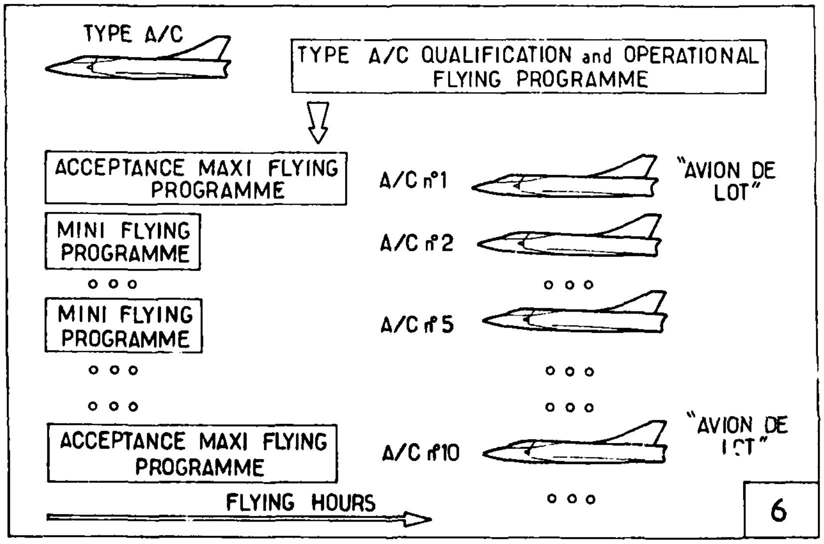
  LIGHT AIRCRAFT
  EXECUTIVE AIRCRAFT
  AIRLINE TRANSPORT
  AIRFORCE TRANSPORT
  FIGHTER HIGH ALTITUDE
  BOMBER
  LOW LEVEL PENETRATION
  HOSTILE DETECTION AND RANGING
  CARRIER OPERATION
  ROTORCRAFT
  VSTOL

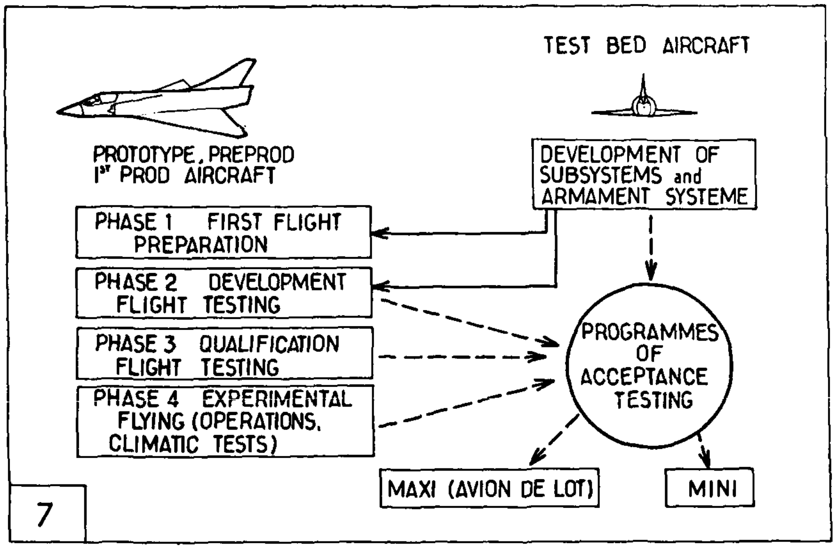
4

CRITERIA FOR ASSESSMENT

- TYPE OF OPERATIONS _ IFR _ VFR FLIGHT DAY OR NIGHT
  - CREW QUALIFICATION NUMBER
  - MINIMUM CREW
  - CREW COMFORT
  - PASSENGER COMFORT
  - LEVEL OF SAFETY
  - FLIGHT BOUNDARIES
  - FLYING QUALITIES
  - HANDLING
  - COMPLEXITY
- PERFORMANCE







8

TECHNICAL CONTENT OF ASS^{T} and ACC^{T} TESTS

A _ AIRCRAFT

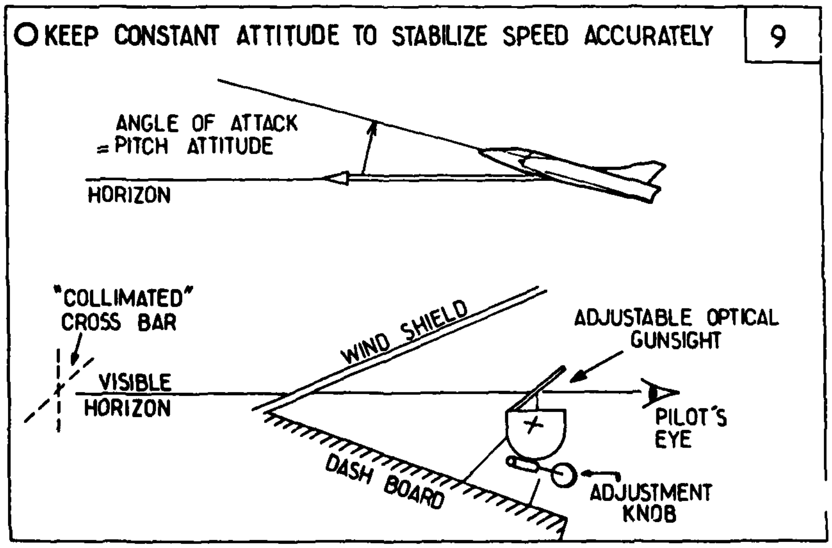
- FLIGHT BOUNDARIES
- PERFORMANCES
- FLYING QUALITIES
- FUNCTIONAL SYSTEMS
- SAFETY DEVICES

o o o

B _ OPERATIONAL SYSTEMES

- NAV SYSTEMS
- AVIONIC SYSTEMS
- TERRAIN FOLLOWING SYST
- TARGET ACQUISITION RADAR
- ARMAMENT TESTING

o o o



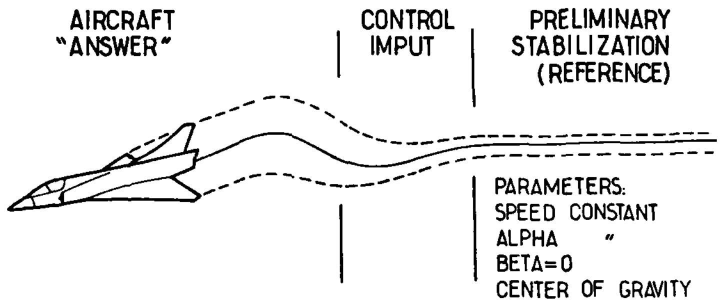
10

O KEEP SPEED CONSTANT FOR ACCURATE MEASUREMENT OF RATE OF CLIMB

$\boxed{\text{RATE OF CHANGE OF ENERGY}} \approx \boxed{\text{RATE OF CLIMB (POTENTIAL GRAVIFIC ENERGY)}} + \boxed{\text{ACCELERATION/G (KINETIC ENERGY)}}$

A VARIATION OF SPEED OF 0.01 G PRODUCES
A VARIATION OF 1% OF THE RATE OF CLIMB
(WHEN THE A/C IS FLOWN AT THE OPTIMUM SPEED FOR CLIMB)

O KEEP SPEED AND ANY OTHER PARAMETER CONSTANT FOR A SMALL TIME BEFORE ANY TEST MANŒUVER

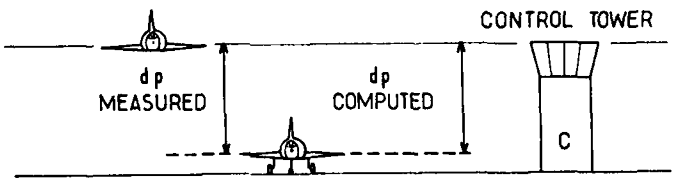


PARAMETERS:
SPEED CONSTANT
ALPHA "
BETA=0
CENTER OF GRAVITY
FLAP SETTING
AUTOSTAB SETTING
INERTIA (EXTERNAL LOAD)
ETC...

11

12

USE DIFFERENTIAL METHODS EXAMPLE ALTIMETER ERROR CALIBRATION

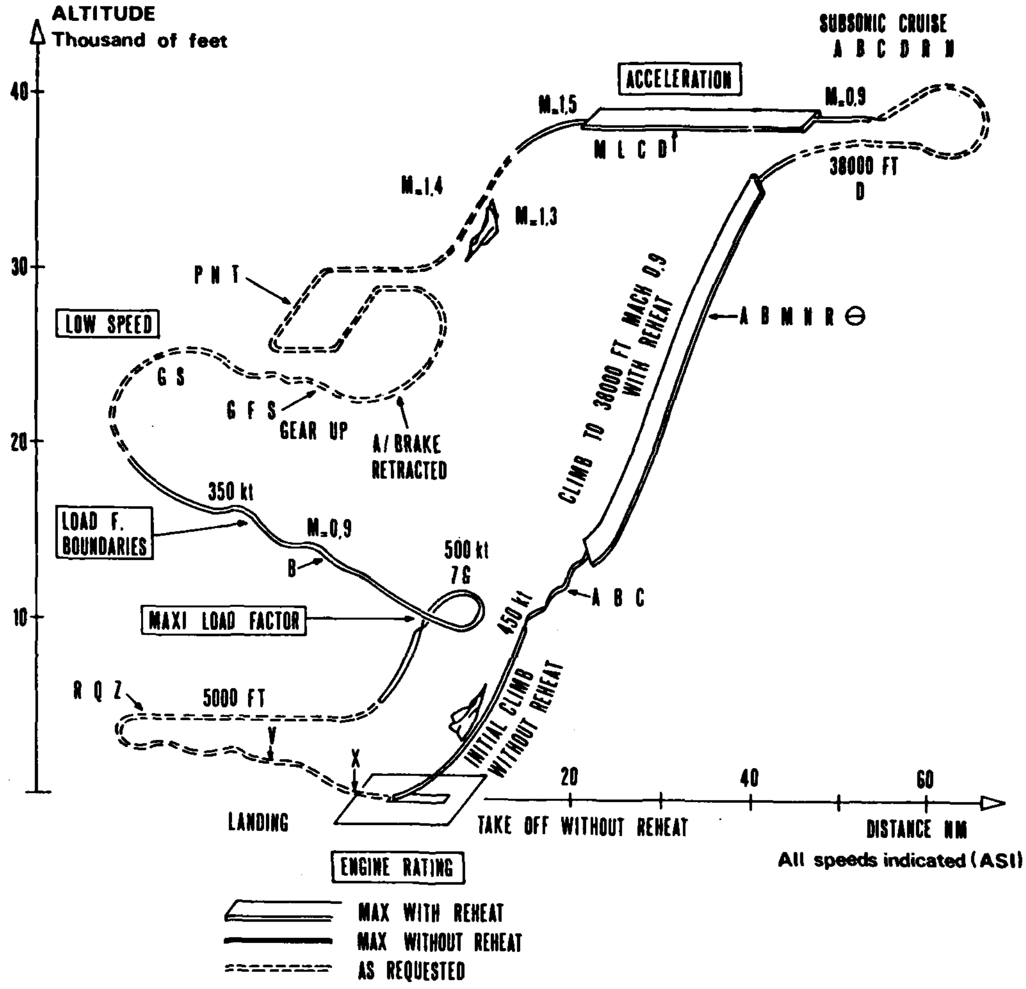


$\boxed{\text{STATIC PRESSURE POSITION ERROR}} = \boxed{dp \text{ MEASURED}} - \boxed{dp \text{ COMPUTED}}$

dp MEASURED:
- O BETWEEN PRESSURE RECORDED DURING EACH RUN AND AT HOLDING POSITION ON GROUND

dp COMPUTED FROM:
- O HEIGHT OF TOWER
- O AIR DENSITY (ATMOSPH PRESSURE and TEMPERATURE)

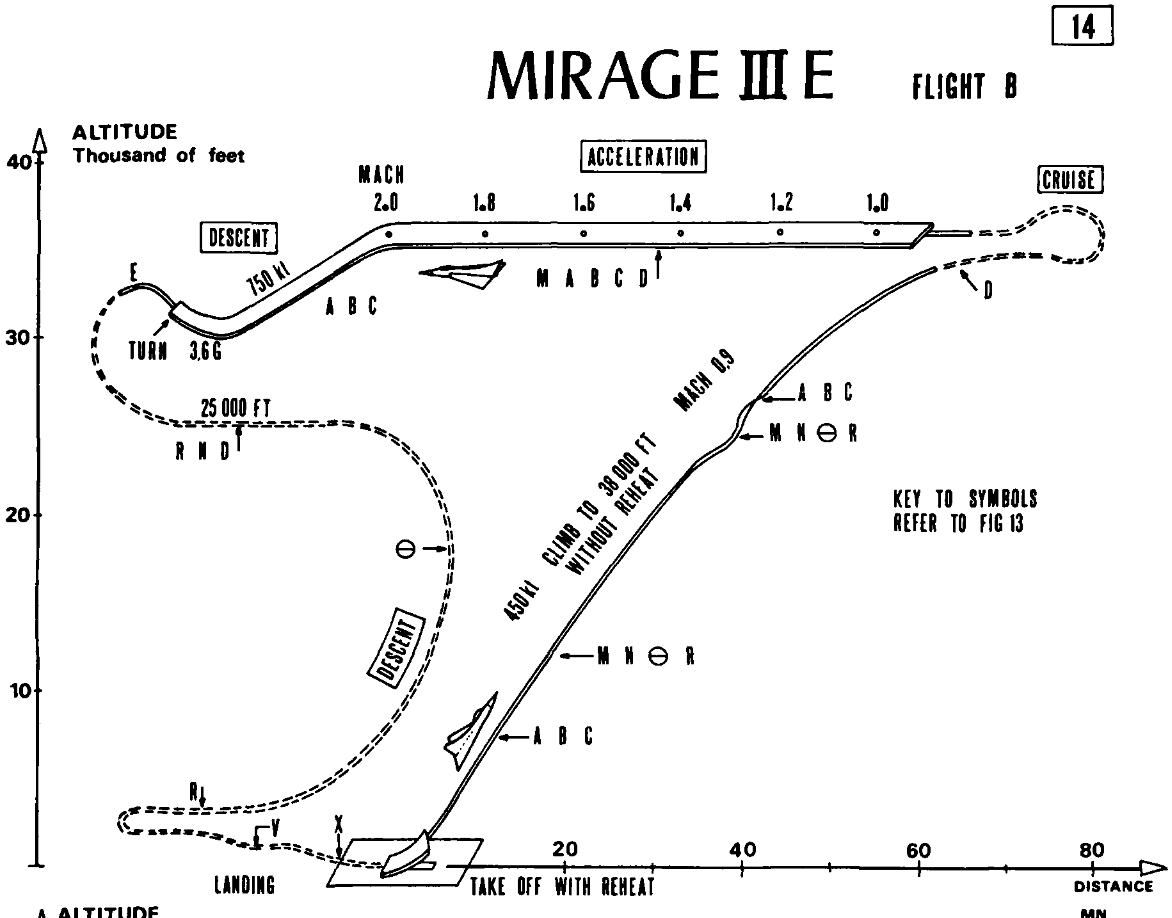
MIRAGE III E FLIGHT A

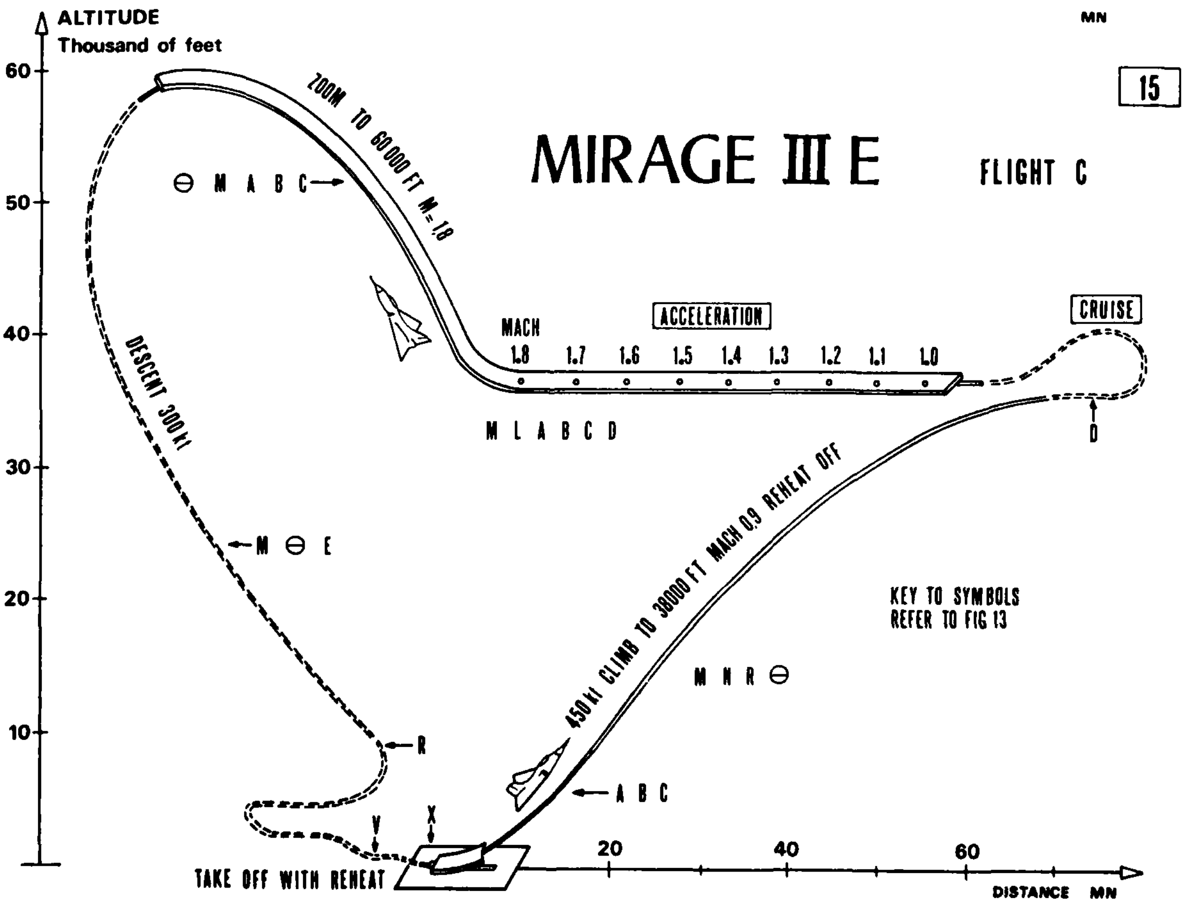


SPECIFIC TESTS

A	Automatic Control Syst.	N	Nav system
B	Pitch,Yaw Damper	M	Engine
C	Roll stabilizer	P	Equipment,Systems
D	Altitude Warning,monit.	T	Radio comm.
E	Airbrakes	S	Low speed warn.syst.
F	U/C Warning	X	Tail parachute
G	U/C extension,retraction	V	Autothrottle
K	Load factor	Q	Fuel quantity indic.
L	Engine overspeed	Z	Radio Altimeter
R	Radar,Weapon syst.	Θ	Air condit.syst.

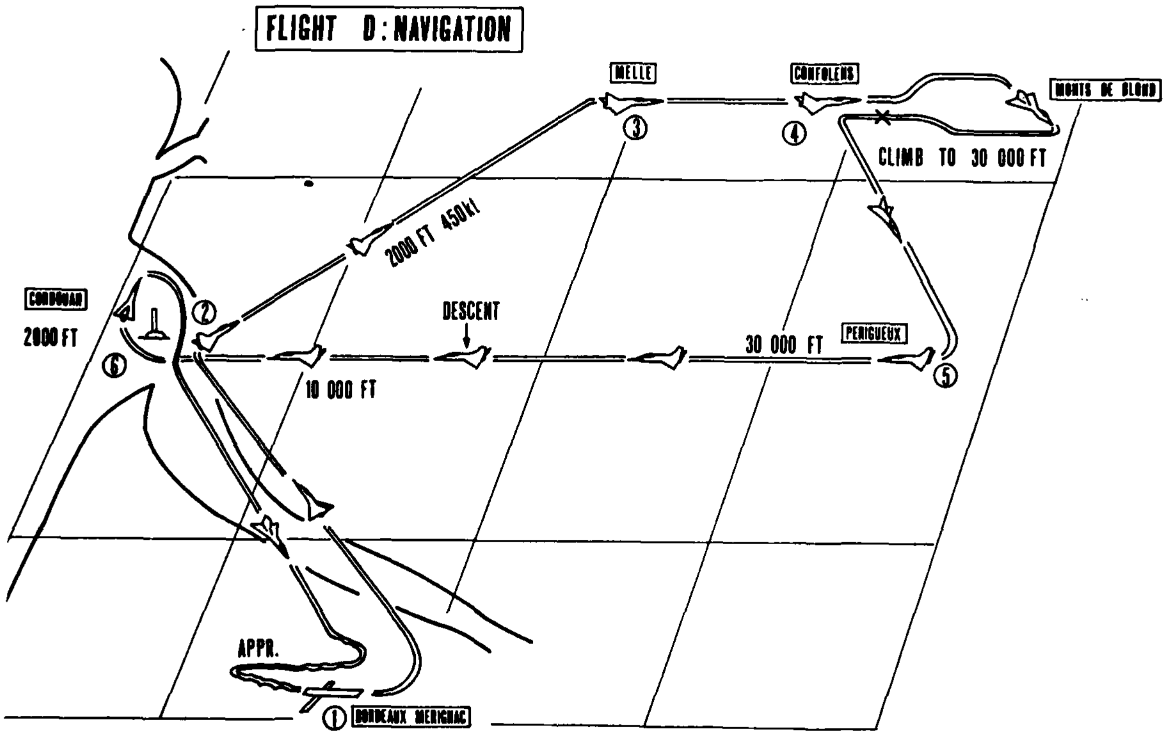
13


14
MIRAGE III E
FLIGHT B
ALTITUDE
Thousand of feet
40
30
20
10
ACCELERATION
MACH
2.0
1.8
1.6
1.4
1.2
1.0
CRUISE
DESCENT
750 kt
E
A B C
M A B C D
D
TURN 3,6 G
25 000 FT
R N D
MACH 0.9
A B C
M N ⊖ R
450 kt CLIMB TO 38 000 FT WITHOUT REHEAT
KEY TO SYMBOLS
REFER TO FIG 13
⊖
DESCENT
M N ⊖ R
A B C
R
V
X
LANDING
TAKE OFF WITH REHEAT
20
40
60
80
DISTANCE
MN


ALTITUDE
Thousand of feet
60
50
40
30
20
10
15
MIRAGE III E
FLIGHT C
ZOOM TO 60 000 FT M=1.8
⊖ M A B C
ACCELERATION
MACH
1.8
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
CRUISE
M L A B C D
D
DESCENT 300 kt
450 kt CLIMB TO 38000 FT MACH 0.9 REHEAT OFF
M ⊖ E
KEY TO SYMBOLS
REFER TO FIG 13
M N R ⊖
R
A B C
V
X
TAKE OFF WITH REHEAT
20
40
60
DISTANCE MN

MIRAGE III E

16



USAF DEVELOPMENT TEST AND EVALUATION

by

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SUMMARY

The test and evaluation of new weapon systems is an integral part of the total acquisition process. The major emphasis in the early development stages is placed on quantitative test and analysis to determine functional adequacy and specification compliance. Highly instrumented aircraft are used to provide data on aircraft performance, flying qualities, structural integrity and subsystem operation, This paper addresses the management and test procedures used at the Air Force Flight Test Center (AFFTC) to plan, conduct and report on the Development Test and Evaluation (DT&E) phase.

INTRODUCTION

The system acquisition process may be initiated through a request for proposal which details the operating characteristics and capabilities of the desired weapon (F-16, A-10) or support system (C-5, KC-10). Interested contractors then submit design proposals intended to demonstrate their solution to the operational requirement. These proposals are then studied from both technical and economic standpoints and a contract awarded for production of prototype article for initial development evaluation. A more recent adaptation of this procedure has evolved as a "fly-before-buy" concept utilized by the United States Air Force (USAF). In this process, the top two proposals are identified and a limited number of vehicles are built in prototype for a competitive fly off. This allows an intensive evaluation of the competing entries in the major areas of performance, flying qualities, structures and system evaluation. Confidence in the final selection is naturally much higher than that achieved from studies of "paper airplanes" because major problems may be corrected earlier in the acquisition process. This procedure has been successfully used in A-9/A-10, F-16/F-17, Advanced Medium Short Takeoff and Landing Transport and Air Launched Cruise Missile developments.

A view of total systems acquisition process (figure 1) may be helpful in putting the major decision points and the interrelationships of participating organizations in perspective. Emphasis in this paper is on the DT&E process which is the test and evaluation conducted to demonstrate that engineering design and development are complete, that design risks have been minimized, and that the system will meet engineering and operational specifications. DT&E is essentially a detailed engineering analysis of a system's performance beginning with individual subsystems and progressing through a complete system, where system design is tested and evaluated against engineering and performance criteria by the implementing command (Air Force Systems Command in the USAF). Operational Test and Evaluation (OT&E) is usually an early and concurrent phase of the acquisition process as depicted in figure 1. OT&E is conducted to estimate a prospective system's operational effectiveness and operational suitability and to identify any operational deficiencies and need for any modifications. In addition, OT&E provides information on organization, personnel requirements, doctrine, and tactics. OT&E is essentially an operational assessment of a system's performance where the complete system is tested and evaluated against operational criteria (requirement and employment concepts) by personnel with the same qualifications as those who will operate, maintain, and support the system when deployed.

The policy on Test and Evaluation is stated in Department of Defense Directive (DoDD) 5000.1 Major System Acquisitions (reference 1). The stated policy is "Test and evaluation shall commence as early as possible. An estimate of military utility and of operational effectiveness and operational suitability, including logistic support requirements shall be made prior to large scale production commitments. The most realistic test environment possible and an acceptable representation of the future operational system will be used in the testing." This policy is further explained in DoDD 5000.3, Test and Evaluation (reference 2) which directs that "In each DoD component there will be one major field agency, separate and distinct from the developing and procuring command and from the using command, which will be responsible for OT&E." In the Air Force, this agency is the Air Force Test and Evaluation Center (AFTEC), a separate operating agency responsible for the overall management of OT&E. AFTEC's role, detailed in Air Force Regulation (AFR) 23-36 (reference 3), is to plan, direct, control and independently evaluate and report on OT&E. While OT&E ideally should be separate from developmental testing, the early phases may need to be combined with DT&E. DoDD 5000.3 policy states: "Operational testing should be separate from developmental testing. However, developmental testing and early phases of operational testing may be combined where separation would cause an unacceptable increase in the acquisition cost of the system." In practice, a great deal of the two test phases are accomplished concurrently for major system acquisitions to conserve costly test resources and reduce test time. The management structure which allows this is shown in figure 2. AFR 80-14 (reference 4) details the operating interrelationships which are discussed subsequently in this paper. A glossary is included to define terminology used in

DT&E is carried out to assess the critical questions and areas of risk of the system and to meet the development objectives specified in the program documents. DT&E should accomplish the following (reference 5):

1. Provide data to the contractor for the development process.
2. Identify deficiencies in the system, deficiencies in the development contract specifications, and determine the degree to which those specifications have been met.
3. Insure the compatibility and performance of the support items (for example: simulators, life support systems, support equipment, computer resources, technical manuals and other data).
4. Provide estimates of deployed system reliability and maintainability to be expected when deployed.
5. Determine if the system is safe and ready for operational testing.
6. Accumulate and provide data to estimate the survivability, vulnerability, and logistics supportability of the system.
7. Provide data for the compatibility and interoperability of the new system/equipment.
8. Provide data for refining estimates of requirements for training programs and training equipment.
9. Provide information on environmental issues to be used in preparing impact assessments.
10. Insure design integrity over the specified operational and environmental range by conducting preproduction qualification tests.
11. Validate and verify technical orders and computer software.

TEST PROGRESSION

Most DT&E programs have a set pattern of evolution. Early participation by AFFTC, using command and logistics personnel includes program and design reviews, reviewing and commenting on documents, such as specifications and hazard analyses, preparation of detailed test plans, monitoring ground tests, and obtaining data for later use. Component qualification tests at the vendor's facility are not usually monitored by the AFFTC.

The overall evaluation begins as the first aircraft is assembled and the various subsystems are integrated for functional operation. Initial ground tests emphasize individual subsystem development and "debugging" which includes the interfaces among all the major subsystems. Early flight tests demonstrate basic vehicle airworthiness, handling qualities and satisfactory function of major subsystems such as flight controls, landing gear, engine, hydraulics and electrical. The instrumentation systems are also evaluated for function and accuracy during the initial flight stages as well as acquiring data for analysis. Flights progress with basic envelope expansion in the structural, flutter and flying qualities areas. Initial functional systems tests are performed concurrently. The contractor normally maintains the aircraft during this period with subsequent Air Force maintenance participation and finally, full Air Force maintenance in subsequent test phases. As the flight test activity progresses, the focus changes to an overall assessment of the total aircraft by subjecting the aircraft to environmental extremes, allowable limits and projected operational usage. Some of the subsystems are evaluated separately to obtain baseline information prior to evaluating the aircraft as a total weapon or support system. These tests are basically a continuation of the initial tests but the emphasis is testing against the operational requirement, testing for specification compliance, and determining the functional characteristics. The tests are designed to quantitatively and qualitatively assess the capability of the total system to perform its design mission. Although not an integral part of the aircraft, support equipment must be evaluated as part of the total system. Reliability, maintainability, human factors, technical order, and other areas are evaluated to complete the assessment of support equipment and man machine interfaces (reference 6).

PROGRAM MANAGEMENT

Planning

Test programs require a considerable amount of planning, which includes defining objectives, organizing and staffing the test organization, identifying and obtaining required support, preparing detailed test plans and establishing reporting requirements. However, there are certain requirements that merit special attention. These include the following:

1. Aircraft, systems and support equipment should be configured as closely as possible to a production version. A frequent problem is that the test agency is usually constrained to the use of preproduction aircraft and equipment which may not be updated in time to allow effective evaluation of the changes.

2. Instrumented aircraft and ranges must be available to provide data for quantitative analysis of test results.

3. Adequate calendar and flying time must be allocated for an effective evaluation.

4. The resources necessary to accomplish the program, including personnel and facilities, must be provided.

5. The Air Force (customer) should maintain the test aircraft with adequate personnel, training, and skills to accomplish the test program effectively.

6. Technical manuals that are adequate for use by maintenance, aircrew, and engineering personnel must be available.

Organization

At the AFFTC, the test organization for a major program is normally designated as a Combined Test Force (CTF) which is organized and manned to accomplish all of the program objectives, including all participating commands, and integrate all test and evaluation activities. A typical CTF organization is shown in figure 2. To simplify the diagram, lines of communication within the CTF are not included. The number of personnel required is dependent upon the objectives, number of aircraft, complexity of the aircraft, and flying and calendar time required.

Present day major flight test programs focus on combined (AFFTC, contractor, AFTEC and operating and support commands) DT&E/IOT&E (Initial Operational Test and Evaluation) programs (reference 7). The AFFTC and the contractor are responsible for the DT&E portion of the program and AFTEC manages the IOT&E portion. Two of the major areas of emphasis in working with personnel from the participating commands are to minimize duplication of testing and to insure testing is oriented toward real world requirements. The focal point for operational test requirements is AFTEC. Although much aircraft testing has an operational flavor, critical parameters are rigorously controlled for an engineering assessment of test results.

One of the key requirements in establishing a test organization is definition of responsibilities. The concept used on the YF-16/YF-17 Lightweight Fighter Prototype Program, which met with considerable success and is being used as a model for other programs included the following:

1. Close working relationships and coordination between DT&E and OT&E test director.

2. Combined DT&E/OT&E test teams. The test team was comprised of personnel from the AFFTC, AFTEC, participating commands and contractors.

3. Combined test plans. Preparation and coordination of specific plans were accomplished jointly by Air Force and contractor personnel. A single plan was published for accomplishment of both DT&E and OT&E objectives.

4. Participation of Air Force pilots in all phases of testing.

5. Integration of Air Force and contractor engineers and pilots. The Air Force personnel were physically collocated with their contractor counterparts.

6. Availability of all data to all members of the joint test team which facilitated independent analyses and reporting.

7. Combined Air Force (AFFTC, AFTEC and participating commands) deficiency reporting represented the total expertise and general consensus of the Air Force test team personnel.

8. Independent analyses and reporting allowed each organization to express its own views relative to its basic assumptions. This also gave the Air Force Management Agency (Program Office) added confidence in the test results on which all organizations agreed. In addition, it gave the Program Office more visibility and the opportunity to follow up on areas of disparity.

Total Program Integration

Numerous tests and evaluations must be accomplished to adequately satisfy program objectives. One of the keys to overall systems testing is the integration of test planning and test activities.

Normally, all of the major systems and components are evaluated. Some are evaluated separately to obtain baseline data before evaluating the total system. As an example, accuracy of the ranging of a forward looking radar is assessed prior to determining air-to-air gunnery or weapon delivery accuracy.

Specific evaluations must also be integrated since many of them overlap. As an example, the human factors coverage must overlap into both the pilots' and the maintenance evaluations of the cockpit. In turn, pilot inputs are an important part of the stability and control evaluation.

Instrumentation and Data Processing

Instrumentation and data processing systems are extremely important components of systems evaluation. The introduction of magnetic tape to airborne instrumentation systems approximately 20 years ago gave rise to a variety of instrumentation hardware and software developments. Many airborne systems were developed in the United States by aircraft contractors to meet their own unique requirements. Compatibility with other facilities, such as the AFFTC, was not a design requirement since it was assumed that most of the data would be processed at the contractor's home plant. However, in many instances, the USAF eventually obtained the aircraft for testing. The AFFTC became involved with almost every type of aircraft and instrumentation system ever built in the U.S. This led to numerous support problems such as incompatibility with data processing hardware and software, maintainability problems, and long data turnaround times because it was necessary to use nonproduction oriented operations to force data through the system. In addition, the system had little or no residual value to the USAF because of the proliferation of many unique, modified and often undocumented systems.

These problems, combined with the evolutionary changes in test philosophy towards combined testing, mandated standardization of data acquisition and processing systems. This led the AFFTC to develop under contract, a modularized standard instrumentation system which could be easily maintained and supported as well as being compatible with AFFTC data processing facilities. This system is currently being used or is planned for use in a wide variety of future aircraft test programs. Its development and use has alleviated many of the past data acquisition system problems which confronted the AFFTC.

Similar problems were encountered with data processing hardware and software. Data acquisition system commonality solved some of the problems in these areas. In addition, action was taken at the AFFTC to develop and document a library of general purpose computer subroutines which can be combined with airplane unique subroutines such as engine thrust calculation procedures. These approaches, the General Subsystem Analysis Program (GSAP) and Uniform Flight Test Analysis System (UFTAS), have alleviated many of the software development, checkout, control and documentation problems which occurred on past programs (reference 8). Additional detail in the use of the computer in flight test is continued in reference 9.

Reporting

Written AFFTC reports on DT&E results typically include deficiency, management and final reports. OT&E results are reported separately and independently by AFTEC. A significant amount of effort is expended on deficiency reports since these are considered action documents that are used to identify problems which should be corrected or studied for possible correction or refinement. The most effective method of preparing the deficiency reports is to have integrated inputs to each report from all interested organizations on the combined test team.

Management reports are submitted on a regular basis to the Program Office and other key organizations. Normally, they are submitted on a monthly basis and include program status and limited or summarized test results.

Final technical reports are published after the program is completed, or when significant portions of the test have been accomplished. Special emphasis is placed on including positive features as well as problem areas. Qualitative information and quantitative data are included. A compact test summary report is also published on some programs. The objective of this report is to present a concise overview of the entire test program in one document.

The AFFTC has also established a "corporate memory" data bank to feed back and compile problem areas with program management and major hardware problems. The intent is to have information available to aid personnel on new programs and thereby maximize utilization of past experience.

TEST AND EVALUATION

Certain tests, common to all aircraft programs, include functional tests of systems such as propulsion and avionics and integrated areas of evaluation, such as testing at various climatic extremes. There are also evaluations such as the reliability and maintainability assessment which do not normally require dedicated flight time. Although not an integral part of the aircraft, support equipment must be evaluated as a part of the total system. Following are discussions of each of the major evaluations accomplished during DT&E.

While space does not permit detailed coverage, examples of instrumented parameters, test methods and generalized analysis procedures are discussed. A major share of the analysis workload is comparison of the quantitative data to contract specifications. This activity is common to all the following subsystem discussion and will not be repeated under each heading.

Performance

Performance testing involves a large number of parameters ranging from as few as 20 parameters on a limited program to as many as 200 parameters for a multiengine aircraft program. Some of the primary parameters include airspeed, altitude, normal and longitudinal flightpath acceleration, angle of attack, total temperature, fuel flow and fuel quantity. Engine instrumentation often accounts for the bulk of the performance instrumentation with engine rpm's, temperatures, pressures and fuel flow.

The objectives of performance testing are to evaluate the aircraft against various performance specifications and to generate a performance model used for Flight Manuals and flight simulators. Basically the task involves generating a thrust-drag-fuel flow model from conventional (climb, cruise, accelerations and turns) and dynamic (roller coaster, windup turns and split-S) maneuvers. The resulting model should be able to accurately estimate the performance (excess thrust, fuel flow) of the aircraft for any set of conditions within the operational envelope. Data from the conventional maneuvers (such as climb rates, specific fuel consumption, turn rates) are also presented to give confidence to the model and for comparison to other aircraft.

Flying Qualities

A representative listing of measurands for flying qualities testing would fill several pages and be dependent on the design of the flight control system of the aircraft being tested. A representative number of 60 to 75 parameters would be valid for a tactical fighter. Primary examples include indicated airspeed, pressure altitude, ambient temperature, angle of attack and sideslip, control surface positions, fuel quantities, aircraft attitudes and angular rates, three-axis accelerations, control forces, and some internal flight control parameters.

The primary objectives are to determine if the aircraft flying qualities meet the Air Vehicle Specification, to evaluate its ability to perform the design missions and to develop a stability and control model of the aircraft for use in flight simulators. Real time monitoring of engineering units data is important during these tests, especially during hazardous missions such as high angle of attack flights. Typical postflight data analysis involves processing of dynamic maneuver data through a parameter identification computer program to extract stability and control derivatives, analyzing pseudosteady state maneuvers such as windup turns and steady sideslips for specification compliance, analyzing rolling maneuver data for time-to-bank, maximum roll rate, roll coordination, and roll coupling susceptability, and evaluation of data from pilot-in-the-loop tasks such as air-to-air and air-to-ground tracking, formation flight, in-flight refueling and Instrument Landing System (ILS) landings for frequency response and task performance information.

Flutter

Primary flutter instrumentation consists of accelerometers placed in wing and empennage locations to record surface movement. This is a test where real time analysis is a must. Data from the accelerometers are telemetered (TM) to a ground station for analysis. The test points are done to examine the trends of frequency and damping of critical modes. Critical test conditions are established from predictive analyses which are based on extensive wind tunnel and ground vibration investigations. Surfaces may be excited by mechanical methods (shakers), aerodynamic vanes or by abrupt inputs through the aircraft control system. Data recorded at the ground station are processed through a minicomputer based time series analyzer with associated software. In near real time the flutter test engineer analyzes damping characteristics and establishes trends with respect to airspeed to determine if the test may progress to more critical conditions (reference 10). Extensive postflight, or posttest, analysis is made against the predictive analysis to determine a safe operating envelope for the clean aircraft or with weapon configurations.

Structures

Structural instrumentation consists of strain gauges located throughout the aircraft on the major load carrying members. The gauges may provide stress levels or may be combined to give shear, bending or torque loads. The test engineer uses buildup techniques in aircraft maneuvers and dynamic pressure to insure that loads do not exceed the limits of the structure. Engineering units conversion and combining of loads, using online minicomputer systems, is extremely helpful. Postflight analysis of the exact test condition provides a correlation data point with the predictive loads analysis.

Airframe Subsystems

The airframe systems include the structure, canopy, landing gear (brakes, steering, etc), flight controls, secondary power (hydraulics and electrical) and environmental (cabin conditioning, oxygen system, pressurization, etc) subsystems. Many of the airframe subsystem evaluations are conducted concurrently with other tests although some require dedicated time.

Instrumentation is mandatory for tests of the brakes, flight controls, secondary power and environmental systems, and the arresting hook. Brake instrumentation typically includes electrical signals, torque, pressures, and temperatures. Hydraulic pressures and temperatures and electrical voltages and frequencies are instrumented. Extensive hydraulic and environmental control system instrumentation is required for climatic tests at extreme ambient temperatures.

The primary objectives of evaluating airframe subsystems are to determine their functional adequacy, operating characteristics, and potential operational effectiveness. Emphasis is placed on testing at flight conditions which closely approximate expected operational usage. Specific tests are also integrated with climatic tests, which are discussed in another section of this paper.

Of primary concern are basic flight subsystems and those which allow the system to accomplish its design mission(s). Evaluating the integration of these subsystems with the airframe system is an important part of the total system test. Evaluations which require special attention because they are potentially hazardous include simulation of degraded modes of operation, maximum performance braking tests, and aircraft/arresting gear compatibility. Braking performance on wet runways is also determined. In addition, certain tests are accomplished to determine the effects of ground operation such as the water spray characteristics induced by the nosewheel tires. Of primary importance are the effects of water ingestion by the engine.

A functional analysis of component design and failure frequency is a major part of the assessment which requires no instrumentation. These problems are documented and recommendations for correction made in the Discrepancy Report previously discussed. These reports form a substantial input to the airframe as well as all subsystem analyses discussed in this paper.

Quantitative data analysis may vary widely for airframe subsystems. Brake temperatures may be evaluated to identify a need for increased brake stack sizing or in determining handbook stopping distances under varying speed/gross weights. Hydraulic pressures and flow rates are utilized to evaluate system capacity to handle varying workloadings. Electrical parameters can pinpoint system transients or overload conditions. Tailhook loads are analyzed to determine barrier engagement types, gross weight limits, speed limits or offcenter maximums. Environmental control system pressures, temperatures and flow rates are used to evaluate system turbine efficiency, heat exchanger sizing and overall system capacity.

Propulsion and Fuel Systems

For purposes of this paper, the propulsion systems include the engine, fuel system, auxiliary power unit, jet fuel starter and emergency power unit. Although extensive testing is accomplished in test cells and wind tunnels prior to flight, many of the variables to which an aircraft is exposed cannot yet be duplicated on the ground. As a consequence, propulsion system flight testing remains the only true test of integration between the engine, flight airframe, and related systems.

Typical parameters to be measured are fuel flow rates, throttle position, compressor/turbine pressures and temperatures, fuel quantity, and jet fuel starter temperatures and pressures.

The primary propulsion system test objectives include determination of functional effectiveness. Test conditions as close as possible to operational usage are stressed. Tests are also integrated with other specialized types of evaluation, such as climatic and icing tests, which are addressed in other sections of this paper.

Of primary concern are the operating characteristics of the engine. They are evaluated during ground and air starting, engine transients, afterburner operation, extreme aircraft maneuvering such as air combat maneuvers, and high angle of attack flight. The air start and afterburner operation envelopes are normally verified. Real time data is essential for certain tests of single-engine fighters such as air starts.

Obtaining a high amount of operating time on one or two engines is normally stressed to identify any problem prior to high usage of the engine in the field. The evaluation of propulsion performance parameters which are directly related to aircraft performance, e.g., fuel flow and specific fuel consumption, is also addressed.

Other areas of investigation include engine/bay compatibility, engine/armament compatibility and external noise level surveys. Exhaust gas signature characteristics are an important consideration for aircraft to be operated in a combat environment. Aerial refueling normally requires developing an operating envelope for inclusion in the aerial refueling handbook.

The jet fuel starter is evaluated during ground and air starts. Evaluations of the auxiliary power and emergency power units also include both ground and in-flight starting and operation, with emphasis placed on their ability to provide sufficient power output to accomplish their intended function under all expected operational conditions.

Analysis of these parameters results in the identification of airspeed/altitude envelopes for engine restart and afterburner relight capability for publication in the Flight Manual. The fuel control may be rescheduled based on engine thermodynamic and mass flow analyses. Jet fuel starter operational limits may be extended or limited as a result of parameter analysis in that subsystem. In-flight refueling envelopes are defined for inclusion in the Flight Manual.

Avionics Subsystems

The ultimate objective of avionics subsystem testing is to verify individual subsystem performance while evaluating the adequacy of the total integrated avionics package. One result of the sophistication of today's aircraft avionics systems is the difficulty in isolating systems for individual evaluation or troubleshooting. The latest technological trend in integration of aircraft avionics systems is to employ digital data busses with internal time division multiplexing. This creates unique and challenging problems in instrumentation and software. Software is an integral part of the complete data acquisition system, and preparation for the evaluations begins with the initial software development and component fabrication. Early correlation of measured parameters with evaluation criteria is required as a definition of the interface between airborne recording systems and ground processing of data. Subsystem components, instrumentation, and software are exercised and refined during preliminary development with dynamic simulation in an integration laboratory. Individual subsystem development dominates the initial flight tests. As the test activities progress, the focus changes to overall assessments of the total integrated avionics systems by subjecting the aircraft to allowable limits, projected operational usage, and environmental extremes.

There are a multitude of individual avionics subsystems and interfaces which may require evaluation and numerous ways of categorizing avionics equipment by functional elements. For discussion purposes, subsystems may be subdivided into the following major elements: navigation/guidance, fire control, penetration aids, communications, reconnaissance, automatic flight control, central integrated checkout and auxiliary electronics equipment.

Rather than give a brief discussion of evaluation requirements for a number of the above subsystems, a system involving a number of interfaces will be treated in some depth. A terrain following system has been selected because it involves numerous avionics interfaces as well as with basic airframe subsystems such as the mechanical flight control subsystem.

Terrain following radar (TFR) subsystems usually provide low altitude terrain following, terrain avoidance and blind letdown capabilities. The TFR may consist of antenna receivers, transmitters, computers and power supplies, a radar scope panel and a control panel. The TFR receives inputs from the radar altimeter, attack radar, bomb-nav system or auxiliary flight reference system and central air data computer.

Typical data for TFR evaluation are gathered from the multiplexer bus on modern subsystems. These might include forward looking radar range and range rate, radar altitude, inertial groundspeed, aircraft g, pitch attitude and rate, and airspeed.

With current TFR systems, the aircraft may be flown manually or automatically at a preselected terrain clearance. Climb and dive signals generated can be coupled into the attitude director indicator (ADI) and heads-up display (HUD). In the automatic mode, the climb and dive signals are coupled into the pitch channel of the flight control system. The TFR mode can also be used to make blind letdowns to a preselected terrain clearance.

TFR test objectives typically include evaluation of manual and automatic terrain following modes, including an evaluation of performance against various terrain features such as sand dunes, trees, water, snow and sea-land transitions. The system must also be evaluated over terrain profiles which vary from gently rolling hills to abruptly changing mountainous terrain.

TFR evaluation provides an excellent area for the utilization of real time data (reference 11). A unique and complex application of the real time system was used to evaluate the B-1 in the TFR environment. The test engineer was provided with displays showing the terrain profile of a test course at the AFFTC and the aircraft flight profile above that course, as well as deviations from predictions. This application merged onboard TM data with range data from a number of sources. The flight test range radar and onboard navigation and control systems data were merged in the central computer to provide these real time displays for analysis.

A TFR evaluation is a prime example of a test which blends an evaluation of integrated subsystem performance with an assessment of aerodynamic characteristics. Avionics subsystems evaluated during TFR runs include the radar, air data computer and cockpit displays, including the HUD. Airplane control characteristics and performance capabilities must be factored into the terrain following system to insure that terrain clearance profiles are compatible with the aerodynamic characteristics of the vehicle.

Armament

The primary objective of armament testing is to determine if the aircraft/store combination can be employed to the criteria established by the technical specifications and user requirements. Aircraft/munition compatibility and delivery accuracy are the final tests in a complex effort starting with the conception of a munition and an aircraft which demonstrate the ability of both to function as a weapon system.

Prior to beginning the certification process on a given aircraft, which is discussed herein, the weapon itself goes through a development cycle. To illustrate, a conventional bomb is designed to meet a given requirement which dictates shape, size, fin configuration, warhead arming and fuzing. The weapon is manufactured as a prototype. Loading, fit checks and electrical compatibility tests are accomplished. The weapon is then tested for effectiveness against appropriate targets, arming/fuzing functions are evaluated and the ability to withstand airborne carriage environments is determined. If these tests prove satisfactory, the weapon is ready for certification on new and existing aircraft.

Separation characteristics are initially investigated in the wind tunnel. Model weapons may be separated from the wing in free fall or in captive mode through sting balance measurements and computer positioning. Pitch, yaw and roll rates are determined for varying airspeeds. The weapons are next separated from the aircraft with these parameters recorded by aircraft mounted cameras. This data is compared with wind tunnel results and a safe separation envelope (airspeed-altitude) determined for the aircraft/store configuration. The primary objective of store separation testing is to determine an airspeed/altitude envelope within which a store may be safely separated from an aircraft. This may be a single store release or ripple (as used in weapon delivery) or jettison of a bomb rack with weapons attached.

In addition to analysis of single store separation, store/store collision after release and jettison of full/partially full racks and fuel tanks are also of interest. Ripple release of bombs is evaluated to determine the minimum release interval to obtain the tightest possible ground pattern without bomb collisions in flight. Determination of ejection velocity is also important for use in accuracy analysis and ballistic evaluation as discussed later. Clearance envelopes must also be determined for emergency jettison of loaded or partially loaded racks or pylons. Careful analysis of center of gravity locations must be accomplished prior to tests of this type. External fuel tanks may usually be jettisoned full with minimum difficulty. Empty tanks normally present problems at higher airspeeds as they tend to fly after release and may contact the aircraft. Partially full tanks may be difficult to jettison due to tumbling tendencies induced by center of gravity variations depending on the method of mounting/jettison.

The instrumentation for armament system testing may be divided into three general categories: airborne systems to record aircraft parameters and photography, range instrumentation for external measurement of aircraft position/release conditions and weather data for computation with range data. Some typical examples of aircraft-instrumented parameters are airspeed, altitude, dive angle, pitch/roll yaw attitude and rate, g, radar range and range rate, and weapon release/fire event. HUD film records pipper position at the weapon event while external cameras provide data on weapon pitch/yaw/roll rates and position during separation. Range instrumentation consists of tracking radar and cinetheodolites to provide aircraft x, y and z position dynamic rates.

Weapon delivery accuracy evaluation comprises two major areas: error analysis and a statistical presentation of weapon scoring. Error analysis is conducted to determine the reasons why the bomb missed the target. Errors may arise from two sources: (1) the aircraft was not at the planned release conditions when the bomb was separated, and/or (2) ejection/separation effects were such that the predicted ballistic trajectory was not obtained.

Aircraft dynamics at time of release are determined by range space positioning instrumentation and the recording of onboard parameters. If the bomb missed the target, the aircraft release conditions are reviewed to determine if planned release conditions were met. Errors on the ground are correlated with release errors. The analysis becomes more complex with computer bombing systems in automatic modes since the aircraft position at release is not preplanned. Here the aircraft conditions at release are compared with where the computer "thought" the aircraft was and the analysis conducted from this point. When aircraft release condition errors are identified, steps may then be taken to correct the problems which caused them.

If the aircraft was at the programmed release conditions and the bomb still missed the target, the problem lies in release dynamics; (i.e., delayed release, ejection velocity error) or basic weapon trajectory. Onboard cameras are used to study weapon separation and determine anomalies in the immediate vicinity of the aircraft. Phototheodolites

are used to record weapon trajectory. The end result is an aircraft which can be positioned properly and a weapon which will hit the target (within the tolerances allowed in the weapon system specifications).

Computer batch processing plays an extremely important role in reduction and analysis of this data. Onboard and HUD data are merged and utilized in a predictive ballistics program to indicate bullet position relative to a target for air-to-air gunnery evaluation. A series of programs is also used in air-to-ground weapon delivery analysis to determine error sources and their effects on weapon impact-to-target relationship.

Finally, statistical analysis is prepared to present actual miss distance data in a form which allows assessment of the system capability. Circular error probable (CEP) is perhaps the most common method utilized. In this analysis, the radius of a circle is defined which will contain 50 percent of all bomb impacts.

Several papers in reference 12 address more specific aspects of range instrumentation and weapon test techniques.

Climatic Test

The overall goal of climatic flight test is to evaluate the aircraft, support equipment, maintenance procedures and related human factors while operating in the various environmental extremes. This is accomplished through four test phases: Climatic laboratory tests at Eglin AFB, Florida, arctic tests at Eielson AFB, Alaska, tropic tests at Howard AFB, Panama, and desert tests at Yuma Marine Corps Air Station, Arizona, or El Centro Naval Air Station, California.

The primary goals of the climatic laboratory tests are to establish subsystem baseline data under controlled conditions and to identify any potentially hazardous climatic related deficiencies prior to flight tests. Any major deficiencies discovered in the laboratory are corrected and the modifications are evaluated at the remote test sites. The laboratory provides a full exposure of temperatures from -54 to +52 degrees C (-65 to 125 degrees F). All systems including the engine are operated in the laboratory. Normally, the tests begin at 21 degrees C (70 degrees F) and are then lowered to -18, -32, -42, and -54 degrees C (zero, -25, -45, and -65 degrees F). Then they are raised to 21 and 52 degrees C (70 and 125 degrees F) with the aircraft exposed to simulated rainfall and solar radiation. Detailed aircraft inspections are accomplished after the cold and hot temperature exposure. Baseline data from the initial 21 degree C (70 degree F) runs and the 21 degree C runs after cold soak and heat soak tests are compared.

The arctic test phase is conducted during January and February when the maximum number of cold days is normally encountered. The overall objective is the evaluation of aircraft operations in an extreme cold environment. The aircraft must be exposed (soaked) at or below -29 degrees C (-20 degrees F) for a sufficient length of time to allow internal stabilization of component temperatures at or near ambient conditions. Tests are then flown to simulate mission profiles while utilizing all of the systems.

The tropic test phase is normally conducted during October and November. The May-June time period may also be an acceptable alternative to the fall period. After periods of rainfall, the aircraft is inspected for entrapped moisture and initiation of corrosion. Special attention is also given to the environmental control system, due to the additional load required to remove humidity in the cooling cycle.

The desert test phase is conducted during July and August to expose the aircraft to maximum ambient temperature conditions, combined with high solar radiation levels. The aircraft is heat soaked for four hours prior to each mission and a 72-hour heat soak test is accomplished. Particular attention is given the environmental control system, hydraulic and engine lubrication system. Special attention is also given the cockpit to exposure to solar radiation.

Some examples of the quantitative analysis output of this test phase are identification of design deficiency in hydraulic heat exchanger capacity, failure of hydraulic actuation times at low temperature due to fall off of flow rate and identification of environmental control system capacity to provide avionics cooling through mass flow rate limit. These are but a small sampling of potential utilization of the many parameters available for climatic test. Much of the data is presented in tabulated or graphical format to aid designers of future systems in coping with realistic extremes of the environment.

Much additional information on climatic, as well as other areas of test and instrumentation outlined in this overview paper, will be detailed in the AGARD Flight Mechanics Panel Symposium on Subsystem Testing/Flight Test Instrumentation. This symposium will be held in Geilo, Norway in October 1980 and proceedings will be available in the spring of 1981.

Adverse Weather

The basic objectives of adverse weather testing are to evaluate the aircraft and its systems during all phases of operation in simulated and actual adverse weather and to establish operating procedures and techniques for inclusion in the Flight Manual.

This is accomplished by evaluating instrument flight characteristics during various phases of flight (descents, instrument approaches, etc) and various simulated and natural climatological conditions (rain, turbulence, thunderstorm activities, etc). Other general evaluations include evaluation of the cockpit, night operations, ground handling on runway surfaces covered with ice, snow, slush, and water, and effects on systems.

The simulation tests are conducted under stringently controlled conditions prior to conducting tests in natural conditions. Rain and icing conditions are simulated by the use of KC-135 and C-130 aircraft water spray tankers. Special emphasis is placed on analysis of ice formation and texture to verify icing severity, water droplet size, and ambient temperature conditions. In the case of the F-16 aircraft, very stringent lightning tests will be accomplished on the ground prior to flying near thunderstorm activities. After the simulation tests are completed, testing is accomplished in natural conditions.

Reliability and Maintainability (R&M)

The R&M evaluation results in qualitative and quantitative information. The latter includes data products such as mean-time-between failure and maintenance man-hours per flying hour. The evaluation is heavily supported by maintenance personnel and must interface with the human factors efforts in a number of areas such as accessibility.

The basic procedure used at the AFFTC to obtain, store, retrieve, and analyze all maintenance and operational data is called the Systems Effectiveness Data System (SEDS). This system is an Air Force Systems Command unique system and is designed specifically for use on weapon systems under test.

The concept of building prototype aircraft to predict operational suitability and reduce risk had a resurgence in the U.S. in the early 1970's. The value of an R&M effort on a prototype program was the subject of considerable debate for a number of reasons. Normally, maintenance is performed by the contractor with very little (if any) USAF "hands on" maintenance activity. This limits the R&M evaluation to an over-the-shoulder observation exercise. Secondly, many features of prototype aircraft are not representative of a production article. As a consequence of these and other factors, objectives that have been developed as realistic expectations for prototype R&M evaluations are to: (1) identify system candidates for R&M improvement; (2) identify areas to retain prototype R&M; (3) identify system and component candidates for reliability verification testing; (4) compute achieved/forecast R&M parameters; and (5) update R&M analytical models. In addition, the maintainability evaluation will result in a qualitative assessment of operational suitability (access, skill level, inspection cycle, etc), analyses of fault isolation equipment, logistics support requirements and maintenance technical data, determination of compatibility with USAF facilities and assessment of training aids. In summary, the R&M evaluations are required to realize the full benefit of the prototype concept and to provide advanced indications of production system operation.

Human Factors Engineering (HFE)

The HFE evaluation is an integral part of the total weapon system evaluation and therefore interacts with all aspects and elements of the CTF. It is concerned with determining whether USAF personnel can operate, maintain, and support the weapon system in its intended operational environment. The specific objectives are to determine if: (1) human engineering requirements and criteria (noise, temperature, access, comfort, visibility, performance and anthropometry) have been incorporated into the system design and are adequate; (2) biomedical and safety criteria (lighting, toxic gas, acoustics, and ventilation) have been met; (3) the system provides for efficient human performance in its intended operational environment; (4) personnel (speciality codes, skills, and number) planning information is appropriate, complete, and adequate; (5) job performance aids are efficient and adequate; and (6) training and training equipment requirements have been met. In addition, there is participation in the technical publication verification effort.

The HFE evaluation is supported by the pilots and maintenance personnel. The methods of acquiring information include observations, interviews, debriefings, photography and cockpit evaluations. In addition, a video tape is invaluable in documenting human tasks and verifying and extracting results. Flight test instrumentation is also used for obtaining cockpit toxic gas data and temperature.

Systems Safety

The systems safety evaluation consists of reviewing hazard analyses accomplished by the contractor, addressing systems safety during planning of hazardous tests and classifying problem areas by hazard level. Although AFFTC personnel do not develop detailed hazard analyses, such as fault tree analysis, reviewing the contractor analyses offers the opportunity of becoming familiar with areas of concern, providing feedback to the contractor and Program Office and monitoring areas of concern throughout the program. A considerable amount of effort is expended in identifying hazards and hazard codes and effects of potential hazardous areas in deficiency reports. On some programs, safety related trends are determined by assigning hazard codes to in-flight discovered discrepancies.

CONCLUSION

The overall DT&E process is both extensive and complex. Properly executed, it results in many subsystem changes and refinements to meet both specification and operational requirements. It provides substantial amounts of engineering data and narrative evaluations and analysis of subsystem performance throughout the operating spectrum of the vehicle. Not all specifications or operating goals will be achieved; however, most shortfalls and their impact will be documented through the reporting process. Any nation intending to procure a weapon or support system which has undergone the extensive DT&E and OT&E cycle will do well to thoroughly examine the results. This examination, properly accomplished, will provide not only a detailed knowledge of system strengths and limitations, but also a baseline of information on which incountry evaluations may be intelligently planned. This will allow a concentration of effort within the areas of potential problems and on aspects of the system most vital to the buyer's intended operational usage.

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GLOSSARY (reference 5)

1. Acceptance Tests. Those tests performed to demonstrate that a specific lot of articles have been manufactured to specification tolerances.

2. Acquisition Process. Normally, it consists of five phases (conceptual, validation, full scale engineering development, production and deployment) with key decision points after each of the first three phases.

3. Availability. Availability is a measure of the degree to which an item is in the operable and committable state at the start of the mission when the mission is called for at an unknown (random) time (inherent availability).

4. Compatibility. The capability of 2 or more operational items/systems to exist or function as elements of a larger operational system or operational environment without mutual interference.

5. Development Test & Evaluation. That test and evaluation conducted to assist the engineering design and development process and verify attainment of technical performance specifications and objectives.

6. Evaluation. The review and analysis of qualitative and/or quantitative data obtained from design review, hardware inspection, testing, and/or operational usage of equipment.

7. Follow-on OT&E. That test and evaluation which is conducted after IOT&E to continue and refine the estimates made during the IOT&E, to evaluate changes, and to reevaluate the system to ensure that it continues to meet operational needs and retain its effectiveness in a new environment or against a new threat.

8. Implementing Command. The command responsible for the acquisition and/or modification of the system, subsystem, or item of equipment. For the U.S. Air Force, this is normally the Air Force Systems Command.

9. Initial Operational Test and Evaluation (IOT&E). That portion of operational test and evaluation conducted prior to Milestone III decision.

10. Logistics Supportability. The degree to which adequate provisions can be made in system's acquisition for support of test equipment, supply support, maintenance manuals, technical data, and support facilities.

11. Maintainability. A characteristic of design and installation expressed as the probability that an item will be restored to a specified condition within a given period of time when the maintenance is performed using prescribed procedures and resources. System maintainability may also be expressed in such terms as Mean-Time-to-Repair, Maintenance Man-hours per Flying Hour, or Mean Downtime.

12. Military Utility. A generic term used to describe the value of an item or system with respect to a current concept of operation.

13. Operational Suitability. The degree to which an operationally effective system can be satisfactorily placed in field use, with consideration being given to availability, producibility, compatibility, transportability, interoperability, reliability, maintainability, safety, human factors, electromagnetic compatibility, logistic supportability and training requirements.

14. Operational Test and Evaluation (OT&E). Test and evaluation conducted to estimate the system's military utility, operational effectiveness, and operational suitability, as well as the need for any modifications.

15. Preproduction Article. An article which is in final form, uses standard parts (or nonstandard parts approved by the agency concerned), and is representative of the final equipment.

16. Production Acceptance Test & Evaluation. Test and evaluation of production items to demonstrate that items procured fulfill the requirements and specifications of the procuring contract or agreements.

17. Prototype. First full scale functional form of a new system, subsystem, or component on which the design of subsequent production items is patterned.

18. Reliability:

a. Hardware Reliability. Hardware reliability is the probability that a part, component, subassembly, assembly, subsystem, or system will perform for a specified interval under stated conditions with no malfunction or degradation that requires corrective maintenance actions.

b. Operational Reliability. The probability that an operationally ready system will perform as required to accomplish its intended mission or function as planned.

19. Survivability. The degree which a system is able to avoid or withstand a manmade hostile environment without suffering an abortive impairment of its ability to accomplish its designated mission.

20. System Acquisition Process. A sequence of specified decision events and phases of activity directed to achievement of established program objectives in the acquisition of defense systems and extending from approval of a mission need through successful deployment of the defense system or termination of the program.

21 System Program Office. The office of the program manager and the single point of contact with industry, government agencies and other activities participating in the system acquisition process.

22. Test. Any program or procedure which is designed to obtain, verify, or provide data for the evaluation: research and development (other than laboratory experiments), progress in accomplishing development objectives, or performance and operational capability of systems, subsystems, components and equipment items.

23. Test Director. A person assigned to conduct a test in accordance with the test plan, and who exercises overall responsibility for achieving test plan objectives.

24. Test Plan. A formal document which gives the complete detailed coordination and integrated plan for the time phased task of giving answers and solutions to the critical questions and areas of risk identified in the program documentation. It must also list the resources required to conduct, analyze and report on the test.

APPENDIX A

FIGURES

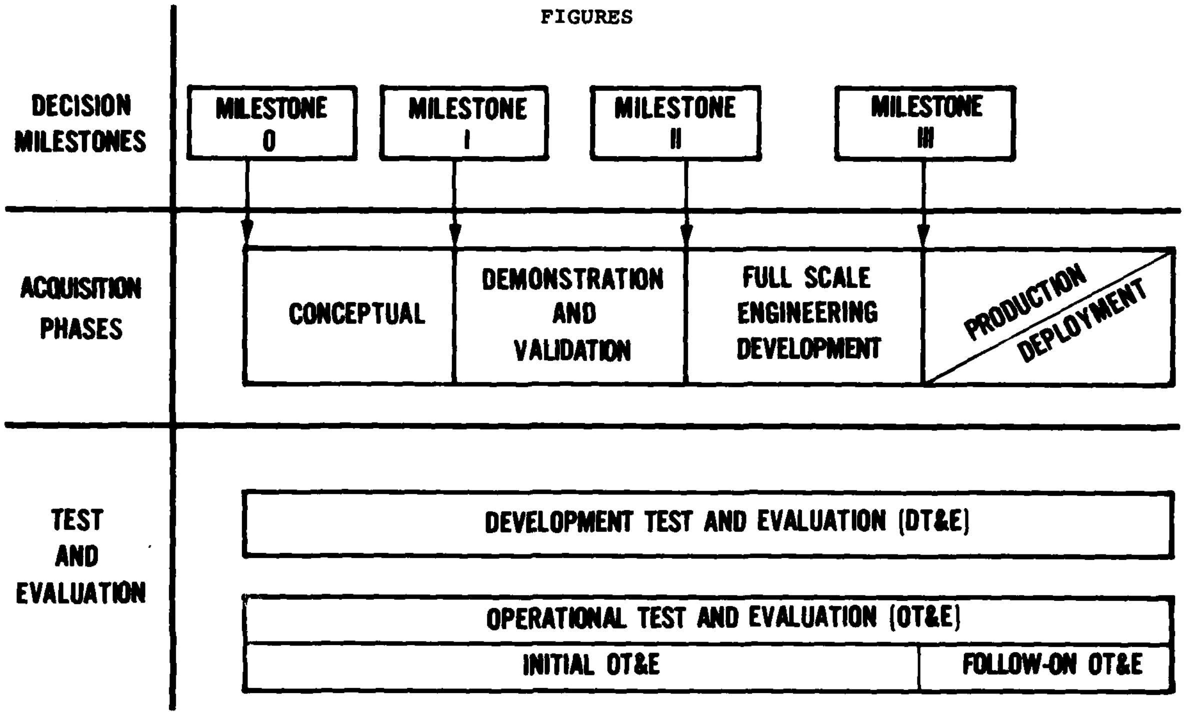

Figure 1 System Acquisition Process

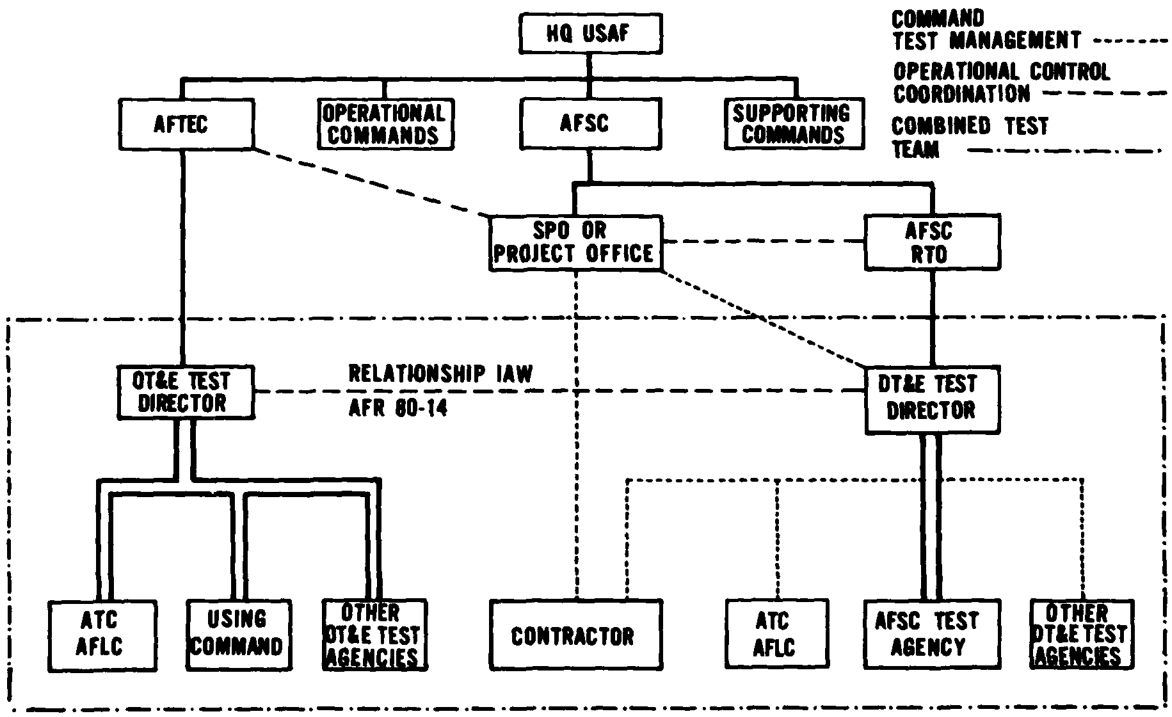

Figure 2 Test and Evaluation Organization

SOME EXAMPLES OF PROCEDURES USED IN U.K. FOR ACCEPTANCE TESTING OF AIRCRAFT PRODUCED BY THE AIRCRAFT INDUSTRY UNDER GOVERNMENT CONTRACT

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SUMMARY

A new aircraft produced by UK contractors is tested to show that it complies with its specification and its acceptability for service use is assessed by the Aeroplane & Armament Experimental Establishment at Boscombe Down.

These assessments are made, mostly on instrumented aircraft, and the scope of the testing carried out is illustrated here in order to indicate the range of data that can be made available to bodies evaluating a production aircraft.

The UK production aircraft are flight tested to a production flight test schedule to ensure consistent handling and performance throughout the fleet and an example of the content of a schedule is presented here.

The lecture is concluded with some general observations to be borne in mind when planning an evaluation.

GENERAL INTRODUCTION

Aircraft manufactured in the United Kingdom for the British Armed Services are built to a specification agreed between the Ministry of Defence (Procurement Executive) and a contractor.

This specification is evolved from studies conducted by departments of "The Ministry" and the Armed Services in conjunction with several contractors and is finally agreed between a contractor and MOD (PE). Sometimes aircraft are designed on a private venture basis by a contractor and are adopted by the MOD after which a specification is prepared (i.e. the BAe Harrier).

The specification includes a number of demonstrations which must be satisfied before the aircraft will be accepted by the customer and these are recorded in a document called the "Acceptance Standard".

The Acceptance Standard requirements may include demonstrations of satisfactory performance, stability and control, handling qualities and airworthiness and the specification is satisfied contractually when these demonstrations are satisfactorily completed. The Acceptance Standard will contain the requirement to meet the standards of airworthiness and stability and control laid down in AvP970 and memos or U.S. "Milspecs" (References 1 and 2) except where they are agreed to be invalid or obsolete by the contractor and MOD (PE).

Until recently all aircraft designed by UK contractors for the British forces were designed to meet or exceed the requirements of AvP970 which has been continuously amended since it was first published in 1935. Future British aircraft will be designed to meet or exceed the requirements of AvP970 and memos which are now very similar to "Milspecs". Some collaborative projects have been required to meet the American "Milspecs" requirements.

Before an aircraft is delivered for use by the Armed Services it must be granted a Controller of Aircraft Release to Service defining the limits to which it may be flown by Service pilots. These limitations are generally the result of recommendations by the Aeroplane & Armament Experimental Establishment who assess the aircraft's suitability for service use normally within structural and handling qualities limits explored by the contractor.

Some aspects of the flight test programme may be conducted jointly by the Aeroplane & Armament Experimental Establishment and the contractor, but the Aeroplane & Armament Experimental Establishment will normally repeat contractors tests where the results are of a subjective rather than quantitative nature. Aeroplane & Armament Experimental Establishment "participation" in the flight test programme may occur early in the test programme and they conduct a "Preview" on early independent assessment of the aircraft before the development programme is complete. The content of the Preview is chosen by the Aeroplane and Armament Experimental Establishment and in consultation with the contractor.

The contract for the aircraft will include the expected cost of the flight trials including the cost of test instrumentation fitted to a number of "development batch" aircraft. For example, the Hawk T Mk 1 programme involved six development aircraft. The aircraft were all instrumented to slightly different standards but with some overlap to allow flexibility in the trials programme, and in order that if one aircraft were lost its tasks could be readily re-scheduled to anoth

The structural design and integrity of a new aircraft is submitt for approval to the Royal Aircraft Establishment structures department, who also vet and approve the load and flutter vibration test results.

FLIGHT TESTING PERFORMED

In order to demonstrate satisfactory airworthiness, performance and handling characteristics the contractor will propose a flight trials programme which will include the following facets :

1. Measurement of aerodynamic loading and inertia loading on the airframe structure and the structural dynamic stability (i.e. freedom from flutter).
2. Measurement of the aircraft stability and control characteristics and assessment of departure behaviour.
3. Measurement of thrust and lift boundaries and take-off, landing, climb, cruise and descent performance.
4. Determination of engine handling characteristics including re-lighting performance and response to rapid throttle movements throughout the flight envelope envisaged for the aircraft.
5. Measurement of electrical, mechanical, hydraulic, avionic and armament system performance over the range of climatic conditions in which the aircraft may be required to operate.

6. Determination of safe carriage, jettison, release and firing envelopes of all stores carried by the aircraft.

7. Clearance of all the aircraft avionics, armament circuits and electronic control devices for freedom from electromagnetic interference.

MEASUREMENT OF AERODYNAMIC AND INERTIA LOADINGS ON THE AIRCRAFT STRUCTURE

From analysis of wind tunnel data and the airload distribution estimated during the structural design of the aircraft, a number of electrical strain gauge systems are applied to the structure of one of the development aircraft. These gauges are positioned wherever possible to record pure shear, pure bending and pure torsion.

The test aircraft is then mounted in a rigid loading frame and a system of hydraulic jacks is arranged to apply loads over the possible range of centres of pressure of the airloads on the airframe. These tests in effect calibrate the strain gauges mounted on the airframe and the structural deformation is related to the applied loads.

The aircraft is then removed from the test frame and flown at progressively more severe conditions culminating in the greatest speed/g envelope the aeroplane will be cleared to fly at.

The extremes of centre of gravity position and inertia distribution are flown during this clearance procedure. The flight recordings of strain gauge output are compared with extrapolations of the strains recorded during the aircraft's loading in the static test frame and the structural loads applied in flight are deduced.

This method has been used with varying degrees of success on many aircraft. Its weakness is that the results are dependent on how accurately the static loadings have been distributed on the airframe in relation to the unknown flight loads and on the extrapolation to flight from calibrated static loadings.

Symmetrical wing and tail-loads are recorded using pull up and turning manoeuvres and fin and rear fuselage torsional loads are obtained from rolling pull-out manoeuvres. On combat aircraft these manoeuvres are extended to include rapid rate rolls where the inertia distribution of the aircraft may allow coupling with the aerodynamic moments causing pitch or sideslip divergence to occur.

Ground resonance testing is used to confirm or revise the estimated structural elastic modes and frequencies and so, if necessary, to update the computed flutter speeds. The flight records of frequency and, in particular, damping of the structural modes are assessed and used to define any possible flutter boundaries. The structure is excited by flight in natural turbulence, by actuation of explosive devices attached to the extremities of the airframe (bonkers) or by rotating eccentric weights.

The behaviour of the structure is determined from the output of accelerometers fitted to various parts of the airframe, and the results in terms of vibration frequency and damping are compared with the ground resonance testing.

Predicted critical external store cases will be tested as well as the basic clean aircraft.

From the results of the aircraft loading and flutter tests the contractor determines a safe structural speed, altitude, normal acceleration, rolling and sideslip operational envelope for the aircraft.

The Aeroplane & Armament Experimental Establishment may decide to recommend a slightly more restrictive envelope for Service operation if stability and control is such that these limits may be easily inadvertently exceeded.

HANDLING & STABILITY & CONTROL ASSESSMENT

UK contractors measure an aircraft's stability and control characteristics for the following reasons:

1. To determine any handling limitations and to develop remedies for unacceptable shortcomings.

2. To demonstrate that the aircraft meets its specification and that it complies with the requirements of AvP970 and U.S. Milspecs where relevant.

3. To provide data for comparison with predicted stability and control characteristics and to thereby refine prediction methods.

4. To provide data for the programming of flight simulators for pilot training and for assessment of piloting techniques before performing them in flight.

The aircraft's longitudinal and lateral/directional handling characteristics are normally assumed to be independent during this testing as is described in many text books (references 3 & 4).

This assumption does break down in some instances, however, where high rotation rates about any of the aircraft axes cause lateral disturbances to generate longitudinal responses. Examples of such effects are inertia coupling during high rate of roll manoeuvres or the development of high incidence during fast spins or post stall gyrations. Unconventional airframe configurations may also exhibit unusual cross-coupling of the longitudinal and lateral behaviour.

Longitudinal assessments include the following aspects and the predicted centre of gravity limits within which the aircraft may be operated in service may be revised in the light of the test results :

1. Assessment of static stability. (Measurement of static and CG margins)

2. Assessment of manoeuvre stability. (Measurement of tailplane/g and stick force/g)

3. Assessment of dynamic stability. (Measurement of period and damping of the Short Period Pitching Oscillation and Phugoid)

4. Determination of handling characteristics at the stall, stall warning magnitude, and the aircraft's controllability near the stall.

5. Behaviour of the aircraft at high Mach No and assessment of trim changes in the transonic region.

6. Assessment of stalling behaviour at high speed. (Measurement of instantaneous usable lift boundary)

7. Assessment of the pilot workload when performing precision tasks such as target tracking and instrument flying and optimisation of the control system.

8. Determination of trim changes due to engine thrust and extension and retraction of flaps undercarriage and airbrake etc. and assessment of the adequacy of trim system ranges.

Lateral/Directional assessments include the following aspects :

1. Assessment of lateral and directional static stability. (Measurement of stability in steady sideslips)
2. Assessment of dynamic lateral and directional stability. (Examination of spiral, Dutch roll and roll subsidence modes)
3. Measurement of rolling performance.
4. Lateral and directional trim range assessment.
5. Assessment of crosswind landing performance and limits.
6. Inertia coupling tests.
7. Spinning and post stall gyration exploration for aircraft capable of achieving these flight conditions.

The contractors testing includes the effect of all likely under-wing stores combinations on the above assessments.

It is rarely possible to design store suspension systems that will allow a large range of stores to be hung under the aircraft wings without affecting the centre of gravity and aerodynamic centre positions. To examine the effects of stores on handling, the aircraft is normally ballasted to the forward and aft cg limits without stores and then flown with the stores that produce the greatest forward and aft cg movements respectively.

As stability is normally decreased by aft centre of gravity and forward aerodynamic centre movements the tests at the aft cg with aft movement due to stores normally produce the most critical handling. The forward centre of gravity limit is usually decided by the requirement to be able to flare the aircraft on landing with full flap at low speed. If the cg is too far forward the tailplane may not be able to control any nose down pitch even at full aft stick.

Forward cg limits may also be defined by rear fuselage or tailplane strength considerations.

From the results of all these tests the contractors may present handling limitations for the pilot to observe when manoeuvring the aircraft as functions of angle of attack or normal acceleration as a function of speed, altitude and store configuration. Alternatively limitations may be expressed in the form of recommendations not to dwell in flight conditions that produce heavy buffet or wing rock or to persist with manoeuvres where more than ½ control authority is used to maintain the flight condition.

The Aeroplane & Armament Experimental Establishment may not recommend the release to service of the contractors flight envelope if they consider that a limitation may be inadvertently exceeded with possible loss of control or structural failure.

Longitudinal and lateral testing is covered in more detail in Lectures 3 and 4 in session 2.

PERFORMANCE MEASUREMENT

An aircraft's performance is measured by contractors for the following reasons :

a) To provide flight measured data on which to base an "Operating Data Manual" (ODM)

b) To determine maximum and optimum performance flight conditions

c) To validate performance predictions

In order to produce an ODM it is necessary to determine any position error corrections that should be applied to the measurement of airspeed, altitude, OAT and incidence. These corrections are normally included in the ODM.

The results of the tests enable the ODM to be written to cover the following phases of flight :

a) Take-off and landing

b) Climb

c) Cruise

d) Descent

e) Sustained turning performance (Thrust boundary)

The detailed description of the methods used in performance measurement is covered in Lectures 1 and 2 of Session 2.

ENGINE HANDLING & RELIGHTING PERFORMANCE

Knowledge of an aircraft's engine handling characteristics throughout all the flight conditions it can achieve is essential for safe operation of the aircraft. For this reason the aircraft manufacturer conducts engine handling tests to determine surge boundaries and re-lighting tests in order to discover any flight conditions at which pilot throttle movement is critical.

When developing an engine the UK engine contractor may perform handling tests including slam throttle openings and re-lighting tests at sea level and at a range of altitudes in an altitude test cell.

From the cell tests surge and re-lighting boundaries are prepared and these give the contractor a guide to flight conditions at which to begin engine testing.

It is usual to investigate the engine's re-lighting performance under favourable conditions and then to establish as broad a re-lighting envelope as possible.

For example, the procedure adopted on the BAe Hawk was to commence re-lighting tests at the highest altitude cleared in the altitude test facility, and then to reduce speed at the constant altitude until the maximum TGT recorded during the light up reached the maximum allowable.

The test altitude was then reduced and minimum re-light speed determined at altitudes down to 10,000 ft.

The success of a re-light attempt is dependent on the engine windmilling speed, and if the speed is too low to achieve an effective light the windmilling speed is easily increased by lowering the aircraft's nose.

Many aircraft have the ability to attempt a hot engine re-light just by switching the ignition on immediately after closing the throttle and before the engine speed has decayed to the steady windmilling value. Such modes of operation are also checked during the aircrafts engine testing programme.

It is advantageous if the engine can be satisfactorily re-started at the best gliding range speed. If this is not possible due to a low engine windmilling speed the re-light may be assisted by the aircraft's self contained ground starting system if the aircraft is fitted with one. Of course the flight envelope for this device must also be determined.

After the engine re-lighting performance has been demonstrated under an easily achievable flight condition its handling conditions under all flight conditions are assessed.

Typical testing consists of slamming to full throttle as rapidly as possible from a range of steady conditions with and without high sideslip and incidence while monitoring the JPT to detect surges.

The effect of rapid throttle closures on the minimum engine speed is also investigated in case rapid throttle closure should extinguish the engine.

Should any critical handling areas be identified the aircrew manual will be written to reflect these characteristics.

STORES JETTISON & RELEASE

Combat aircraft carry a wide variety of weapons and additional external fuel tanks and sensors in order to perform their operational duties.

This equipment can be divided into 4 classes as far as flight testing is concerned :

1. Bombs and other offensive free fall devices
2. Guided and unguided missiles
3. Guns
4. Tanks and reconnaissance pods and unfired missiles etc. not normally released in other than level flight steady conditions.

Satisfactory separation of free fall devices is obviously dependent on the airflow distribution over the aircraft which is in turn dependent on the flight condition at the point of release.

The contractors carry out flight separation tests to determine a release speed, height and normal acceleration envelope (up to the limits required by the aircraft specification) at which each particular device can be released without striking the aircraft and without being disturbed to the extent that its accuracy is unacceptably impaired.

Under-wing stores are normally ejected from the pylons by rams driven by gas impulses generated by explosive devices. The distribution of the ejection force is optimised to result in the store clearing the aircraft flow field with the minimum disturbance.

Stores in class 1 require the greatest practical release envelope for operational effectiveness and flexibility. Stores in Class 4 need only be released in level flight but the release envelope is often more restricted than class 1 stores because they tend to produce more aerodynamic lift (especially empty drop tanks).

The firing envelope for guided and unguided missiles may be limited by the effect of their exhaust gases on the aircraft engine which may be prone to surge if it ingests hot gas flows from missiles passing close to and in front of the intakes. Unguided missiles are normally launched under approximately "1g" flight since they are normally used for ground attack with the pilot performing an essentially "1g" target tracking task.

Guided missiles which are used for air combat require a larger normal acceleration firing envelope since it is desirable to be able to launch these missiles under any flight condition.

Contractors whose aircraft are required to carry guns conduct in-flight firing trials for the following reasons :

1) To discover the location of any concentrations of unburnt cartridge gases in the airframe and to assess the efficiency of the ventilation of airframe cavities.
2) To determine any vibration effects on the aircraft's avionics or instrumentation and automatic flying control or stability augmentation systems.
3) To demonstrate that the engine is unaffected by gun firing gas ingestion or pressure and temperature effects.
4) To demonstrate that the aircraft's structure adjacent to the gun is not damaged by blast effects.

Stores release and jettison tests are normally commenced in level flight at speeds and altitudes where a safe separation is predicted from wind tunnel test results and calculations. The separation is filmed with high speed cameras from a chase aircraft or an on-board cine pod and the records analysed to obtain the store trajectory and the minimum separation. Measured results are then compared with the estimated separation and a sensible incremental increase in launch condition severity is formulated.

This procedure is repeated until the results indicate that the minimum practical separation has been achieved.

A jettison release envelope can then be presented supported by photographic evidence.

The effects of hot gas ingestion on the engine resulting from rocket and gun firing are determined from on-board engine JPT, RPM and intake temperature measurements. As in the case of releases the firings are made in increasingly severe conditions from a low risk starting point and the engine conditions are monitored at each step.

Gun firing effects on the engine are determined from on-board engine instrumentation as on missile launching trials and the firing is normally filmed from a chase aircraft if the gun discharges links and cartridge cases to observe the clearance from the aircraft structure.

Most store release trials are made with ballistic test vehicles or BTV's which are inert stores ballasted to accurately represent live stores in terms of weight and CG and inertia distribution.

SYSTEMS TESTING

The major systems incorporated in a new aircraft are normally tested individually on ground rigs built for the purpose of demonstrating the designed performance and reliability.

The behaviour of the systems in their entirety and any intentional or unintentional interaction between them is then determined on flight test.

Major systems that most combat aircraft embody are listed below :

1) Electrical generation and distribution system
2) Hydraulic system
3) Pneumatic and cabin conditioning systems
4) Fuel systems
5) Avionic systems
    i) Navigation
    ii) Weapon aiming systems
    iii) Communications
    iv) Acquisition systems (Radar etc)
    v) Stability systems (i.e. flying controls)
6) Armament systems

The effects of climate on the behaviour of these systems (particularly 1-4) are determined during tropical and cold weather trials. Detailed descriptions of testing some of the above systems is given in the following lectures.

PRODUCTION FLIGHT TEST SCHEDULE

UK contractors are required to flight test each new aircraft from the production line to ensure that it is fully serviceable, that it's performance is satisfactory and that it's performance and handling are consistent with other aircraft of the same type.

A standard flight test schedule is drawn up for the aircraft type by the contractor which is then agreed by the MOD (PE) in the case of RAF and RN aircraft. In the case of export aircraft a production flight test schedule may not be required by the customer, but the contractor will probably produce one as a standard to ensure consistency of performance of the production aircraft.

An example of the contents of a production flight test schedule for a single engine class IV trainer aircraft is presented below :

<u>Pre Start Up</u>

External	Check of external finish, fit of panels, footsteps etc.
Cockpit	Check optical quality of the transparencies, canopy locking, mirrors, labels and switches etc. Operation of LP fuel cock and re-light button.

<u>Start Up</u>

Engine Start	Check engine start up time to light, max JPT during start up and idle RPM and JPT
Hydraulics	Check hydraulic system and brake system pressures
Controls	Check tailplane range. Check tailplane, rudder, aileron and tailplane standby trim ranges and times for full travel
Flaps	Check operation of flaps and cockpit indications and time up and down
Airbrake	Check operation of airbrake and cockpit indicators
Communications	Check radio volumes, transmit switches, mute switches etc.
Navigation Equipment	Check compass, ground check navaids i.e. TACAN, VOR, ADF & DME
Altimeters	Check main and standby altimeter zeros and operate built-in altitude test facility
Taxy	Check steering and braking operation and behaviour

<u>Flight</u>

Take-off	Check engine accel time, max RPM and JPT. Stabilized max RPM and JPT
Undercarriage	Measure time to retract and extend undercarriage
Trims	Check longitudinal, lateral and directional trims at fast cruising speed low level
Climb	Measure climb performance to near max operational altitude (Record Altitude, time, RPM, TGT, cabin altitude and fuel state every 5000 ft on the climb) Climb at IAS/IMN schedule
Cabin Conditioning	On the climb check operation of cabin conditioning and airflow distribution system
Navigation	On the climb check navaid ranges and bearings and standby compass heading
High Mach No	Dive to max Mach No and note transonic trim changes and pull-out acceleration
High Altitude Cabin Pressurisation	Check cabin pressurisation at standard high altitude and engine RPM. Momentarily switch "off" cabin conditioning and check altitude warning.
Engine Handling at High Altitude	Check acceleration from idle to near max RPM
Engine Relight System	Check airborne ignition of engine starter
Medium Altitude Accel/decel Trim Check	Check maximum RPM achieved on a level acceleration at full throttle. Check satisfactory lateral and directional trim change with speed.

Stalling	Check stalling speeds and buffet warnings and behaviour on clean aircraft and full flap undercarriage down
Ram Air Turbine	Check deployment of the RAT and the minimum airspeed at which RAT functions
Spin Recovery	Check aircraft enters and recovers from spins in each direction normally
Inverted Flight	Check that engine oil system and fuel pressure are satisfactory in inverted flight
Max IAS/Max g Handling	Check anti-g system at max g at max IAS Check max level airspeed at low level and corresponding RPM and JPT
Airbrake	Check airbrake extension at high speed low level.
ILS	Check ILS for correct function of indications
Landing	Check anti-skid and braking operation
Fuel Low Level	Check low level fuel warnings within limits

Aircraft fitted with complex flying control, engine control and weapon systems will require further detailed checks to prove satisfactory operation, but the test schedule should include most of the elements of the preceding example.

On receipt of a new aircraft from the production line the customer may fit additional equipment or make some modifications and may then repeat a similar flight test schedule.

UK contractors do not normally submit aircraft at intervals on the production line to assessment bodies for quality control checks.

MISCELLANEOUS TESTING

Contractors must execute most of the tests described up to this point, however individual aircraft may have additional systems or capabilities that enable them to perform their primary roles more efficiently.

These features must also be examined by the contractor and/or the regulating authority (Aeroplane and Armament Experimental Establishment.)

Some examples of such features are :-

a) Air to air refuelling systems

b) Operation from grass and unprepared sites

c) Operations using revolutionary techniques and equipment such as the BAe Harrier ski-jump

d) Operation from aircraft carriers

EVALUATION CONTENT

It is not necessary for a customer evaluating a UK aircraft to perform all the assessments noted herein, but it is useful to know what testing is performed by the contractor and acceptance authority in order to understand the flight test information that is readily available.

Some of the acceptance testing can only be satisfactorily carried out on a comprehensively instrumented aircraft which may not be available for evaluation. The lectures that follow will, however, suggest test techniques that enable useful data to be gathered from uninstrumented aircraft using basic tools such as stopwatch, kneepad and pilot display camera records. (A cockpit voice recorder is also a useful tool for gathering test data and recording qualitative assessments during flight).

The content of the evaluation sortie should come fully planned in advance with due regard for the task the aircraft is required to perform and after consideration of the contractors brochure data and published flight test data.

The evaluation should ideally be carried out by a pilot with test pilot training, supported by flight test engineers with experience of handling qualities, stability and control, performance data measurement and avionic systems appraisal.

If the aircraft can achieve and sustain high g forces the evaluation pilot should be physically fit enough to be unaffected by this level of performance.

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A REVIEW OF FLIGHT TEST INSTRUMENTATION SYSTEMS FOR ACCEPTANCE TESTING OF THE PRODUCTION AIRCRAFT

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SUMMARY

As long as the aircraft was only a simple vehicle of a limitated complexity, the acceptance flight testings, which has to be done either by the manufacturer or the customer, was concentrated on the assessment of the pilot and his crew. Nowadays, the increasing complexity of the aircraft and its system changes completely the problems : a compromise has to be found between the largest increase of the flying hours necessary to the acceptance or assessment of an aircraft and a limitation of these flying hours by means of a flight test instrumentation installed on board.

1. INTRODUCTION

More and more, the development of a new aircraft is a long and expensive venture. Obviously, as soon as the first flight is reached, and during all the phase of the flight testing of the aeroplane, an extensive flight test instrumentation is installed aboard. And, as it is described in J.F.RENAUDIE's presentation papers of this lecture series, the number of parameters recorded during the development of the projects has tremendously increased during these last 25 years. To summarize this trend, one can say that the number of recorded parameters has been multiplied by 3 or 4 hundred during this period. This, of course, once again, corresponds to the main flight testing phase of the prototype or the pre-production aircraft. The result of that, is that, independently of the large increasing complexity of the aircraft and its system, the duration of the development phase has not been proportionally increased and, sometimes, like for instance in the Airbus program (A 300), this development phase has been reduced with respect of time.

As far as the assessment or the acceptance testing of the production aircraft is concerned, the same evolution appears. Indeed, 20 or 25 years ago, for the aircraft manufacturer or for the customer, only two or three flights were sufficient without any instrumentation to control or assess the performances and handling of the vehicle and its system. Now, for military or civil aircraft with their always increasing sophistication, the same guidelines cannot be followed. Thus, the aircraft manufacturer on the one hand, for his own control flights before proposing his aeroplane and the user himself on the other hand, the need of the flight test instrumentation installed on board of the production aircraft slowly appeared.

This need in fact appeared, to begin with, on the military single-seater aircraft. Indeed, on this type of machine, it is very difficult for a single pilot to check visually with the cockpit instruments the functioning of all the systems or to appreciate the accuracy of some of them.

When no instrumentation is available, this has the consequence to increase the number of flights for the acceptance testing of the production aircraft. Very often, the problem is not so serious on a large aircraft like, for instance, airliners on which a full crew can note the information of the various instruments.

2. FLIGHT TEST INSTRUMENTATION DEFINITION

In the case of the pilot assessment or the acceptance testing of the production aircraft, the definition of the flight test instrumentation cannot be made, generally speaking, through the same approach than the one which is used to define the main flight testing instrumentation on board the prototype. The definition is the result of a sort of a compromise.

The ideal would be to begin with a list of parameters. This parameters'list should be built without considering the difficulty corresponding to each one of them.

So, let's suppose that the specialists in charge of the acceptance testing of the production aircraft have reached among them an agreement on this parameters'list.

One must note, at this stage, that this parameters'list has to be prepared by the aircraft manufacturer for his customer. Of course, on the customer's side, one can always ask for addition or modifications. At last, about this parameters'list, one must never forget, during its establishment, that the proposed aircraft is a production one and has already been flight tested, as far as the type is concerned ; therefore, the purpose of the instrumentation is only to check that the production aircraft is in accordance with the agreed specifications.

Let's then suppose that the parameters'list has been agreed. The next stage is to consider how each one of these parameters can be acquired and recorded.

Two cases can be considered :

- the aircraft is fitted with a crash recording system,

- no special recording means are available on the aircraft.

The case of the aeroplane or helicopter fitted with a crash recording system is of course the easiest one. In that opportunity, one just has to compare the parameters'list which has been selected for the acceptance testing to the parameters'list and the capacity of the crash recording system. Very often, most of the parameters included in the new parameters'list are among the recorded parameters of the crash recording system. Just a few complementary parameters will have to be added. For this, the problem of the transducers (sensors) and their wiring has to be set up. For the extra transducers which have to be installed, it must be decided if they have to be removed or not after the flight test. As a general rule, the users are very reluctant to admit that on their aircraft, some extra weights which are not necessary for the normal flights or normal use of the aircraft will remain on board of it. But, through the experience we have in France, the main problem on production aircrafts is the installation of transducers and their wiring to the data acquisition.

At the other hand, there is the case of the helicopter or aeroplane without any recording means on board. More and more, a removable package is prepared and installed aboard for the few acceptance flights. Ne cesseraly, in this case, some extra wirings have also to be installed keeping in mind that most of them would be removed after these assessment flights.

You will see, in the following examples, that this simple division in these two cases doesn't correspond to the exact reality. Sometimes, due to the complexity of the aircraft which has already been equiped with the crash recording system, one has to complete it or to add a special system. You will see that this was the case of the SST Concord and the Airbus A 300.

3. GENERAL DESCRIPTION OF PRESENT ACCIDENT RECORDING SYSTEMS

About since 1975, there has been among the main airlines an agreement on the definition of Aircraft Integrated Data System (AIDS) which is connected to the crash recorder required by nearly all the official national organisations as :

- FAA - NTSB in U.S.A.,
- CAA in Great Britain,
- DGAC in France.

The characteristics of this system must be considered because they represent the actual possibilities of data acquisition equipments which are currently available.

It must be precised, that the AIDS specified either in accordance with ARINC 573 or with ARINC 717 are used both for civil and military applications as far as the format and recording principles are concerned.

It is thus worthy to analyse their main features :

- Digital form of recording to be able to protect the datas against fire, shocks etc...

- As a consequence, a sampling rate for each parameter is imposed.

From these two basic points, the international authorities agreed on detailed specifications. We would like to summarise them :

- 30 to 35 parameters,
- total sampling rate 64 words of 12 bits per second (768 bits/sec.),
- recording format : 4 subframes of 64 words (subframe duration : 1 second),
- autonomy of the recorder : 25 hours.

Besides, survivability characteristics are imposed to the recorder in accordance with the FAA : TSO C 51 standard.

Obviously, such a system can be used as the basic element of an acceptance flight testing instrumentation.

4. EXAMPLES

4.1. Military aircraft

About military aircraft, we would like to take the examples of modern single-seaters and air-carrier.

4.1.1. Mirage F1

In the case of the Mirage F1, there is aboard a full crash recording system. This system, developped by two french firms, consists of :

- Data Acquisition P/N ED 3330 - SFIM
- Crash Protected Recorder P/N PE 6010 - Schlumberger.

These two equipments have been adopted for military aircraft (volume, power supply...) But, in principle, they are in accordance with the international regulations in this field.
The main modification is the possibility to multiply by 6, the sampling rate of all the parameters at the Data Acquisition level, and to increase the magnetic tape transport speed in the same ratio, in order to keep a similar packing density on the tape. The only drawback of this modification is to reduce to a few hours, the recorder autonomy which is acceptable for acceptance flight testing.

4.1.2. Mirage 2000

On this last version of the DASSAULT fighters, the same philosophy is followed.

Once again, the Data Acquisition (P/N ED 3333 developped by SFIM) is connected to a crash recorder (P/N PE 6015 developped by Schlumberger).

There is also an increase of the sampling rate :

- for the normal flights, nominal sampling rate multiplied by 3,
- for the acceptance flights, the sampling is multiplied by 6.

This choice permits to use the same ground equipments in the Airlines, and in the Airforce.

4.1.3. Transall

This Germano-French Aircraft is, for the moment, equiped with a SFIM trace recorder P/N A 26.

The A 26 has a function of crash recorder, but for the acceptance flights, the photographic tapes are processed, and a dozen of parameters controlled.

4.2. Civil Aircraft

4.2.1. Falcon 10 - 20 - 50

This type of aircraft is either equiped with photographic recorders or with Digital Systems depending on the type of flights they will have to do (private or public).

In both cases, with no modifications, the recording is processed after the acceptance flights.

4.2.2. SST Concord

For this Aircraft, which of course, was the first Supersonic Transport to fly, a rather long list of parameters was published.

The normal crash recording system, installed according to the Anglo-French regulation, was not always sufficient to accept the parameters selected for the first production aircrafts.

Therefore, a special Data Acquisition unit was designed by SFIM, matched with a cassette-recorder.

But, this system, with an increased capacity, was designed, keeping the same recording system of the Aircraft. Thus, the same ground facilities were used for the data processing.

4.2.3. Airbus A 300

A removable package was designed with the same principle than the one used for CONCORD : the SFIM SERAC system. This package is installed in the fuselage during the acceptance flights, or for complementary flight testing.

But, here again, as the same recording format is kept, no special ground means are used.

EXAMEN DES INSTALLATIONS D'ESSAIS UTILISEES POUR LES VOLS D'EVALUATION ET DE RECEPTION D'AVIONS DE SERIE

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SOMMAIRE

Tant que les aéronefs étaient de simples véhicules de complexité limitée, les vols de réception qui devaient être faits, aussi bien par l'avionneur que par l'utilisateur, étaient basés sur l'évaluation du pilote et de son équipage. Maintenant, la complexité accrue, aussi bien des avions que de leur système, pose le problème de façon totalement différente : un compromis doit être trouvé entre une importante augmentation du nombre d'heures de vol nécessaires à l'évaluation ou à la réception d'un avion et une limitation de ce même nombre d'heures de vol au moyen d'une installation d'essais installée à bord.

1. INTRODUCTION

De plus en plus, le développement d'un nouvel avion est une longue et coûteuse aventure. Evidemment, depuis le premier vol et durant toute la durée des essais en vol de l'avion, une installation d'essais très développée est installée à son bord. Et, ainsi qu'il est exposé dans le document de présentation de cette série d'exposés rédigés par J.F.RENAUDIE, le nombre de paramètres enregistrés durant la phase de développement des prototypes a augmenté de façon surprenante durant ces 25 dernières années. Pour résumer cette tendance, on peut dire que le nombre de paramètres enregistrés a été multiplié par trois ou quatre cents durant cette période. Ceci, bien sûr, une fois encore, correspond à la phase principale des essais en vol du prototype ou de l'avion du pré-série. Le résultat de cela est que, indépendamment de l'énorme accroissement de complexité de l'avion et de ses systèmes, la durée de la phase de développement n'a pas été accrue dans les mêmes proportions et de surcroît, parfois, comme par exemple dans le programme Airbus, cette phase de développement a pu être réduite.

En ce qui concerne l'évaluation ou la réception des avions de série, une évolution identique s'est produite. En fait, il y a 20 ou 25 ans, aussi bien pour le constructeur que pour l'utilisateur, deux ou trois vols étaient suffisants pour effectuer l'évaluation ou le contrôle des performances et qualités de vol de l'aéronef et de ses systèmes, vols qui ont été effectués sans aucune installation d'essais. Maintenant, que cela soit pour des avions militaires aussi bien que pour des avions civils dont la sophistication ne cesse d'accroître, la même philosophie ne peut pas être poursuivie. Ainsi, l'avionneur d'un côté, pour ses propres vols de contrôle effectués avant de présenter l'avion à son client, et pour l'utilisateur lui-même, de l'autre côté, le besoin d'une installation d'essais embarquée sur avions de série est apparu peu à peu.

Ce besoin, en fait, se fait sentir au départ sur les mono-places militaires. En fait, sur ce type de machine, il est très difficile pour un pilote seul à bord de vérifier, uniquement avec ses instruments de vol, le fonctionnement de tous les systèmes et, en particulier, d'apprécier la précision de certains d'entre eux.

On comprend bien pourquoi, lorsqu'aucune installation d'essais n'est disponible, l'inévitable conséquence est d'augmenter le nombre de vols nécessaires à la réception des avions de série. Notons cependant que très souvent, le problème ne se pose pas de façon si aiguë sur des avions de fort tonnage, par exemple les avions de transport civils sur lesquels un équipage complet peut noter les informations de nombreux instruments de vol.

2. DEFINITION DE L'INSTALLATION D'ESSAIS

Dans le cas d'une simple évaluation ou de la réception d'un avion de série, la définition de l'installation d'essais en vol peut être faite, d'une façon générale, de la même manière que celle utilisée pour définir l'installation d'essais principale montée à bord des prototypes. Cette définition est le résultat d'une certaine forme de compromis.

L'idéal est de commencer par établir une liste de paramètres. Cette liste de paramètres doit être rédigée dans l'absolu sans se soucier provisoirement des difficultés correspondant à l'acquisition de chacun d'entre eux.

Supposons ainsi que les spécialistes responsables des vols de réception des avions de série sont arrivés entre eux à un accord sur cette liste de paramètres.

Notons, à ce stade, que cette liste de paramètres doit être préparée par l'avionneur pour son utilisateur. Bien entendu, cet utilisateur, de son côté, peut toujours demander des compléments ou des modifications. Enfin, à propos de cette liste de paramètres, il ne faut pas perdre de vue, au cours de sa

rédaction, que l'avion considéré est un avion de série et qu'il a déjà été largement essayé en vol en ce qui concerne son type ; c'est pourquoi, le but de cette installation d'essais n'est que de vérifier que l'avion de série est conforme aux spécifications originales.

Supposons donc qu'un accord a été trouvé relatif à cette liste de paramètres.

Deux cas peuvent être considérés :

- l'avion est équipé d'un système d'enregistrement,
- aucun moyen d'enregistrement particulier n'est disponible à bord.

Le cas de l'avion ou de l'hélicoptère équipé d'un système d'enregistrement est naturellement le plus facile. Dans ce cas, il suffit de comparer la liste de paramètres retenue pour les vols de réception à la liste de paramètres et la capacité du système d'enregistrement d'accident. Le plus souvent, la plupart des paramètres qui font partie de la nouvelle liste se trouvent dans le groupe de ceux qui sont enregistrés sur l'enregistreur d'accident. Seuls, donc, quelques paramètres complémentaires seront nécessaires. Pour cela, le problème des capteurs et de leur câblage doit être traité. Pour les capteurs complémentaires qui devront être installés, il devra être décidé si ils seront démontés ou pas après les vols d'essais. Notons, que d'une façon générale, les utilisateurs hésitent beaucoup à admettre que sur leur avion, quelques poids morts non utilisés pour les vols normaux ou l'emploi normal de l'avion seront conservés à son bord.

Mais, au travers de l'expérience que nous avons en France, le problème principal sur les avions de série réside dans l'installation de capteurs et de leur câblage vers l'acquisition de données.

D'un autre côté, il y a le cas de l'hélicoptère ou de l'avion dont l'équipement normal ne comporte pas de moyens d'enregistrement. De plus en plus, un ensemble mobile est préparé et monté à bord pour les quelques vols utiles à la réception. Nécessairement, dans ce cas, quelques câblages complémentaires devront être utilisés sans perdre de vue toutefois que la plupart d'entre eux devront être retirés après ces vols d'évaluation.

Ainsi que l'on pourra le voir dans les exemples qui suivent, cette simple division en deux cas ne correspond pas rigoureusement à la réalité. Ainsi, de par la complexité de l'avion qui est déjà équipé cependant d'un système d'enregistrement d'accident, celui-ci doit être complété ou un système supplémentaire ajouté. Vous verrez que ceci a été le cas du TSS Concorde et de l'Airbus A 300.

3. DESCRIPTION GENERALE DES SYSTEMES D'ENREGISTREMENT ACTUELS

Depuis 1975, les principales compagnies mondiales se sont accordées sur la définition d'un système d'acquisition des données intégrées (AIDS) associé à l'enregistreur d'accident exigé par la plupart des organisations officielles nationales comme par exemple :

- FAA - NTSB aux U.S.A,
- CAA en Grande Bretagne,
- DGAC in France.

Les caractéristiques de ce système sont à considérer puisqu'elles représentent les possibilités actuelles des équipements d'acquisition des données qui sont disponibles facilement.

Précisons de plus que les AIDS spécifiés soit selon l'ARINC 573 ou selon l'ARINC 717 sont aussi bien utilisés sur des avions civils que sur des avions militaires, au moins en ce qui concerne les principes et le format d'enregistrement.

Il vaut donc la peine d'en analyser les principales caractéristiques :

- Enregistrement sous forme digitale afin dêtre en mesure de protéger les données contre le feu, les chocs...
- En conséquence, une cadence d'échantillonnage est imposée pour chaque paramètre.

De ces deux points de base, les autorités internationales se sont mises d'accord sur des spécifications détaillées. Nous souhaiterions les résumer :

- 30 à 35 paramètres,
- échantillonnage total 64 mots de 12 bits par seconde (768 bits/sec.),
- format d'enregistrement : 4 cycles courts de 64 mots (durée du cycle court : 1 seconde),
- autonomie de l'enregistreur : 25 heures.

En dehors de cela, les caractéristiques de survivabilité sont imposées à l'enregistreur selon les normes de l'AFA : TSO C 51.

De toute évidence, un système peut être employé comme élément de base pour une installation d'essais destinée aux vols de réception.

4. EXEMPLES

4.1. Avions militaires

En ce qui concerne les avions militaires, nous aimerions prendre comme exemples les intercepteurs mono-places militaires et les avions de transport.

4.1.1. Avion Mirage F1

Dans le cas du Mirage F1, il y a à bord de l'avion un système complet d'enregistrement d'accident (système de sécurité). Ce système, développé par 2 firmes françaises, se compose de :

- une acquisition de données type ED 3330 fabriquée par la SFIM,
- un enregistreur d'accident protégé type PE 6010 fabriqué par Schlumberger.

Ces deux équipements ont été adaptés à un avion militaire (volume, alimentation...) Mais, de façon générale, ils sont conformes aux réglements internationaux dans ce domaine.

La principale modification est la possibilité de multiplier par 6 la cadence d'échantillonnage de tous les paramètres au niveau de l'acquisition de données et d'accroître également la vitesse de la bande magnétique dans le même ratio afin de garder sur celle-ci une densité d'enregistrement similaire. Le seul inconvénient de cette modification est de réduire à quelques heures l'autonomie de l'enregistreur, ce qui est acceptable dans le cas des vols de réception.

4.1.2. Mirage 2000

Sur cette dernière version des intercepteurs DASSAULT, une philosophie identique a été retenue.

Une fois encore, l'acquisition de données ED 3333 développée par SFIM est reliée à l'enregistreur d'accident PE 6015 développé par Schlumberger.

Il y a aussi un accroissement de la cadence d'échantillonnage :

- pour les vols normaux, la cadence d'échantillonnage nominale est multipliée par 3,
- pour les vols de réception, la cadence nominale est multipliée par 6.

Ce choix permet d'utiliser au sol les mêmes équipements, soit dans les compagnies aériennes ou dans l'armée de l'air.

4.1.3. Transall

Cet avion franco-allemand est, pour le moment, équipé avec un enregistreur photographique SFIM type A 26.

L'enregistreur A 26 a pour les vols normaux, une fonction d'enregistreur d'accident, mais pour les vols de réception, la bande photographique est traitée et une douzaine de paramètres contrôlés.

4.2. Avions civils

4.2.1. Falcon 10 - 20 - 50

Ce type d'avions sont, ou bien équipés d'enregistreur photographique, ou de systèmes numériques suivant le type de vols qu'ils intreprennent (privé ou public).

Dans les deux cas, sans aucune modification à porter au système, les enregistrements sont traités dans le cas des vols de réception.

4.2.2. TSS Concorde

Pour cet avion qui, naturellement, était le premier avion supersonique de transport à voler, une liste plutôt longue de paramètres avait été établie.

Le système normal d'enregistrement installé conformément aux règlements anglo-français ne permettait pas d'acquérir un nombre suffisant de paramètres ainsi qu'ils avaient été sélectionnés sur les premiers avions de série. C'est pourquoi une acquisition de données spéciale fut développée par la SFIM couplée à un enregistreur à cassette de sa fabrication.

Mais, ce système, avec une capacité accrue , fut étudié en conservant le même format d'enregistrement que celui de l'enregistreur d'accident de l'avion. Ainsi, les mêmes moyens sol étaient utilisés pour le traitement des données.

4.2.3. Airbus A 300

Un ensemble mobile fut étudié avec les mêmes principes généraux que ceux utilisés pour CONCORDE : le système SFIM SERAC.

Cet ensemble est installé dans le fuselage pendant les vols de réception ou dans le cas d'essais en vol complémentaires.

Ici aussi, le même format d'enregistrement est conservé afin de ne pas avoir à élaborer de système particulier de traitement.

PERFORMANCE TESTING PRODUCTION AIRPLANES

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SUMMARY

Efficient flight testing and data analysis techniques can be used to determine test-day airplane performance levels. Basic performance, instrumentation, and data analysis concepts are discussed in this paper as well as ground and flight tests. These concepts and tests can quickly, yet adequately, demonstrate Flight Manual performance using production planes with little or no special flight test equipment.

Specifically, the paper includes a discussion of basic performance testing and data analysis concepts along with pretest information requirements. Ground tests such as instrumentation calibrations, installed static thrust calibrations and airplane gross weight and center of gravity checks are outlined. Flight tests which enable static source position error determination, as well as an assessment of takeoff, cruise, acceleration and turning performance are discussed. Also described are tests that enable assessments of the engine handling characteristics of the airplane. The presentation and analysis of the data acquired during each type of tests are also outlined with emphasis on using test-day data as opposed to test-day data corrected to standard-day conditions.

SYMBOLS AND ABBREVIATIONS

A/B	Afterburner
AFFTC	Air Force Flight Test Center
AOA	Angle of attack
C_D	Drag coefficient
cg	Center of gravity
C_L	Lift coefficient
deg	Degrees
deg C	Degrees Celsius
deg/sec	Degrees per second
DT&E	Development Test and Evaluation
EPR	Engine pressure ratio
F_{ex}	Excess thrust
F_g	Gross thrust
FOD	Foreign-object damage
ft	Feet
g	Acceleration of gravity
H_c	Calibrated pressure altitude
H_E	Energy-height
H_i	Indicated pressure altitude
HUD	Head-up display
ΔH_{pc}	Static pressure position error
Δh_t	Height difference between ground reference and test airplane
in. Hg	Inches of Mercury
K	Temperature probe recovery factor
kt	Knots
lb/hr	Pounds of fuel per hour

MAC	Mean aerodynamic chord
M_c	Calibrated Mach number
M_i	Indicated Mach number
ΔM_{pc}	Mach number position error correction
NAMPP	Nautical air miles per pound of fuel
NLF	Normal Load Factor
pct	Percent
PLA	Power Lever Angle
P_s	Specific Excess Power
$\Delta P/q_{cic}$	Static source position error coefficient
R/C	Rate of climb
RF	Range Factor
RPM or rpm	Engine speed in revolutions per minute
RTO	Rejected takeoff
sec	Seconds
T_a	Ambient temperature
TIT	Turbine inlet temperature
TSPI	Time-space positioning instrumentation
V_c	Calibrated airspeed
V_i	Indicated airspeed
ΔV_{pc}	Airspeed position error correction
W_f	Engine fuel flow
W_t	Airplane gross weight
W_t/δ_a	Weight-pressure ratio
δ_a	Ambient pressure ratio (P_a/P_{asl})
θ_a	Ambient temperature ratio (T_a/T_{asl})
ρ	Atmospheric density
σ_a	Ambient density ratio (ρ_a/ρ_{asl})
γ	Climb angle

FOREWORD

This paper contributes to the portion of this lecture series devoted to helping the countries involved assess and choose new aircraft. It discusses performance evaluation of production airplanes equipped with little or no specially-installed flight test instrumentation.

INTRODUCTION

This paper addresses basic flight testing concepts and techniques that can be used effectively to assess the performance of military jet-propelled airplanes. The emphasis will be on the practical application of theory and concepts so that the participating countries can be more able to understand and solve the many problems involved with performance flight testing of modestly instrumented or uninstrumented airplanes. Specific test techniques will also be discussed, but only to the extent that they clarify concepts.

The basic assumption of this paper is that the task at hand is to evaluate the performance of a production airplane that has little or no special flight test instrumentation. Specifically,

1. **The airplane is a production model.**

2. Optimum operating techniques (i.e., best speeds for climb, cruise, etc.) have previously been determined.

3. The evaluators possess, or desire to use, minimum resources for data processing. All calculations are made on programmable desk-top calculators.

4. Development Test and Evaluation (DT&E) performance reports are available from the manufacturer and country of origin.

5. A developed Flight Manual is available that is based upon flight test results.

This last assumption is very important. The performance flight tests discussed herein are intended to provide limited test-day data with which to demonstrate and validate the airplane's Flight Manual. If the Flight Manual proves valid, it can be used to assess the airplane's total performance capabilities against those of other airplanes as described by their respective Flight Manuals. The purpose of these tests is <u>not</u> to build or even fully check the airplane's Flight Manual, only to provide sufficient information to assess its validity. The expensive flight testing required to develop a Flight Manual is assumed to be the responsibility of the manufacturer and country of origin. The customer should contractually demand an accurate Flight Manual as part of the airplane purchase.

Seven general topics will be discussed. They are;

1. Some basic performance concepts
2. Pretest data requirements
3. Minimum parameter requirements
4. Instrumentation systems
5. Data analysis considerations
6. Ground tests, and finally ...
7. Flight tests and evaluation

For each topic, general concepts will be discussed, test objectives will be specified, key parameters will be identified, some test techniques will be recommended, and data analysis considerations will be discussed.

BASIC PERFORMANCE CONCEPTS

Performance test and evaluation is a <u>quantitative</u> business. There is no such thing as <u>qualitative</u> performance. There are a large number of uncontrollable variables such as ambient temperatures and pressures, windspeeds and directions, etc., that must be measured and then considered when any performance data are analyzed. Not all are "corrected for" by mathematical manipulations to the performance data, but all are considered so that their impact on these data can be assessed. Therefore, the performance of an airplane can never be fully defined until the quantitative data are: acquired, reduced, edited and thoroughly analyzed.

Airplane performance is defined as information that defines the actual performance of the airframe/engine combination and is presented in terms of rate-of-climb (R/C), nautical air miles per pound of fuel (NAMPP), etc. Airframe data used to calculate airplane performance, such as drag polars or lift curves, will be referred to as aerodynamic data. Engine operating characteristics such as gross thrust, airflow, etc. will be referred to as engine performance data.

When flight testing to demonstrate an airplane's performance against the Flight Manual, the data need <u>not</u> be corrected to standard conditions since most current Flight Manuals present airplane performance information at off-standard atmospheric and aircraft gross weight conditions. Also, when comparing two different airplanes i.e., a direct side-by-side comparison, correction need not be made to standard atmospheric conditions since the airplanes are operating in the same air mass, although gross weight corrections might still be required. Actually, current thinking by performance flight test engineers at the United States Air Force Flight Test Center (AFFTC) is that mathematical manipulations and corrections to test-day data should be kept as minimal as possible because many so-called "standardization" corrections are based on assumptions which are not always valid. This is perhaps what is so attractive about airplane performance demonstrations against a Flight Manual or direct side-by-side comparisons: little complex mathematical manipulations to the test results are required.

It is appropriate at this point to define three more terms as they will be used in this paper. They are performance evaluations, demonstrations, and comparisons. A thorough <u>evaluation</u> of the performance of an airplane involves the analysis of each performance <u>element</u> (i.e., lift, drag, thrust) so that the total capability of the airplane

can be determined and optimum operating procedures can be developed. Evaluations, as defined in this paper, require long flight test programs, sophisticated instrumentation and special aerodynamic analysis-oriented maneuvers.

A Flight Manual performance demonstration, as defined here, involves only the determination of airplane performance (R/C, NAMPP, etc) under known, measured conditions. No attempt is made to thoroughly analyze the aerodynamic or engine characteristics that combine to generate this performance. Once the performance data has been acquired, it is used to determine if the test airplane's performance matches some reference document such as the Flight Manual's performance supplement or the airplane performance reported in its DT&E reports.

A comparison, by contrast, is defined as a direct, side-by-side flyoff between two or more airplanes. Since the airplanes are operating at the same time through the same air mass, a direct comparison of specific performance can be made, however sufficient additional data must still be acquired to define the atmospheric conditions and the configuration of the airplanes.

Both demonstrations and side-by-side comparisons can be accomplished with little or no special flight test instrumentation and a limited data acquisition system. The production cockpit instrumentation can be the primary data source. These tests can also be successfully accomplished with limited data processing and analysis resources. Furthermore, they require no specially developed maneuvers designed to acquire aerodynamic data. The test maneuvers used are quite operations-oriented and include level or constant-altitude accelerations, stabilized turns and stabilized cruise points. The successful completion of demonstrations and comparisons does require, however, that the test airplane be at an advanced stage in its development cycle and performance limitations, optimum operating techniques and documentation be known and available.

PERFORMANCE DATA

The objective of this section is to discuss pretest performance information and aerodynamic data requirements, parameters to acquire during testing, instrumentation system requirements, data recording requirements, and data analysis concepts that will facilitate a successful performance flight test demonstration or comparison.

A. PRETEST DATA BASE

Before any flight testing is attempted, the following documentation should be acquired from the manufacturer and/or the country of origin.

1. Development Test and Evaluation performance reports.
2. The Flight Manual complete with performance supplement.
3. The Flight Manual Performance Substantiation Document

This last document is very important and should be acquired, if available. It documents what aerodynamic and propulsion data were used, how they were used, and what assumptions were made on developing the airplane performance (R/C, NAMPP, etc) specified in the Flight Manual. Some of the important information contained in these documents is:

1. Lift curves
2. Drag polars
3. Thrust and fuel flow relationships
4. Inlet pressure recovery characteristics
5. Drag associated with configuration changes
6. External store drag
7. Weight and center of gravity relationships
8. Airplane physical dimensions such as wing area, mean aerodynamic chord location, etc.

Remember, as with the Flight Manual, performance flight testing of the production airplane will also help to validate many of these data.

B. DATA REQUIREMENTS

The following parameters can be considered the minimum for a successful performance demonstration or comparison.

1. Time
2. Transmittable event tone
3. Event light
4. Total air temperature (T_T)
5. Airspeed (V_i)
6. Pressure altitude (H_i): Set altimeter at 29.92 in. Hg
7. Normal acceleration (n_z)
8. Total fuel flow (W_f)
9. Power lever angle or position (PLA)
10. Engine speed (rpm)
11. Some engine parameter that can serve as a surrogate for power and/or thrust level, such as Turbine Inlet Temperature (TIT), Engine Pressure Ratio (EPR)
12. Fuel quantity remaining
13. Airplane configuration
14. Airplane loading

The above parameters are essential for any performance analysis. The following parameters should be obtained if available; however, they can be calculated if they are not.

1. Mach number (M_i)
2. Angle of attack (AOA)
3. Pitch attitude (θ)
4. Bank angle (ϕ)

C. INSTRUMENTATION SYSTEMS

The above performance parameters must be acquired and recorded. Instrumentation systems can be either remote to the test airplane or onboard the aircraft. If takeoff tests are to be accomplished, some form of remote time-space positioning instrumentation (TSPI) is very useful. These include cinetheodolite cameras, single-station solution cameras, radar or laser trackers. These systems are not absolutely necessary, but experience has proven that higher quality data results with their use as opposed to only onboard TSPI equipment. A remote source of weather data (i.e., ambient temperature, pressure, windspeed and direction) is also desirable for test control and data analysis.

For the purposes of Flight Manual performance demonstrations and/or side-by-side comparisons, the production cockpit panel instruments and the head-up display (HUD) system should suffice as the onboard instrumentation system. A quick scan of the required parameters previously listed indicates that most are available in production airplanes either on the panel or on the HUD. Head-up displays, of course, are relatively new, but will likely be found on all future military airplanes, especially in fighter and attack types. They are very useful for flight testing because they present essential information on easily readable and recordable displays.

Of course, all these data presented by the instrumentation systems must be recorded for later analysis. If remote TSPI and atmospheric conditions reporting systems are used, their output can be hand recorded; however, most current systems have some form of analog or digital permanent output.

The onboard cockpit instruments or HUD may be hand recorded, voice recorded, recorded on film or video tape, or most desirably, a combination of these. The simplest technique is to hand record the applicable parameters. If stabilized maneuvers are properly flown, this can be a very effective data recording technique. The more stable the test conditions, the better the results.

Recently, AFFTC engineers have obtained increasingly satisfactory results using a pilot's voice recorder to supplement hand recorded data. These systems are extremely simple and inexpensive and consist of nothing more than a small cassette tape recorder interfaced with the communications system of the airplane and are usually carried by an aircrewmember. During one test series, the small microphone of the cassette recorder was simply taped inside the pilot's oxygen mask while the recorder was carried in his flight suit pocket. During the tests, the pilot turned the recorder ON and described all the test conditions and read the cockpit instruments.

Cameras designed to record instrument panel presentations have been used for many years with mixed success. These cameras are usually hand-held when testing larger airplanes or helicopters and mounted to the airframe when evaluating small or single-seat airplanes. Adverse sun angles appear to be the biggest problem. It always seems as though the airplane has turned into the sun just at the moment data are required. However, if the tests are planned carefully, these cameras can be used with good results. Again, the more stable the test maneuvers, the easier it is to plan the sun angles so that the cameras are not affected.

Most current fighter or attack airplanes, and probably all future ones will have a HUD and a HUD recording system that uses either film or video tape. The HUD recording system is very useful during pilot training for target scoring and as permanent documentation of training maneuvers. These same HUD recording systems can be used as an outstanding source of flight test data. Since these are designed to be visible to the pilot under all light conditions, sun angles are not as critical as with cockpit cameras. Also, a recorder is often standard equipment. If the airplane has a HUD, but no recorder, small, lightweight and reasonably priced video recorders are currently available that are easily interfaced with the production HUD. These video recorders are also available with optional time code generators whose output is displayed on the production HUD. FORMIDABLE!

D. DATA ANALYSIS CONSIDERATIONS

If valid conclusions are to be made from any flight test effort, all airplane performance data must be edited before any analysis is attempted. These data must be checked for "wild points" or inconsistencies utilizing knowledge of airplane performance limits such as maximum speeds at altitude, airplane acceleration limits and pilot-reported atmospheric and airplane conditions. While editing and analyzing a relatively small amount of data, time histories of all performance parameters (i.e., the parameter of interest plotted against time) must be plotted and faired. When the airplane performance values (such as R/C or NAMPP) are being calculated, the smoothly <u>faired values</u> of these parameters at the points or times of interest and <u>not</u> the actual <u>data point values</u> should be used. By smoothly fairing the time histories of the pertinent parameters and using the faired values for the calculation of airplane performance, the effects of random errors, as indicated by scatter, in the data can be minimized. Although time consuming, this time history fairing technique is probably still the best way to produce high quality performance data. Even the most sophisticated computerized data editing and curve fit routine cannot compare with the experienced flight test data analyst armed with a sharp pencil and flexible curve when it comes to determining good data. Typical time history plots are illustrated in figure 1.

Indicated pressure altitude must be recorded for any airplane performance analysis. Indicated pressure altitude (H_i) is obtained by setting the altimeter "window" at 29.92 inches of mercury. At this setting, the altimeter will read pressure altitude whether in a standard or nonstandard atmosphere.

GROUND TESTS

Carefully planned ground tests are an essential part of every flight test program, regardless of scope. The objective of this section is to discuss three of the most important: instrument calibrations, weight and center of gravity determination, and installed static thrust engine calibrations.

A. INSTRUMENT CALIBRATIONS

No valid performance characteristics can be determined without calibrated instruments. The objective of this section is not to describe how to calibrate aircraft instruments, but to discuss some of the types of instrument error and how these errors impact flight test data. The three principal instrumentation errors that must be accounted for during flight testing are: bias, lag and hysteresis. Simply stated, bias presents itself as an incorrect instrumentation reading under static parameter conditions. Lag is indicated by an instrumentation error under dynamic or varying parametric conditions. Hysteresis is present when the instrumentation reading is a function of its past values, i.e.; the levels presented are a function of the previously reported values. All three of these principal error sources are present, to some degree, in all flight test or production airplane instruments. They cannot be eliminated, but they can be accounted for in subsequent data analysis if the error characteristics of the instruments are known. Instrumentation calibration is the task of determining these error characteristics.

Once the instrumentation has been calibrated, it is relatively easy to account for instrumentation errors during data analysis. First, for the actual flight tests, pick the specific instruments with low, repeatable error characteristics. Indeed, this is the very purpose of the calibration: not to "fix" the instruments, but to determine their error characteristics so that the most predictable instruments can be used. During data analysis, instrument bias is corrected for by using the results of the instrument calibration. A typical instrument calibration is presented in figure 2. Note that at an

indicated airspeed of 200 knots, a 5-knot instrument error correction must be made. Because of instrument hysteresis, the correction must be obtained by averaging the upscale and down scale readings. Instrument correction is the very first step in data editing and analysis, and it is accomplished prior to plotting of time histories.

Lag constants can be determined by applying step input functions to the instruments. These techniques are documented in reference 1. Applying lag constants, however, is difficult. For the purposes of an airplane performance demonstration or comparison, the best way to handle instrument lag is to first, pick instruments with low lag values and second, plan stable test maneuvers so that the key parameters are held as constant as possible.

Hysteresis is also represented in figure 2 as the difference in the correction to be added to indicated airspeed when going upscale or down scale. Notice that instrument hysteresis results in errors as high as 10 knots at 200 knots indicated airspeed. Hysteresis cannot be accounted for by applying a simple correction as in the case of bias or by mission planning as in the case of lag. Since it is essentially impossible to track the previous history of the parameter being displayed, hysteresis presents itself as random scatter on the data. Therefore, during data analysis, hysteresis is accounted for by using an instrument calibration correction that is an average of the upscale and down scale readings and by putting a smooth fairing through the resulting time histories as discussed in the Data Analysis Section.

One preflight test procedure that fits loosely under the heading of instrumentation calibrations is checking for and eliminating pitot-static system leaks. These are very important checks and they are often overlooked. The exact method to leak check the pitot-static system of a given type airplane is specified in the Technical Orders for that aircraft so methods will not be discussed here. The point to be emphasized, however, is that pitot-static system leaks, if present, must be brought within reasonable limits or the resulting airplane performance data will be erratic and inconsistent. At the AFFTC, acceptable maximum equivalent leak rates are considered to be: 1 knot per minute at 300 knots indicated airspeed, and 50 feet per 3 minutes at 15,000 feet indicated altitude.

At the AFFTC flight tests are not attempted without first accomplishing instrument calibrations and leak checks. If we go to a remote site on a quick-response test, calibrated instruments for key parameters are taken and used to replace the airplane's production instruments. In later sections of this paper, key parameters will be identified whose instruments must be calibrated for a given performance test. Unfortunately, instrumentation calibrations require comparisons of the test instruments with some repeatable, accurate standard and this usually means a sophisticated and complex calibration laboratory. Experience has indicated that any shortcuts taken in the calibration process usually result in erroneous performance data. If the testing organization does not have access to these facilities (either within government or in the private sector) then the manufacturer of the test airplane must provide either instrument calibrations or data that have already been adjusted for instrument calibrations.

To summarize, all instruments which will display key parameters must be calibrated. The instruments to be used for the tests are selected for low, repeatable error levels. Pitot-static system leaks are brought within limits. Missions are flown to minimize the effects of instrumentation lag. The first step in data editing is to apply corrections to the indicated parameter values for bias and then the parameter time histories are plotted and smoothly faired to account for instrumentation hysteresis characteristics.

B. WEIGHT AND CENTER OF GRAVITY DETERMINATION

The performance of any airplane is a direct function of gross weight, and to a lesser degree, center of gravity (cg) position because of longitudinal trim drag. The gross weight of the airplane must be known for all test conditions and any error in the determination of gross weight will be reflected directly in the final airplane performance results. At the AFFTC, fuel tank calibrations are conducted to determine gross weight and cg as a function of internal fuel quantity and airplane pitch attitudes. For a production airplane, however, it is usually sufficient to use the manufacturer's recommended weight, cg and fuel quantity curves as represented by figure 3. The manufacturer must, however, provide instrument calibrations for all cockpit fuel quantity indicators, including the totalizer, so that instrument error can be removed from the calculation of gross weights.

The test airplane must be weighed before and after each performance flight to determine engine start and shutdown gross weight and cg location. This is required because the airplane's gross weight and cg will change from mission to mission because of differences in fuel specifics, loadings and configurations. Once the engine-start measurement for gross weight and cg are used to adjust the manufacturer's recommended curves, standard fuel burn sequences and cg variation with fuel quantity are always assumed, as illustrated in figure 3. Gross weight is determined for test conditions by subtracting fuel used from the measured engine-start gross weight. Center of gravity is obtained by using the manufacturer's curves adjusted for the engine-start cg measurement.

Postflight gross weight and cg measurements are used to validate the in-flight weight and cg computations. If an unacceptable difference exists between the postflight measured and the calculated values for gross weight and cg, a simple linear correction as a function of time is applied to the calculated values as illustrated in figure 3. If test experience indicates no significant difference between postflight measured and calculated values, the requirement for the postflight weighings may be relaxed.

Special equipment is required to determine the weight and cg locations of the test airplane. Recently, a number of manufacturers have marketed portable scales designed to accomplish this task. These portable scales have been used at the AFFTC with excellent results. Specifications for these scales are given in reference 2. The key parameters that must be measured are;

1. Fuel quantity from totalizer
2. Airplane gross weight (W_t)
3. Airplane center of gravity (cg)

The importance of an accurate determination of in-flight gross weight cannot be overly emphasized. Probably the two biggest problems in assessing "field" reports on airplane performance at the AFFTC is the uncertainty in instrumentation calibrations and erroneous gross weight information.

C. STATIC INSTALLED THRUST DETERMINATION

When conducting flight tests, there always exists a concern as to whether or not the test airplanes are a representative sample from population or aircraft fleet of interest. One series of ground tests that must be accomplished to assist in this "representative sample" determination is a static thrust run. Specifically, the objectives of static thrust calibrations are to determine if:

1. The test engine is representative of a production unit
2. Thrust levels before, during and after the performance tests are consistent
3. Airframe installation thrust losses are as advertised assuming that the uninstalled engine performance is known
4. The engine information provided in the DT&E reports and performance data substantiation documents is valid for the test airplane

Obviously, only a limited amount of data can be acquired toward objective 4 since a static thrust run permits exploration of only a small portion of the engine's operating envelope. However it is an important portion since it covers the takeoff envelope which is always of concern. The data obtained also forms an excellent base from which to assess the data indicated for the remaining positions of the envelope.

To conduct static thrust runs, some method is required of measuring the horizontal component of installed thrust from idle to maximum power. This is normally accomplished with a thrust stand where the airplane can be secured to allow maximum power operation. Thrust stands vary from complex balance-beam affairs with real time computational capabilities as represented by the AFFTC horizontal thrust stand to simple cable and load cell arrangements as presented in figure 4. The airplane is secured with a cable attached to the landing gear struts and a ground tiedown. The cable is quite long (approximately 50 feet) to both keep the angle between the cable and the thrust vector low, and to allow the load cell to be placed in a position of relative protection from the engine exhaust. The load cell is a standard model calibrated prior to use and its output feeds into a signal conditioning system that allows for both a real time digital output and an analog permanent record of the thrust levels attained. Excellent test results have been attained by AFFTC personnel using this simple and inexpensive thrust measurement system.

The key parameters that must be recorded are;

1. Pressure altitude (H_i)
2. Ambient temperature (T_a)
3. Ambient windspeed and direction (V_w)
4. Engine fuel flow (W_f)
5. Power lever angle (PLA)
6. Engine speed (rpm)

7. An appropriate power indicating parameter such as turbine inlet temperature (TIT), engine pressure ratio (EPR)

8. Measured static thrust (F_g)

After the airplane has been secured with the cable, run the engine to maximum power for a few seconds to remove any slack from the system and "set" the cable. Shut down the engine and null the load cell output to zero. For the actual tests, traverse the engine static thrust range from idle to maximum thrust and back to idle in approximately 5 percent increments of either rpm, EPR or measured thrust. Allow 3 to 5 minutes of engine stabilization after the point has been set or until the strain gauge output indicates a stable thrust level. Record all atmospheric data and key engine parameters at each increment. If the test airplane is equipped with afterburners, either PLA, W_f or nozzle position (if available) can be used to supplement measured thrust as test control parameters. Of course, if only minimum and maximum afterburning thrust data are required, PLA is sufficient to set repeatable test points. Usually the thrust range is traversed twice with single-engine airplanes, and once for each engine then once with both engines for two-engine airplanes if fuel permits. For large airplanes with two or more engines mounted relatively far from the centerline of the airplanes, tests should be planned to minimize yawing moments created by asymmetric engine operations.

These tests should be conducted at least at the start and end of the flight tests during periods of calm or very light winds. The load cell should have a range of operation and the cable be of sufficient strength so that data can be acquired with all engines at maximum thrust. For safety considerations and test control, radio contact must be maintained with the pilot at all times. It is important that foreign-object damage (FOD) screens not be installed over the airplane's engine air inlets during these thrust calibrations.

The installed thrust run data are presented as illustrated by figure 5. The basic concept is to obtain engine installed thrust as a unique function of engine parameters. During the tests, these plots are developed for test control by using indicated values for the parameters. After the tests, however, instrument calibrations are applied and an effort is made to develop unique functions between installed thrust and rpm, EPR, TIT and fuel flow. Parameter identification techniques have indicated that the engine data can be normalized (i.e., reduced to a unique function) using the parameters presented in figure 5B, however, other coefficients can properly be used to normalize a given data set. For example, $\theta_a^{0.688}$ has been used rather than $\theta_a^{0.5}$ as indicated in figure 6. Other coefficients may be valid as long as it is recognized that they are good only for a particular data set.

Performance demonstrations and comparisons could be successfully conducted without using the static thrust run results to develop or confirm existing thrust and engine parameter relationships since no attempt is made to establish absolute levels of thrust and drag. What makes static thrust runs absolutely necessary for any performance flight tests, is the determination of: whether or not the airplane is a representative fleet sample; any engine performance changes during the flight tests; and finally, the extent of losses in engine performance due to the airframe installation.

FLIGHT TESTS

Finally, we arrive at the purpose indicated by the title of the paper: The performance flight testing of production airplanes. These flight tests can be used to evaluate new models or subsequent production machines. It should be emphasized, however, that no flight testing can be successfully accomplished without information from the ground testing previously discussed.

A. AIRSPEED CALIBRATIONS

All airplanes are calibrated for pitot-static system error during development. However, some export and virtually all later production airplanes have small configuration changes around the nose or empennage that may cause these initial calibrations to become invalid. There is no way to obtain quality performance data if a high level of uncertainty exists in the speed and altitude of the airplane. In flight testing for pitot-static system errors, it is usually assumed that all the airspeed and altitude errors result from the positioning of the static pressure source and that the pitot probes are free of total pressure reporting error. This assumption greatly reduces the magnitude of the pitot-static position error determination and has been proven to provide performance data of suitable quality.

The objective, therefore, is to determine static pressure source errors due to the positioning of the static ports on the airframe. Total pressure errors and total temperature errors are usually not investigated. Analytical methods do exist for the calculation of static source position errors but their applications have met with mixed success. Most performance flight test engineers still rely on, indeed insist upon, flight tests to determine the static source position errors. The task, therefore, is to determine the

actual pressure altitude at which the test airplane is flying and compare that altitude to the altitude indicated by the instrumentation system after the proper instrument calibrations have been made.

Volumes have been written about this subject. On the surface, the conduct and analysis of static source position error tests would appear quite straightforward. In actual practice, however, these tests and the subsequent data analysis is often quite complex. There are two principal methods that provide adequate data to assess subsonic static source position error. These are the pacer and the ground reference flyby techniques. The pacer technique involves flying the test airplane in formation and at the same altitude as a "pacer" airplane with a known, previously calibrated, static pressure position error. The ground reference flyby technique involves flying the test airplane past some ground-based reference point at a known, and recorded pressure altitude. The static source position error of the test airplane is then determined by comparing its reported pressure altitude, connected for instrument error, with that of calibrated pacer or the known pressure altitude at the reference. In equation form, this process can be represented as:

$$\Delta H_{pc} = Hc_{Ref} - H_{ic_{Test}} \tag{1}$$

where ΔH_{pc} = static pressure position error of test airplane

Hc_{Ref} = True pressure altitude of reference, either pacer or a ground point

$H_{ic_{Test}}$ = Reported pressure altitude of test airplane after correction for instrument error

Since the ground reference is usually at a lower altitude than the test airplane, equation (1) must be expanded to account for a difference in height. The ground reference flyby technique and the variables involved are represented in figure 6. For this technique equation (1) becomes:

$$\Delta H_{pc} = Hc_{Ref} + \frac{T_{a_t}}{T_{a_s}} \times \Delta h_t - H_{ic_{Test}} \tag{2}$$

where T_{a_t} = Test-day ambient temperature (deg K)

T_{a_s} = Standard-day ambient temperature (deg K)

Δh_t = Measured height difference between ground reference and test airplane

Once ΔH_{pc} has been determined, the effect of this static source position error on airspeed, Mach number, and static pressure for both subsonic and supersonic flight can be calculated using equations presented in reference 1. Only the subsonic equations are listed below.

$$\Delta V_{pc} = \Delta H_{pc} \times \frac{\sigma_s}{58.566} \times \frac{a_{sl}}{V_{ic}} \left[1 + 0.2 \left(\frac{V_{ic}}{a_{sl}}\right)^2 \right]^{-2.5} \tag{3}$$

$$\Delta M_{pc} = \Delta H_{pc} \times 0.007438 \left(\frac{1 + 0.2\, M_{ic}^2}{T_{a_s}\, M_{ic}}\right) \tag{4}$$

$$\frac{\Delta p}{q_{cic}} = \frac{\left\{1.4\frac{V_{ic}}{a_{sl}}\left[1+0.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]^{2.5} \times \frac{\Delta V_{pc}}{a_{sl}}\right\} + \left\{0.7\left[1+0.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]^{1.5}\left[1+1.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]\left(\frac{\Delta V_{pc}}{a_{sl}}\right)^2\right\}}{\left[1+0.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]^{3.5} - 1.0} \tag{5}$$

Where ΔV_{pc} = Airspeed position error correction

σ_s = Density ratio (ρ/ρ_{sl}) measured at H_{ic}

a_{sl} = Speed of sound at sea level - standard day conditions

V_{ic} = Instrument corrected airspeed

ΔM_{pc} = Mach number position error correction

M_{ic} = Instrument corrected Mach number

T_{a_s} = Standard day temperature at H_{ic} (deg K)

To employ the ground reference flyby technique, one requires equipment to measure pressure altitude at the ground reference, and a method of determining the height between the reference and the test airplane (Δh_t). At the AFFTC, 4 to 6 standard altimeters are used to simultaneously record pressure altitude as the test airplane passes the reference ground station. The readings from these altimeters are then averaged in a simple but effective attempt to reduce the impact of hysteresis and random errors. Any height measuring system can be used to measure Δh_t. At the AFFTC we use geometrical triangulation with the tower and flyby line as references but stadiametrics, cinetheodolites, laser trackers, ground-based radars, etc., can be used. One very promising new concept is the use of the onboard radar altimeter to measure Δh_t. In the past, these radar altimeters have not been of sufficient accuracy or resolution to enable their use in static pressure position error determination. Newer radar altimeters, however, promise both greatly improved accuracy and resolution. The advantage of radar altimeters is that they are standard equipment on the airplane so no modification to the airframe is required, and they eliminate the need for any ground-based Δh_t measurement system.

The pacer method is probably the easiest to analyze and requires no special ground equipment, however, it naturally requires the expense and maintenance of a calibrated pacer airplane with approximately the same performance envelope as the test airplane. At the AFFTC, the maintenance of a fleet of four pacers is a full-time effort. Each pacer must be flown against a calibrated reference, such as another pacer or the ground reference, approximately every 4 months so that its pitot-static system characteristics remain know in spite of frequent instrumentation or airplane configuration changes. The instruments used in the pacers are frequently removed for calibration updates.

Regardless of which technique is used to obtain the static source position error calibration the following key parameters must be recorded for each data point;

1. Ambient pressure altitude from ground reference or pacer airplane ($H_{c_{Ref}}$)

2. Ambient temperature (T_a)

3. Indicated pressure altitude of test airplane (H_i)

4. Indicated airspeed of test airplane (V_i)

5. Gross weight of test airplane (W_t)

6. Height from ground reference to test airplane: flyby method only (ΔH_t)

The reference flyby technique as proven through experience to be the most precise method. It is, however, time consuming since only one data point is obtained per pass by the reference, and requires a relatively complex analysis. It also produces data at only one, rather low, altitude. This does not usually present a problem when calibrating fighter or attack airplanes that have both high airspeed and Mach number limits. It does, however, present a problem when testing larger airplanes that usually have rather high Mach number speed limits but are restricted to relatively low values of airspeed, because they cannot be flown to true airspeeds high enough to enter the transonic region when position errors are usually the largest. When conducting reference flyby tests, care must be exercised to keep the test airplane at sufficient height so that the static source position error calibration is not influenced by the proximity of the ground. At the AFFTC, a rule of thumb is to fly at least 1½ wingspans above the ground while recording data.

The pacer method can be used throughout the altitude envelope of the test airplane. Because of the difficulties associated with flying in close formation at very precise airspeeds, however, the pacer method usually produces static source position error data of rather less quality than the ground reference flying method. It is recommended, however, if the performance flight test organization has access to a calibrated pacer airplane. It is a very time-efficient technique since many points can be obtained per mission. The pacer method also allows ΔV_{pc} to be directly measured so that the measured values can be compared with those calculated from ΔH_{pc} using equation (3).

There are other techniques that can be used to determine static source position errors, but they are not recommended. These techniques, such as the groundspeed course, usually require either complex analysis or are very dependent on the measurement of atmospheric conditions such as windspeed and direction. Supersonic static source position error calibrations are possible, however, they will not be discussed in this paper due to their rather limited application and because supersonic static or total pressure position errors are very small above the transonic speed range. Usually the ground reference flyby technique is used with ground-based radars required to determine Δh_t. Since the test airplane is usually at very much higher altitudes than the ground-based radars, great care must be taken to account for atmospheric conditions when attempting to obtain supersonic position error calibrations with radar data.

Typical static source position error data are presented in figure 7. The recommended technique is to convert the ΔH_{pc} values obtained during flight testing to values of $\Delta P/q_{cic}$ (which is called the static source position error coefficient) using equation (5). These values of referred pressure differences are then plotted versus lift coefficient and faired as in figure 7. Lift coefficient calculated using instrument-corrected Mach number is labeled C_{Lic}. The determination of lift coefficient is discussed in the Cruise Performance Section and defined by equation (12). Once a consistent fairing for position error coefficient has been attained, the fairing can be converted back into ΔH_{pc} fairings using equation (6), and into ΔV_{pc} and ΔM_{pc} fairings using equations (3) and (4), respectively.

$$\Delta H_{pc} = \frac{\Delta P}{q_{cic}} \times \frac{29.92126 \left[1+0.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]^{3.5} - 1}{0.00108\ \sigma'_s} \tag{6}$$

where σ'_s = Density ratio at $(H_{ic} + \Delta H_{pc}/2)$

Curve and table lookup values for the relationships expressed in equations (3), (4), (5), and (6) can be found in reference 1.

The basic concept is to obtain ΔH_{pc} from flight test, convert ΔH_{pc} to values of the static source position error coefficient $(\Delta P/q_{cic})$ and obtain a consistent fairing. This fairing is then converted back into fairings of ΔV_{pc}, ΔM_{pc} and ΔH_{pc}. Although this process appears relatively simple, obtaining consistent, accurate static source position error calibrations is one of the most basic, yet most challenging tasks in performance flight test. Almost every conceivable phénomenon from angle of attack to ground effect to fuselage flexure around the static ports can and does effect the position error calibration data. Static source position error calibrations, however, must be conducted if there is any indication that the test airplane does not share the calibrations of the production airplanes as indicated in the Flight Manual.

B. TAKEOFF PERFORMANCE

The takeoff performance of an airplane is very important since it determines, among other things, from where the airplane can be operated. The objective of these tests is to determine the horizontal ground distance that the airplane requires to accelerate from brake release to takeoff (lift-off) and from takeoff to some obstacle clearance height, usually 50 feet (the airphase distance).

Of course, to fulfill the objective, some technique or equipments are required to measure horizontal distances. Single-station solution cameras, cinetheodolites and laser trackers have all been used with success. Some larger airplanes have downward-looking cameras mounted on the fuselage bottom to photograph runway features such as painted stripes or tar strips that are at known distances along the takeoff roll. One technique that has been successfully employed in the past during remote site testing is to station several observers along either side of the runway at the anticipated takeoff point. After the actual takeoff each observer independently walks to the spot he or she thought the airplane lifted off. The distances from brake release to each observer are recorded and averaged for a single takeoff distance. Although not terribly sophisticated, this technique can produce data of suitable quality for the purposes of a performance demonstration or comparison. This technique cannot be used to determine obstacle clearance distance. In the absence of TSPI, obstacle clearance distances must be calculated using onboard production or flight test instrumentation and can be approximated by:

$$S_{50} = \frac{50}{\operatorname{Tan}\gamma} = \frac{50}{\tan\left[\sin^{-1}\left(\frac{50}{t_{50}-t_{Lo}}\right)/\left(\frac{V_{Lo}+V_{50}}{2}\right)\right]} \tag{7}$$

Where S_{50} = Distance to 50 feet (ft)

t_{50} = Time from brake release to 50 feet (sec)

t_{Lo} = Time from brake release to lift-off (sec)

V_{Lo} = True airspeed at lift-off (ft/sec)

V_{50} = True airspeed at 50 feet (ft/sec)

A simple TSPI system to measure obstacle clearance distances can be implemented by stationing several manually-sighted theodolites and operators at predetermined distances along the runway between the Flight Manual-specified takeoff and obstacle clearance distances. These theodolites are used to determine the height of the airplane as it passes each station. Each operator would also be equipped with a portable radio and a stopwatch. At brake release the pilot activates the event tone as a signal for each operator to start their stopwatch. As the airplane passes each theodolite station, time and height above the ground are recorded. The pilot must record takeoff and climbout airspeeds either manually or with the onboard instrumentation. Using this technique, a reasonably accurate obstacle

clearance performance analysis can be accomplished. Brake release to takeoff distance would be determined by additional observers who would converge on their estimated lift-off point as previously discussed.

Maximum performance landing or rejected takeoff (RTO) tests are not recommended for performance demonstrations and comparisons. Few airplanes can accomplish these tests without presenting hazards both to the aircrew and ground observers. Lightweight maximum performance landings present the possibility of blown tires due to skids and subsequent loss of control of the airplane. Heavyweight maximum performance landings or RTOs can result in fused brakes and flat or even blown tires due to the tremendous heat generated by the brakes during the stop. Maximum performance landing or RTOs are among the most hazardous tests conducted at the AFFTC and should be attempted only under very controlled conditions by experienced flight test personnel.

The key parameters required to demonstrate Flight Manual takeoff performance are:

1. Indicated airspeed during takeoff roll and airphase (V_i)
2. Aircraft indicated pressure altitude (H_i)
3. Ground roll distance to takeoff (S_{TO})
4. Height of airplane above runway, if TSPI is available (h_t)
5. Airphase distance from takeoff to 50 feet, if TSPI is available (S_{50})
6. Airplane gross weight (W_t)
7. Local meteorological data such as;
   a. Windspeed (V_w)
   b. Wind direction
   c. Ambient temperature (T_a)
   d. Ambient pressure altitude (P_a)
   e. Runway slope

If takeoff tests are conducted to provide data for comparison with the Flight Manual, the airplanes should be operated as recommended by the Flight Manual. No-wind conditions are by far the best for these tests, but they can be conducted in steady winds if the Flight Manual provides charts to account for winds during takeoff. Equations do exist to allow takeoff and airphase distances to be corrected to no-wind conditions, but they require considerable experience to apply correctly and are not recommended. Equations also exist to allow correction of test day takeoff distances to standard day conditions. They are not required for these performance demonstrations because the basic objective is to determine if the Flight Manual accurately represents the takeoff performance of the airplane over a wide range of test-day conditions. Since takeoff performance data tends to be quite scattered (large values of standard deviation), many tests with several repeat points are required before any defendable conclusions can be made. Therefore, every takeoff accomplished during a performance demonstration or comparison should obtain ground roll and airphase distances. If takeoff performance is a prime concern for the test airplanes, consideration should be given to reserving one or two flights for the sole purpose of conducting takeoff tests. In this manner, not only can the data base be expanded, but the effects of gross weight and temperature can be examined.

C. CRUISE OR RANGE PERFORMANCE

Cruise performance or range is an important factor in the assessment of the combat potential of an airplane and tests to determine cruise performance must be included in any performance demonstrations. The accepted method currently used to determine cruise performance is the stabilized speed-fuel flow method commonly called the speed-power technique.

The key parameters that must be recorded at each stable speed-power point are listed below. Remember that each must be corrected for instrument error and that airspeed and pressure altitude must also be corrected for static source position error before any data analysis is attempted.

1. Indicated airspeed (V_i)
2. Indicated pressure altitude (H_i)
3. Total temperature (T_T)
4. Total fuel flow (W_f)
5. Total fuel remaining

Speed-power data can be obtained during any flight condition with stable airspeed and altitude, however, the two most popular methods are at constant altitude, and at constant weight-pressure ratio (W_t/δ_a). The constant-altitude method is very useful since most Flight Manuals have cruise performance presented as a function of airplane speed (Mach number) and weight at a constant altitude. Constant-altitude cruise data are obtained by flying the airplane at selected speeds from the maximum speed attainable or allowable at the nonafterburning power setting to near-minimum flying speed. Each speed-power point must be stable in airspeed, pressure altitude and power setting. Airplane trim or configuration must not be changed during the speed-power point. From three to five minutes of stable flight are required per point with data recorded at the beginning, middle and end of the stabilized period. It is important to fly as slow as possible when obtaining the near-minimum flying speed point. This not only allows a qualitative assessment of the airplanes flying qualities at slow speeds, but insures that the airplane is flown on the "backside" of the power curve during the speed power tests. The backside points are not required if only optimum cruise is of interest, but allow a concurrent assessment of the endurance or loiter performance of the airplane.

As previously noted, total temperature or ram air temperature must be recorded in flight. The Flight Manual will present cruise performance as a function of ambient temperature (T_a). To convert total temperature (T_t) to ambient temperature (T_a), use the relationships in equations (8) and (9).

$$T_T = T_a\left[1 + K\left(\frac{\gamma-1}{2}\right) M^2\right] \tag{8}$$

where T_T = Total air temperature (deg K)

T_a = Ambient air temperature (deg K)

K = Temperature probe recovery factor (usually from 0.95 to 1.00)

γ = Ratio of specific heats: C_p/C_v

M = Mach number corrected for static source position error

If K is assumed to be equal to 1.00, which is an appropriate assumption for the purposes of a performance demonstration, and the ratio of specific heats (γ) is assumed to be constant 1.4, which is usually valid for Mach numbers less than 2.0, then equation (8) reduces to:

$$T_a = T_T/(1 + 0.20\ M^2) \tag{9}$$

If one of the objectives of the performance tests is to either determine the optimum cruise conditions or validate the Flight Manual's recommended optimum, the constant weight-pressure ratio technique is used. The test points are flown in the same manner as the constant-altitude technique with stable airspeed, pressure altitude and power levels, but successive points are flown at higher altitudes to maintain a constant value of W_t/δ_a as the airplane becomes lighter due to fuel consumption. The ratio of test to sea level standard pressure (δ_a) is calculated as shown in equation (10).

$$\delta_a = P_{a_t}/P_{a_{sl}} \tag{10}$$

where P_{a_t} = Test ambient pressure

$P_{a_{sl}}$ = Standard-day sea level pressure

By flying the airplane at stabilized speed-power-altitude points and at constant W_t/δ_a with various Mach numbers intervals, the optimum cruise conditions can be validated or determined. Optimum cruise performance will occur when NAMPP is maximized. Cruise performance is usually expressed in terms of the parameter Range Factor (RF) as defined

$$RF = NAMPP \times W_T = \frac{V_T}{W_f} \times W_t \tag{11}$$

where V_T = True airspeed

W_f = Total fuel flow

W_t = Airplane gross weight

by equation (11). For jet-propelled airplanes, fuel flow is essentially a function of the thrust required for level flight (as opposed to power required for level flight as is the case for propeller-driven airplanes). Thrust required for level flight is equal to total drag for that flight condition, which can be expressed as the nondimensional parameter, drag coefficient (C_D). Drag coefficient, in turn, can be defined by lift coefficient (C_L) and Mach number. Because of these relationships it can be shown that the cruise performance of a jet airplane, i.e., the speed-fuel flow relationships NAMPP and RF can be expressed in terms of speed and lift coefficient. As indicated by equation (12), lift coefficient is a function of Mach number and the weight-pressure ratio for a given type of airplane. Therefore, by flying speed-power test points at constant values of W_t/δ_a and

at various Mach numbers, we are essentially attempting to find the lift coefficient and its related Mach number which produces the lowest drag, and hence fuel flow, for the highest speed. Best endurance or loiter performance is simply the speed, at any specified altitude, where minimum fuel flow occurs. Best endurance speeds can be determined at any altitude by merely reducing speed until either fuel flow stops decreasing and begins to increase, or until flying qualities dictate that no further reduction in speed is advisable.

$$C_L = \int \frac{n_Z}{M^2 S} \times \frac{W_t}{\delta_a} \tag{12}$$

where n_Z = Normal acceleration

S = Airplane reference wing area

Constant W_t/δ_a speed-power data are usually analyized by preparing plots like those in figure 8. The recommended optimum cruise Mach number, altitudes, and gross weights are obtained from the Flight Manual. Tests are then conducted at a number of weight-pressure ratios at and around this recommended optimum value, as illustrated in figure 8A. The full speed range from maximum speed in nonafterburning power to near-minimum flying speed should be investigated at each weight-pressure ratio. The number of speed-power points flown during each weight-pressure ratio series depends on the speed range, but six to eight is normally considered a minimum. The airplane should be flown at the high speed point first and then speed reduced for each subsequent point in the weight-pressure ratio series. After the weight-pressure ratio series have been plotted on the RF versus Mach number plot, the data are faired as illustrated in figure 8A. The peaks of these constant weight-pressure ratio fairings can then be plotted as shown in figure 8B to aid in further data analysis.

During a demonstration or comparison flight test, probably both the constant-altitude and weight-pressure ratio techniques will be used to determine cruise performance. Regardless of which technique is used, there are several corrections that can be made to adjust the resulting speed-fuel flow relationships for off standard or aim flight conditions. Theoretical relationships exist for correcting test fuel flows to standard ambient temperatures, desired test weight-pressure ratios and to zero values of excess thrust. However, for the purposes of flight test to demonstrate Flight Manual performance, these corrections are not usually required or recommended. Cruise performance is not exceptionally sensitive to small variations in ambient temperature and, indeed, Flight Manuals only present NAMPP or RF for standard day temperatures. The requirement for corrections to account for variations in weight, altitude and power levels during any given test speed-power point can be minimized through proper pilot technique directed toward insuring a stable point. If subsequent data analysis indicates that a particular point was unacceptable, it can easily be reflown since cruise performance tests can readily fit into any test scenario. If program constraints require that a limited amount of unstable data must be analyzed, the equations and techniques required to apply temperature, weight, altitude and energy corrections to cruise data are presented in reference 3. These corrections can become quite complex, however, and are not recommended for performance flight demonstrations or comparisons. By far, the best results can be obtained by expending effort to fly good, stable speed-power data rather than to attempt correction of unsuitable data.

D. ACCELERATION AND TURNING PERFORMANCE

Sometimes referred to as "combat performance" in Flight Manuals, acceleration/deceleration and turning performance is of interest in all military airplanes, but especially in fighter and attack airplanes. The objective is to determine if the Flight Manual is correct and appropriately depicts the ability of the airplane to change its energy level and direction of flight. The basic concept for the performance demonstrations and comparison of production airplanes is to analyze dynamic performance characteristics, such as accelerations and turns, using as stable maneuvers as possible.

Energy-Height Concepts

Excess thrust F_{ex} is the essential parameter for the analysis of acceleration and deceleration performance. Unfortunately, unless the airplane is equipped with flightpath or body mounted accelerometers and a high sample rate data recording system, excess thrust cannot be measured accurately and must be calculated. Further, it must be calculated indirectly by first calculating Specific Excess Power (P_S) which is the time derivative of energy-height (H_E). These calculations are based upon the measurement of the speed and height of the test airplane obtained from either onboard cockpit instrumentation or ground-based TSPI.

The primary data to be obtained from acceleration (or deceleration) testing are time required to obtain a given Mach number and P_S. Time required to accelerate to a given Mach number can be determined directly from faired time histories (our first data analysis step) of either cockpit instrumentation-derived true airspeed and Mach number or ground-

based TSPI. Remember that the cockpit-derived values must be corrected for both instrument and static source position errors. These times to accelerate or decelerate can be compared directly with Flight Manual values for test day conditions.

Energy height, P_S and F_{ex} are found by considering the total energy of the test airplane to be the sum of Potential and Kinetic energies at any point during the acceleration.

$$E = W_t \; (H + V_T^2/2g_r) \tag{13}$$

where E = Total energy

H = Geopotential height, which is merely tapeline height (h_t) corrected for local gravity

g_r = Universal gravity constant 32.174 ft/sec^2 or 9.806 M/sec^2

By dividing both sides of equation (13) by airplane test gross weight, Energy-Height (H_E) can be obtained, and P_S is merely the time derivative of Energy-Height. We must, however, calculate $\dot{H}$ using either the time differential of tapeline altitude ($\dot{h}$) as obtained from ground-based TSPI, or the time differential of pressure altitude ($\dot{H}_C$) as obtained from onboard cockpit instrumentation. For ground-based TSPI, equation (16) can be used. If, however, instrument and static source position error corrected onboard instrumentation are used, equation (17) is appropriate.

$$H_E = \frac{E}{W_t} = H + \frac{V_T 2}{2g_r} \tag{14}$$

$$P_S = \dot{H}_E = \dot{H} + \frac{V_T}{g_r} \times \dot{V}_T \tag{15}$$

$$P_S = \frac{g_L}{g_r} \times \dot{h} + \frac{V_T}{g_r} \times \dot{V}_T \tag{16}$$

where g_L = Local acceleration due to gravity

$$P_S = \frac{g_L}{g_r} \dot{H}_C \left(\frac{T_{std}}{T_{test}}\right)^{½} + \frac{V_T}{g_r} \times \dot{V}_T \tag{17}$$

where T_{std} = Standard-day ambient temperature at test altitude (deg K)

T_{test} = Test-day ambient temperature at test altitude (deg K)

For most flight test work, and especially for performance demonstrations or side-by-side comparisons as defined, herein, both ratios g_L/g_r and $(T_{std}/T_{test})^{½}$ can be assumed quite small and will not significantly effect the calculation of specific excess power. If this assumption is made, equation (17) reduces to equation (18). Excess thrust can be

$$P_S = \dot{H}_C + \frac{V_T}{g_r} \times \dot{V}_T \tag{18}$$

calculated using equation (19).

$$F_{ex} = P_s \times \frac{W_t}{V_T} \tag{19}$$

With this last equation, all the essential elements have been obtained to evaluate Flight Manual acceleration, deceleration and turning performance. Specific excess power data are used to determine optimum climb and acceleration schedules and F_{ex} data are used to determine best turn speeds.

Key parameters that must be recorded during acceleration and turning performance tests are:

1. Indicated airspeed (V_i)
2. Indicated pressure altitude (H_i)
3. Total fuel quantity remaining
4. Total temperature (T_T)

5. Normal acceleration (g)

6. Time

TEST MANEUVERS

Test techniques will be discussed for level accelerations, stabilized and windup turns. Level decelerations will not be discussed since they are essentially the reverse of level accelerations. Level accelerations are particular productive performance maneuvers that can be used to obtain direct side-by-side comparisons of the performance of two different airplanes, the minimum practical flying speed as well as the thrust-limited maximum speed, times to accelerate to given Mach numbers at selected normal load factors, and P_s and F_{ex} levels throughout the speed and altitude envelope at selected normal load factors. Level accelerations can be conducted at 1 g or at elevated normal accelerations and load factors if the performance of the test airplane permits. Level accelerations at elevated normal load factors are accomplished in a turn at constant altitude. To accomplish the acceleration, the airplane is first stabilized at near the minimum flying speed. It is often useful to use speed brakes and/or a turn at a load factor higher than the planned acceleration to help stabilize the airplane at a low speed while keeping engine power at a relatively high level. This is especially helpful when testing airplanes equipped with slow-accelerating engines. When the airplane and engine(s) have stabilized and the desired heading is reached, simultaneously retract the speed brakes, advance power to the level desired for the acceleration, and roll out wings level for a 1-g acceleration or to the bank angle appropriate for the selected normal load factor. Accelerate to maximum or thrust-limited speed at constant altitude. Onboard cockpit instrumentation, be it pilot comments and notes or cockpit and/or HUD cameras or video recorders, should be used to acquire data at the maximum sample rate possible. Regardless of the data acquisition system, the pilot should call out speeds and times during the acceleration for recording by ground-stationed test personnel.

One of the items of interest that result from 1-g level acceleration tests is the maximum speed of the test airplane at given test gross weights, altitudes, and temperatures. It is often desirous to determine the maximum speed of the airplane at <u>standard temperatures</u> and gross weights to evaluate guarantee compliance. This can be quite simply, yet effectively done by plotting test-day ambient temperature versus maximum speeds attained as functions of airplane test gross weight as presented in figure 9. The plot can then be entered with standard-day values of ambient temperature, a specified gross weight, and the maximum speed for those conditions determined with reasonable accuracy.

Stabilized turns are designed to determine the thrust-limited turning performance of an airplane. For any given airplane, the level of normal acceleration that can be sustained in a turn is a function of the test gross weight and thrust available. These are excellent maneuvers to acquire data for performance demonstrations and comparisons because the results are fairly insensitive to small variations in atmospheric conditions or airplane gross weights. The Flight Manual values for sustained maneuvering capability, therefore, can easily be validated for the test conditions flown without corrections for fuel used during the maneuver or small changes in pressure altitude and ambient temperatures. The airplane is stabilized at the test values of speed (Mach number) and altitude. The stabilized turn is initiated by the pilot simultaneously selecting maximum power and rolling into a turn to increase bank angle until normal acceleration, Mach number, and altitude are stabilized. When speed and altitude are stable in maximum power, the resulting normal acceleration is the thrust-limited turning capability. These turns can offer considerable pilot challenge at the higher normal accelerations and load factors. The exact technique used to stabilize in speed and altitude varies with pilot preference. AFFTC test pilots have learned that it is helpful to think of using bank angle to control speed and pitch altitude to control altitude while attempting to stabilize.

It is often useful to follow a level acceleration or thrust-limited turn with a windup turn to the maximum useable lift coefficient. After the acceleration or stabilized turn the Mach number is maintained as constant as possible while normal acceleration is increased at approximately 0.5 g per second to the maximum useable lift. The pilot notes buffet characterisitics as normal acceleration and load factor, and therefore angle of attack increase. These lift-limited turns can be compared with the thrust-limited performance to form a good quantitative picture of the maneuvering potential of the test airplane. Note that the buffet levels and maximum useable lift as functions of angle of attack and C_L are <u>not</u> affected by variations in airplane gross weights or atmospheric conditions.

Specific excess power and F_{ex} data presentation and analysis should be initiated by a time-history presentation of the instrument corrected and static source position error calibrated key parameters as is the case for all the test data discussed in this paper. At selected times throughout the acceleration, equations (13) through (19) are used to calculate P_S and F_{ex} from the data fairings. Specific Excess Power and F_{ex} values are usually presented as shown in figure 10. Both P_S and F_{ex} are plotted as a function of Mach number and pressure altitude, with accelerations resulting in positive values of P_S and F_{ex} and deceleration in negative values. These values of P_S and F_{ex} will be at the test altitudes and ambient temperatures and should be so noted. The resulting data fairings can be used to determine agreement with the Flight Manual which will usually contain hot day, standard day and cold day performance. The data can also be used to; validate or modify speeds for

optimum climb and maneuvering since correction to standard day ambient temperatures change the levels of P_S and F_{ex} but usually do not significantly alter the shapes of the curves, determine maximum speeds and, along with pilot comments on handling qualities, minimum useable flying speeds.

Stabilized thrust-limited turning peformance can be presented as illustrated by figure 11. Normal acceleration (n_z) can be converted into thrust-limited turning rate (ω) in degrees per second using equation (20).

$$\omega = \frac{57.3 g_r \left(n_z^2 - 1\right)^{1/2}}{V_T} \tag{20}$$

If the windup turns are accomplished the resulting data can be presented as illustrated by figure 12. Notice that lift-limited turning performance and buffet boundaries are always presented in terms of C_L or angle of attack since these parameters are independent of test atmospheric conditions or airplane gross weights. Once these data have been plotted and faired, the fairings of maximum useable C_L can be changed into values of normal acceleration or load factor and maximum turn rate using equations (12) and (20). The thrust-limited turning performance can be combined with the lift-limited or maximum turning performance and buffet boundaries as illustrated in figure 13.

E. ENGINE HANDLING CHARACTERISTICS

No performance demonstration would be complete without an attempt to validate the engine handling characteristics of the test airplane. A certain amount of engine handling information will be acquired when flying the test maneuvers discussed previously in this paper, however, it will be limited since most of the maneuvers are designed to facilitate stable engine operation. Therefore, special tests must be conducted to determine the dynamic response characteristics of the test engine(s). Engine handling tests are quite easily accomplished but they do require flight time and detailed planning, especially with single-engine airplanes, because unrecoverable engine stalls can possibly result. These are essentially "yes or no" tests and do not require any sophisticated analysis of the resulting data. Since modern turbojet engines and their controls are extremely complex, the results of engine handling tests are not always repeatable. Because of this, a large sample size, i.e., a large number of test points, must be obtained if any valid conclusions are to be made. Experience has proven that these tests cannot be successfully integrated into the airplane performance tests because of safety considerations and separate flight time should be allocated for this task. All tests should be conducted in a buildup fashion: test points in the middle of the engine's operating envelope must be accomplished first, progressing to the outer portions of the envelope.

As in all the tests discussed in this paper, certain key parameters must be recorded at or just before the engine handling characteristics tests.

1. Indicated airspeed (V_i)
2. Indicated pressure altitude (H_i)
3. Total temperature (T_T)
4. Power lever angle or throttle position (PLA)
5. Engine speed (rpm)
6. Turbine inlet temperature or exhaust gas temperature (TIT or EGT)
7. Total fuel flow (W_f)

Three engine handling tests are recommended. These are air starts, Bodes or engine acceleration tests, and afterburner lights. Air starts are accomplished as recommended by the Flight Manual throughout the flight envelope with special emphasis on the high altitude, low speed corner. During the starts, data should be recorded at as rapid rate as possible. Pilot comments are particularly important because these are handling qualities evaluations and require qualitative as well as quantitative assessments. Tests at lower altitudes must be planned so as to allow sufficient height for acceleration to higher speeds should the initial air start attempt prove unsuccessful. For single-engine airplanes, the initial air start tests should be conducted within gliding distance of the runway should a "flameout" landing become necessary. At the AFFTC, simulated flameout landings with the engine operating at idle power are usually practiced before engine start tests are conducted on single-engine airplanes.

Bodes are rapid power lever or throttle movements (less than one second) between various power levels to examine the in-flight engine acceleration and deceleration characteristics. These are accomplished throughout the flight envelope. Data should again be sampled at the maximum rate possible. Pilot comments are especially important since some engine phenomena, such as compressor stalls (not enough airflow) or surges (too much airflow) are apparent to the pilot but often occur too rapidly to be indicated by the instruments. Any rapid power lever movement can be used to determine engine response. One sequence used is:

1. Idle-to-Military rated (maximum nonafterburning) power
2. Military-to-maximum afterburning power
3. Maximum-to-Military
4. Military-to-idle
5. Idle-to-maximum
6. Maximum-to-idle

The engine should be allowed to stabilize after each rapid power lever movement. These tests should also be performed in the power approach or landing configuration at typical landing pattern and final approach airspeeds, but of course, at higher than pattern altitudes.

If the test airplane is equipped with an augmented or afterburner (A/B) engine, the in-flight operation of this augmentation should be examined. These can be accomplished with the engine response tests (Bodes) just mentioned or independently. Again the full A/B-lite envelope should be explored with emphasis on the low speed, high altitude and low speed, low altitude portions. It is always a good idea to accomplish some of these tests during ground runs before flight tests are attempted. This is also true for the Bodes described above. By accomplishing these engine handling tests in conjunction with the static installed thrust tests previously discussed, the basic engine handling characteristics can be validated or determined prior to flight. As in the previous engine handling tests, pilot comments are essential in assessing the "feel" of the A/B-lite characteristics.

Data presentation is straightforward. Instrument corrected time histories of key engine parameters are plotted as illustrated in figure 14A. By plotting PLA, rpm, fuel flow, and TIT or EGT versus time, the air start and response characteristics can be assessed. The envelope for successful air starts, stall-free Bodes, or successful A/B lights can be determined by preparing a "bookkeeping" plot as presented in figure 14B. At each point tested, a data point indicating satisfactory or unsatisfactory engine response is noted as a function of pressure altitude and Mach number. This presentation can be used to either spot check the Flight Manual engine operating envelopes or if accomplished in a thorough fashion throughout the envelope, define the acceptable engine operating regions.

F. CONCLUSIONS

When assessing a new airplane for purchase or determining production airplane capabilities vis-a-vis a tested prototype it is important that some performance tests be conducted. Performance data need not be corrected to standard-day conditions if the intent is to use these data to demonstrate that the Flight Manual reflects the performance capabilities of the airplane because current manuals present performance levels for off-standard day conditions. Excellent performance data can also be obtained by flying test maneuvers with the test airplane in formation with a similar airplane to determine the extent of the differences between the two.

All performance testing is quantitative, and data must be acquired during each test maneuver. When onboard production cockpit instruments are used as test instrumentation, they must be first calibrated so that instrument errors are known. If possible, highly sensitive, calibrated flight test instruments should be used in place of the production units. Even assuming that the Flight Manual fuel quantity, gross weight and cg relationships are correct, the airplane should be weighed and its cg determined before and after each performance flight. Installed static thrust tests are required to assess whether or not the test airplane is representative of the fleet, at least from a thrust standpoint.

If there is any reason to believe that the static source position errors as indicated by the Flight Manual may not be representative of the test airplane, airspeed calibrations are required. The ground reference flyby technique produces the most precise results, however, pacer missions may be required depending upon the speed envelope of the test airplane. Test day data acquired during takeoff and cruise performance tests can be successfully used to assess the validity of the Flight Manual.

Dynamic performance maneuvers such as level accelerations, and thrust-and lift-limited turns are quite easily performed but analysis of the results may be difficult because of the limited instrumentation and data recording systems available on production airplanes. Again, test data are clearly adequate for validating off-standard day performance indicated by the Flight Manual and for assessing basic airplane operating characteristics such as best climb and maneuvering speeds. Acceptable engine handling characteristics are essential for a combat airplane and an assessment of these characteristics should be made during any performance test program.

Several traditional performance test maneuvers were omitted from the discussions in this paper because they either require extensive, high-sample-rate instrumentation, sophisticated atmospheric condition measuring equipment, or extensive data analysis. Correction of test-day data to standard-day conditions is not recommended since, once validated by test-day performance levels, the Flight Manual can be used to obtain standard-day performance. If the Flight Manual is validated, it can also be used with the manuals of other airplanes to assess the performance of the test airplane against its contemporaries. If the test data indicates that the Flight Manual is incorrect and does not represent the test airplane, the manufacturer should be instructed to either change the Flight Manual or change the airplane, depending upon the requirements of the customer.

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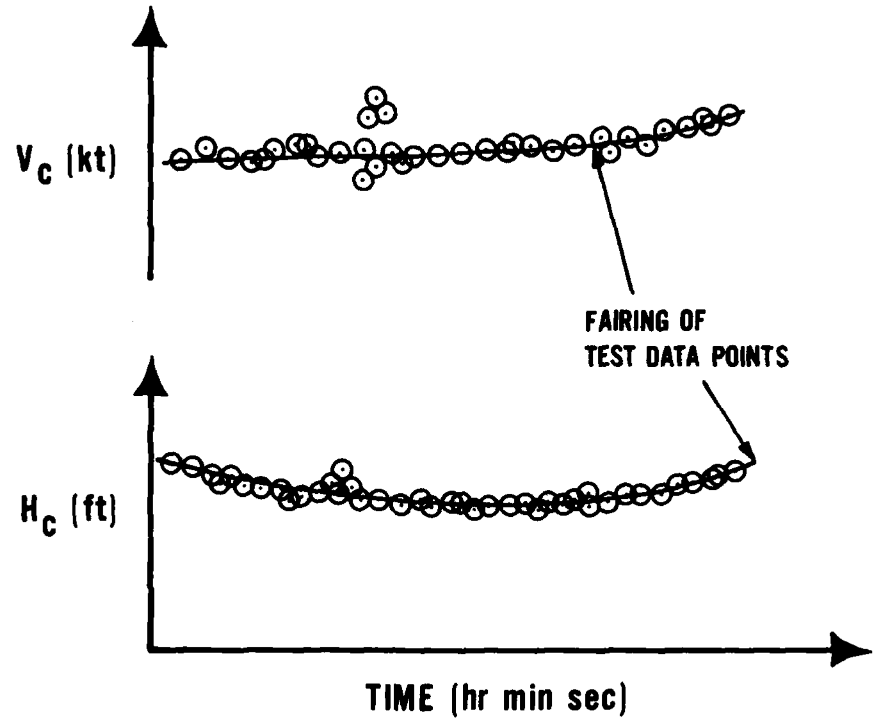


Figure 1 Typical Time Histories

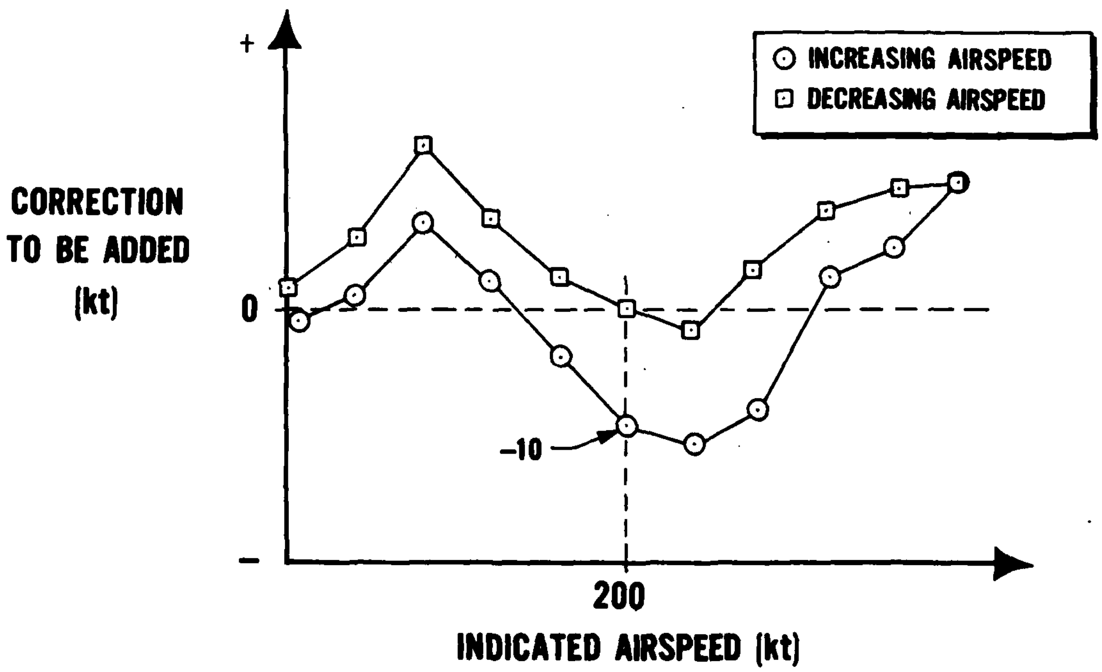


Figure 2 Typical Instrument Calibration

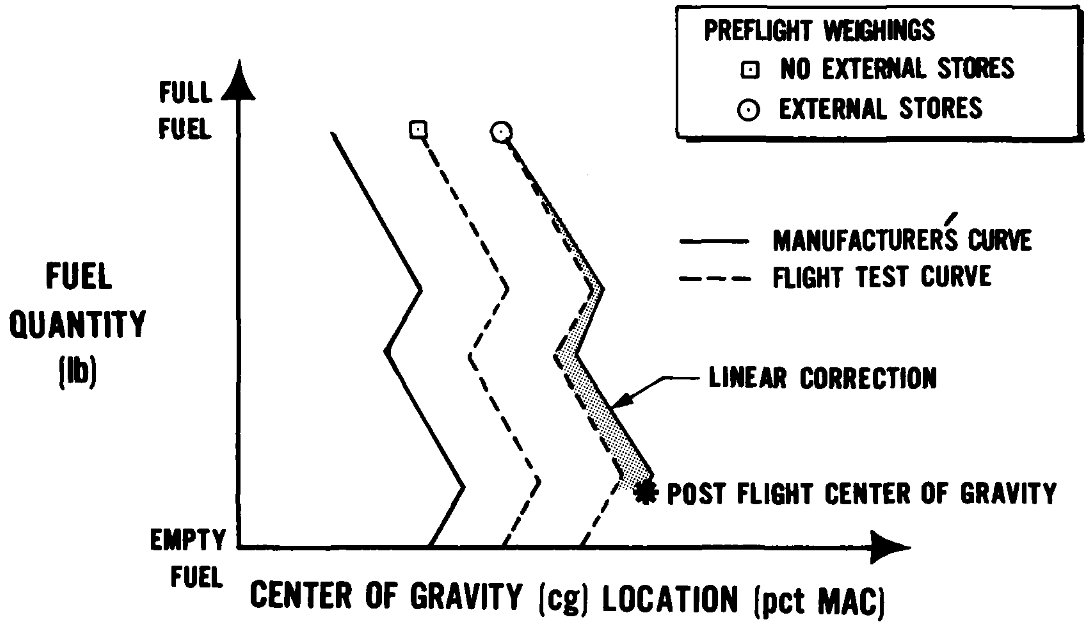


Figure 3 Airplane Center of Gravity Locations

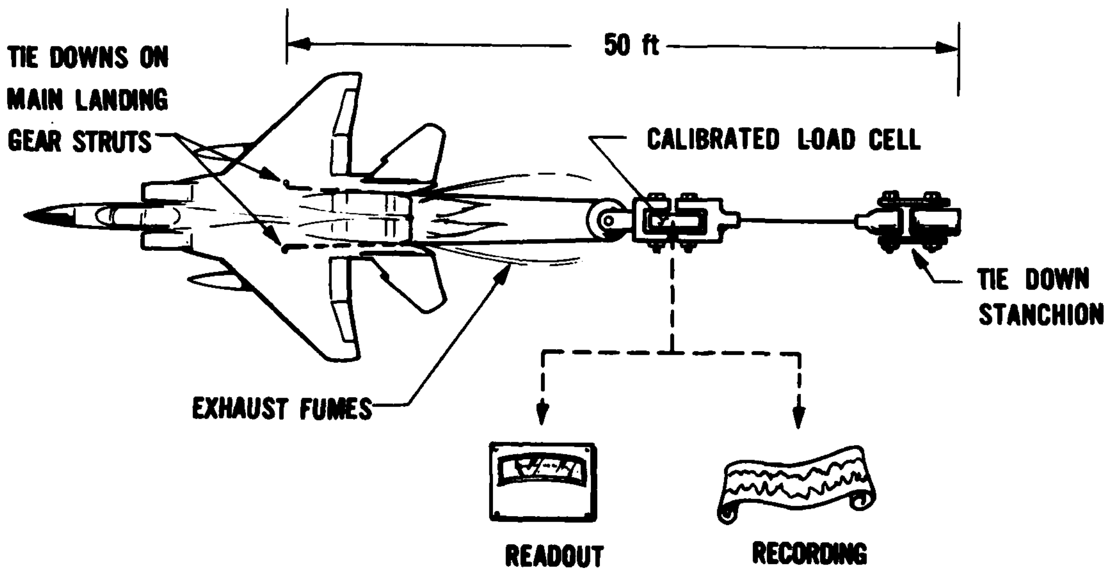


Figure 4 Simple Thrust Stand

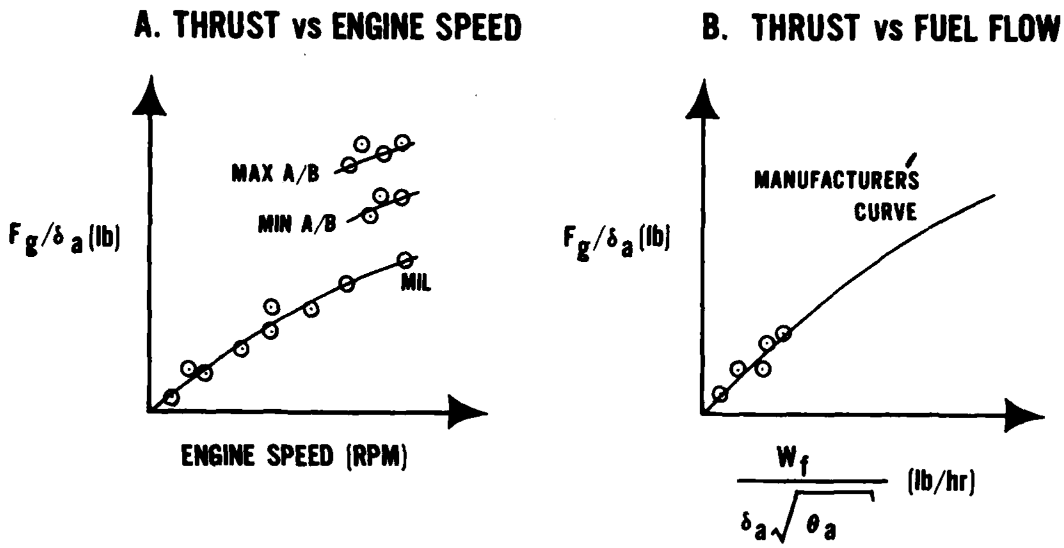


Figure 5 Static Installed Thrust Data

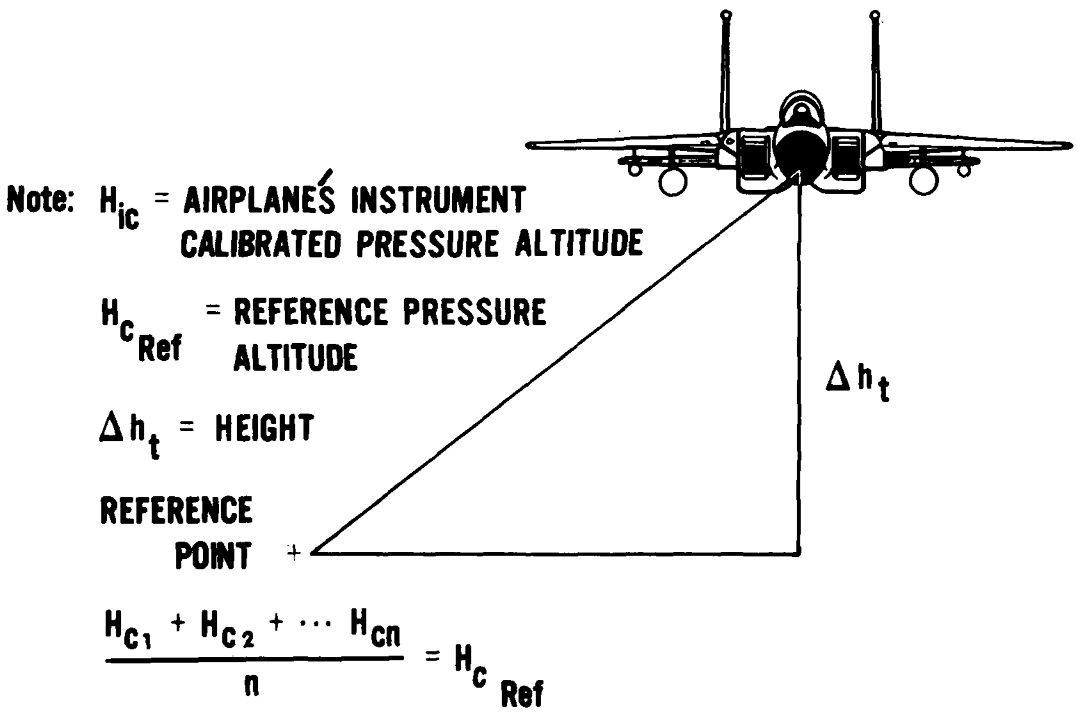


Figure 6 Ground Reference Flyby Variables

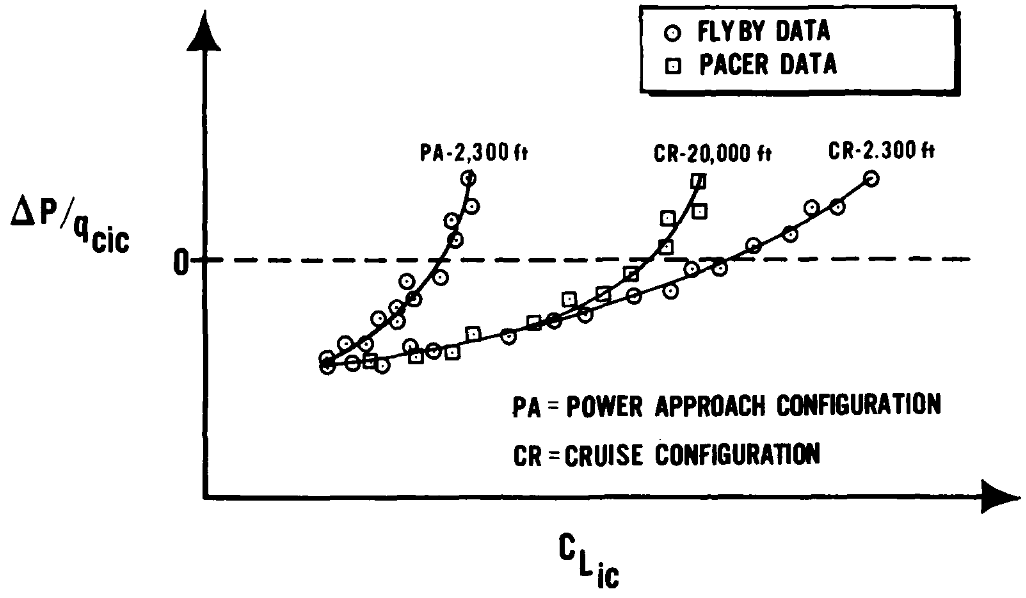


Figure 7 Static Source Position Error

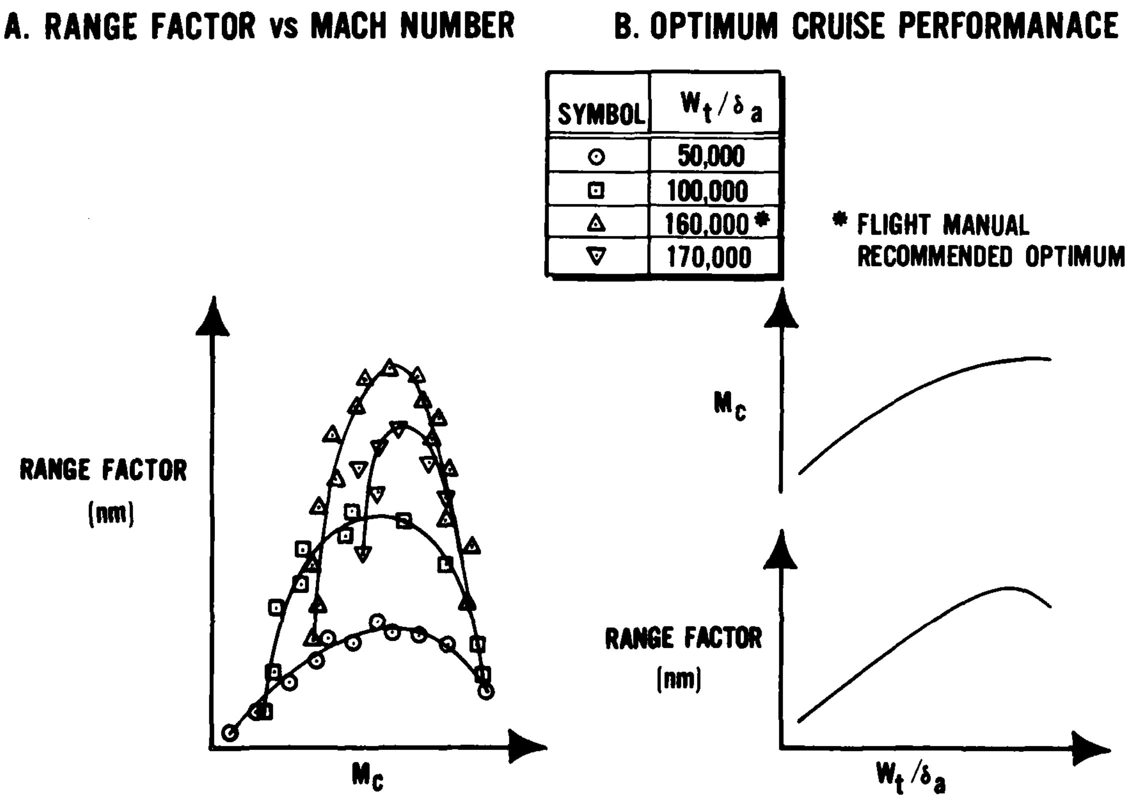


Figure 8 Cruise Performance

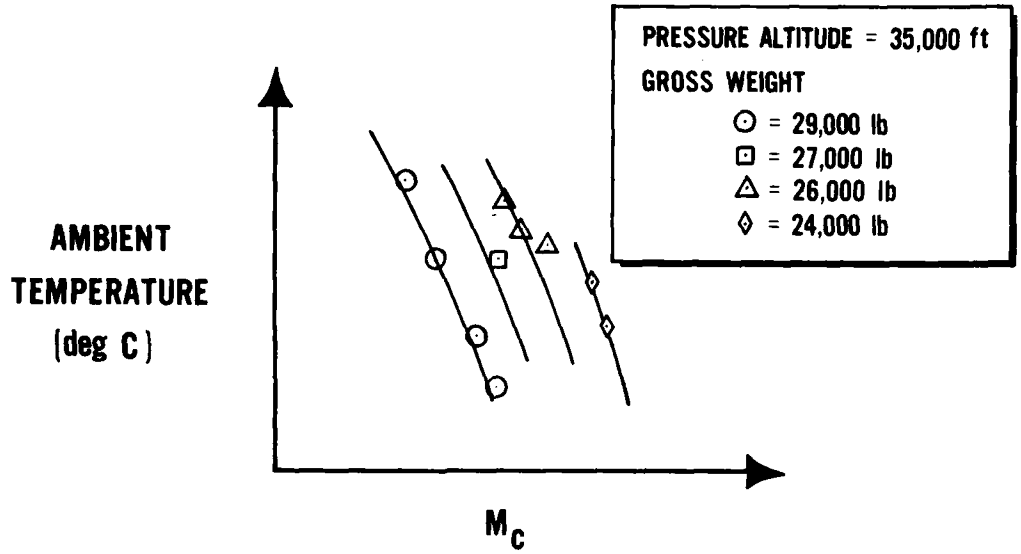
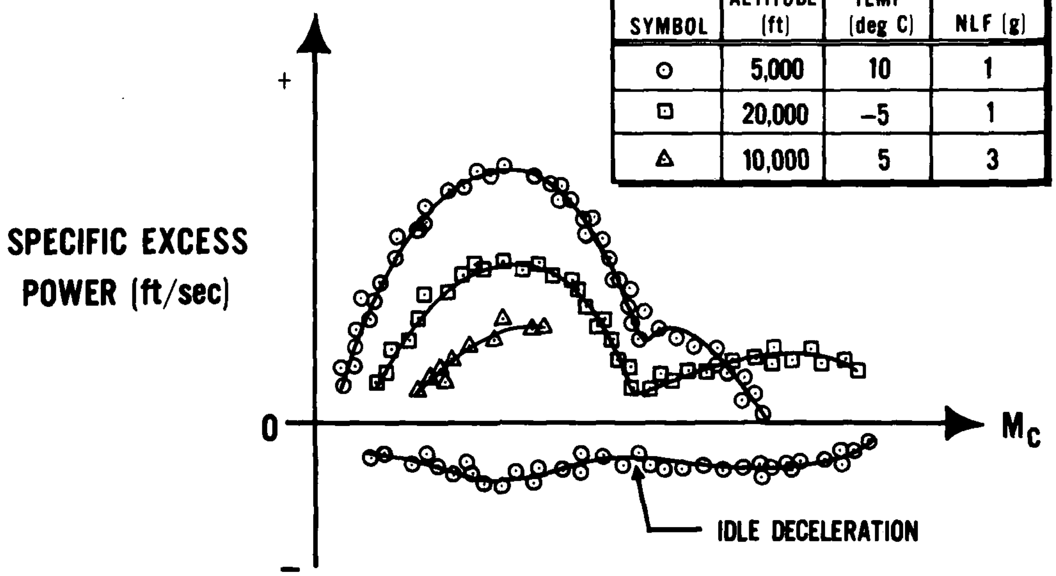


Figure 9 Maximum Speed Determination



SYMBOL	PRESSURE ALTITUDE (ft)	AMBIENT TEMP (deg C)	NLF (g)
⊙	5,000	10	1
⊡	20,000	–5	1
△	10,000	5	3

Figure 10 Specific Excess Power

SYMBOL	PRESSURE ALTITUDE (ft)	AMBIENT TEMP (deg C)
⊙	5,000	10
⊡	20,000	−5
△	30,000	−15

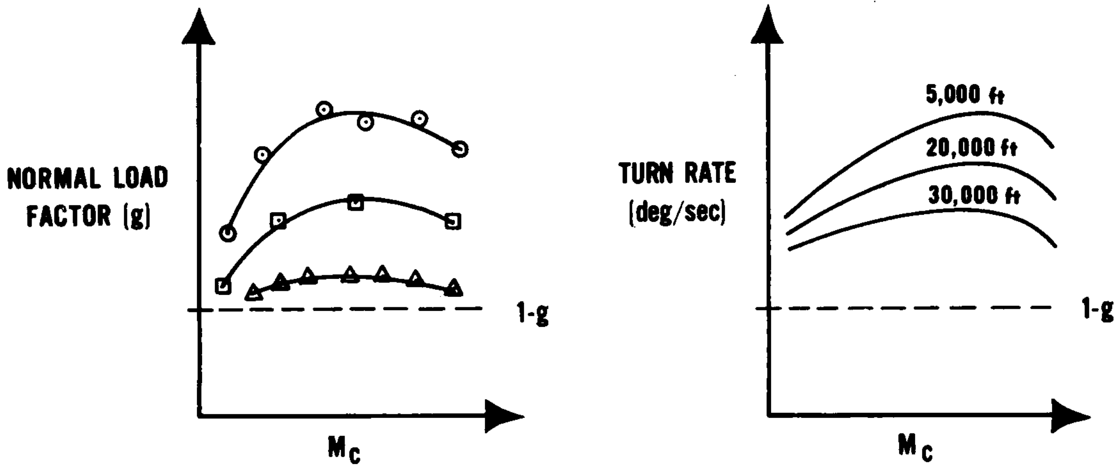
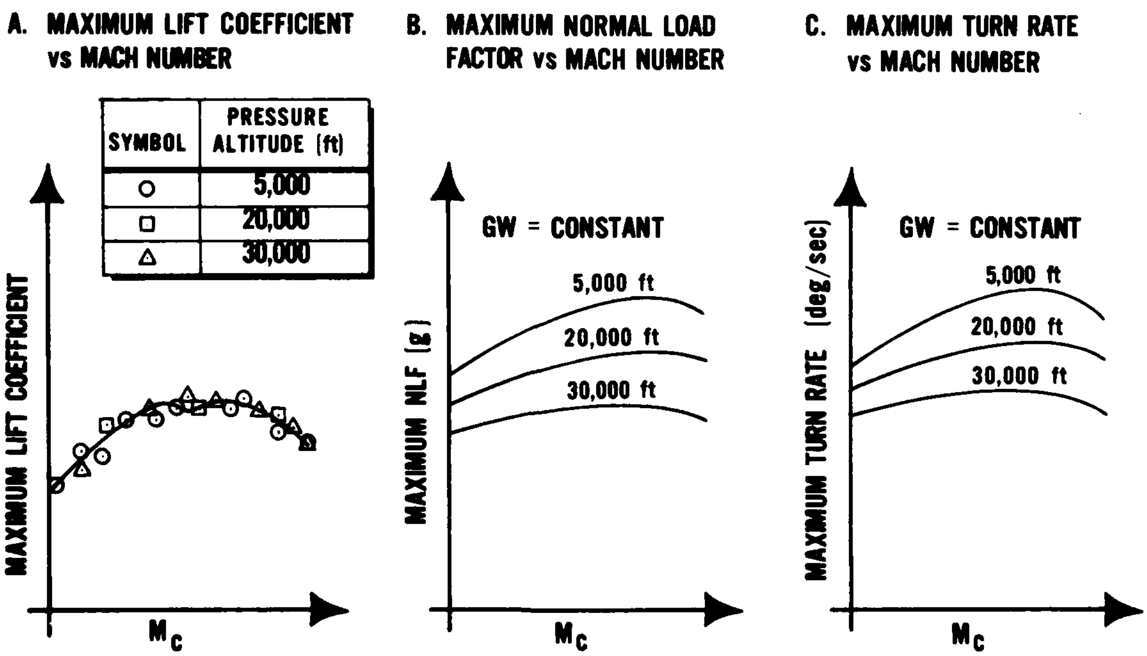


Figure 11 Thrust-Limited Turning Performance



SYMBOL	PRESSURE ALTITUDE (ft)
○	5,000
□	20,000
△	30,000

Figure 12 Lift-Limited Turning Performance

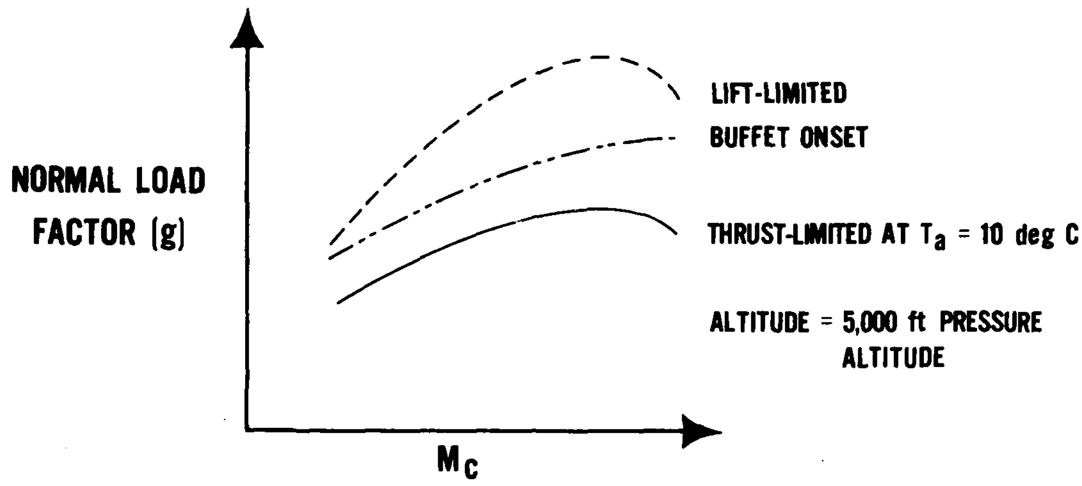


Figure 13 Turning Performanacc at 5,000 Feet

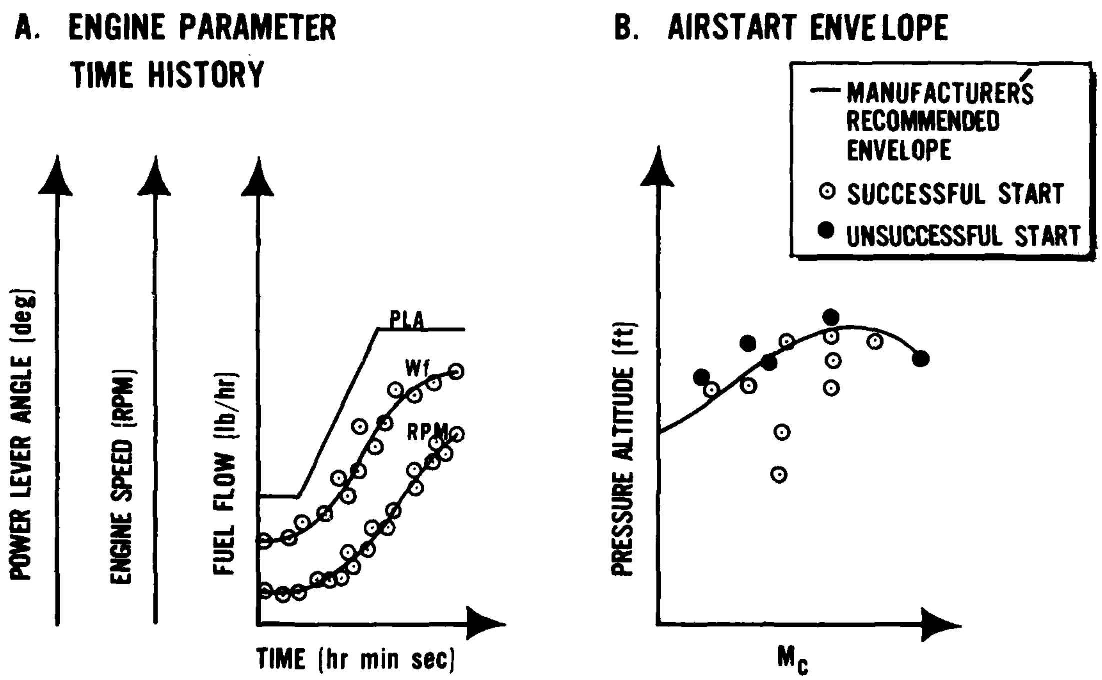


Figure 14 Engine Handling Characteristics

EVALUATION OF LONGITUDINAL CHARACTERISTICS INCLUDING STABILITY HANDLING AND CG RANGE

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SUMMARY

Tests used by contractors to assess the longitudinal stability and control characteristics of an aircraft are enumerated here to give an indication of the measured flight test data that can be made available to an evaluating body.

Simple flight tests to obtain qualitative and quantitative data on static, manoeuvre and dynamic stability, low speed handling, turning performance and high Mach No effects during an evaluation are also described herein

1. INTRODUCTION

Any evaluation of the longitudinal characteristics of an aircraft will depend on the class of aircraft (i.e. transport, fighter or bomber etc) and the type of control system fitted to it (i.e. power, manual or power with manual reversion)

For the purposes of the ensuing discussions the aircraft being evaluated will be assumed to be a highly manoeuvrable aircraft classified as belonging in class IV of References 1 and 2. Class IV includes fighter/interception, attack, tactical reconnaissance and advanced training aircraft.

The following discussion will also be limited to aircraft with powered control systems with artificial feel and hence stick forces generated by springs, bob weights and Q-feel pots. It is considered that manual control systems except for perhaps the rudder are no longer appropriate for this class of aircraft.

The handling qualities required of an aircraft are dependent on the role the aircraft is required to perform and this must be borne in mind when interpreting flight test data. For example a ground attack aircraft should possess sufficient stability to enable accurate target tracking in turbulence to be performed, where as an interceptor requires agility at the expense of some static stability.

A number of the flight tests that are described here are performed by contractors for the following reasons

a) To show that an aircraft meets specified stability and control requirements

b) To provide data to aid the design of future aircraft

c) To provide correlation data with wind tunnel results

d) To provide aerodynamic data for mathematical modelling of the aircraft's characteristics on simulators.

In addition the results of some of the tests are used to approve increasing increments of performance when approaching handling limitations in a progressive manner.

References 1 and 2 present levels of stability and handling characteristics required by Class IV aircraft in the various stages of flight and most aircraft designed in the U.S. and U.K. will meet most of these requirements. Occasionally these requirements will not be met but this is usually due to a design inovation which renders a requirement obsolete or not applicable, (e.g. stalling speeds of a vectored thrust VSTOL aircraft in semi-jetborne flight).

2. LONGITUDINAL STATIC STABILITY

Aircraft manufacturers measure the longitudinal stability of their products for the following reasons

a) To demonstrate that the aircraft satisfies any stability requirements specified in the requirements to which it has been designed

b) To determine the position of the stick fixed neutral point in order to establish a satisfactory aft CG limit for stable flight

c) To determine any degradation in static stability due to external stores.

A potential customer evaluating an aeroplane may use the results of static stability tests when comparing the stability of two designs.

The general theory of static stability is well documented in references 3, 4 & 5 and stability equations derived therein will be used in the following discussion of static stability.

If an aircraft returns to a trimmed condition after a disturbance in speed at constant incidence or incidence at constant speed it is said to possess longitudinal static stability. The static stability of the aircraft is related to the centre of gravity position to which the aircraft is loaded and if the c.g. is moved sufficiently far aft the stability reduces to zero. The c.g. position at which the stability is zero is called the neutral point and the stick position to trim is constant at all speeds.

If an aircraft has irreversible powered controls it's stability is independent of whether the pilot is holding the controls fixed or not and it is only necessary to consider the "stick fixed static stability". (When considering the stability of aircraft with non-powered controls the effect of a disturbance with the controls free must be considered since the aerodynamic balance of the control surfaces causes them to "float" at different positions depending on incidence and speed.)

If an aircraft is stable with respect to speed and incidence it is necessary to move the stick aft to trim at decreased speed and increased incidence, and forward at increased speed and decreased incidence. Aircraft with static instability can be flown by the human pilot but this condition induces high workloads which are not acceptable for tasks such as instrument flying for long periods. Slight static instability is sometimes accepted for example during the transonic period of an acceleration to supersonic speed.

Some recent combat aircraft such as the F16 are statically unstable and rely on the use of advanced electronic stability and control systems to [illegible]

is further complicated by the use of stick force rather than displacement to signal the pilot's intentions to the stability system.

An aircraft's longitudinal static stability is measured under one of the following conditions

a) In steady flight at constant altitude over a range of constant speeds with increased engine power setting at each speed increment

b) At constant engine power adjusting the rate of climb or descent to hold a range of steady speeds

c) Performing a steady acceleration at full power to maximum speed and a steady deceleration at idle power.

Engine power settings have a relatively small effect on the tailplane angle to trim unless the tailplane is immersed in the engine jet efflux like the Harrier. If the power setting effect is thought to be significant method b) with the aircraft in a glide at idle power should give the best estimate of the tailplane trim to balance the pure aerodynamic pitching moments on the aircraft.

Since most aircraft of Class IV are flown at power settings above idle most of the time method c) which in effect produces data at two power settings is more productive in terms of flying time.

The contractor will gather stability data with an aircraft carrying flight test instrumentation data recording equipment but useful results can be obtained from cockpit gauge readings when evaluating an uninstrumented machine.

At each flight condition of a) and b) or at set increments of IAS during acceleration/deceleration of method c) the pilot should record IAS, Fuel State, tailplane angle altitude and engine rpm.

After reading instrumentation records or applying cockpit gauge calibrations to the cockpit instrument readings a curve of tailplane angle to trim with IAS/IMN at which it was measured may be drawn. See Figure 1.

Examination of the shape of the tailplane angle to trim $\sim$ IAS(IMN) curve shown as Figure 1 will indicate whether the controls are moved in the conventional sense to trim with speed. Curve 1 is the result expected for a stable aircraft, curve 2 shows that at speed V_2 the aircraft is neutrally stable with respect to speed. Curve 3 is typical of the aircraft of curve 1 at a more aft C.G.

A quantitative assessment of the aircraft's static stability is obtained from the flight test results as follows.

The aircraft weight is computed from the fuel state and zero fuel weight and the C.G. position is determined from the aircraft load sheet on weight $\sim$c.g. diagram.

From the measured flight data the normal lift coefficient C_n is computed from the relation.

$$C_n = \frac{2W}{\rho V^2 S} = \frac{2W}{\gamma M^2 p S}$$

and is plotted with the corresponding tailplane angle to trim η_T (See Figure 2)

From the simple theory of static stability neglecting the effects of compressibility and airframe distortion these curves will be straight lines intersecting at the η_T axis but in the general case they may be similar to Figure 2.

In the simple case the slope of the straight lines and in the general case the local slope of the curves at any chosen CN is indicative of the stability of the aircraft. It can be seen that an aft c.g. is less stable than a forward c.g. (The slope of the $\eta_T \sim C_n$ curve is +ve for an unstable aircraft and zero for neutral stability when the c.g. is at the neutral point).

From the standard theory the aircraft is stable when the change of the overall aircraft pitching moment with the normal force coefficient, i.e. dC_m/dC_n is -ve . The slope of the $\eta_T \sim C_n$ curve is related to dC_m/dC_n as shown below.

$$dC_m/dC_n = \bar{V} A_1 d\eta_T/dC_n \qquad \text{where} \qquad \bar{V} = s'\ell'/S\bar{c}$$

A_1 = tailplane lift curve slope (a function of Mach No)
s' = tailplane area
ℓ' = moment arm between C.G. and tailplane aerodynamic centre
$\bar{c}$ = standard mean chord of the wing
S = area of the wing

Measurements of longitudinal static stability are normally made with the aircraft ballasted at two or more c.g. positions as far apart as considered safe. The c.g. positions tested should include the furthest practical aft and forward loadings.

At each c.g. the tailplane angles to trim and the corresponding normal force coefficients are plotted and the aircraft's neutral point or effective aerodynamic centre is determined by extrapolution at a range of values of C_n. (Note:- the neutral point is the c.g. position at which the stick movement to change speed is zero).

The c.g. margin or the difference between the measured neutral point and the test c.g.s can be determined and can be compared with any minimum requirements in the aircraft specification.

The aircraft's static margin is the measure of it's overall static stability and hence the control movement to change speed. In general the static margin is a function of speed and is proportional to the c.g. margin such that $K_n = \psi H_n$ where ψ is a constant of proportionality and a function of speed.

The results of the tests may be presented as a function of C_n as shown in Figure 3.

An example of the process of computing static margins is shown in appendix 1.

When evaluating an aircraft's longitudinal static stability it should be flown on the most aft c.g. of the production tolerance and fitted with the stores configuration that gives the most rearward increment in c.g. in conjunction with the most forward increment in aerodynamic centre position.

Some ground attack and air to air combat aircraft have very little static stability at moderate and high speeds and consequently exhibit very small trim changes while accelerating or decelerating during weapon delivery or combat. Adequate static stability is desirable for low speed tasks, instrument flying and very low level high speed flight.

3. LONGITUDINAL DYNAMIC STABILITY

When an aircraft is trimmed level flight is disturbed in pitch with the controls fixed it normally returns to steady flight in an oscillatory manner.

Standard theory in References 4 and 5 shows that a statically stable aircraft may exhibit two simultaneous damped oscillatory motions while returning to steady flight after a disturbance. These are:-

a) A high frequency heavily damped oscillatory motion (The short period pitching oscillation (S.P.P.O.))

b) A low frequency lightly damped oscillatory motion (The Phugoid).

References 1 and 2 lay down requirements for the damping of these modes and contractors measure the damping and period to demonstrate compliance.

An aircraft's damping in pitch is measured with the following test technique.

The aircraft is trimmed in straight and level flight at the required speed and altitude condition which should be noted by the pilot. The stick should then be deflected smartly forward and returned to the trim condition with the objective of achieving as near a triangular pulse stick input within 0.5 secs as possible. Immediately the stick is returned to the trimmed position it should be released and allowed to remain free until the resultant motion damps out. The above procedure should be repeated by deflecting the stick aft. The stick deflection in each case should be sufficient to give an increment of approximately $\pm 1g$ and the ailerons and rudder should be kept at the trimmed condition throughout the manoeuvre. An example of a high altitude relatively lightly clamped SPPO is shown in Fig 4.

The S.P.P.O. is normally damped out very quickly and is assumed in simple theory to be an oscillation in pitch at constant speed. If the test is continued beyond the point when the S.P.P.O. has ceased the characteristics of the Phugoid can be assessed. From simple theory the Phugoid oscillation is manifest by an exchange of potential and kinetic energy at constant incidence.

The S.P.P.O. is normally very well damped and is best recorded with a continuous trace recorder recording pitch rate and altitude and longitudinal control surface deflection. The Phugoid however can be assessed from cockpit gauge readings.

The S.P.P.O. test results are normally presented as plots of the periodic time and the number of cycles or the time to halve the amplitude of the oscillation against Mach No or IAS at each test altitude. See Figure 5 . Satisfactory damping can be qualitatively assessed by performing this test at high and low speed where the damping is most likely to be deficient if at all. Lack of adequate pitch damping in the transonic flight region is normally overcome by fitting an automatic pitch damper that senses the pitching acceleration and applies a suitable damping tailplane input independant of the pilot's stick.

The characteristics of the S.P.P.O. are dependent on the aircraft's pitching inertia, aerodynamic damping in pitch, and the static stability and is therefore related to the c.g. position. Aft movement of the c.g. normally reduces the damping in pitch and therefore the dynamic stability.

The Phugoid can be quantitatively assessed using a stopwatch once the S.P.P.O. has ceased since the period may be as great as 80 secs or more. The pilot should note the maximum and minimum speed and altitude and measure the time elapsed between instances when the initial trimmed conditions are achieved. If the maximum and minimum values of speed and height are compared over successive cycles the convergence or divergence of the oscillation can be determined. See Figure 6 for a typical Phugoid.

The greatest difficulty encountered during these assessments is that the results are dependent on the pilot's ability to return the stick to the initial condition following the disturbance. A small residual longitudinal trim error may make a stable Phugoid appear unstable.

4. LONGITUDINAL MANOEUVRE STABILITY

An aircraft's longitudinal manoeuvre stability is measured in order to determine the control deflections and hence the control forces that must be exerted by the pilot to manoeuvre the aircraft.

Contractors measure manoeuvre stability to demonstrate compliance with the requirements of Ref. 1 and 2 and to provide data to establish the practical c.g. range.

In order to manoeuvre a stable conventional aircraft the pilot is required to move the stick aft to increase the wing incidence and hence the normal acceleration.

The tailplane deflection required to produce an incremental change of normal acceleration is related to the aircraft's manoeuvre stability and may be expressed in terms of tailplane/g.

Typical Class IV aircraft with powered controls use springs, bob weights and pilot-static pressure difference systems to provide "feel" or stick force which increases with the normal acceleration demanded.

References 1 and 2 suggest maximum stick forces which the pilot can be expected to exert during various requirements of flight and U.K. and U.S. aircraft will normally satisfy their requirements. The longitudinal stick force should not exceed 50lb.

The aircraft is normally designed so that the stick force and stick force gradient required to achieve the maximum allowable normal acceleration with the aircraft trimmed for 1g flight is great enough to avoid the maximum normal acceleration being inadvertently exceeded.

A minimum stick force gradient is also set by References 1 and 2 to ensure that the aircraft is not so sensitive that the pilot will be over controlling in pitch when making small changes in incidence and flight path during target tracking and other precision tasks.

If the aircraft c.g. is moved far enough aft no change in the tailplane angle to trim or stick force is required to produce a change in normal acceleration. This position is termed the "stick fixed manoeuvre point" or h_m . The difference between any c.g. position at which the aircraft is flying and the manoeuvre point is a measure of the aircraft's manoeuvre stability and is termed the manoeuvre margin $H_m = h_m - h$.

The general theory of manoeuvre stability and the derivation of the equations describing the manoeuvre margin are presented in references 3, 4 and 5 and the results only are quoted here.

In manoeuvring flight the incidence of the wing is increased relative to a 1g flight condition at the same speed. The trimming effect of the tailplane is related to it's local incidence which is dependant on the increased down wash from the wing and a nose up pitching velocity resulting from the curved flight path of the manoeuvring aircraft.

As in the case of static stability the aircraft's overall pitching moment change with normal lift coefficient dC_m/dC_n must be -ve for manoeuvre stability.

The manoeuvre stability is quantified at c.g. h by $H_m = -\left(dC_m/dC_n\right) = h_m - h$ where h_m = the nondimensional position of the manoeuvre point.

From comparison of the pitching moment equations for an aircraft in manoeuvring and 1g flight it can be shown that the static and manoeuvre stabilitys are related by $h_m = h_n + \dfrac{\bar{V}A_1}{\mu_1 \ell}$

$$H_m = H_n + \frac{\bar{V}A_1}{\mu_1 \ell}$$

where $\mu_1 = \dfrac{W}{g\rho S\ell}$ = Relative Aircraft Density Parameter

W = aircraft weight
g = gravitation at constant
ρ = air density
S = wing area
l' = Distance between C.G. and aerodynamic centre

It should be noted that the stick fixed neutral point is normally ahead of the manoeuvre point and the manoeuvre margin is normally greater than the static margin.

The manoeuvre stability can be quantified from the following flight tests

a) Steady pullouts from dives at constant speed

b) Spiral turns with increasing normal acceleration at constant speed

The latter method is more productive since each manoeuvre produces a number of data points per test.

The spiral manoeuvre is most productive when performed with aircraft fitted with continuously recording flight test instrumentation but reasonable results can be obtained from cockpit gauges for normal accelerations up to 5g.

For the spiral manoeuvre the aircraft is trimmed to fly straight and level at the test altitude and target air speed and the the pilot should note altitude, air speed/Mach No, fuel state, tailplane, RPM and incidence. The aircraft is then placed in a turn with gradually increasing normal acceleration. If performing the test with cockpit gauges only the pilot should increase the normal acceleration in steps reading the gauges for incidence, tailplane angle and normal acceleration at each increment. As the normal acceleration is increased the lift dependent drag also increases and it is necessary to allow the aircraft to lose height in order to keep air speed constant. The addition of some power on entry to the turn allows the rate of descent to be reduced.

At high values of normal acceleration the aircraft may have descended significantly below the target test altitude and the manoeuvre should be repeated from a higher entry altitude so that the limiting normal acceleration is achieved at the target altitude.

It may not be possible to fly the aircraft steadily enough at the limiting normal acceleration to record a steady tailplane angle to trim from the cockpit gauges. The pilot should however be able to sense stick force lightening or pitch up and the resulting reduction of tailplane angle to trim although he may not be able to quantify the effect.

The spiral manoeuvres should be executed at a range of Mach No /IAS and altitudes and at two c.g. positions if it is intended to determine the stick fixed manoeuvre point.

From the aircraft fuel state the test weight and c.g. position are determined and the normal force coefficient is computed, i.e. $C_n = 2nW/\rho V^2S = 2nW/\gamma M^2PS$. The test results are then plotted as curves of tailplane to pull g, $\eta_T \sim$ normal force coefficient C_n. See Figure 7. At discreet values of C_n the local slope of the $\eta_T \sim C_n$ curve is computed and the resulting tailplane angle/C_n is plotted against the c.g. position and is extrapolated to the point where $d\eta_T/dC_n$ is zero to determine the manoeuvre point for the aircraft at the particular value of C_n. See Figure 8.

The position of the manoeuvre point at each speed and altitude can be determined as a function of CN.

The manoeuvre margin H_m at c.g. = h is computed as $H_m = h_m - h$.

The furthest aft c.g. to which the aircraft should be loaded and flown should be limited to a set distance in front of the most forward position of the manoeuvre point measured during flight trials and this minimum manoeuvre margin is normally not less than $0{\cdot}05\bar{c}$. The manoeuvre point is normally aft of the neutral point.

From the piloting point of view the stick force to manoeuvre the aircraft is of more significance and is more readily understood than the manoeuvre margin stick fixed. The latter is of value when comparing the stability of two aircraft quantitatively.

The stick force/g can be computed from the stick force and tailplane deflection with stick position calibrations at each increment of tailplane applied during the flight tests. The stick force movement from the trimmed 1g flight condition is plotted against normal acceleration at each test altitude c.g. and each speed/Mach No as shown on Figure 9.

The stick force/g for the example can be seen to reduce at high g but the aircraft has positive manoeuvre stability since it is always necessary to move the stick aft to increase g.

If an aircraft is to comply with Ref. 2 requirements for manoeuvre stability the slope of the curves of stick force/g must be measured at low g and the corresponding normal acceleration/unit of applied incidence (n/α) computed. The results are then plotted against the Ref. 2 requirements shown typically on Figure10.

The Ref. 2 requirements are dependent on the design maximum normal acceleration and are shown on Figure 10 for an aircraft with a designed maximum of +8g.

As described in Ref. 2 the aircraft's characteristics are required ideally to lie within level 1 but they may lie within level 3 where the pilot workload may be excessive or the mission effectiveness inadequate or both. Within level 3 manoeuvres such as air to air combat can be terminated successfully and take off, landing, cruise and climb etc. can be completed.

4. LOW SPEED HANDLING AND STALLING

Knowledge of an aircraft's stalling speed, stallwarning and it's handling characteristics at low speed are of great importance to the manufacturer and operator alike.

The minimum take off, approach and landing speeds are normally related to the measured stalling speed in the appropriate air frame configuration for these phases of flight with due regard to the stall-warning available.

The manufacturer aims to achieve the lowest stalling speeds possible because they govern the ground run distances for take off and landing and therefore the maximum payload that can be carried from a given run.

The contractor measures the stalling speeds in the cruise, take off, landing and approach configurations in order to furnish themselves, assessment authorities and customers with operating data. The results of the stalling tests are normally presented in the form of charts of stalling speed against aircraft weight for each of the air frame configurations as shown in Figure11.

Stalling speeds and incidences achieved at the stall are dependent on

a) Amount of aft stick travel available

b) The c.g. position

c) Aircraft flap and undercarriage positions

d) External stores configurations

e) Rate of aft stick movement and hence rate of deceleration

Traditionally warning of an impending stall is given by the buffet generated by air flow separation from the wing magnified by the separated flow impinging on the tailplane. This warning is termed buffet warning or stall warning and is quantified as the buffet margin, i.e. Margin = $V_{BUFFET\ ONSET} - V_{STALL}$

Typically the buffet margin should not be less than 5 knots.

Some aircraft designs do not produce sufficiently prominent buffet warning and an artificial stall warning system is provided to alert the pilot. The artificial warning may be audio or tactile and is likely to be engineered to be a function of the rate of approach to the stall.

It is desirable that the air flow separation on the wing at the stall should develop from inboard to prevent wing drop and therefore reduce height loss on recovery. In addition pitch up at the stall will not occur if wing tip stalling is avoided on swept wing aircraft.

If a wing tip stall and hence pitch up cannot be avoided the aircraft's control system may include a stick pusher that will push the stick forward automatically if the pilot exceeds the stall warning incidence by more than a tolerable level. Such a device will prevent the aircraft achieving a condition in which a high tail aircraft cannot recover because the tailplane is stalled in the wake of the wing and fuselage.

The maximum incidence that the aircraft can achieve is sometimes limited by the aft stick travel alone but the aircraft's manoeuvrability may be impaired if this is done.

Unlike slow speed aircraft with straight wings those of category IV may not pitch down at the stall but may develop a divergent dutch roll motion or pitching oscillation while in a high rate of descent. Such oscillations should cease immediately when the stick is returned to the trimmed position.

When U.K. contractors perform stalling trials they are required to equip the aircraft with an anti-spin parachute or recovery rockets as specified in Reference 1.

Should the aircraft depart at the stall and enter a developed flat spin these devices will aid recovery by producing an anti-spin pitching moment and reducing incidence.

U.K. contractors are required to examine an aircraft's stalling behaviour from straight and turning flight and to demonstrate that the stall can be averted without significant height loss on receipt of the stall warning. This examination should be performed at the extreme forward and aft c.g. positions at which the aircraft is intended to operate.

Additional stalling tests should be made with all store configurations to be carried in order to determine the effect of stores on the stalling speeds.

The contractor is also required to assess and report the aircraft's handling characteristics in straight flight and in 30° banked turns at speeds close to the stall.

The following techniques are typical of those used for the examination of the aircraft's stalling characteristics and speeds in straight and turning flight

STRAIGHT FLIGHT

The aircraft should be flown at 1.4 X the expected stalling speed with the control forces trimmed out and the minimum engine thrust set to allow a sufficiently slow deceleration to be achieved. The aircraft should then be decelerated at a rate of less than 1Kt/second in level flight by slow aft stick movement until stick hard back or some uncontrollable pitching rolling or yawing motion occurs.

Every effort should be made to stop excursions in roll and yaw occuring with rudder and aileron and the pilot should make an assessment of the amount of control required to arrest any excursion before stick hard back is reached.

The pilot should note fuel state, configuration and altitude at the initial trim speed, buffet onset speed and minimum speed achieved. The minimum speed will normally be achieved with the aircraft in a descent and the normal acceleration (less than +1.0) should be recorded to correct the stalling speed.

ie $V_{STALL} = V_{MIN}/\sqrt{n}$

If the aircraft is fitted with incidence or airstream direction detector gauges, their readings at buffet onset and minimum air speed should be recorded. Some pilots fly incidence rather than speed during circuit flying and are therefore interested in the buffet margin in terms of incidence.

TURNING FLIGHT

With all the control forces trimmed out and the engine power for level flight at 1.3 X the expected stall speed in the turn the aircraft should be put into a turn at a target angle of bank. Once established at the target bank angle the speed should be reduced by aft stick movement until the aircraft stalls. The test should be repeated at increasing angle of bank up to that corresponding 0.4 x maximum design normal acceleration, e.g. if maximum design g is +8 then AOB = $\sec^{-1}$ 0.4 x 8.0 = 72°

The pilot is required to record the same data as on the straight stalls and to comment on the following

a) The stall warning

b) Whether the stick was pulled to the aft stop before the stall occurred

c) The behaviour of the aircraft at the stall and the severity and extent of any wing drop

d) The method of recovery and the height lost from the stall to recovery to level flight

On both straight and turning stalls the pilot is required to determine whether the stall can be averted if recovery action is taken at the receipt of the stall warning.

BEHAVIOUR NEAR THE STALL

Once the aircraft's stalling speed has been established it should be determined if it can be trimmed to fly laterally level at constant speed at 1.2 and 1.1VS with hands and feet off. If not, the amount of control required should be recorded.

LOW SPEED TURNS

After measuring the turning stalling speed the pilot should determine the minimum speed at which sustained 30° AOB turns can be made and the conditions which impose the limit. The pilot should assess the amount of control required to sustain the turn and whether any loss of control effectiveness is exhibited.

A U.K. contractor supplying aircraft to the U.K. Armed Services will have been required to perform most of the previously mentioned tests. The results recorded from the tests will be presented in a similar manner to Figure 12.

In general any evaluation of the low speed handling characteristics of an aircraft should include stalls to check stall speed and buffet margins claimed by the contractors in the operational flap/undercarriage configurations both on forward and aft c.g. positions.

In addition the adequacy of the buffet margin and the aircraft's handling after receipt of stall warning and before the stall should be assessed in case the aircraft does not respond to recovery action with the necessary rapidity.

Finally the effect on the stalling speeds and buffet margins of under wing stores should be checked during the evaluation of the aircraft.

The standard stalling tests do not use much flying time and reliable data is easily gained from the cockpit gauges.

5. TURNING PERFORMANCE MEASUREMENT

A combat aircraft's instantaneous turning performance is of vital importance and is dependant on the maximum normal lift coefficient that it can achieve. The limiting normal acceleration or the incidence at which it is achieved is dependent on height and Mach No and can be presented as a manoeuvre boundary curve in terms of achievable incidence v Mach No or IAS at each altitude tested, e.g. Figure 13

The minimum achieveable radius of turn is attained at the maximum useable lift coefficient for the aircraft since it's lateral/directional characteristics may not allow the maximum lift coefficient to be generated with the pilot retaining control.

The manoeuvre boundary is found from the limiting conditions of the spiral manoeuvres flown to determine the manoeuvre margin (See Section 4).

The aircraft should be placed in a spiral up to the maximum normal acceleration achievable turn and engine power increased as required to maintain speed as constant as possible and to minimize height loss.

The pilot should note fuel state at the start of the manoeuvre and the normal acceleration, incidence, IAS or IMN, and altitude on receipt of any buffet warning and at maximum normal acceleration.

The test results are then presented as Figure 13 and are readily interpreted.

The aircraft's manoeuvre boundary may be defined by the following conditions

a) Full aft stick movement

b) Achievement of the aircraft's structural maximum allowable normal acceleration

c) The onset of lateral/directional oscillatory motions which the pilot cannot control and which may diverge rapidly enough to cause failure of the fuselage or fin

d) Very heavy buffet leading to reduced fatigue life

e) Build-up of excessive lateral or directional out of trim.

The pilot should qualitatively assess which of the above factors limited the maximum achieved normal acceleration and incidence and the amount of buffet warning received before departure from controlled flight.

If an aircraft's turning performance is limited by the build up of some form of oscillatory motion it may be fitted with a form of stability augmentation system. For example, an aircraft may develop an uncontrollable wing rock at high incidence which could be tamed by a roll stabilizer and hence allow the pilot to retain control until a greater incidence, and therefore more lift, is achieved.

Some aircraft are fitted with control systems that limit the normal acceleration and incidence achievable at any flight condition automatically. Incidence and normal acceleration are accurately sensed and a computer restricts the control deflections to prevent the aircraft from reaching conditions where a departure from controlled flight can occur.

The results of the manoeuvre boundary tests can be refined to produce more interesting results than just an incidence Mach No curve at a given altitude as follows.

A typical plot of the maximum normal acceleration achieved at one weight/c.g. position and altitude with Mach No is shown as Figure 14 . It should be noted that the maximum normal acceleration that can be achieved at a given flight condition (IMN and Altitude) is dependent on the aircraft weight and a more general result is of greater value.

The maximum normal acceleration that can be achieved at any condition is a function of the maximum lift coefficient of the wing which is also a function of Mach No as is illustrated in Figures 15 & 16 .

The maximum normal acceleration can be expressed as $\hat{n} = L/W$ where L = lift in the turn & W = Weight

In non-dimensional form $\hat{n} = \gamma P M^2 S C_{LMAX}/2W$ where C_{LMAX} = Max lift coefficient, M = True Mach No
γ = Ratio of specific heats for dry air
Po = Sea level standard pressure

Since δ Po and S are constants and C_{LMAX} can be assumed to be a function Mach No only in general $\hat{n}W/\delta = f(M)$. If $\hat{n}W/\delta$ is computed from the measured flight test results and is plotted against Mach No the resulting unique curve will be independent of the test height and aircraft weight (Neglecting Reynolds No effects) See Figure 16 .

If the normal acceleration at which buffet was recorded is also treated in the same manner a buffet boundary independant of height and weight may also be presented. See Figure 16 .

Having measured the maximum available g at any one height the maximum available g can be predicted for any other height. Assuming constant weight at a particular Mach No (and hence C_{LMAX}) the maximum g at the 2nd altitude is given by $\hat{n}_2 = \hat{n}_1 \delta_2/\delta_1$. The results of the test example figure are extrapolated to other altitudes. See Figure 17. The above technique for extrapolating the instantaneous turning performance is not strictly valid when Reynolds No effects are significant. To overcome the effects of Reynolds No the tests should be conducted over a range of altitudes and the extrapolation confined to small increments of altitude.

The ability of a combat aircraft to perform it's allotted task is critically dependant on the turning performance it can achieve and also by it's handling qualities.

In an operational environment the pilot will be required to divide his attention between the inside and outside of the cockpit and if he is manoeuvring the aircraft it is desirable that he has some natural indication that he is close to limiting flight conditions. This is especially important when the limiting condition is a sudden departure from controlled flight rather than the aircraft's refusal to generate more lift because the stick is on the aft stop.

Any evaluation of a combat aircraft should include both qualitative and quantitative assessment of the manoeuvre boundary. The former is necessary because the manufacturer may be optimistic in describing the limiting handling qualities and the latter is desirable since the results can be used to test the validity of the contractors data.

The carriage of external stores will affect the manoeuvre boundary in the following ways:-

a) By changing the value of $C_{L_{MAX}}$ for the aircraft

b) By altering the aircraft's handling qualities which may be severely degraded in some instances i.e. producing early onset of uncontrollable wing rock

c) By changing the centre of gravity position on the aircraft

In general if the c.g. is moved forward the tailplane carries a greater down load at high incidence and therefore the normal acceleration achievable at a given weight is reduced.

6. EFFECTS OF MACH NO ON THE LONGITUDINAL STABILITY AND CONTROL

Aircraft designed to fly at high subsonic, transonic and supersonic speeds experience trim changes and control effectiveness variations due to the rearrangement of the air pressure distribution on the air frame resulting from the formation of shock waves.

Contractors are required to show that the aircraft remains controllable through the transonic speed range and in supersonic flight. Some relaxation of the static stability with respect to speed in the transonic region is however tolerated.

An aircraft's behaviour in acceleration during high speed dives and the trim changes associated with recovery must be demonstrated to be innocuous for operational aircraft flown by average pilots.

Reference 1 and 2 lay down desirable control force limits for recovery from high speed dives.

As the flight Mach No is increased towards Mach 1.0 shock waves begin to form on the aircraft where the local air pressure is lowest and the local Mach No has first become unity.

The shock waves build in strength as the aircraft speed reaches Mach 1 and they cause the pressure distribution on the wing to change such that the centre of pressure moves aft. The overall aerodynamic centre migrates from approximately $\frac{1}{4}$ SMC at low speed to $\frac{1}{2}$ SMC when the aircraft is supersonic.

The aft aerodynamic centre movement causes a nose down trim change relative to subsonic conditions when the aircraft is truly supersonic.

In the transonic region the aircraft may experience trim changes in any direction about all three axes which may require unorthodox control movements to trim. Any disturbance in pitch especially, but also in roll and yaw, may cause the air flow which has separated because of the shock waves to reattach and a further separation to occur elsewhere on the air frame. In re-trimming the aircraft at transonic speed the pilot may cause a similar variation in shock wave distribution.

At low speeds the change in pitching moments and lift forces with speed are neglected and static stability is limited to the incidence effect at constant speed, i.e. the static margin and c.g. margins are numerically equal.

At high speed the effect of speed on the moments becomes significant and the pitching moment with forward speed derivative Mu, which is normally destabilizing, may become sufficiently large to cause the direction of stick movement to trim to reverse as speed increased.

Under these conditions the static margin Kn is assumed to be made up of two components

i) Contribution to the static margin due to incidence - dC_m/dC_L

ii) Contribution due to speed $= (M/2C_L)dC_m/dM$ where dC_m/dM is proportional to $\frac{dC_m}{dV}$ or Mu

It can be seen that an aircraft that is statically unstable may be stable with respect to incidence disturbances whilst being unstable with respect to disturbances in speed at one and the same time. While this phenomena is not very satisfactory an aircraft whose instability is small with respect to speed in the transonic region may be regarded as satisfactory provided it is statically stable at low speeds.

On some aircraft the transonic trim changes are automatically reduced by a mach trimmer, a device which applies a tailplane input independent of stick movement as a function of Mach No to check the transonic trim change. Use of a mach trimmer enables the designer to keep the pilot's stick input direction in the conventional sense as the aircraft is accelerated through the transsonic speed range.

Contractors flight trials include flight at speeds up to the design maximum and to the design maximum normal acceleration and the envelope explored is presented typically as Figure 18.

The aircraft may not be controllable at the limits of this envelope however, and any evaluation of an aircraft by a potential customer should include a flight along the boundary of the envelope paying attention to the aircraft's handling. On such an assessment the pilot should be on the look out for signs of pitch up, pitching oscillations that are difficult to damp out, wing drop or yawing disturbances.

Should any gross trim changes that are difficult to control be encountered it is important to consider whether they will prevent the pilot satisfactorily completing the mission for which the aircraft is intended.

For example, a highly manoeuvrable trainer may drop a wing at a particular Mach No above the maximum level Mach No during a transonic dive. Such an occurrence will not prevent a student completing the acceleration to supersonic speed if he can pick up the wing using say $\frac{1}{4}$ aileron and therefore the wing drop can be considered insignificant.

On the other hand if the trim changes on the pullout from the dive result in a pitch up such that the pilot can only just avoid exceeding the maximum normal acceleration at the speed the pitch up occurs the handling characteristics should be judged as unacceptable.

7. TARGET TRACKING

Aircraft designed for air to air combat and ground attack should exhibit good target tracking characteristics since their operational effectiveness can be compromised if the pilot cannot track targets accurately.

An aircraft's target tracking performance is related to the following factors

a) Longitudinal static stability

b) Longitudinal manoeuvre stability

c) Longitudinal dynamic stability

d) Harmony of longitudinal and lateral controls

e) Sensitivity of the longitudinal control system

The contractor will assess the aircraft's target tracking performance during the test programme and may modify the control circuit gearing to increase or decrease the control sensitivity or the harmony of the longitudinal and lateral controls.

This assessment is largely qualitative and subjective since different pilots exhibit different levels of skill in target tracking. Quantitative data for the assessment of target tracking accuracy can be obtained from gun and weapon aiming sight recording camera results.

Any potential procurer evaluating such an aircraft should examine the target tracking in the operational role and carrying typical external store loads. This assessment ideally should occur in both turbulent and non-turbulent weather conditions and the pilots display recorder should be run throughout each attack.

The acceptability of an aircraft's target tracking behaviour depends on the weapons the aircraft is required to deliver. For example, a degraded target tracking ability may be satisfactory when delivering missiles with wide "look" or target acquisition angles but not when firing guns or unguided missiles.

Aircraft intended primarily for ground attack can usefully be more stable than for air to air combat since they are often required to operate in turbulent conditions which will make target tracking harder for the pilot. A ground attack manoeuvre is essentially a 1g manoeuvre in the tracking phase so the manoeuvre stability is not so important as in air to air combat.

GROUND ATTACK TARGET TRACKING

The pilot should carry out simulated ground attack manoeuvres over a range of speeds and dive angles (related to the type of weapon to be used).

If the gunsight has a fixed cross the pilot should track the target with the fixed cross, since disturbances in the aircraft altitude related to aircraft stability will show in the displacement of the target from the cross.

From the gunsight camera records the displacement of the cross from the target in pitch and azimuth should be measured and the following data deduced.

The control system performance is good and the controls well harmonised if the tracking errors in azimuth and elevation are small and of similar magnitude and the pilot reports equal workload in azimuth and elevation.

If there is a great difference between elevation and azimuth the harmonisation of longitudinal and lateral controls may be responsible and control surface/stick movement gearings may be modified

If either elevation or azimuth tracking exhibits an oscillatory error of similar frequency to the aircraft's longitudinal S.P.P.O. or directional (Dutch Roll) oscillation then it is possible that the aerodynamics damping in pitch or yaw is not adequate. Deficiency in these areas may be reduced with the incorporation of pitch or yaw dampers.

If the aircraft exhibits any oscillatory tracking problem this will be accentuated with destabilizing external stores and more attention should be paid to target tracking in these configurations.

If the aircraft's static stability with respect to incidence is too low then the pilot will have to work harder in tracking the target in pitch. Low static stability with respect to speed however is an advantage when target tracking since the trim changes associated with speed changes are less significant and annoying.

AIR TO AIR TARGET TRACKING

Air to air target tracking is assessed by tracking a target aircraft orbiting at steady increments of normal acceleration up to the maximum sustained value.

The gunsight camera film is analysed as for ground attack manoeuvres and similar observations can be made.

In air to air target tracking the aircraft's manoeuvring stability and stick force/g assume greater proportions than in ground attack. Any errors in tracking at high g may be related to the basic manoeuvre stability of the aircraft. The stick force/g to correct tracking errors should not be too large for the pilot to smoothly adjust g and not so low that he is likely to overcontrol and induce oscillations.

The ability of the pilot to track a target at increasing normal acceleration should also be examined up to the maximum normal acceleration that can be obtained.

8. CONSIDERATIONS ARISING FROM RELAXED STABILITY

Some recent combat aircraft have been designed with -ve static stability and have advanced control systems to translate the pilot's control demands into unconventional control surface movements to trim.

The advantages occurring from relaxed stability are:-

a) Reduced trim drag and therefore increased range or speed

b) Enhanced turning performance

c) Reduced landing and take off speeds due to generation of more lift.

A statically stable aircraft as discussed in Section 2 normally carries a down load on the tailplane during the cruise. Under these conditions the wing must develope enough lift to balance the weight and the tailplane down load. An aircraft with negative longitudinal static stability carries an up load on the tailplane and so at a given weight the wing is required to produce less lift. It follows that at a given weight the aircraft's induced drag and therefore overall drag is reduced.

Turning performance is enhanced from reduced induced drag for a given weight.

Landing and take off runs are reduced from reduced drag and increased lift.

Unlike the stable aircraft discussed in Section 2 an aircraft with negative static stability will exhibit the following characteristics

a) The tailplane angle to trim curve with speed will have a negative slope over most of the speed range

b) A tailplane angle to trim with lift coefficient will have a +ve slope

c) The manoeuvre stability curve of tailplane angle/increment in normal acceleration is unchanged

The aircraft's control system is engineered so that if the pilot wishes to trim the aircraft at higher speed he must move the stick forward and apply a forward force just as if the aircraft were longitudinally statically stable.

When evaluating a relaxed stability aircraft the pilot should assess whether the controls move in the sense required of a stable aircraft and should determine how much of the control surface deflection range is used during normal flight. In addition he should determine whether the control forces can be trimmed out using the trimmers in all normal phases of flight.

9. CONCLUDING REMARKS

A pilot evaluating the longitudinal stability and control characteristics of an aeroplane can easily obtain quantitative data from cockpit gauges on the following topics

a) Stalling speeds and stall warning margins and trimming demands at low speed

b) Manoeuvring performance and maximum useable normal acceleration and incidence boundaries

c) Longitudinal static stability, manoeuvre stability and the acceptability of the C.G. range

d) Trimming demands in high speed flight

e) Effect of external stores on the above

Qualitative data can be readily obtained for

a) Dynamic stability

b) Manoeuvre stability

c) Controllability and stability in high speed flight

d) Target tracking performance

An evaluation of an aircraft should wherever possible begin with a thorough assessment of the operational requirements of the replacement candidate.

REFERENCES

1 Design Requirements for Aircraft for the Royal Air Force and Royal Navy, Ministry of Defence (P.E.) Volume 1 Books I & II and Memos

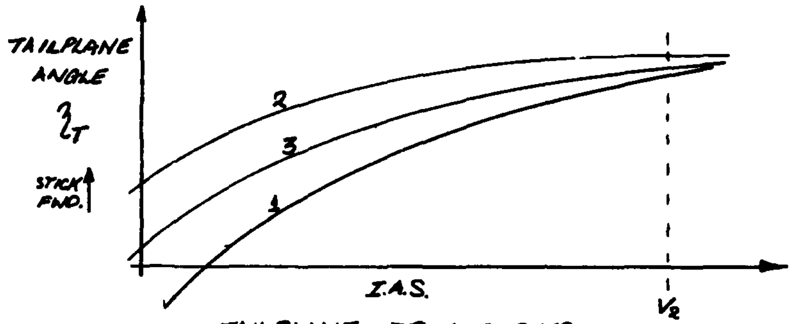
2 Military Specification Flying Qualities of Piloted Airplanes, MIL-F-8785B (ASG) 16 September 1974

3 AGARD Flight Test Manual Vol II, "Stability and Control"

4 A. W. Babister, Aircraft Stability and Control, Oxford, Pengamon Press, 1961

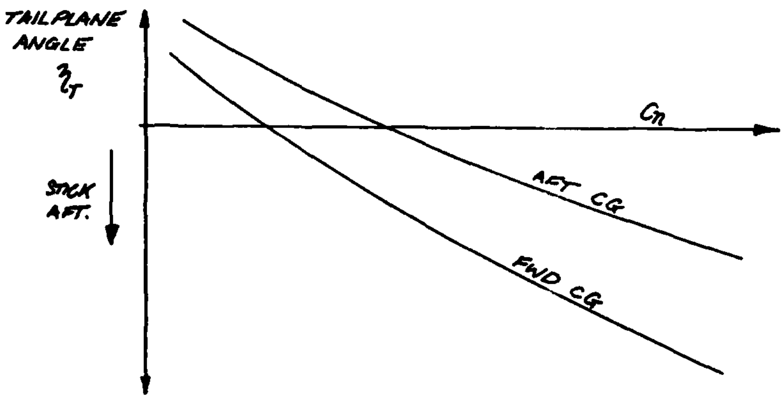
5 B. Dickinson, Aircraft Stability and Control for Pilots and Engineers, London, Sir Isaac Pitman & Sons Ltd, 1968

FIGURE 1



TAILPLANE TRIM CURVES

FIGURE 2



TAILPLANE ~ NORMAL FORCE COEFFICIENT FOR TRIMMED FLIGHT

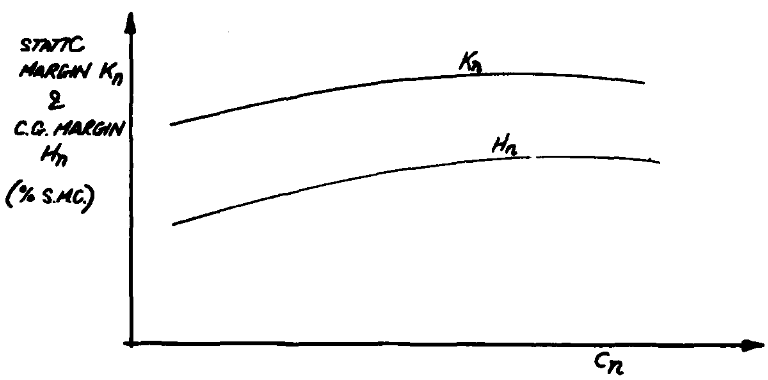


FIGURE 3

STATIC & C.G. MARGINS

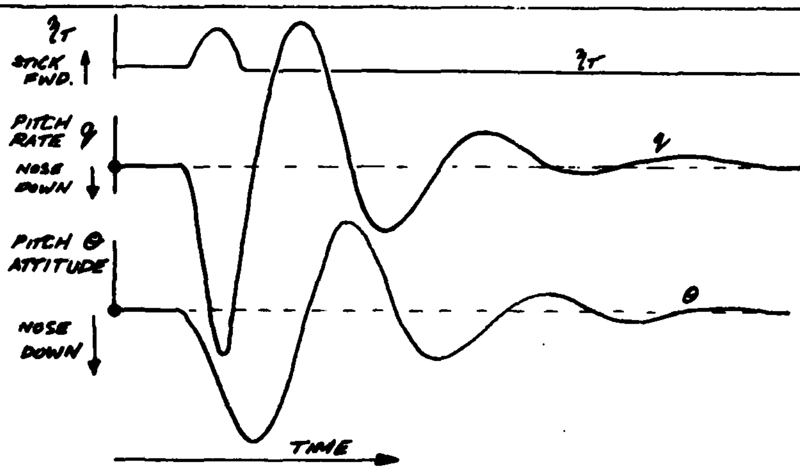
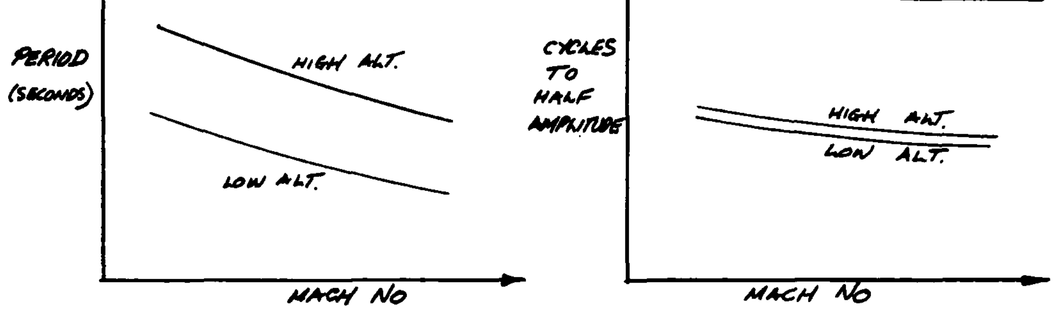


FIGURE 4

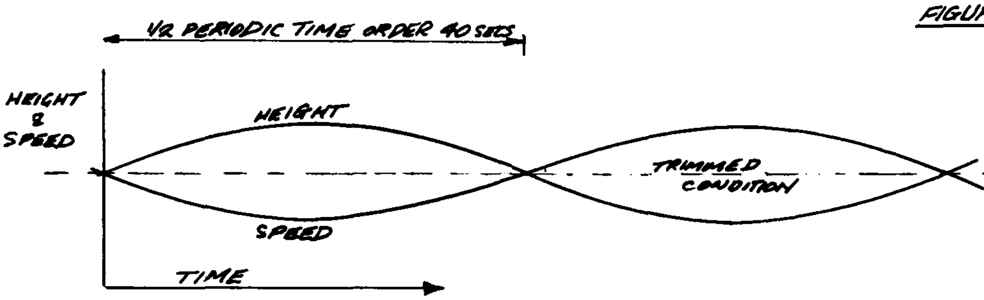
S.P.P.O. TIME HISTORY

FIGURE 5



S.P.P.O PERIOD & DAMPING

FIGURE 6



PHUGOID OSCILLATION

FIGURE 7

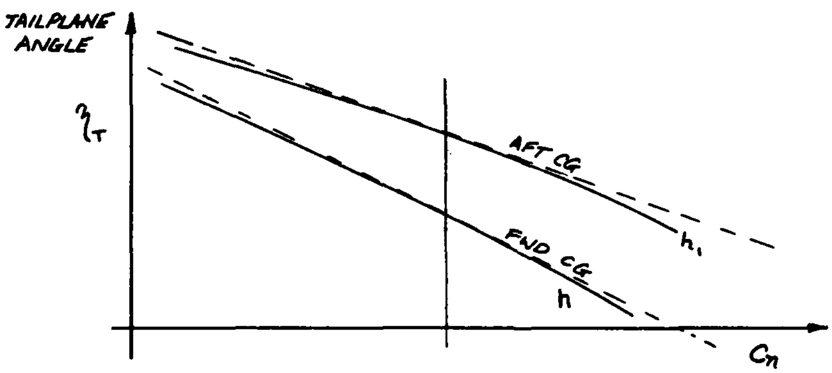
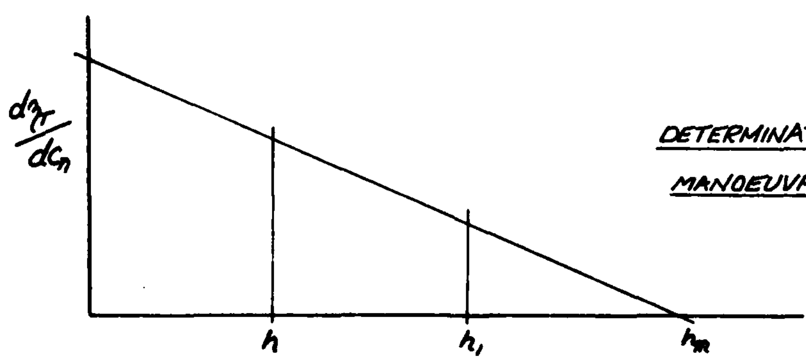


FIGURE 8



DETERMINATION OF THE MANOEUVRE POINT h_m

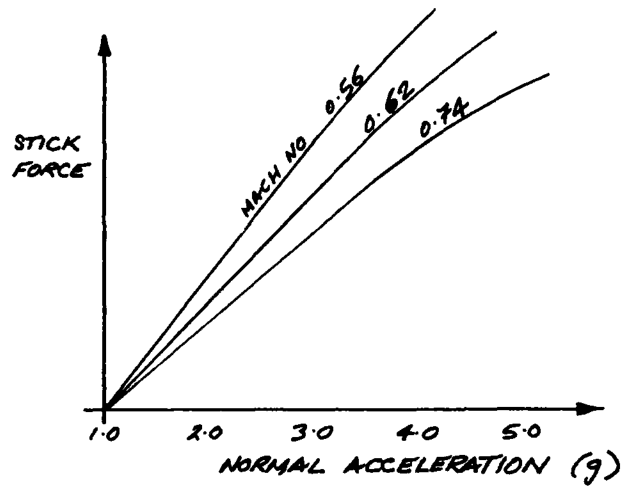


FIGURE 9

EXAMPLE OF STICK FORCE ~ NORMAL ACCELERATION RELATION

EXAMPLE OF STICK FORCE/g TEST RESULTS WITH RESPECT TO REF 2. REQUIREMENTS FOR SYMMETRICAL FLIGHT LIMIT LOAD FACTOR $n_L = 8$

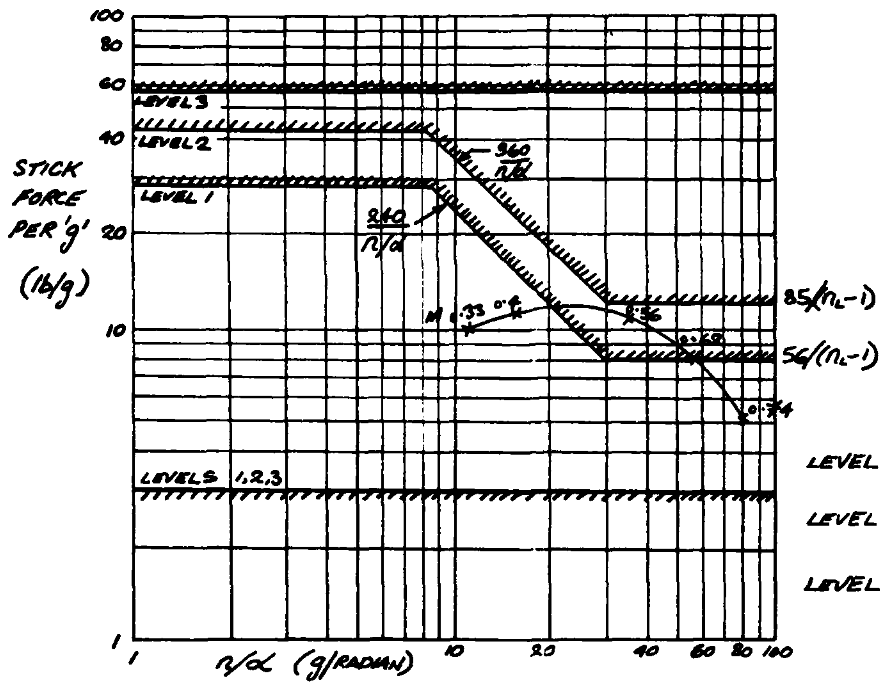


FIGURE 10

LEVEL 1 GREATER OF $\frac{21}{n_L-1}$ AND 3lb

LEVEL 2 GREATER OF $\frac{18}{n_L-1}$ AND 3lb

LEVEL 3 3lb

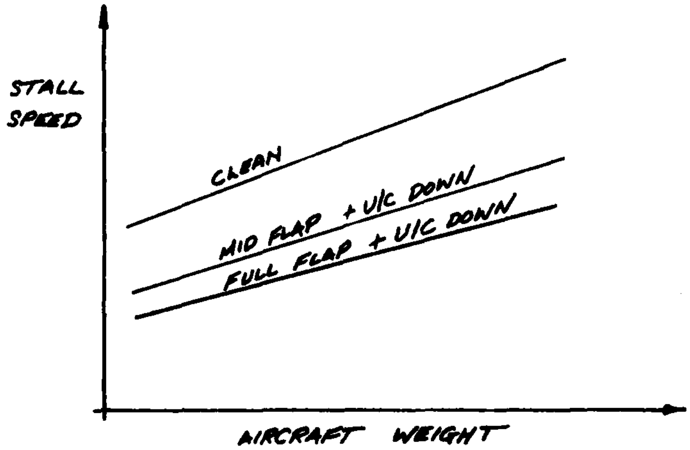


FIGURE 11

TYPICAL STALLING SPEED ~ WEIGHT PRESENTATION

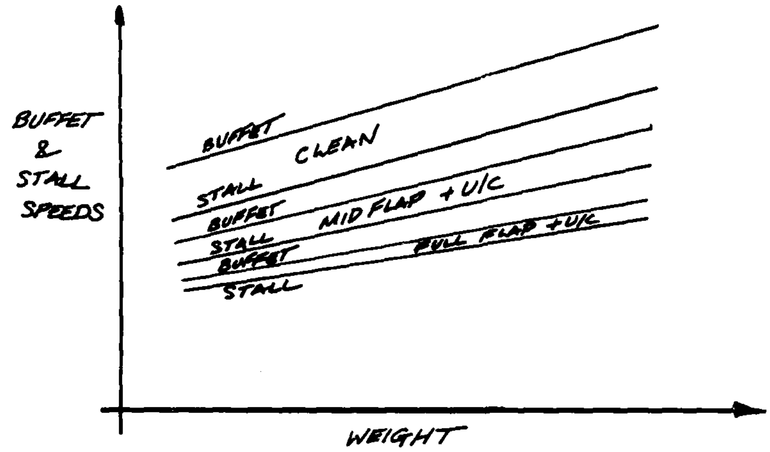


FIGURE 12

STALL SPEED & BUFFET SPEED PRESENTATION

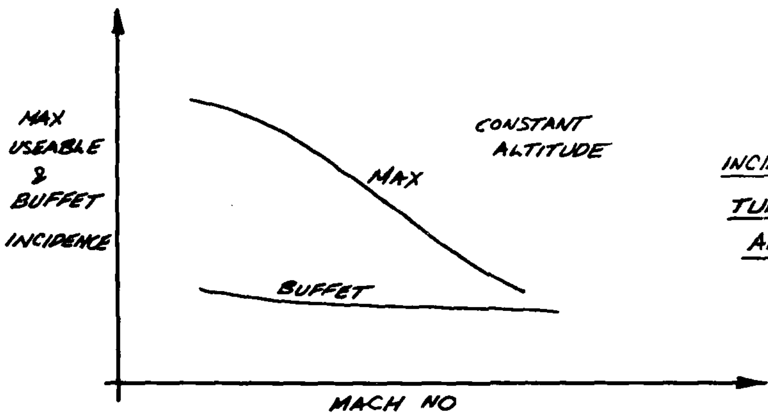


FIGURE 13

INCIDENCE FOR MAXIMUM TURNING PERFORMANCE AND BUFFET MANOEUVRE BOUNDARY

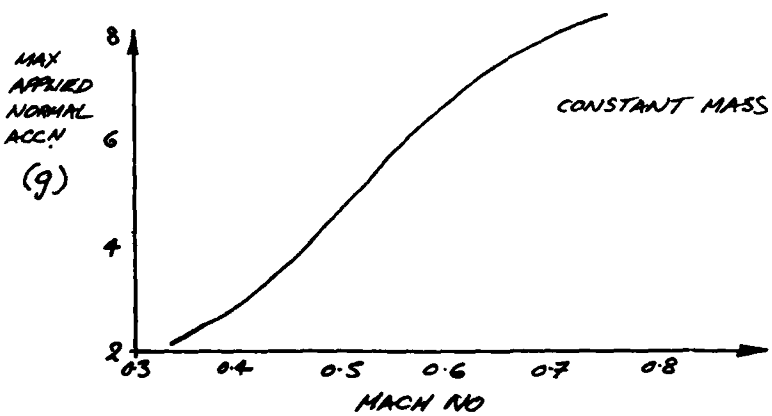


FIGURE 14

TYPICAL MAXIMUM NORMAL ACCELERATION WITH MACH No

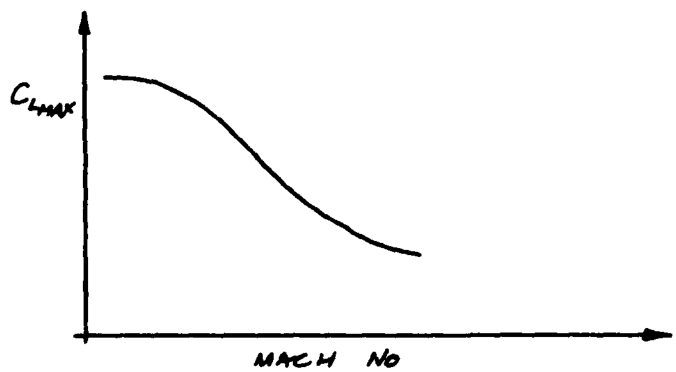


FIGURE 15

VARIATION OF MAX LIFT COEFFICIENT WITH MACH No

AD-A088 530 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
AIRCRAFT ASSESSMENT AND ACCEPTANCE TESTING.(U)
MAY 80
UNCLASSIFIED AGARD-LS-108 NL

2 OF 3
AD A
088530

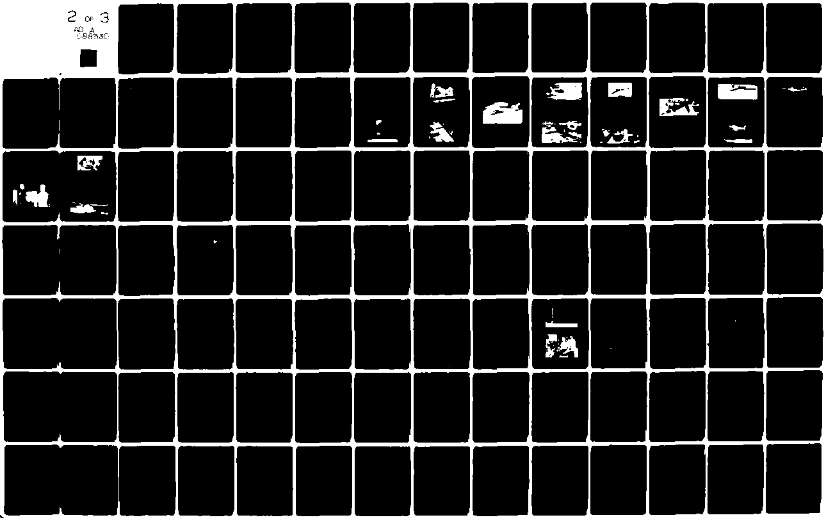
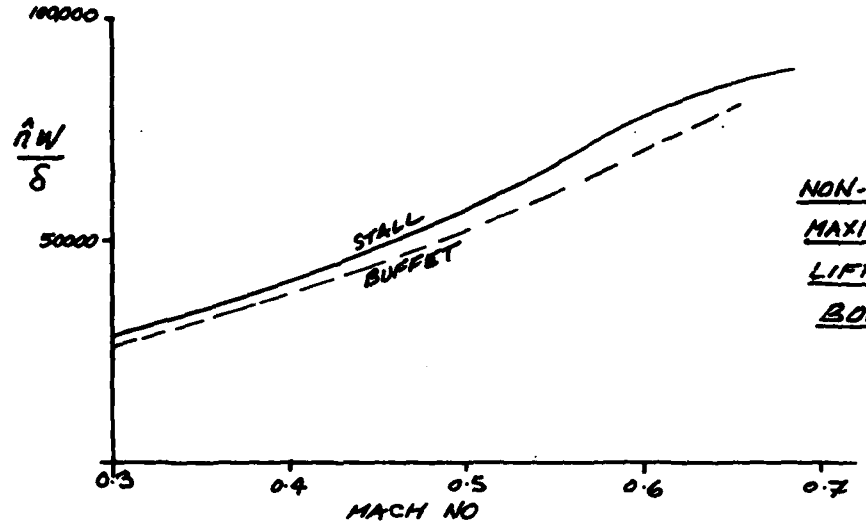
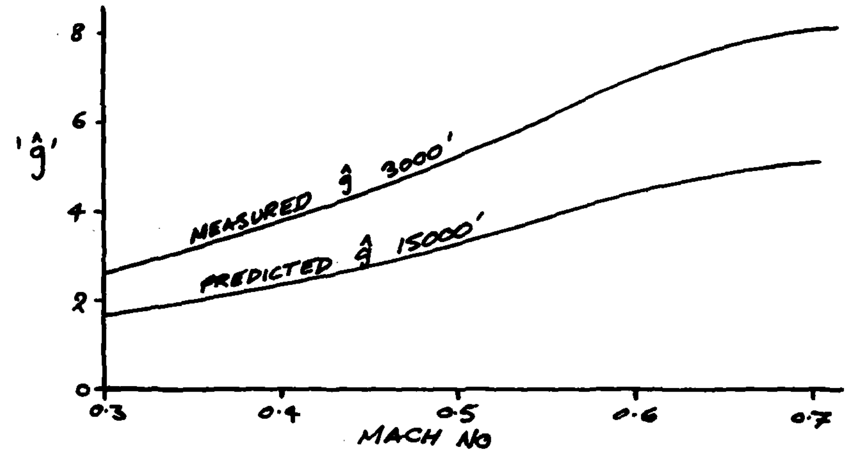


FIGURE 16



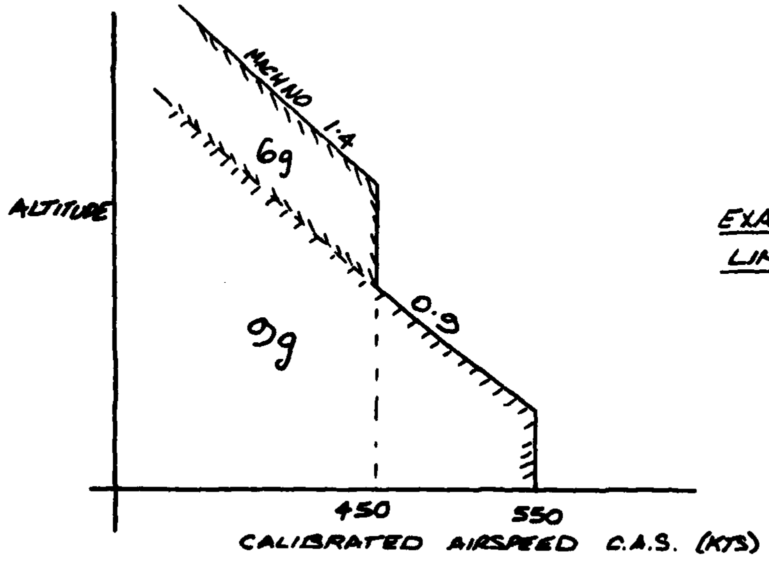
NON-DIMENSIONAL MAXIMUM INSTATANEOUS LIFT & BUFFET BOUNDARIES

FIGURE 17



EXAMPLE OF MAX NORMAL ACCELERATION EXTRAPOLATION

FIGURE 18



EXAMPLE OF OPERATING LIMITATIONS PRESENTATION

APPENDIX I

Example of the computation of Static Margin, CG Margin & Position of Neutral Point

From trim curve tests with the centre of gravity at 0.2 and 0.3 $\bar{c}$ the aircraft normal force coefficient was computed and plotted as Figure 1.

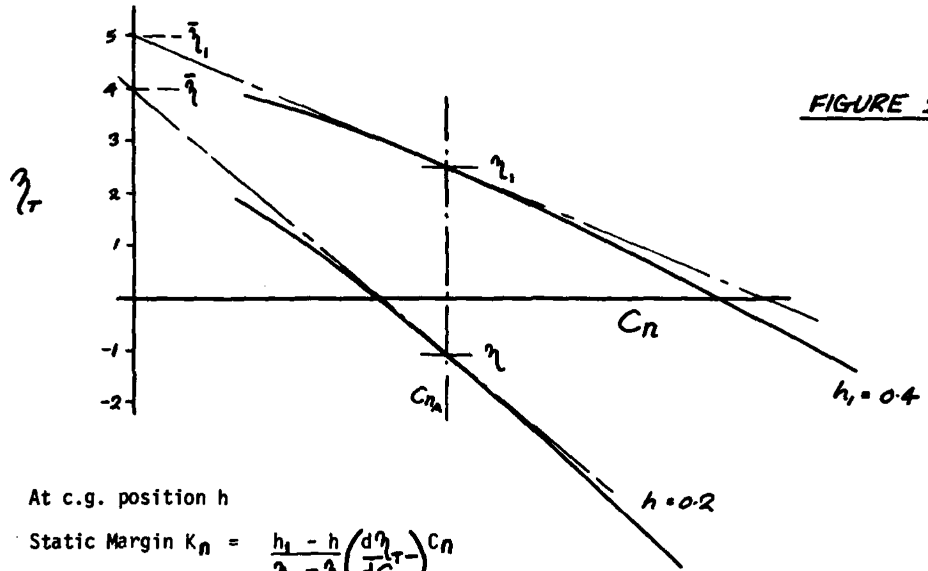


FIGURE 1

At c.g. position h

$$\text{Static Margin } K_n = \frac{h_1 - h}{\eta_1 - \eta}\left(\frac{d\eta_T}{dC_n}\right)C_n$$

$$= \frac{h_1 - h}{\eta_1 - \eta}(\bar{\eta} - \eta)$$

$$= \frac{(0.4 - 0.2)}{+5 - (-1)}(4 - (-1))$$

$$= 0.167$$

and at c.g. position h_1 (move aft)

$$K_{n_1} = \frac{h_1 - h}{\eta_1 - \eta}\left(\frac{d\eta_T}{dC_n}\right)C_n$$

$$= \frac{h_1 - h}{\eta_1 - \eta}(\bar{\eta}_1 - \eta_1)$$

$$= \frac{(0.4 - 0.2)}{+5 - (-1)}(5 - (-1))$$

$$= 0.083$$

The neutral point h_n is found at C_{n_A} by extrapolation as shown in Figure 2 below.

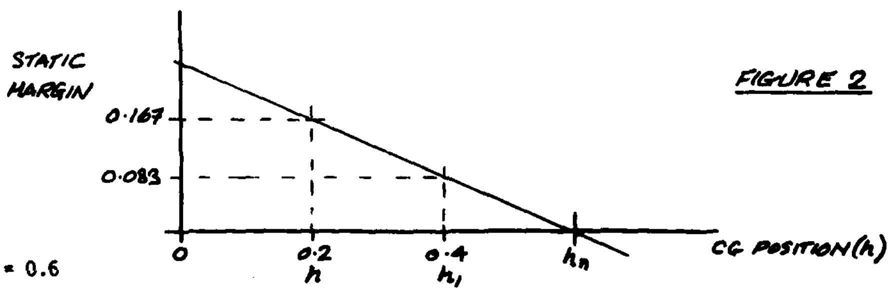


FIGURE 2

$h_n = 0.6$

C.G. margin $H_n = h_n - h$

$= 0.6 - 0.2 = 0.4$

$H_{n_1} = h_n - h_1$

$= 0.6 - 0.4 = 0.2$

APPENDIX I CONT.

The relationship between the static margin and CG margin ψ is then computed for the value of C_{n_A}

$$\psi_1 = Kn_1/Hn_1 \quad \text{and} \quad \psi = Kn/Hn$$

$$\psi_1 = 0.083/0.2 \quad \text{and} \quad \psi = 0.167/0.4$$

$$\psi_1 = 0.415 \quad , \quad \psi = 0.418$$

In this instance aft movement of the cg reduces the static stability at a slower rate than the CG movement itself. (See reference 5 page 66).

EVALUATION OF LATERAL AND DIRECTIONAL CHARACTERISTICS AND SPINNING BEHAVIOUR

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SUMMARY

The lateral and directional handling characteristics that should be investigated during an evaluation of an aircraft already tested by a contractor and test techniques utilising limited flight test instrumentation to gather quantitative data are described herein. Finally an approach to the assessment of spin entry and recovery behaviour is also presented.

INTRODUCTION

Aircraft designed in the U.K. and U.S. are required to comply with the lateral and directional handling requirements stated in AVP970 and Milspecs respectively (References 1 and 2).

Aircraft of Class IV of these requirements will normally be capable of achieving large roll rates and hence have the ability to encounter large increments of sideslip that may exceed the design limits if the control system is not powerful enough to limit it.

An aircraft's lateral and directional characteristics are normally assessed separately from the longitudinal characteristics except in the cases of high rate rolling in conjunction with applied normal acceleration and spinning and post stall gyrations.

Lateral and directional characteristics are very closely associated and at high incidence and many Class IV aircraft resort to the secondary effects of their control surfaces to maintain steady flight conditions and for manoeuvring. For example the use of rudder for roll control at high incidence with the ailerons in the neutral position.

High performance aircraft of previous generations only needed rudder controls to overcome crosswind effects on landing, to hold heading when balancing the effect of a failed engine on a multi engined aircraft and for checking drift during crosswind ground attack manoeuvres. On later aircraft the pilot was required to use rudder for roll control when a certain incidence had been exceeded and he had to be trained to think in terms of rudder usage at high incidence.

Some recent aircraft are fitted with control systems that mix the rudder and aileron inputs as a function of incidence such that the pilot need only move the stick laterally or apply lateral force for roll control.

Contractors conduct rolling and other lateral and directional handling assessments for the following reasons

1. To demonstrate that the dynamic and static stabilities are acceptable
2. To demonstrate that the aircraft complies with the specific limits of References 1 and 2
3. To determine any control limitations necessary to prevent departure from stable flight during rapid rolling and other manoeuvres
4. To demonstrate spin or post stall gyration behaviour if the aircraft can achieve these flight conditions
5. To determine any handling limitations on crosswind take off and landing performance

LATERAL AND DIRECTIONAL STATIC STABILITY

Lateral and directional static stabilities are closely related since they both depend on the generation of restoring moments due to sideslip in order to return the aircraft to wings level straight flight after encountering a disturbance.

If an aircraft has too little lateral and directional stability it will impose a high work load on the pilot and may make precision flying tasks such as instrument flying difficult to perform.

Aircraft classed as belonging to Class IV of Ref. 1 are usually fitted with irreversible powered flying controls and their stability with fixed control surfaces will be considered here.

Some recent aircraft have artificial static stability bestowed on them by advanced electronic control systems while others just use electronic systems to augment their stability in critical flight conditions. For example the BAe Harrier uses roll and yaw stabilizers to input rudder and ailenon deflection demands to the control system to minimize the effect of sideslip and roll disturbances when in semi-jet borne flight.

The restoring moments when the aircraft is disturbed are developed as a result of design features such as dihedral, sweep back and fin geometry and are modified by the position of the wing on the fuselage. At high incidence the dihedral effect of a swept back wing may become over powering and the aircraft may be designed with anhedral to counter the increasing dihedral effect due to sweep back as incidence is increased.

Contractors measure these restoring moments by conducting steady side slip tests from which the yawing and rolling moments due to side slip angle (or lateral velocity) are determined.

For positive static stability the yawing and rolling moments generated from side slip must oppose any disturbance from straight and level flight. If the aircraft is in a positive side slip to the right the yawing moment due to sideslip should be positive (to the left) and the rolling moment should be negative (right wing up). See Figure 1 .

The yawing and rolling moments due to sideslip can be determined if the rolling and yawing

due to rudder and aileron inputs are known, either from calculation, from wind tunnel tests or from flight-test measurements.

The yawing moments due to rudder can be determined by streaming a wing tip parachute with known drag characteristics and measuring the rudder angle needed to trim the aircraft in straight and level flight.

The rolling moment due to aileron can be determined by loading the aircraft asymmetrically and measuring the aileron angle required to trim the aircraft to fly straight and level.

When an aircraft is established in steady sideslip the net rolling and yawing moments are zero, i.e.

$$L_\zeta + L_\xi + L_\beta = 0$$

and

$$N_\zeta + N_\xi + N_\beta = 0$$

in non-dimensional form

$$l_\xi \xi + l_\zeta \zeta + l_\beta \beta = 0$$

$$n_\xi \xi + n_\zeta \zeta + n_\beta \beta = 0$$

For most aircraft l_ζ and n_ξ are small except at high incidence since they represent the secondary effects of rudder and aileron. So these equations can be reduced to

$$\xi = \frac{-l_\beta \beta}{l_\xi} \quad \text{and} \quad \zeta = -\frac{n_\beta \beta}{n_\zeta}$$

if l_ζ and n_ξ are neglected.

From the above equations l_β and n_β can be found if sideslip angle and the control angles are measured in flight using the following technique.

The aircraft is trimmed straight and level at constant speed and rudder and aileron are applied to produce incremental changes in sideslip angle while holding heading constant. The test is continued until the maximum allowable sideslip or full rudder is achieved. At each increment of sideslip the pilot should note sideslip, rudder aileron and bank angle and also the fuel state at the start of the test. Reference 2 statesthat control forces for a centre stick aircraft should not exceed 25lb for full aileron and 175lb for full rudder when executing normal manoeuvres within the flight envelope. These forces should be checked during sideslip tests.

Indicated sideslip angle should be connected for any position error effect and then plotted against rudder and aileron angle, as shown in Figure 2 .

If the control derivatives are in the conventional sense, (i.e. left rudder produces right sideslip and left aileron produces left roll) the aircraft is stable if left rudder and right aileron are required to hold the aircraft in a sideslip to the right as indicated in the figure.

Quantitative data cannot normally be satisfactorily gathered on aircraft without flight test instrumentation since sideslip, rudder and aileron angles are often not displayed in the cockpit.

A qualitative assessment of lateral and directional stability can be made from the direction of the control inputs required to hold the steady sideslip and from the response of the aircraft when the controls are released in a steady sideslip. If the aircraft returns to wings level flight quickly when the controls are released it has strong stability and if the maximum achievable sideslip angle is small with full rudder control, it is deficient in control power. Should a large sideslip angle be generated by a small rudder control input when the aircraft has been shown to be stable by it's response on releasing the controls the rudder may be judged to be too powerful.

The aileron and rudder angle relationships with sideslip angle will not normally be linear over the full aileron/rudder range. In the case of the rudder control this may be due to changes in n_β , n_ζ or n_ξ but if n_ξ is neglected the ratio ζ/β is still equal to $\frac{-n_\beta}{n_\zeta}$ and the aircraft is not unstable directionally unless the sign of ζ/β changes (i.e. point 1 on figure 7 2).

A similar state of affairs exists with the aileron control where the aircraft would be laterally unstable at point 2 on the figure.

Stick and rudder pedal forces required to hold an aircraft in a steady sideslip will be non-linear if the relationships between aileron and rudder with sideslip are non-linear. These forces may be optimised with the incorporation of non-linear springs or changes of stick to control surface deflection gearings. The control forces required to manoeuvre the aircraft may also be modified as a function of speed or Mach No by use of a pilot-static pressure difference sensor or "q" feel pot.

LATERAL AND DIRECTIONAL DYNAMIC STABILITY

If an aircraft is disturbed from a steady flight condition by a momentary pilot input or turbulence it will exhibit a response that can be considered as three independent motions occuring simultaneously.

i) A normally heavily damped rolling motion where angle of bank changes occur with virtually no change in yaw and sideslip. This is known as the Roll Subsidence Mode and determines the aircraft's response to lateral upsets and aileron.

ii) A normally lightly damped or slightly divergent motion in yaw and bank with virtually no sideslip. This is known as the spiral mode and it determines the long term stability of the aircraft and it's lateral trim.

iii) A lightly damped or divergent oscillation in roll yaw and sideslip of relatively short period (typically 2-4 seconds) that determines the aircraft's long term steadiness in turbulent air. This motion is described as the Dutch Roll Motion if the rolling component is large and snaking if the rolling component is small and the motion is predominantly in yaw.

The above motions can be described by the solution of the three lateral equations of motion for side force, yawing moment and rolling moment after making the assumption that they can be divorced from the longitudinal equations by neglecting cross coupling terms. Sideways velocity, yaw rate and roll

rate are assumed to be exponential functions of time, i.e. $V = V_o e^{\lambda t}$

$$P = P_o e^{\lambda t}$$

$$r = r_o e^{\lambda t}$$

The above expressions are substituted in the three lateral equations of motion and V_o, P_o and r_o are eliminated by simultaneous solution of the three equations in terms of λ

The stick-fixed lateral motion of the aircraft is then represented by the characteristic quartic equation $A_e\lambda^4 + B_e\lambda^3 + C_e\lambda^2 + D_e\lambda + E_e = 0$ where the constant coefficients are composed of the lateral aerodynamic derivatives and inertias as described fully in Reference 3.

The solution of the characteristic equation normally yields a large real root, a small real root and a pair of complex roots which represent the three motions described above.

The large real root represents the Roll Subsidence Mode which is damped very quickly (typically within ½ second) due to the damping in roll derivative lp generated principally by the difference in local incidence between the rising and falling wings as the aircraft rolls.

The small real root represents the Spiral Mode which may be divergent and which if uncontrolled will cause the aircraft to enter an ever tightening spiral. A spirally unstable aircraft will necessitate the pilot often retrimming laterally which may be annoying when instrument flying. If an aircraft is spirally unstable the instability should not be great enough to double the amplitude of the motion in less than 20 seconds.

The complex pair of roots represent the Dutch Roll Mode which may on some designs be very lightly damped and require a stability augmentation system to reinforce the damping.

Swept wing aircraft have a tendency to exhibit poor Dutch Roll damping at low speed on the approach with wing flaps extended but the length of time that combat aircraft operate in this condition is short and the deficiency will not necessarily increase pilot workload significantly.

With an instrumented aircraft all three dynamic stability modes can be examined by exciting the aircraft with the controls and observing it's response using the following technique.

The aircraft is trimmed for straight and level flight and the rudder is then smartly deflected in either direction and immediately returned to the neutral position. The controls are then held fixed and the resulting oscillation observed until it damps out or diverges sufficiently for pilot to consider that the sideslip structural limit may be reached.

Typical flight test instrumentation records of a damped Dutch Roll oscillation with spiral instability and strong roll subsidence are presented as Figure 3 . The period and cycles and time to ½ amplitude are determined using the envelope on long decrement methods of Reference 4

The pilot cannot easily control a lightly damped Dutch Roll Motion and consequently the design requirements specify a minimum level of damping. For U.K. designs the oscillation is required to decay to ½ amplitude within one cycle of removal of the excitation force. Roll rate/Yaw rate ratio should be less than 3.0.

If the aircraft is fitted with a gyro gunsight with a fixed cross the dynamic stability can be observed from the relative movements of the pipper and the fixed cross when the Dutch Roll Mode is excited and a qualitative assessment can be made. Spiral instability can be seen by the aircraft's gradual change of heading.

Quantitative data can be obtained if a pilot display gunsight camera is fitted and run during the test.

Accurate free fall bombing and high definition aerial photography and good target tracking are difficult to achieve if the aircraft does not have good dynamic stability either with or without augmentation and particular care should be taken to examine the Dutch Roll tendency when evaluating an aircraft for these tasks.

An aircraft that carries it's payload externally may be dynamically stable in the clean configuration but it's stability may be seriously degraded when the external stores are fitted and the side area ahead of the centre of gravity is increased disproportionately. An evaluation of such an aircraft should include at least one flight with an aerodynamically destabilizing set of external stores.

Reference 2 details various Dutch Roll damping criteria which they consider appropriate for each aircraft type.

ROLLING PERFORMANCE ASSESSMENT

An aircraft's rolling performance may be described in terms of the maximum roll rate that it can achieve under a given flight condition and the time constant of the rolling motion (i.e. the time taken to achieve 63% of the final steady roll rate after application of the roll control).

With an instrumented aircraft the roll response may be obtained using the following technique.

The aircraft is established in wings level unaccelerated flight at the target test speed and altitude and the ailerons are then deflected fully in one direction with the rudder held neutral. When the aircraft has rolled through 180° the controls are returned to the initial position and recovery to level flight completed. The fuel state just before entry should be noted in order to establish the inertia in roll (A).

Normally with a fighter type aircraft the maximum rate of roll will have been achieved within this bank angle change and a typical trace record of the manoeuvre is presented as Figure 4 .

The aircraft's roll response to aileron is not normally affected by the Dutch Roll or Spiral Mode characteristics since the time constant is so small and the roll response can be considered simply from the rolling moment equation

$$A\dot{p} = L$$

The initial slope of the response curve Figure 4 corresponds to the initial rolling acceleration $\dot{p} = L_{\xi}/A$ at the start of the manoeuvre.

Once the aircraft starts to roll the roll rate is opposed by the damping in roll derivative L_p which arises from the difference in incidence between the rising and falling wings as the aircraft rolls, provided that the incidence is not close to the stall where L_p changes sign.

The rolling moment equation can be written in the form

$$A\,\dot{p} = L_{\xi} + L_p \quad \text{neglecting other rolling derivatives.}$$

When a steady roll rate is achieved the equation can be written in non-dimensional form as

$$\hat{P} = -\frac{-l_{\xi}\xi}{l_p}$$

The dimensional roll rate is $\hat{P} \times 2V/b$

(If the control derivative l_{ξ} is known the derivative l_p may be estimated from this test).

Simulation research into pilot opinion as to the rolling performance required by different types of aircraft in various phases of flight has led to the formulation of opinion plots such as Figure 5.

The initial roll acceleration $\dot{P}_0$ and time constant T_R of the motion for several qualities of performance are presented in the figure and the performance achieved by an aircraft under evaluation is also shown. The figure also illustrates that for a given roll acceleration the time constant for good handling must not be too great or too small.

If handling qualities diagrams such as Figure 5 and prepared for various phases of flight for an aircraft type under evaluation the merits of one or a number of different aircraft can be deduced.

Rolling performance may also be examined on the basis maximum roll rate achievable at each flight condition and the time taken to roll through a given bank angle. The later technique being more applicable to non-aerobatic aircraft.

LATERAL AND DIRECTIONAL CONTROL DURING TAKE OFF AND LANDING

The maximum crosswind component in which an aircraft may take off and land safely is limited by the adequacy of the lateral and directional controls at low speed and the aircraft's directional controllability on the ground.

The adequacy of the lateral and directional controls at low speed in the take off and landing configurations can be determined from the sideslip angles achieved during wings level steady sideslip tests at low altitude using the following technique. The aircraft is flown straight and level at the expected approach speed and configuration. Sideslip is then generated incrementally using rudder whilst the wings are held level with the ailerons. The maximum sideslip generated when either full rudder or aileron is achieved reflects the maximum crosswind component W in which the pilot will be able to "kick off the drift" on landing or hold the wings level on take off

$$\text{i.e. } W = V\sin\beta_{TRUE}$$

The pilot should not actually be required to use full lateral control under limiting conditions since the aircraft may in practice experience gusts above the crosswind limits at lift off or just before touchdown. Crosswind limitations should therefore be chosen with reference to the sideslip that can be held with $\frac{1}{2}$ or $\frac{2}{3}$ of the critical lateral or directional control available.

The crosswind limit may be determined by the effectiveness of the aerodynamic controls or other steering devices such as nosewheel steering or differential braking during the ground run on take off, or the roll out on landing. The accuracy of the directional control on the ground may also cause the maximum allowable crosswind to be scheduled in terms of runway width and to be further restricted for night operations.

Crosswind limitations are determined during the contractors flight trials and are presented in the Aircrew Manual. They may be qualitatively examined during an evaluation by landing and taking off on a crosswind runway in a moderate wind and estimating the amount of aileron and rudder travel used.

External stores, especially asymmetric configurations will affect the crosswind handling and a typical worst case should be assessed during an evaluation.

EFFECTS OF MACH NO

As in the case of longitudinal handling the lateral and directional handling characteristics change with increasing Mach No as shockwaves form on the aircraft distorting the pressure distribution.

Changes occur in the stability and control derivatives from variations in the pressure field and airfame elastic distortion and these may occur rapidly in the transonic region.

Most aircraft suffer some stability degredation in the transonic region but may recover and exhibit improved handling when truly supersonic.

An evaluation should include examination of trim changes both at high Mach No at altitude and at the altitude where the flight envelope allows the greatest dynamic pressure to be achieved.

Structural elastic distortion can cause control reversal which is undesirable and this is most likely to occur at high speed.

Reduction of an aircraft's stability with Mach No may be countered by use of a stability augmentation system or an entirely artificial stability and control system. In the latter case, whatever the flight condition, a computer decides the direction and magnitude of control surface input necessary to prevent any flight path deviation. Pilot inputs to manoeuvre the aircraft are translated by the computer to demand the appropriate surface deflection and to limit any requirement that would result in departure from stable flight or structural overload.

An evaluation of an aircraft should include examination of flight test results of contractors handling tests throughout the permissible speed range and demonstration of the most critical cases.

INERTIA COUPLING AND ROLLING PERFORMANCE

Highly manoeuvrable combat aircraft in Class IV of References 1 and 2 are generally required and have the ability to achieve high roll rates (in excess of 150°/sec) at moderately high speeds (400Kts+).

Under these conditions they are prone to the phenomena of inertia coupling which is manifest by sudden increases in yaw or pitch and hence sideslip or incidence due to the aircraft's inertia forces overcoming the stabilizing aerodynamics. The resulting excursions in incidence or sideslip if not limited can cause complete loss of control and/or structural failure at high speed.

Any assessment of aircraft in Class IV particularly should include a demonstration of the manufacturers stated limits in rolling performance and consideration should be given to the ease with which these limits may be inadvertently exceeded (e.g. increasing rate of roll with rudder perhaps).

Flight records should be examined to observe any rapid sideslip and incidence changes during maximum performance rolls over the normal acceleration range that the aircraft may be flown at.

The effectiveness of the controls in returning the aircraft to level flight after the pilot has taken recovery action should be examined.

Contractors measure the aircraft's tendency to inertia coupling from roll and stop tests using full lateral control applied over a range of bank angle changes (180°, 360°) over a range of increasing speeds and entry normal accelerations.

It is usual to commence the investigation with 1g, 360° rolls and then to increase the speed incrementally to the maximum.

When investigating rolling behaviour with applied normal acceleration the aircraft is initially rolled through 180° once the intended g is achieved. If flight test instrumentation records show that roll rate has reached its peak value before recovery action was taken it is considered safe to perform 360° rolls at the same flight condition on a subsequent flight.

The favoured flight technique used for these tests depends on the normal acceleration required at entry.

For positive normal acceleration the aircraft is placed in a steady banked turn at the required altitude speed and normal acceleration. (For speeds above max level the aircraft may be established in a descending spiral). Once the target flight condition is achieved the aircraft is rolled through the required bank angle.

For negative acceleration the aircraft is flow inverted in a shallow dive to the test speed and altitude and the pilot then pushes the stick forward to achieve the target normal acceleration before rolling through the target bank angle change. -1g is achieved by rolling the aircraft from inverted level flight and zero g from a pushover from erect level flight before rolling.

These tests are normally considered as high risk exercises and comprehensive continuous data recording is essential for safe conduct of the trials. Sideslip and incidence should be prominently displayed to the pilot in the cockpit and ideally a telemetry system conveying the flight data to trace recorder for ground monitoring should be employed. A safety pilot on the ground where the data is displayed can be of great value to advise the pilot of recovery actions by radio if the test pilot becomes disoriented or loses control.

A telemetry system is more economical in terms of flight time when it is necessary to inspect flight records before attempting a more severe test flight condition.

The advent of electronic computers has enabled the aircraft's characteristics to be rapidly assessed over a wide range of flight conditions using mathematical models and the response in roll to various lateral control inputs and at flight conditions including +ve and -ve normal acceleration to be examined before flight trials commence.

From the simulations sideslip and incidence excursions can be related to maximum values tolerable from a structural point of view and a theoretical flight condition boundary can be drawn for rolling manoeuvres for each lateral control input.

As stated previously an aircraft's longitudinal and lateral stabilities can be assessed separately if some of the cross coupling terms in the six equations of motion are neglected. When the aircraft is subjected to a high rate of roll the product of inertia terms become significant and longitudinal and lateral stabilities cannot be considered separately.

If an aircraft is rolled rapidly about its flight path axis the inertia can be seen to transfer the aircraft rotation axis to the principal inertia axis at a rate dependent on the mass distribution of the aircraft and the roll rate.

This effect is most marked on aircraft whose mass distribution is such that the principal axis of inertia is greatly inclined to the horizontal fuselage datum and the flight path. Inertia coupling susceptibility is greater for aircraft with small wingspan and fuselages with mass distribution such that large masses are concentrated at the nose and tail e.g. heavy nose radar and high mounted heavy tailplane.

The manner in which the mass distribution affects the aircraft's tendency to rolling instability through inertia coupling is shown in Figure 6 and is described in detail in Reference 6
If it is assumed that the aircraft's mass is concentrated at the extremity of wings, nose and tail and the aircraft is rotating about an axis coincident with the flight path it can be seen that

i) The centrifugal forces acting on the wing masses due to rotation tend to yaw the aircraft into wind

ii) The centrifugal forces acting on the nose and tail masses tend to yaw the aircraft out of wind. If the aircraft's mass is largely concentrated in the fuselage the out of wind moment may be the greatest and the aircraft will only remain stable if the aerodynamic forces can balance the resultant inertia forces.

A similar condition can be seen to exist in the pitching plane as shown in Figure 6 The inertia effect in the pitching plane is destabilizing.

The aerodynamic moments which balance the destabilizing inertia forces are n_v in the yawing plane and m_w in the pitching plane.

If the aircraft's configuration is such that inertia coupling may occur at a low value of roll rate the weathercock stability and effectiveness should be carefully optimised during the design. If the aircraft's control system enables it to generate unnecessarily high roll rates at full aileron the aircraft may be fitted with high speed stops to limit the aileron travel, activated by undercarriage retraction for example, so that full aileron travel is available at low speed where required.

Inertia coupling effects are not always only found at the extremes of the flight envelope and speed so the contractors investigation must cover intermediate flight conditions.

SPINNING AND RECOVERY

Intentional spinning of many modern highly manoeuvrable aircraft is prohibited but nevertheless spins may result when these aircraft are manoeuvring near the extremes of their flight envelopes.

Departures from stable flight occasioned by wing rock, wing drop or excessive sideslip at high incidence, can develop into wildly oscillatory or very fast flat spins from which recovery after a turn or two is not possible.

U K. contractors investigate the departure characteristics of their designs in order to determine satisfactory recovery drills and to measure height loss during recovery from incipient and fully developed spins.

From the results of trials, conducted initially by the contractors, the Aeroplane and Armament Experimental Establishment may recommend that the Services be granted a clearance to spin. They may alternatively recommend prohibition in which case they will offer advice on the recognition of an incipient spin and the best technique for recovery to steady flight.

A spin is a more or less steady motion which may result when an aircraft is disturbed in roll or yaw at incidences at or beyond the stall.

At incidences above that at which the stall occurs, some of the more important aerodynamic stability derivatives change sign or magnitude suddenly and enable the aircraft to enter a self-sustaining rotary motion or spin. The damping in roll derivative lp, which normally damps rolling disturbances, becomes positive and consequently accelerates the aircraft when it is disturbed in roll, i.e. the falling wing, although at a greater incidence than the rising wing, produces less lift and therefore carries on falling. Disturbances in yaw will also generate disturbances in roll, which can cause wing drop and entry into a spin.

Normally, an aircraft will enter a spin with the nose above the horizon and will take several turns before it attains a steady helical flight path about a vertical axis.

If the spin axis is above the aircraft, the spin is said to be erect and if below the spin is termed inverted.

The spinning motion is a complex one involving pitching, rolling, yawing and sideslipping, but if it is steady all the moments and forces acting on the aircraft must balance and the aircraft will be in a state of equilibrium.

Sometimes, a steady spin is not attained and the aircraft oscillates in pitch, roll and yaw, but more normally an aircraft with a conventional planform and inertia distribution, will orientate itself in such a way as to reach equilibrium.

In the spin, the resultant aerodynamic force normal to the plane of the wings acts through the axis of the spin. The aircraft drag is equal and opposite to the gravitational force and the lift force is reacted by the centrifugal force generated by the rotation of the aircraft about the spin axis. Sideforces are small since sideslip is usually small and sideforce balance plays little part in determination of the spin characteristics.

The type of spin and the aircraft's orientation are principally determined by the balance of the moments about the aircraft axes. For equilibrium, the inertia moments are exactly balanced by the aerodynamic moments, the balance being effected by the aircraft's adjustment of the rate of spin rotation and the sideslip angle.

Pitching moment balance:-

The inertia moment about the pitching axis is $M = (C-A)pr$

For the BAe Hawk (C-A) is positive (See Figure 7) and in an erect spin, where p and r have the same sign, is a nose up pro-spin moment. This is balanced by the net aerodynamic nose down pitch moment of the wings and tailplane.

Rolling moment balance:-

The inertia moment about the rolling axis is $L = (B-C)\, qr$

L is negative and in an erect spin it is balanced by a net pro-spin aerodynamic rolling moment composed of l_p, l_r, l_v and l_ξ some of which are pro-spin and some anti-spin. For a standard spin l_p, l_r and l_ξ are pro-spin and l_v is anti-spin.

Yawing moment balance:-

The inertia yawing moment about the yawing axis is $N = (A-B)\, pq$

N is -ve in an erect spin and is balanced by a net pro-spin aerodynamic moment composed of n_p, n_r, n_v and n_γ.

In a standard spin n_p is anti-spin and n_v, n_r and n_γ are pro-spin.

Alterations in any of the flying control settings will change the balance of the moments and will result in a change in spin rate or sideslip angle or both.

Moving the stick forward will produce a nose down pitching moment and gyroscopic precession in yaw normally increases sideslip and spin rate. Increased spin rate increases the pro-spin nose up inertia moment and incidence will increase until balance is attained.

Application of aileron in the direction of the spin (in-spin aileron) will increase the roll rate and if the ailerons produce adverse yaw, will tilt the inner wing further below the horizon and slow the spin rate. If the ailerons produce proverse yaw, the wing tilt will be reduced and the spin rate will increase. See Figure 8 for aircraft orientation in a positive spin.

Application of aileron against the direction of spin (out-spin aileron) will produce the opposite effect, i.e. adverse yaw will decrease wing tilt and increase spin rate.

An increase in spin rate will usually result in an increase in incidence and will flatten the spin.

Before investigating spinning characteristics in flight contractors normally conduct tests with models dropped from helicopters or launched into the working sections of vertical wind tunnels such as the Spinning Tunnel at the Institut de Mechanique des Fluids de Lille, France.

The effect of control positions on spin entry and recovery are investigated using models whose control surfaces may be actuated in flight by radio control and the most likely recovery procedure is predicted.

The severity of the spin and it's likely tendency to become oscillatory is also assessed from these tests.

Spinning trials may be hazardous and contractors take steps to minimise the possibility of the pilot not being able to recover the aircraft by fitment of additional cockpit instrumentation and antispin devices.

Typical precautions include the following

a) Fitment of antispin parachute or rockets at the aircraft's tail to provide a nose down pitching moment and reduce the wing incidence should the pilot run out of longitudinal control.

b) Development of the engine relight system before the spinning trials to ensure that the pilot will be able to relight the engine if it is necessary to shut it down if a surge occurs.

c) Provision of additional hydraulic accumulator capacity in order to allow the pilot more control activity for recovery should the engine stop and the engine driven hydraulic pumps be unable to supply adequate pressure.

d) Provision of prominent aileron and rudder control position gauges and indications of direction of roll and yaw to enable the pilot to accurately centralize the controls and easily determine the direction of aircraft rotation.

The use of telemetry to display the aircraft's behaviour to engineers and a safety pilot on the ground to enable the safety pilot to offer advice to the test pilot if he becomes disoriented is highly desirable and greatly enhances the trials safety.

The flight trials can be conducted in three stages and may be terminated at any stage if recovery can only be effected with the assistance of an antispin parachute or other recovery aid.

Stage 1 :- Exploration of the incipient spin, the tendency to enter a spin with application of any of the controls and early recovery indication

Stage 2 :- Exploration of the fully developed spin, assessment of the effect of in-spin and out-spin aileron and a range of tailplane inputs and recovery techniques

Stage 3 :- Exploration of oscillatory spin modes and the effect of control mishandling at spin entry and recovery. Determination of effects of manoeuvring flap, airbrake and engine power settings and spin entry from mishandled aerobatic and high rate turning manoeuvres. Examination of the likelihood of entering an inverted spin and recommendation of recovery procedure to be adopted.

The trials are normally commenced at altitudes above 30,000 ft with the aircraft ballasted to the mid C.G. position.

The aircraft is slowed to the unaccelerated stall with wings level and engine at idle and the effect of full aileron or rudder applied at the stall is determined. The controls are centralized after approx. 180° bank angle change if the wing drops. If recovery is satisfactory the test is repeated with the pro-spin controls held for and increasingly greater bank angle changes until a developed spin is achieved. If the aircraft is spin resistant the effect of applying pro-spin control in a steady banked turn or in conjunction with a rapid aft stick movement in level flight on receipt of stall warning may be more successful in inducing a departure.

It is not necessary for a potential purchaser evaluating an aircraft to repeat the contractors spinning trials but any evaluation of an aircraft cleared for spinning should include a demonstration of the range of spin behaviour exhibited by the aircraft.

From the results of the spinning programme the contractor will be able to describe and demonstrate the following :-

1. The characteristics of the incipient spin, recovery procedure and expected height loss on recovery.
2. The effects of the following variables on the spin characteristics, entry and recovery and height loss
    a) Power setting
    b) In-spin and out-spin aileron input
    c) Rudder input
    d) Longitudinal control input
    e) Effects of combinations of b) c) and d)
    f) Centre of gravity
    g) External stores
    h) Extended manoeuvring flaps and air brake
3. The aircraft's behaviour in an inverted spin if it can be achieved.

If the contractors trials have shown that the aircraft cannot be recovered from a departure from normal flight or a developed spin, the pilot may be given limitations to fly to in order to avoid loss of control. Alternatively, the aircraft's control system may be designed to limit it's manoeuvrability in order to prevent incidence, sideslip and normal acceleration reaching levels at which departure will occur.

An evaluation should question the ease with which a pilot may inadvertently exceed the limitations and thereby enter a spin.

Most aircraft are more reluctant to enter an inverted spin since the fin and rudder are not easily blanked by the turbulent wakeof the wing and tailplane. The contractors will determine the conditions under which an inverted spin may be achieved and the recovery procedure to be adopted. It should be noted that if the aircraft has a manual rudder it may be blown into a pro-spin position by the airflow at the rear of the aircraft and recovery from a well developed inverted spin may be dependent on the maximum foot force that the pilot can apply.

SYMBOLS

L_p	Rolling Moment due to roll rate
L_ζ	" " " " rudder angle
L_ξ	" " " " aileron angle
L_β	" " " " sideslip
L	Total rolling moment
l_p	Rate of change of rolling moment coefficient due to rolling
l_ζ	" " " " " " " " " rudder
l_ξ	" " " " " " " " " aileron
l_β	" " " " " " " " " sideslip
l_r	" " " " " " " " " yawing
N_p	Yawing moment due to roll rate
N_ζ	" " " " rudder angle
N_ξ	" " " " aileron angle
N_β	" " " " sideslip
N	Total yawing moment
n_p	Rate of change of yawing moment coefficient due to rolling
n_ζ	" " " " " " " " " rudder
n_ξ	" " " " " " " " " aileron
n_β	" " " " " " " " " sideslip
n_r	" " " " " " " " " yawing
M	Total pitching moment
m_w	Rate of change of pitching moment coefficient due to vertical velocity
ζ	Rudder angle
ξ	Aileron angle
β	Sideslip angle
v	Sideways velocity
p	Roll rate
r	Yaw rate
A, B, C	Inertia Moments in pitch, roll and yaw

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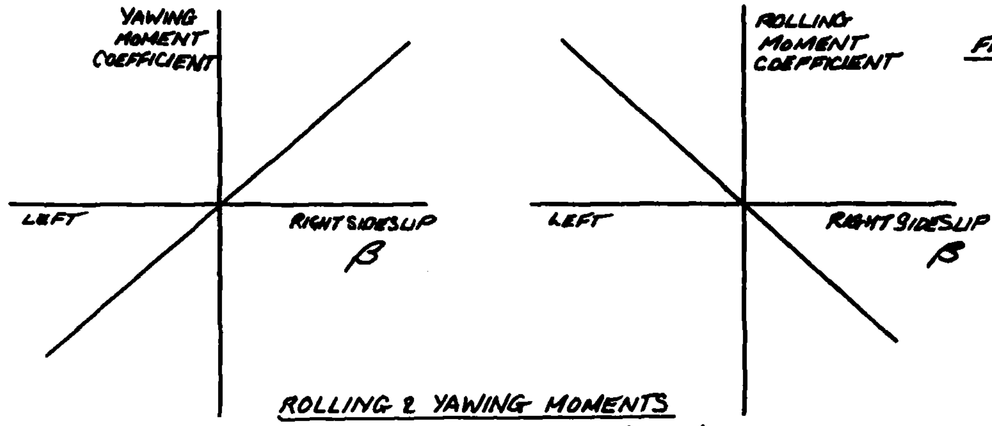
FIGURE 1



ROLLING & YAWING MOMENTS FOR STATIC LATERAL STABILITY

AILERON & RUDDER IN STEADY SIDESLIPS

FIGURE 2

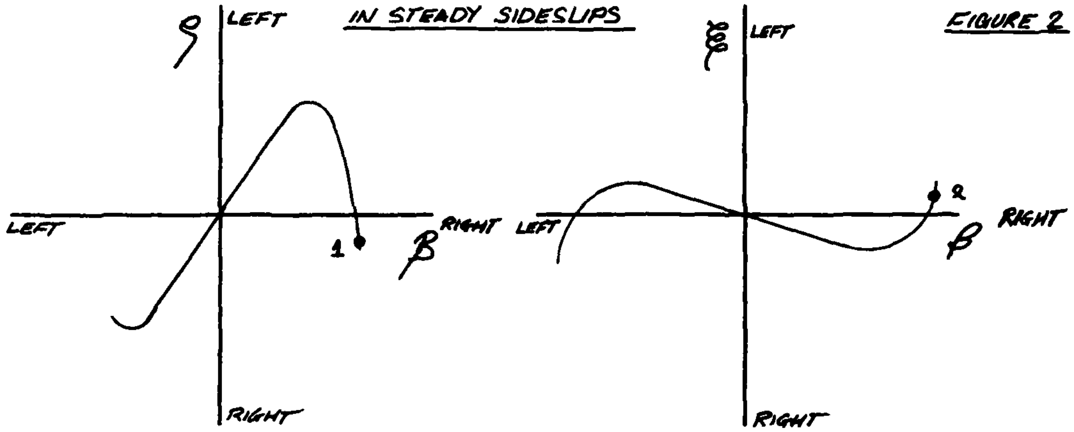
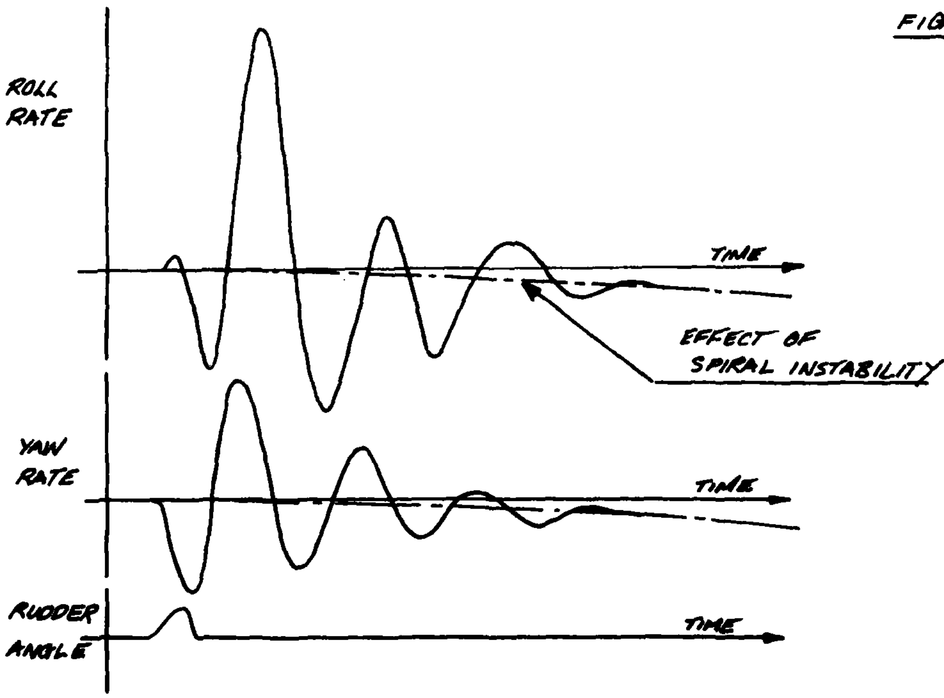
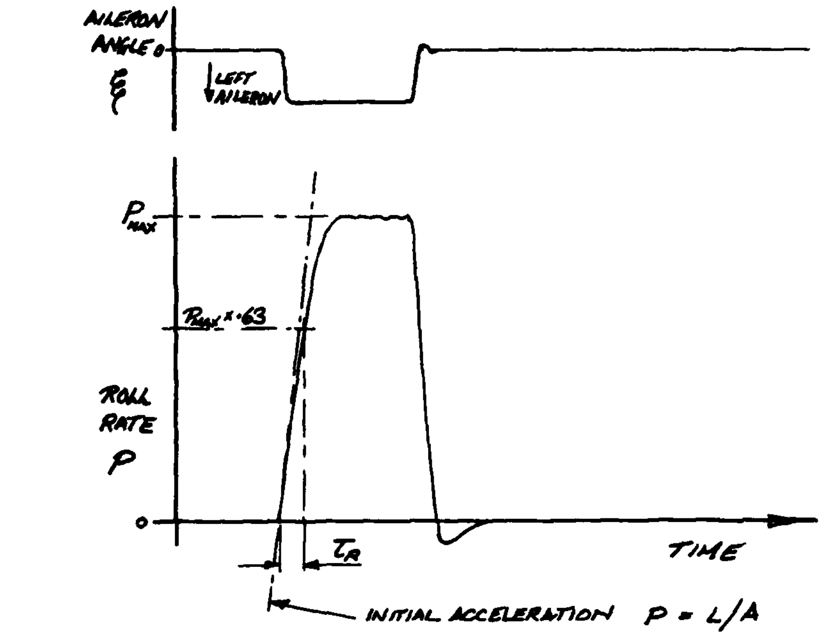


FIGURE 3



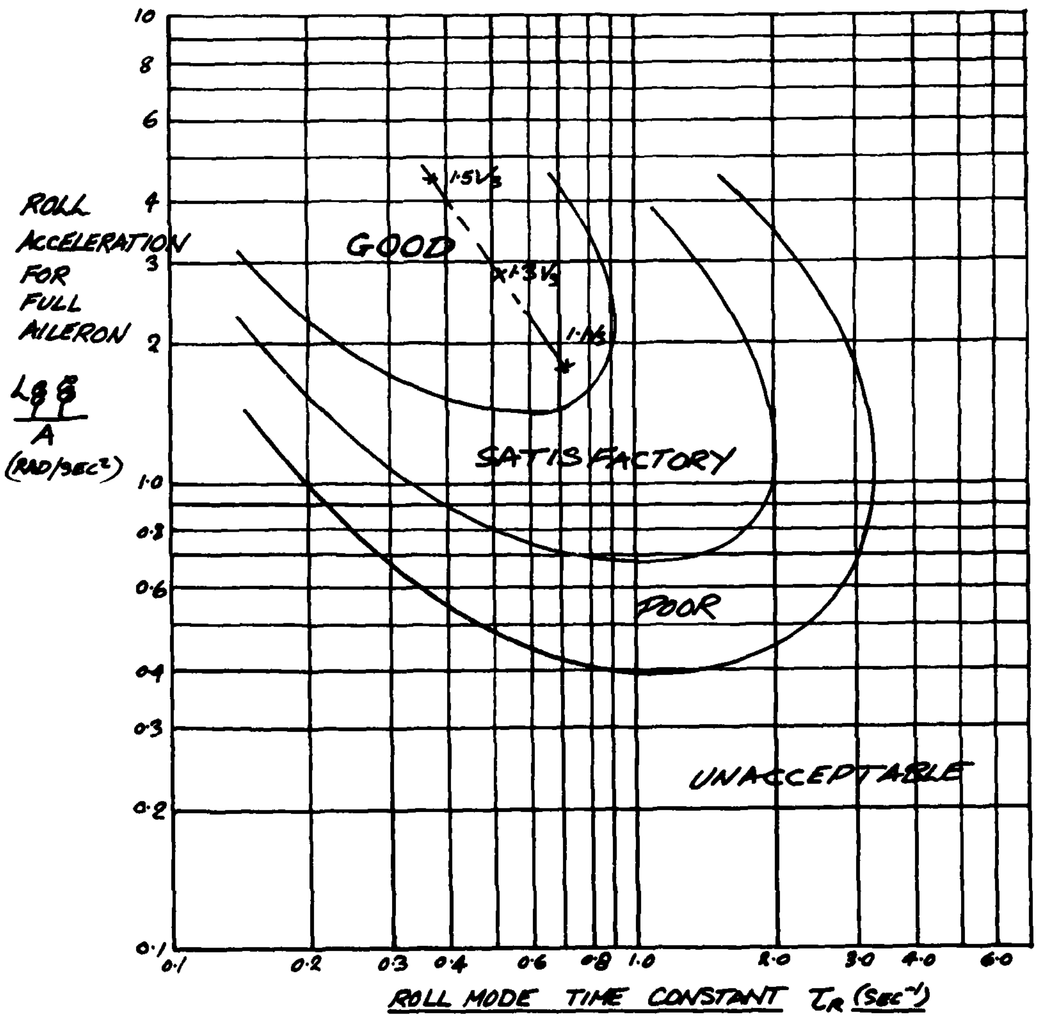
ROLL RATE & YAW RATE DURING DUTCH ROLL OSCILLATION

FIGURE 4



ROLL RESPONSE TO FULL AILERON

FIGURE 5



PILOT OPINION: BOUNDARIES FOR APPROACH CONDITION ON ROLL ACCELERATION AVAILABLE ~ LIGHT STRIKE AIRCRAFT

FIGURE 6

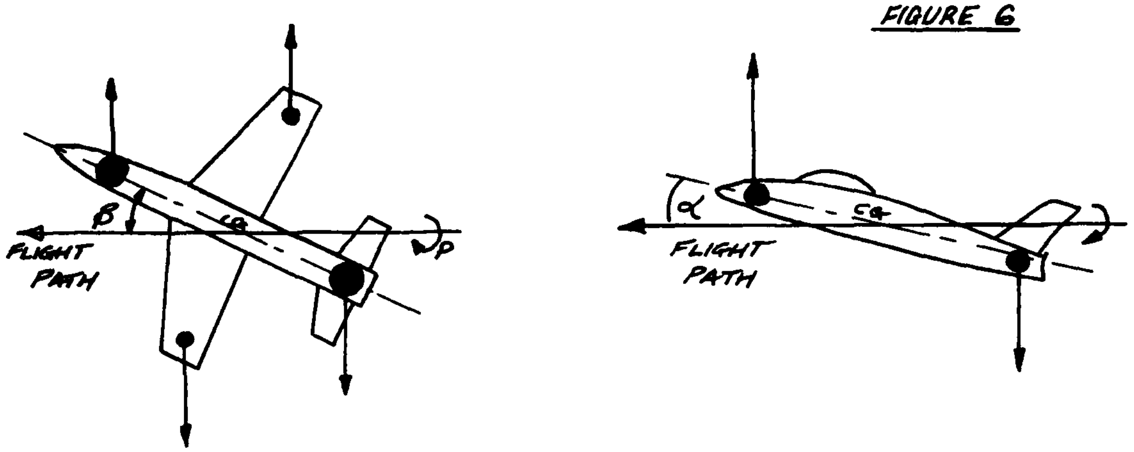
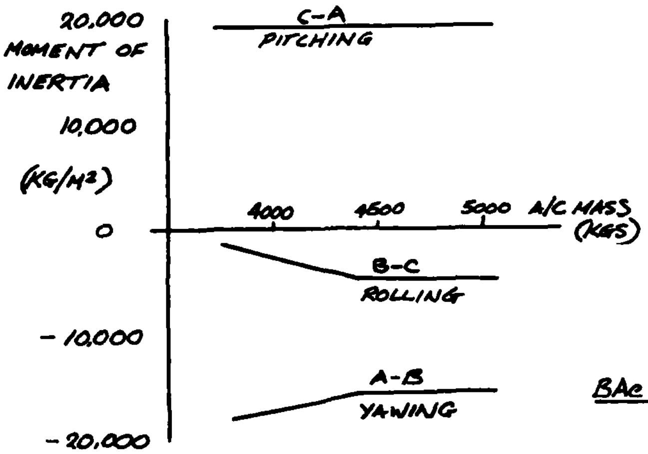


FIGURE 7



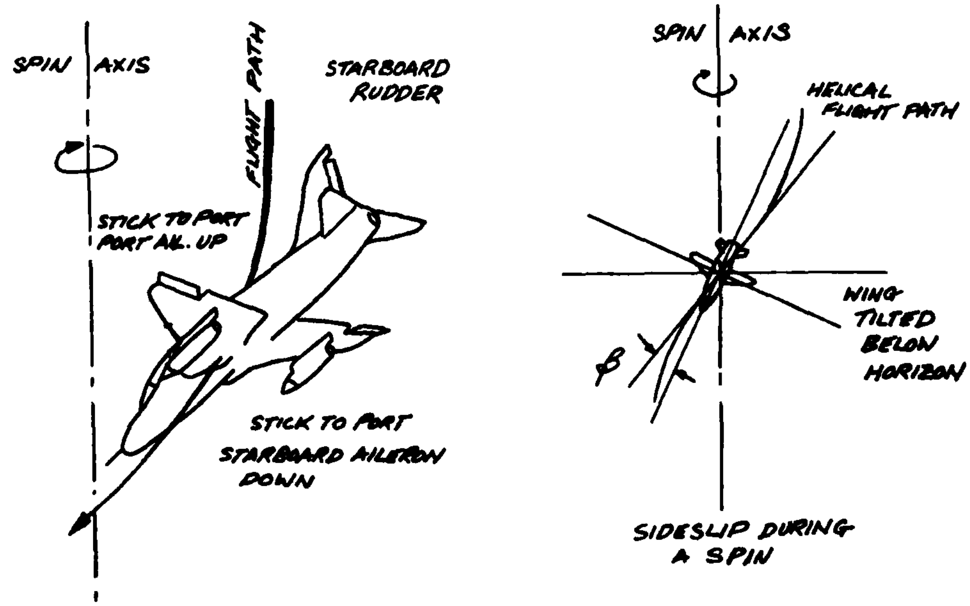
INERTIA ROLLING MOMENT $(B-C)qr$ = ANTI-SPIN
" YAWING " $(A-B)pq$ = ANTI-SPIN
" PITCHING " $(C-A)pr$ = PRO-SPIN

NOTE:- IN A RIGHT HAND SPIN
p, q, r ARE +VE

BAe HAWK INERTIA DIFFERENCES VS AIRCRAFT MASS

ORIENTATION OF AN AIRCRAFT IN A POSITIVE SPIN
(OUTSPIN AILERON AND PROSPIN RUDDER APPLIED)

FIGURE 8



FLIGHT RESEARCH TECHNIQUES UTILIZING REMOTELY PILOTED RESEARCH VEHICLES

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SUMMARY

This lecture presents a survey of the use of the remotely piloted research vehicle (RPRV) in aeronautical research. The paper emphasizes the flight test experience that has been acquired at the NASA Dryden Flight Research Center with several types of RPRV's, including those with a pilot in the loop, a concept developed at Dryden. The paper discusses the application of RPRV's to various test objectives; the approaches utilized range from the simplest and least expensive of vehicles, such as the Minisniffer, to the very sophisticated and complex highly maneuverable aircraft technology (HiMAT) RPRV.

The advantages and disadvantages of RPRV's are discussed, as well as safety considerations. The ground rules set early in a program can profoundly affect program cost effectiveness and timeliness.

1.0 INTRODUCTION

Experiments in remotely piloted vehicle (RPV) flight testing began at the NASA Dryden Flight Research Center approximately 10 years ago. Those early tests started by adding a test pilot and digital computer elements to state-of-the-art drone technology. Each succeeding RPV program made greater use of these elements in meeting research objectives.

The RPRV (the word research was first added to RPV in Ref (1)) became increasingly popular with NASA engineers and program managers because of its greater flexibility and because they had greater control over what the aircraft did in flight. This control was achieved from an aircraft-type cockpit that was on the ground and incorporated a full instrument flight rules (IFR) panel, a forward-looking TV, and variable stick-force gradients. A programmable ground computer functioned as a part of an experimental aircraft control system. The RPRV was less popular with NASA test pilots, on the other hand, because they had fewer opportunities to fly RPRV's and because more simulation time was necessary to prepare for flying RPRV's. The pilots' skill and knowledge were often quite highly taxed in order to successfully complete an RPRV mission.

In the meantime, arguments in favor of the RPRV met with success in the promotion of new programs.

We originally put the test pilot on the ground and an RPRV in the air because the cost and time necessary to develop new research aircraft the conventional way had become prohibitive. Much more ground system, structures, and wind tunnel testing goes into today's aircraft: in 1950, a new aircraft underwent an average of 1200 hours of wind tunnel testing; in 1970, approximately 12,000. The additional wind tunnel test time is due partially to the sophistication of the aircraft, which makes it difficult to duplicate configuration perturbations and flight conditions accurately in the systems, structures, and materials ground tests. The result is that many additional hours of wind tunnel facilities are necessary to give confidence in the data. And the enormous investment involved was discouraging the research community from making bold moves into new technology.

We at Dryden found the RPRV attractive because it built confidence in new technology by demonstrating its capabilities in the real and dynamic environment of flight. Use of RPRV's seemed especially advantageous because it permitted testing to be done at low cost, in quick response to demand, and at no risk to the pilot.

The RPRV has the potential for low cost because of its smaller size, lack of life support systems, and lower requirement for redundant systems. The quick response time and reduced cost result from the elimination of many manrating tests and from the ability to use simple and modifiable structures. The use of programmable ground-based control systems also provides quick response, as well as flexibility. Finally, hazardous testing is possible because the vehicles may be considered expendable or semiexpendable.

The RPRV differs from the military drone or RPV in that it gives a test pilot exactly the same responsibilities and tasks as if he were sitting in a cockpit on board a research airplane. As in manned flight testing, the pilot has complete responsibility for performing data maneuvers, evaluating vehicle and systems performance, and determining the appropriate action to take in emergencies or if the aircraft does not respond as expected.

The mission of a military RPV, on the other hand, is so distinct that an autopilot can be programed. The craft's aerodynamic performance is accurately defined in extensive wind tunnel and flight testing, and the design of the autopilot is based on these tests. Of course, the flexibility of the autopilot is limited to certain routines, such as cruise, 15° turns, and 30° turns. The controller fine adjusts the autopilot, or several autopilots at once, for he may have a whole formation under his guidance.

In contrast, the RPRV is designed to venture into unexplored engineering territory. It does not perform a stereotyped routine, and part of its mission is to explore the aerodynamic performance of the vehicle. Versatility is necessary for this type of testing, and a pilot is the most versatile system we have. Completely responsible for vehicle control, a pilot can handle only one RPRV at a time. Versatility proved to be a significant selling point for RPRV's.

"Sending more commands up to the vehicle and getting a large quantity of high quality data back constantly with no dropouts or glitches takes broader and more reliable radio-link bands than for an RPV. The RPRV control system would be vulnerable to electronic countermeasures, thus would be unsuitable in a military situation." Experience is always the best teacher. Six years of RPRV flight experience at Dryden have been logged since the words above appeared in Reference (1). The purpose of this lecture is to pass on the lessons learned in RPRV flight testing during that time.

2.0 RPRV FACILITY

The RPRV has in its control loop a powerful ground-based digital computer (Fig 2.0-1). Programing the computer substitutes for the expensive building of new control system design features into the vehicle itself. The computer, located in a ground-based RPRV facility along with a ground cockpit, serves as part of the RPRV simulator as well. Unlike a manned aircraft control system, it can be used in several successive vehicles.

3.0 NASA RPRV PROGRAMS

Figure 3.0-1 illustrates the eight RPRV programs that have been conducted at the Dryden Flight Research Center since 1969. The Big G Parawing and Minisniffer vehicles were operated more like conventional drones and were not considered true RPRV's.

Tables 3.0-1 and 3.0-2 summarize the objectives and characteristics of these RPRV programs. Hardware qualification (Table 3.0-1) signifies the testing of experimental system components intended for use in follow-on programs. From Table 3.0-2, it is obvious that the scope and cost of the RPRV programs vary widely. For example, the very limited scope of the Big G Parawing program permitted its cost to be orders of magnitude less than that of the highly maneuverable aircraft technology (HiMAT) program. The message is that RPRV programs can be designed to match facilities, funds, and personnel to the resources available.

A brief description of each RPRV program follows.

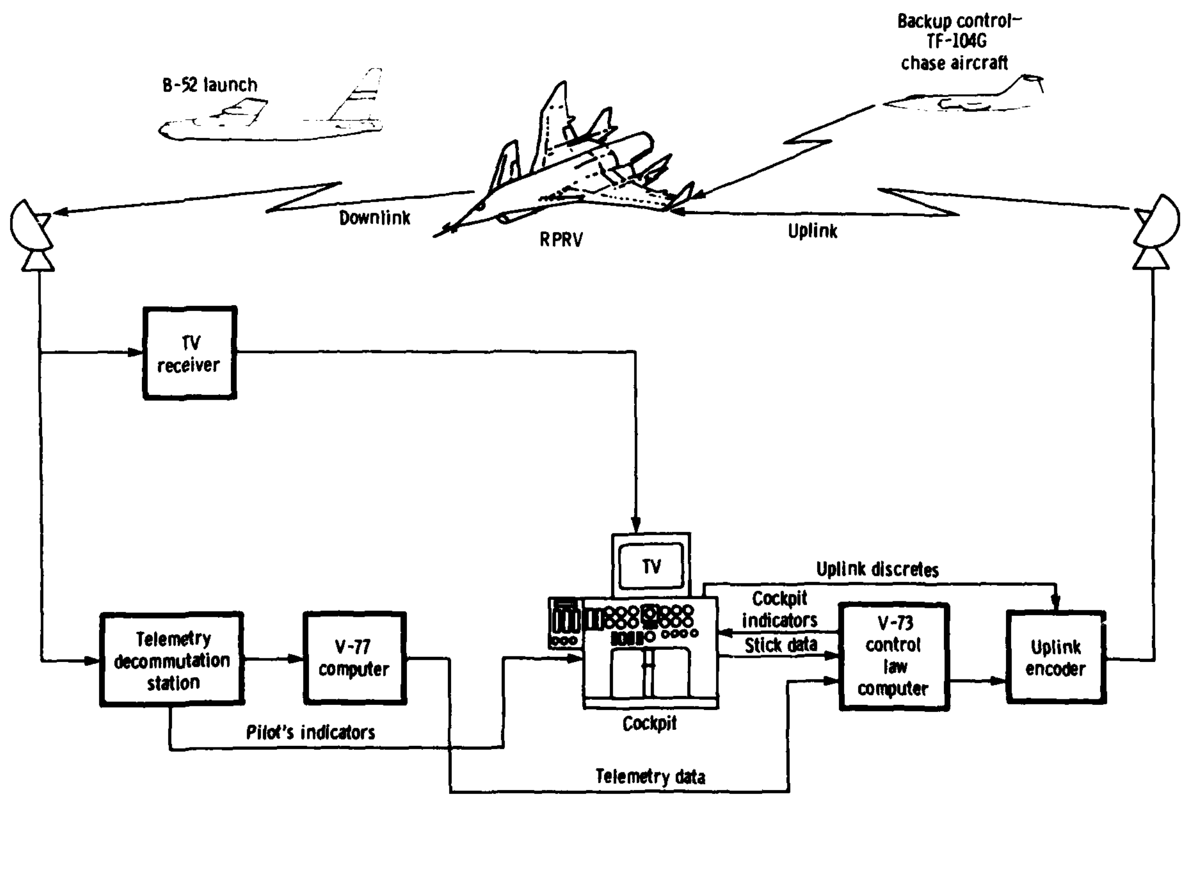

Fig 2.0-1 RPRV control system.

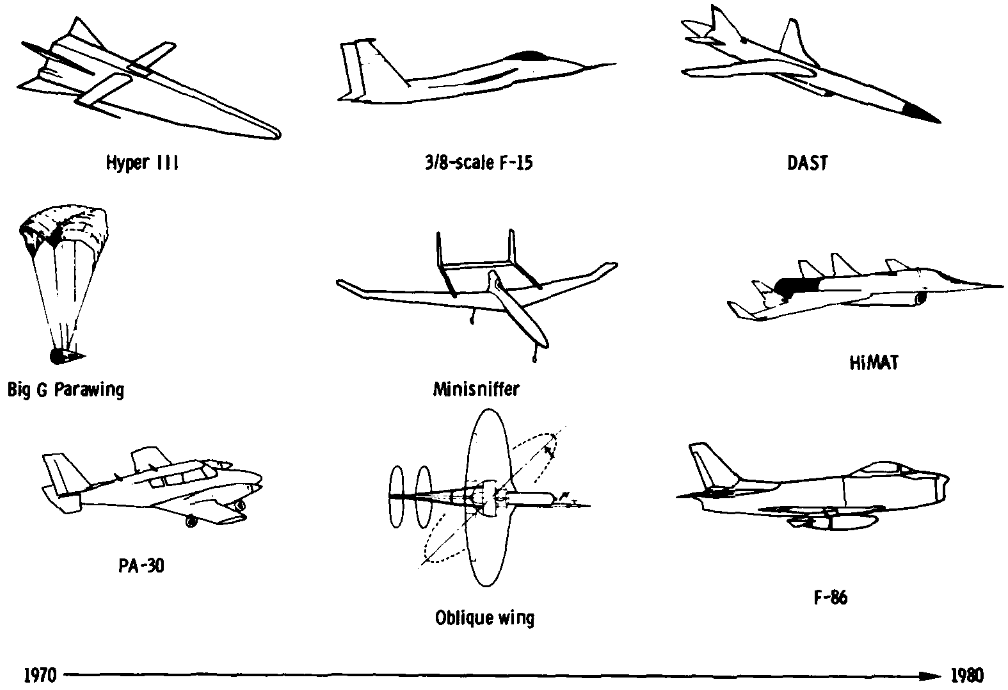

Fig 3.0-1 NASA RPRV programs.

Table 3.0-1 RPRV Program Objectives

RPRV program	Objectives					
	Basic research			Proof of concept		Hardware qualification
	Aerodynamics	Structures	Propulsion	System demonstration	Aircraft configuration	
Big G Parawing	---	---	---	---	X	X
Hyper III	X	---	---	X	X	---
PA-30	---	---	---	X	---	X
3/8-scale F-15	X	---	---	X	X	---
Minisniffer	X	X	X	X	X	X
Oblique wing	X	---	---	X	X	---
DAST[1]	X	X	---	X	X	---
HiMAT[2]	X	X	X	X	X	---
F-86	X	---	---	---	---	---

[1]Drones for aerodynamic and structural testing.

[2]Highly maneuverable aircraft technology.

Table 3.0-2 RPRV Aircraft Information

RPRV vehicle	Propulsion	Parachute equipped?	Type of control system	Vehicle weight, kg	Structure	Number of data parameters	Vehicle cost, dollars
Big G Parawing	None	Yes	Direct, electric	270	Steel tube, aluminum	12	2000 (2 vehicles)
Hyper III	None	Yes	Direct, hydraulic	450	Steel tube, fabric, fiber glass, aluminum sheet	16	5000 (1 vehicle)
PA-30	Propeller	No	Direct, hydraulic, ground computer	1600	Aluminum	35	Available for use
3/8-scale F-15	None	Yes	Direct, hydraulic, ground computer	1100	Wood, foam, fiber glass	77	1,000,000 (2.5 vehicles)
Minisniffer	Propeller	No	Direct, electric, wings leveler	100	Wood, foam, fiber glass, Kevlar	17	100,000 (2 vehicles)
Oblique wing	Ducted propeller	No	Direct, electric	270	Wood, fabric, fiber glass	16	200,000 (1 vehicle)
DAST	Jet	Yes	Direct, hydraulic, autopilot	950	Aluminum, fiber glass	120	500,000 (1 vehicle)
HiMAT	Jet	No	Ground computer, onboard computer, programer, hydraulic	1360	Composite: carbon, Kevlar, aluminum, steel	450	17,300,000 (2 vehicles)
F-86	Jet	No	Direct, hydraulic, SAS[1]	8400	Aluminum	17	Surplus

[1]Stability augmentation system.

3.1 Big G Parawing

The Big G Parawing program was initiated to explore the piloting problems involved in steering a limp-parawing spacecraft configuration to a precision landing on the ground. In 1967, the NASA Johnson Space Center was seriously considering the development of a large version of the Gemini spacecraft that would be capable of returning 12 astronauts to a landing on earth by means of a gliding parachute.

The Big G Parawing program at Dryden had two phases. The first, an RPRV phase, was intended to qualify the parawing system, the structure, and the pilot control system, as well as to measure the loads imposed on an anthropomorphic dummy (Fig 3.1-1) during parawing deployment and ground contact (Ref (2)). In the second phase, the anthropomorphic dummy was to be replaced by a test pilot (Fig 3.1-2) to explore the piloting problems involved in steering the craft to a landing while looking through a viewing port similar to that in the Gemini spacecraft. Forty successful RPRV flights were conducted from 3000 meter drops to precision landings by a visual pilot (a pilot watching from the ground) using a model airplane transmitter. The second phase of the program was cancelled when NASA decided to abandon the drop concept in favor of the horizontal landing (shuttle) concept.

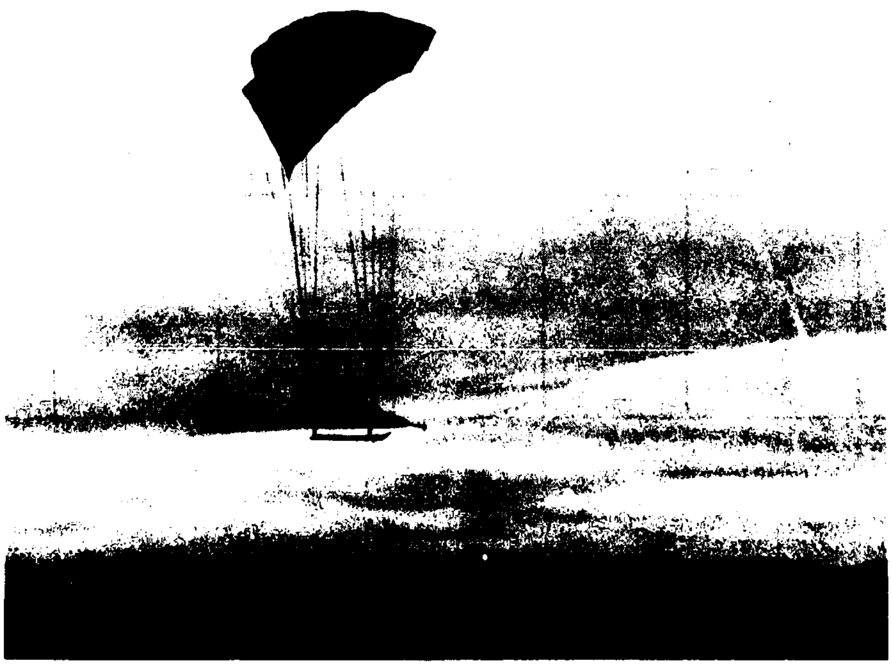

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Fig 3.1-1 Parawing test vehicle.

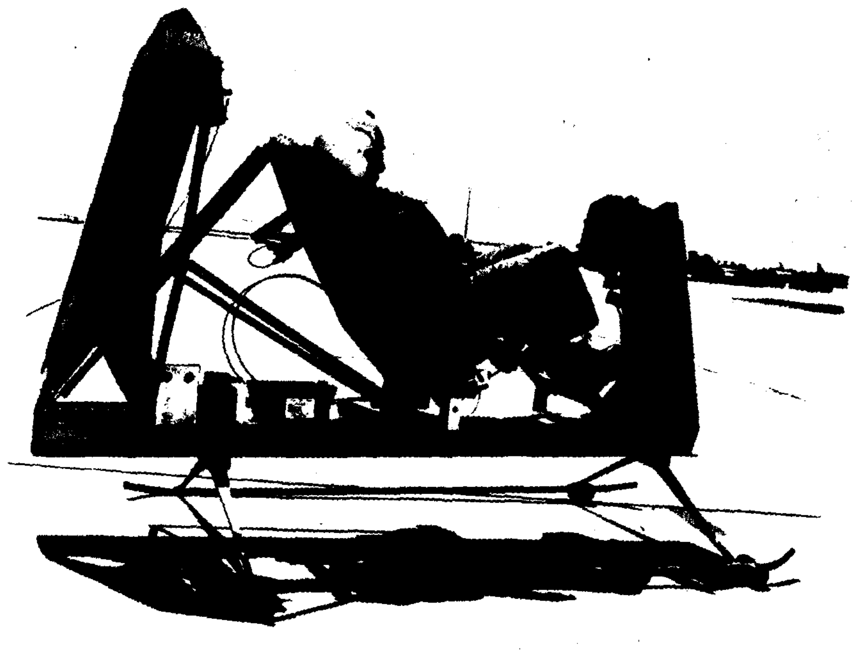

Fig 3.1-2 RPRV test vehicle with anthropomorphic dummy.

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3.2 Hyper III

The objective of the Hyper III program (Ref (3)) was to acquire flight data and investigate the unaugmented subsonic flying characteristics of a reentry spacecraft capable of flight at hypersonic speeds and at high lift-to-drag (L/D) ratios (L/D = 3). The spacecraft utilized a deployable single-piece skewed wing (Fig 3.2-1). The vehicle was towed by helicopter to 3000 meters, launched, and then flown down to about 200 meters by a pilot using instrument flight rules from a ground cockpit. At that altitude control was transferred to a visual pilot, who conducted an unpowered flare and landing. The biggest operational problem occurred in towed flight; the Hyper III had a tendency to make S turns below and behind the helicopter. The IFR pilot found that by experimenting he could damp out these turns by referring to the attitude indicator in the ground cockpit and making the appropriate control inputs. The rest of the flight went according to plan as practiced on the simulator for the IFR pilot and as practiced on speed-scaled radio-control models by the visual pilot.

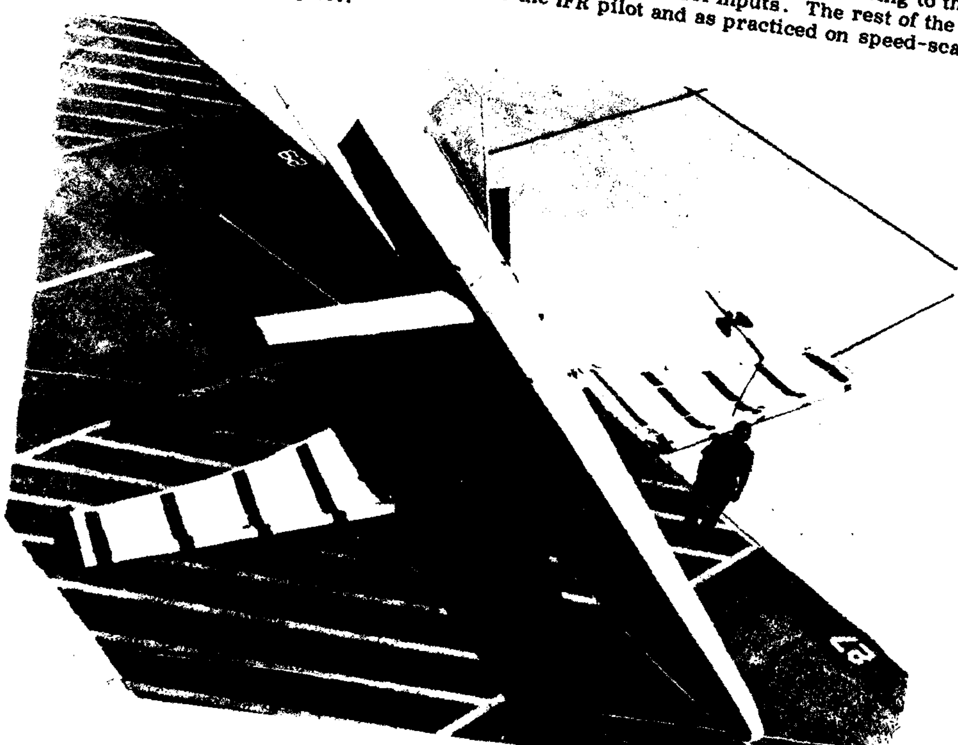

Fig 3.2-1 Hyper III with skewed wing installed and experimental folding flex-wing on ground.

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3.3 PA-30

The control system for the light twin-engine PA-30 airplane (Fig 3.3-1) was originally developed at the NASA Dryden Flight Research Center for experimental control systems work. The controls on the left seat are rigged electrically for fly-by-wire control through a hydraulic control system. The control system commanded from the right seat is the basic mechanical system, and it employs a safety cutout system so the safety pilot can take immediate control of the aircraft at any time. The downlink data transmission system in the aircraft made it natural to install a control system uplink and a TV downlink for RPRV development and research. The PA-30 has been used to develop several RPRV operational concepts, including the ground-based computer control system (Ref (4)) and automatic backup landing system for the HiMAT vehicle. The aircraft is never flown without a safety pilot on board.

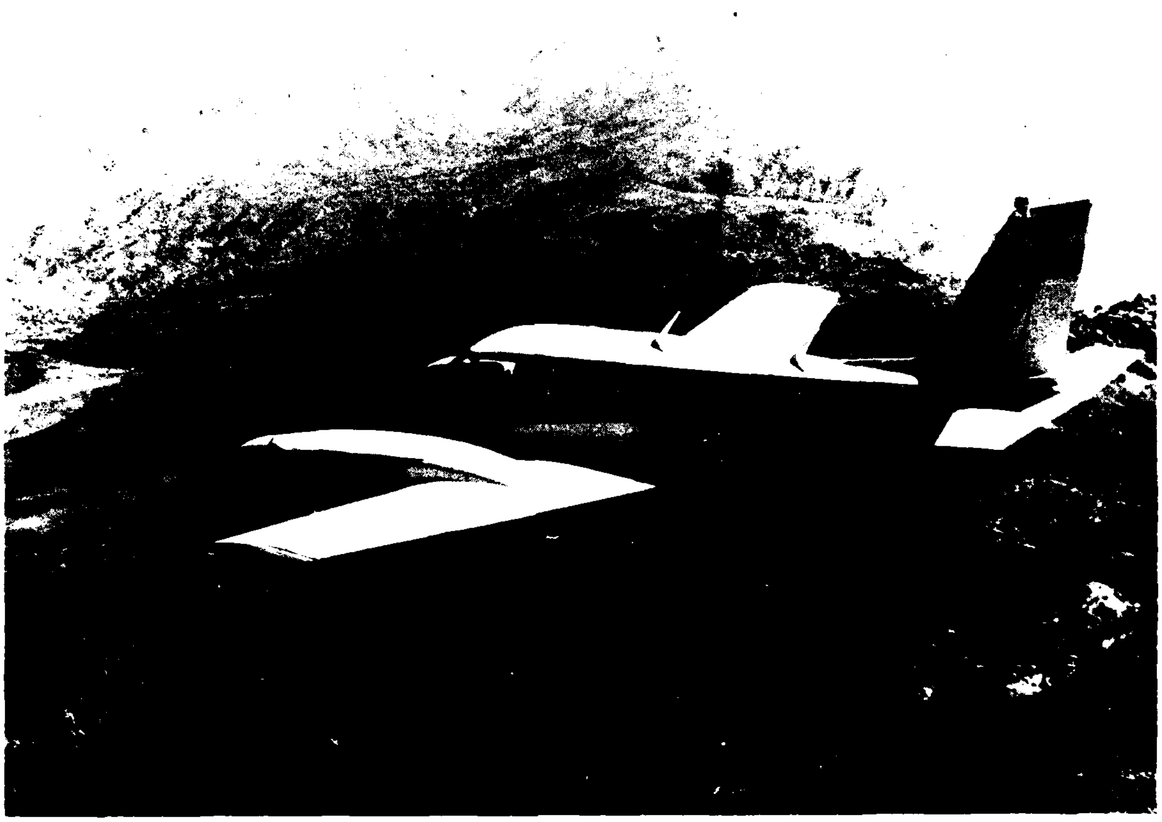

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Fig 3.3-1 PA-30 airplane used for RPRV development and simulation.

3.4 3/8-Scale F-15

The objective of the 3/8-scale F-15 program (Fig 3.4-1) was to explore the aerodynamic and control system characteristics of the F-15 aircraft in spins and high-angle-of-attack flight. The program was designed to make maximum use of existing equipment at the Dryden Flight Research Center. For example, hydraulic, gyro, and telemetry systems available from the retired lifting body programs were used for the aircraft's control systems. The proportional uplink then being used by researchers for transmitting radar data to pilot director instruments on board aircraft for curved instrument landing system (ILS) experiments was incorporated in the 3/8-scale F-15 aircraft for uplink control. Ground data processing computers were also pressed into service for the programmable ground-based control system. A general purpose simulator cockpit being used for stability and control studies was utilized for the RPRV pilot control station. A mid-air recovery system (MARS) Firebee II parachute system was utilized for vehicle recovery during the first flights. Later flights utilized horizontal landing for recovery.

A contract was let to construct three models to contain the NASA-supplied equipment.

The complete familiarity of the NASA crew with the aircraft equipment, combined with easy access to the equipment and uncrowded space inside the vehicle to work on the equipment made the operation of this vehicle relatively straightforward. In total, 35 launches were made from 14,000 meters. Data were acquired to explore the effects of different nose shapes and aerodynamic devices as well as the effects of various control system schemes on vehicle spin characteristics (Refs (5) to (7)).

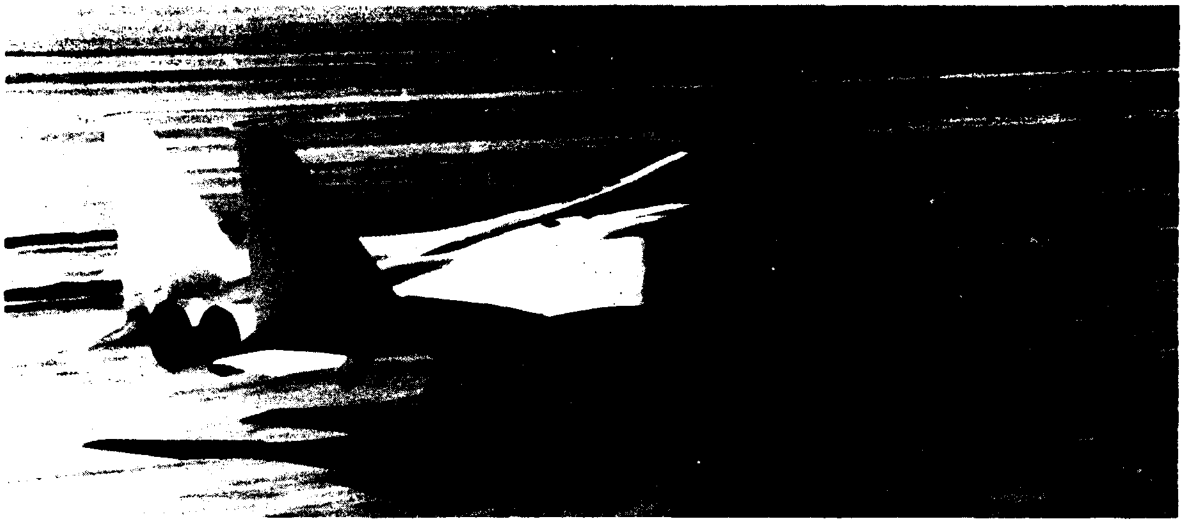

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Fig 3.4-1 3/8-scale F-15 being guided to horizontal landing by RPRV TV link.

3.5 Minisniffer

The Minisniffer program, which began in 1975, was initiated to develop a small unmanned atmospheric survey aircraft capable of sensing turbulence and of measuring both natural and man-made atmospheric pollutants at altitudes up to 27,000 meters (Ref (8)). The vehicle was to be able to fly at low speeds and to be able to maneuver precisely at stratospheric altitudes to conduct atmospheric research on a routine basis. The design missions called for the vehicle to carry an 11 kilogram air sampler to 21,000 meters and to cruise at that altitude for an hour over a range of about 320 kilometers, or to glide back from a 27,000 meter climb.

An essential element in the Minisniffer concept was the development of a reciprocating monopropellent hydrazine engine to drive a large, slowly turning propeller. The NASA Johnson Space Center took responsibility for the development of the hydrazine engine; Dryden was responsible for the development of the complete system. While Johnson worked on the hydrazine engine (Fig 3.5-1), Dryden built the Minisniffer with conventional gasoline propulsion for early flights (Fig 3.5-2) so work could proceed on the vehicle's aerodynamics, structure, and guidance and control systems.

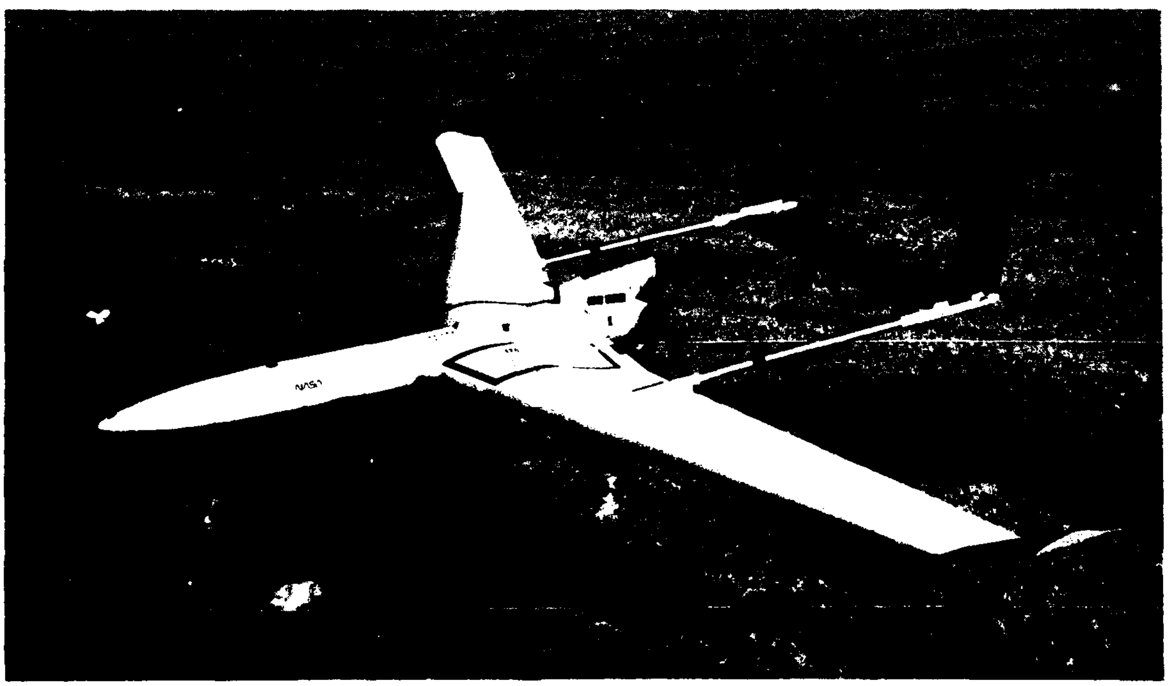

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Fig 3.5-1 Minisniffer with hydrazine propulsion.

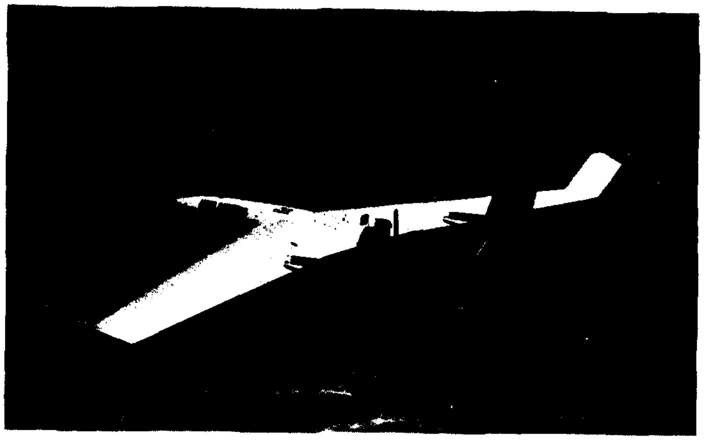

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Fig 3.5-2 Minisniffer with gasoline propulsion.

The first flight tests were conducted with model aircraft radio control systems. Actuators were doubled up when greater hinge moments demanded it. Later, a special lightweight radio control system characterized by longevity and high reliability at high altitudes was developed. A simple wings-leveler system was found to be necessary in turbulent air. A yaw-rate gyro drove the rudders that served as the wings-leveler system. The system worked through dihedral effect at all altitudes and was designed to serve as a Dutch-roll yaw damper at altitudes above 15,200 meters. The hydrazine engine was demonstrated in flight to 6000 meters with a fixed-pitch propeller. However, program funds were not available to develop the variable-pitch propeller needed to climb to higher altitudes.

A later study with the NASA Jet Propulsion Laboratory showed that the Minisniffer could perform well in the rarified carbon dioxide atmosphere of the planet Mars. The 38 percent earth gravity on Mars makes less horsepower necessary for flight, giving the hydrazine-propelled Minisniffer a theoretical 8000 kilometer range over the Mars surface. Use of a Minisniffer-type vehicle for Mars exploration is still under consideration.

3.6 Oblique Wing

The NASA Ames Research Center decided to investigate the oblique wing concept primarily because of its potential for enhancing transonic cruise efficiency. The idea is to position the straight-across wing at right angles to the fuselage for takeoff and landing and to swing one wingtip forward for cruise flight. Wind tunnel data acquired in the Ames wind tunnels indicated that an oblique wing configuration might have lower drag than, for example, variable sweep wings as well as have lower sonic-boom potential on the ground track.

After a radio-controlled model was built and flown by Robert T. Jones at Ames, the Ames engineers devised an RPRV program to further demonstrate the configuration. A contract was awarded for the design and development of a subsonic-only oblique-wing RPRV. The vehicle was to be capable of flight with no tail or with minimum tail so the aircraft could be made as compact as possible. A duct around the propeller was part of the overall structural and aerodynamic scheme.

The resulting vehicle (Fig 3.6-1) was turned over to Dryden for flight testing. The flight test program was short and simple, with a small team of Ames and Dryden personnel working together to

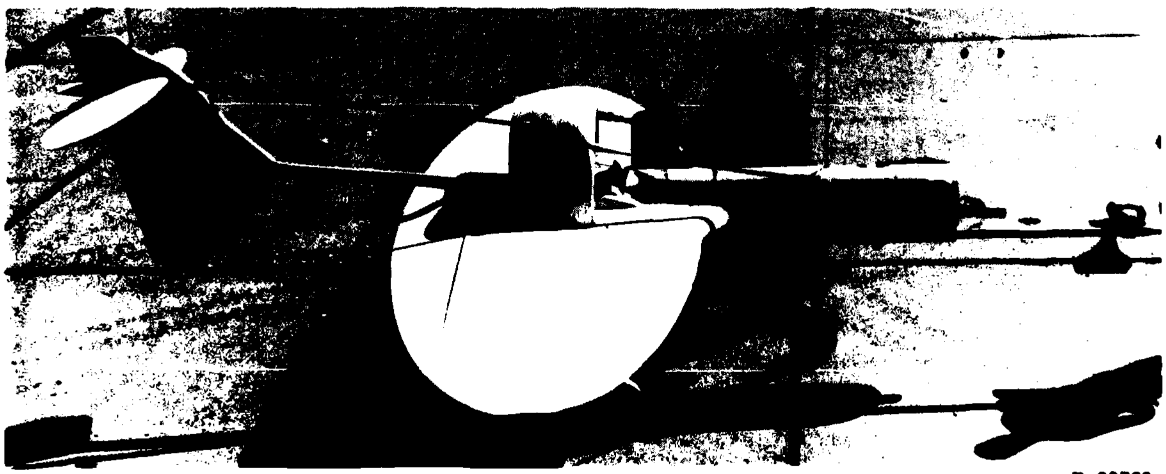

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Fig 3.6-1 Oblique wing RPRV.

acquire stability and control data (Ref (9)). Three flights were made, and the wing was placed at angles up to 45°. The same model airplane uplink system used in the early phases of the Minisniffer program was used to control the vehicle; however, bigger electric actuators were required.

3.7 Drones for aerodynamic and structural testing

The drones for aerodynamic and structural testing (DAST) program was designed to test large-scale models of wings designed for high efficiency cruise in flight at transonic speeds in combination with experimental flutter suppression systems. The high fuselage fineness ratio and supersonic capability of the Firebee II drone made an ideal testbed for these experimental wings. A Firebee II drone loaned to NASA by the Air Force was equipped with an experimental wing (Fig 3.7-1) at the NASA Langley Research Center. The drone was modified at the NASA Dryden Flight Research Center to incorporate an RPRV flight control system in which the test pilot has direct control over the maneuvers performed with the aircraft. The DAST I wing has a Whitcomb supercritical airfoil section and incorporates small aileron-like surfaces controlled by an electronic-hydraulic flutter suppression system. The DAST I vehicle is intended to be flown beyond the flutter boundary of the wing in order to demonstrate the effectiveness of the flutter suppression system. A parachute system is available to recover the aircraft in case of an unpredicted failure of the wing.

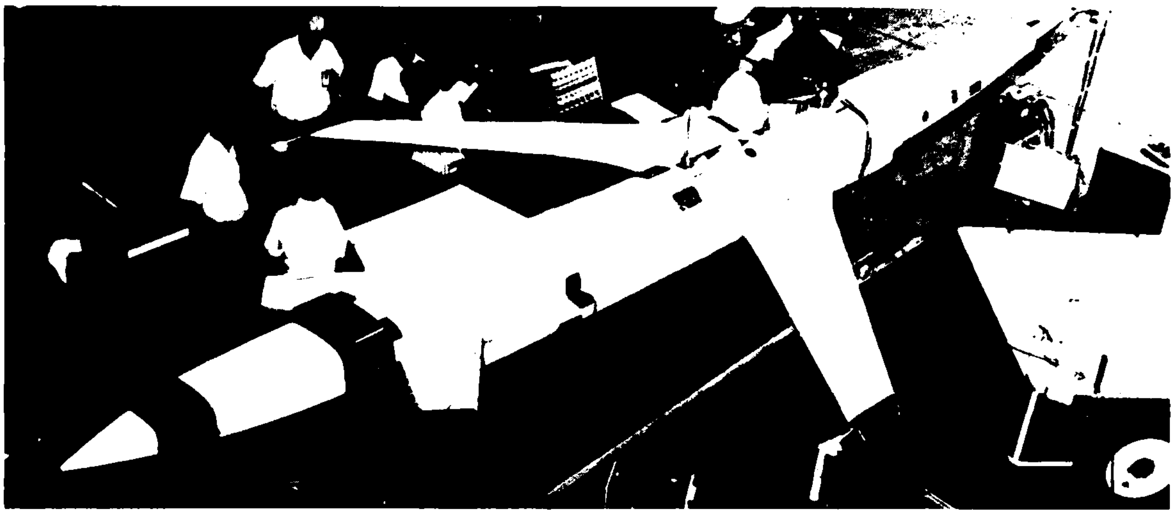

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Fig 3.7-1 DAST RPRV with experimental wing.

3.8 Highly maneuverable aircraft technology

The HiMAT project utilizes a 44 percent-scale model of a 7700 kilogram fighter. It has a wingspan of just over 4.6 meters and a length of 6.9 meters. It was designed to be air launched from a B-52 airplane, and it should be capable of speeds in excess of Mach 1.5. Two of the research vehicles have been built.

The HiMAT RPRV (Fig 3.8-1) is an experimental vehicle in which a synergistic approach is being used to accelerate the development of a new fighter aircraft (Refs (10) to (12)). This approach involves combining many new high-risk technologies into one vehicle to provide information on the interaction between the systems. One of the technological advances incorporated in the HiMAT vehicle is the composite material used for approximately 30 percent of its construction. In addition to weight savings, the composite material allows the wings and canards to be aeroelastically tailored for increased maneuverability and performance. Aeroelastic tailoring uses the unique directional properties of the graphite composite materials to control bending and twisting under aerodynamic loading. In the process of manufacturing the composite, the fibers in the material are oriented in the direction that results in favorable wing twisting as aerodynamic loading increases. The HiMAT's composite wing can be compared roughly to a wood veneer that is stiff in one direction but pliable in another. Under g stresses, the composite structure deforms enough to give the vehicle about 10 percent additional maneuvering capability, even in very tight turns

The HiMAT control system is of the digital fly-by-wire type, which is lighter in weight than a conventional control system. Pilot commands are fed via telemetry to an onboard computer, which sends electronic commands to the flight control surfaces. Another technology being tested is an integrated propulsion system. Instead of a conventional hydromechanical system, this system uses a digital computer to control the aircraft's entire propulsion system. The HiMAT vehicle is powered by a J58 jet engine. The research vehicle also incorporates active control technology that causes the flight control system to provide the aircraft's basic stability. Use of this technology saves weight and increases performance, since the size of the stabilizing surfaces can be reduced.

One of the design requirements for the HiMAT vehicle was that no single failure should permit the loss of the vehicle. Because of this design philosophy, dual systems were incorporated throughout the aircraft. This applied to the microprocessor computers, hydraulic and electric systems, servoactuators, uplink receivers and antennas, downlink transmitters, and antennas.

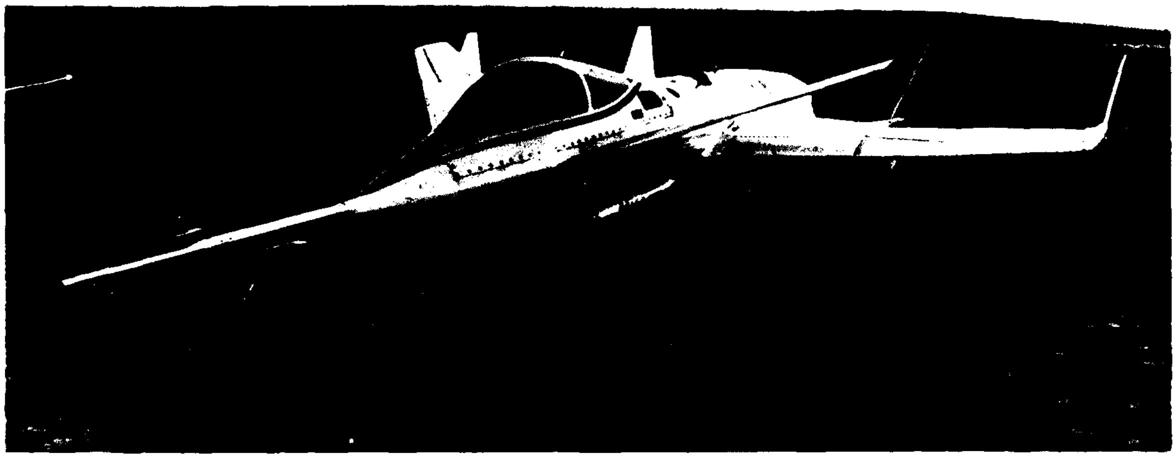

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Fig 3.8-1 HiMAT RPRV after landing on dry lake.

3.9 F-86

Dryden has been participating in a joint program with the Federal Aviation Administration (FAA) to minimize or eliminate the hazard to small aircraft of encountering the wingtip vortices generated by large jet aircraft. In this program, an instrumented manned T-37 jet trainer at Dryden had been used to probe the wake of a B-747 aircraft at altitude. It was deemed too dangerous to encounter these vortices close to the ground.

The Naval Weapons Center at China Lake was then asked to participate in tests using one of their F-86 RPV's (Fig 3.9-1). The Navy had developed an RPV target system around surplus F-86 fighter aircraft, using a system very similar to the one used for the 3/8-scale F-15 system. A surplus F-86 ground simulator is used in conjunction with a transmitted TV image, and the pilot controls the F-86 directly, with full aerobatic maneuvering capability.

A flight test program was then developed in which the F-86 RPRV was used to probe the visible wake of a B-747 (Fig 3.9-2) during landing and takeoff. Twenty-four encounters with B-747 wingtip vortices were made with the F-86 RPRV. The RPRV pilot prevented the F-86 from contacting the ground several times through his ability to respond quickly to aircraft upsets; the value of the RPRV technique was demonstrated through these tests alone, because the data could not have been acquired in any other way.

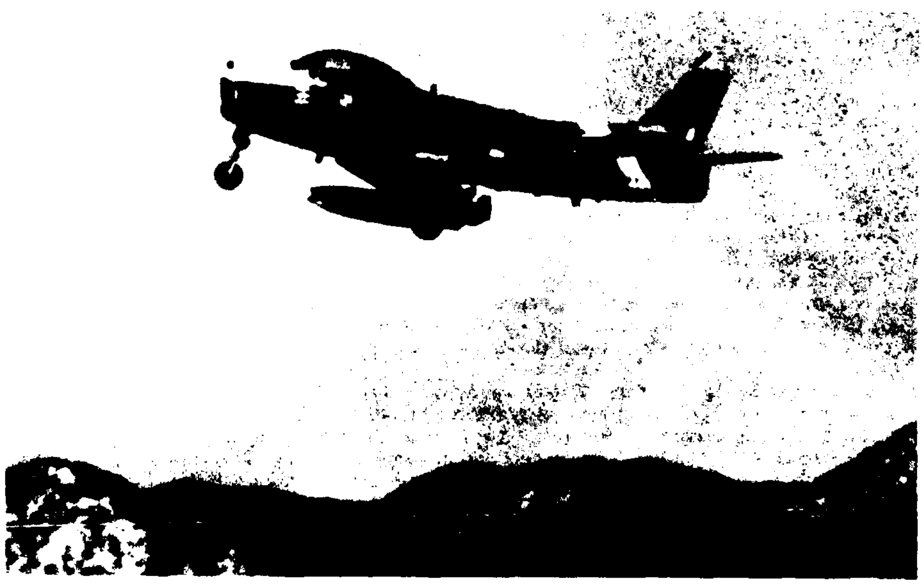

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Fig 3.9-1 F-86 RPV used to probe wingtip vortices generated by B-747.

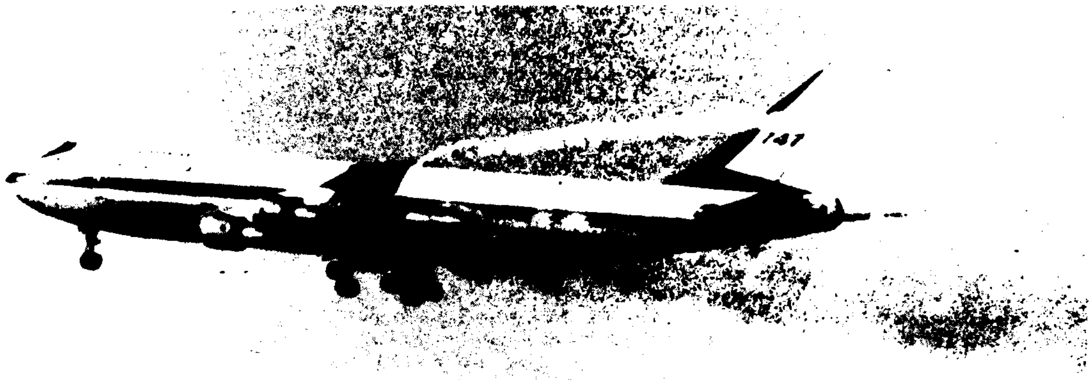

Fig 3.9-2 B-747 with wingtip vortices made visible by smoke generators.

4.0 RPRV OPERATIONAL FEATURES

Table 4.0-1 summarizes some of the operational features of the RPRV programs conducted at the Dryden Flight Research Center. The ground cockpit with downlinked TV is the most popular piloting technique. Piloting through visual contact with the radio-controlled airplane has been used only in small programs, where low cost and simplicity are of primary importance.

Table 4.0-1 RPRV Operational Features

RPRV vehicle	Launch technique	Recovery technique	Piloting technique
Big G Parawing	Helicopter drop	Gliding parachute	Visual, stationary
Hyper III	Helicopter tow	Horizontal landing skids, parachute backup	Ground cockpit, visual, stationary
PA-30	Horizontal takeoff	Horizontal landing	Ground cockpit (TV), safety pilot
3/8-scale F-15	B-52 drop	MARS and horizontal landing	Ground cockpit (TV)
Minisniffer	Horizontal takeoff	Horizontal landing	Visual car chase, radar, TV
Oblique wing	Horizontal takeoff	Horizontal landing	Ground cockpit (TV)
DAST	B-52 drop	MARS	Ground cockpit, F-104 chase
HiMAT	B-52 drop	Horizontal landing	Ground cockpit (TV), F-104 chase, automatic
F-86	Horizontal takeoff	Horizontal landing	Ground cockpit (TV), T-33 chase

The B-52 airplane is the vehicle currently being used to launch RPRV aircraft at Dryden. Although it is much larger than necessary to carry RPRV's aloft, it has proven to be cost effective in the manner in which it is being used: a low launch frequency (one 2-hour flight per month for the DAST and HiMAT programs) reduces its operating costs to a small proportion of the total program operating cost.

The midair recovery system was used for the 3/8-scale F-15 and DAST RPRV's. This recovery technique was chosen because the Edwards Air Force Base 6514th Test Squadron was willing and able to furnish a helicopter and crew to capture the RPRV's in the parachute MARS mode. A high degree of crew proficiency and skill is necessary to make consistently successful captures. The Air Force MARS crew maintains proficiency by practicing with dummy payloads and by retrieving Air Force drones and cruise missiles. It would not be cost effective for NASA to maintain such a capability for its limited number of RPRV's.

5.0 SAFETY CONSIDERATIONS

Each RPRV program has its own set of circumstances and approaches to safety. There are some basic guidelines that are consistently followed, however. The most fundamental of these is that the safety of the people in the launch aircraft and on the ground has priority over the preservation of the RPRV.

The ground rules set up for a particular RPRV program may specify that the vehicle is expendable or semiexpendable. These guidelines are followed when the acquisition of data or flight results under conditions that put the vehicle at risk is considered more important than the loss of the vehicle. Under these circumstances, if all of the data objectives are achieved in one flight and the vehicle crashes at the end of the flight, the program is still considered successful. The ground rules set up for the oblique wing program were similar to this. However, all three planned flights were flown without vehicle loss.

The ground rules for the oblique wing were set up in this way because personnel safety and data objectives did not require vehicle recovery and because the program's cost and time constraints did not permit the precautions that would ensure the safety of the vehicle. Personnel safety could be ensured because the oblique wing was flown within the airspace over Edwards Air Force Base, an airspace from which commercial air traffic is excluded. Further, there are no homes or businesses in the region where the flight tests were conducted. The vehicle was at risk because it had no autopilot functions and was basically spirally unstable. It had no redundant systems whatsoever, not even a backup visual pilot in case the IFR pilot in the cockpit using downlink TV lost control of the vehicle. Thus, many system failures would necessarily have resulted in a crash. However, as planned, the lack of redundancy permitted the flight test program to be conducted at low cost and in a timely manner.

In direct contrast, nothing was spared to avoid the loss of the HiMAT vehicle. The design philosophy for HiMAT was "No single failure shall cause the loss of the vehicle." This project ground rule, of course, did not take priority over personnel safety. The foremost safety concern in any of the B-52 launch operations is the safety of the B-52 air crew in a possible collision between the RPRV and the B-52 after launch. Launch dynamics are carefully analyzed for every air-launch vehicle. Much thought goes into the operational schemes to ensure a clean launch, especially in the case of unstable vehicles, and precautions vary from locking the controls to installing jettisonable RPRV nose ballast.

6.0 INSTRUMENT FLIGHT RULES PILOT TRAINING

6.1 Cockpit characteristics

For RPRV's, most vehicle control is done from a ground cockpit not unlike a ground-based simulator cockpit. The cockpit contains aircraft controls and IFR instruments that are tied into the RPRV through a data link and tracking system. Ground computers are used with the more sophisticated RPRV's (Fig 2.0-1). The cockpit is equipped with a TV screen only if horizontal takeoffs or landings are planned.

At Dryden, RPRV's are usually flown from cockpits located in the so-called RPRV facility, an area in the main building. However, on occasion cockpits have also been installed in vans or trailers near the flight testing in order to shorten the radio range for landing and takeoff operations. The Hyper III and oblique wing cockpits were portable and used at the remote landing sites (Fig 6.1-1).

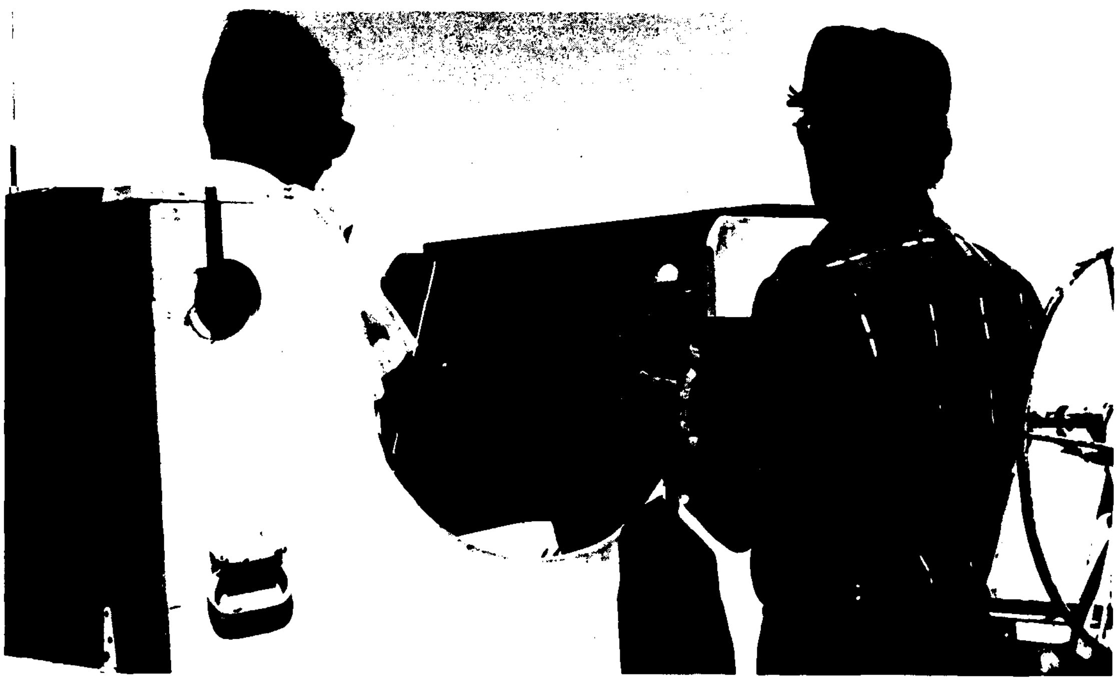

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Fig 6.1-1 Portable outdoor Hyper III cockpit with IFR and visual pilots in place.

6.2 Ground support crew

A flight test engineer often sits next to the cockpit (Fig 6.2-1) to assist the pilot in reading checklists, timing maneuvers, and setting up vehicle control configurations through the ground computer systems. Making up a third member of the team is the flight controller in the Dryden control room. The flight controller is in charge of the operation, and all of the operational information is available to him on plot boards, including vehicle telemetry data and radar tracking information. The controller is always in direct contact with the RPRV pilot, by hard wire if the cockpit is in the RPRV facility or by radio if the cockpit is at a remote site.

ECN 10925

Fig 6.2-1 HiMAT RPRV cockpit with pilot and flight test engineer at flight stations.

6.3 Iron bird simulation

A simulator cockpit identical to the RPRV cockpit is used for pilot training. In the case of the HiMAT vehicle, the signals from the actual RPRV cockpit are fed into a general purpose simulation computer. At some times, signals are fed into the actual (full-scale) aircraft, which is sitting in an adjacent hangar. When the actual airplane is hooked into the simulation (iron bird simulation), the flight crew, pilot, flight test engineer, and controller develop procedures and techniques for verifying aircraft computer software programing before flight to detect possible system failures during flight.

6.4 Training for ground controlled approaches

The controller and sometimes the flight test engineer may assume the responsibilities of a ground controlled approach (GCA) controller to steer the pilot through a landing pattern. Many hours of practice on the simulator involving the RPRV pilot, flight test engineer, and flight controller are necessary to develop flight plans, practice research maneuvers, and develop emergency procedures.

7.0 VISUAL PILOT TRAINING

Radio-controlled model investigations have been conducted at the Dryden Flight Research Center with models weighing less than 18 kilograms for the preliminary investigation of advanced concepts (Refs (13) to (15)), but they were not considered RPRV projects in and of themselves. Radio-controlled models did play a major support role, however, in providing the visual pilot with training for the larger scale Hyper III and Minisniffer RPRV's. The so-called mother ship (Fig 7.0-1), a large 3 meter span model

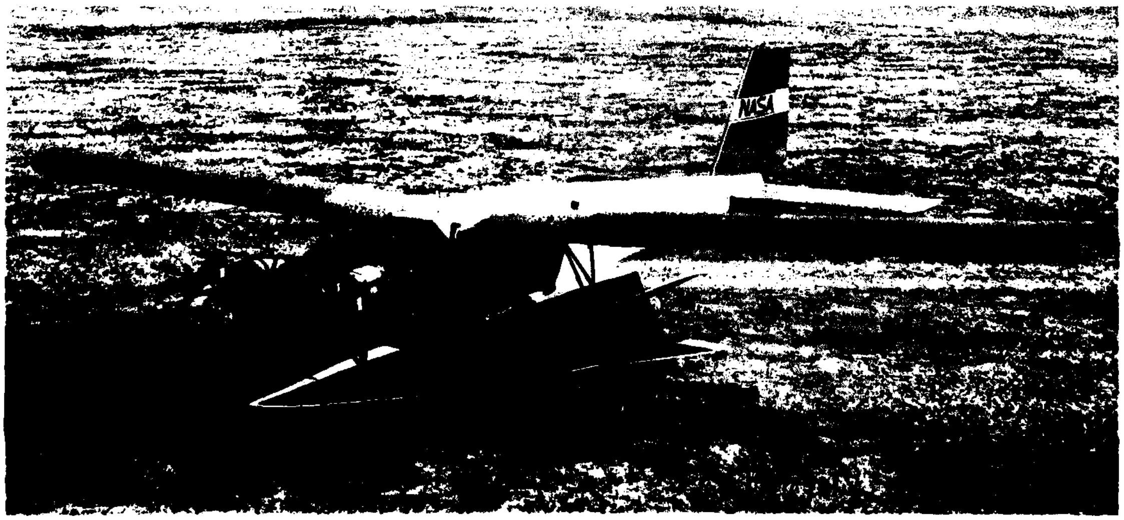

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Fig 7.0-1 Radio-controlled mother ship, a model used to launch experimental models and to develop RPRV techniques.

airplane originally used to launch model lifting bodies, was equipped with a vertical gyro for pitch and roll control, an airspeed indicator, and a radar beacon to develop control transfer techniques between the visual pilot and the IFR pilot.

A second radio-controlled model, a 1/6-scale model of the Hyper III that was weight scaled for speed (body lengths per second), provided an excellent training aid for the visual pilot, who had the task of landing the relatively low lift-to-drag-ratio ($L/D \approx 5$), unpowered, 450 kilogram, 10.7 meter Hyper III RPRV.

A similar 40 percent-scale model was used for visual pilot training for the Minisniffer RPRV.

8.0 LESSONS LEARNED IN RPRV PROGRAMS

8.1 Program planning

By definition, the purpose of the RPRV's is to acquire experimental data. It is important that research engineers spell out the data objectives for the program in as much detail as possible early in the program. The vehicle design and program operational scheme should be selected that achieve these objectives in the most cost effective way. Any revisions of the program to match available personnel, facilities, and funds should be made only after a preliminary vehicle design and operational scheme have been selected. This type of activity is also important in developing manned research aircraft. However, the impact on RPRV programs is greater because decisions on operational techniques (air launch versus ground launch, parachute recovery versus horizontal landings, and so forth) are highly dependent on research objectives.

8.2 Flexibility: an advantage of RPRV programs

More easily than most flight testing, an RPRV flight test program can be tailored to match available personnel, facilities, and funds. This is possible because one RPRV program may require only the simplest of vehicles (one that uses a drone-type control system, for example), whereas another RPRV program may require a much more elaborate vehicle, such as one with a control system that requires several control systems people, for the data objectives to be met. Almost every RPRV program at Dryden was an experiment in operations and was designed to match the available operational personnel and equipment.

8.3 Staffing

Assigning an operations engineer and crew chief early in the vehicle design phase prevents the loss of much time later in the program. These are the people who must make the vehicle work later, and they will make sure it will if they are able to make their needs known early in the vehicle design phase. This is true for any research aircraft, but it is no less important for RPRV's.

8.4 Simulation: a vital tool in RPRV programs

Because the IFR RPRV pilot lacks the motion cues, visibility cues, and sound cues that a test pilot sitting in a manned aircraft cockpit enjoys, he must work much harder to extract information from the cockpit instruments. Simulation is vital to RPRV programs for both systems development and pilot training. The more complex an RPRV is, the more simulation time is necessary. RPRV flights are usually planned in such a way as to extract as much data as possible from each flight because of the higher risk of vehicle loss. As a consequence, every minute of flight time is used to produce as much data as possible during the flight. Precise training for the maneuvers on a simulator is necessary to give this data return.

A small RPRV program such as that for the Minisniffer made use of a very simple and minimal simulation. However, the simulation proved to be very valuable in developing the wings-leveler yaw-damping system and in providing pilot training. Pilot training was especially important in that only a turn-rate and airspeed indicator could be used in yaw-damper-off data maneuvers.

8.5 Advantages of modular approach

Probably the most time-consuming effort in an RPRV program is the design, development, and ground testing of the special or newly developed systems required due to the small size of the RPRV. If known systems can be used, a structure can be sized and designed to utilize them much more quickly than if special systems must be developed to fit a particular structure. (The minimum size of the structure of a manned vehicle is dictated by the cockpit and life support systems.) A good analogy is wind tunnel testing. The wind tunnel itself, the wind tunnel measuring systems, and the data reduction systems are to wind tunnel personnel what already established RPRV module systems are to flight test personnel. The aircraft structure is analogous to wind tunnel test models.

Building an RPRV that has all-new systems as well as a new structure is similar to building a new wind tunnel test facility as well as a wind tunnel test model. The time necessary to accomplish the task increases accordingly.

8.6 The synergistic approach

The HiMAT is the RPRV in which the synergistic approach has been used. As of this writing, three flights have been conducted with the HiMAT vehicle. More flights must be made before the full potential of the concept can be demonstrated, with all systems working together to result in greater vehicle performance than can be provided by the sum of the individual systems.

9.0 ADVANTAGES AND DISADVANTAGES OF RPRV'S

In recommending the RPRV approach to a flight test program, it is easy to list the advantages of the approach over manned flight testing, such as the ability to take higher risks in flight and to eliminate man-rating tests. The RPRV approach has the potential for reducing costs. In the real world, however, some

of these benefits may fail to materialize. For example, many systems actually become manrated because of the danger of collision with manned launch aircraft and the fear of losing the RPRV.

The advantages and disadvantages of the RPRV approach to flight testing may be summarized as follows.

9.1 Potential advantages

The potential advantages of the RPRV approach are lower program cost because of smaller vehicle size, the elimination of manrated tests, and the elimination of life-support systems. Further, higher risks can be taken in RPRV's than in manned aircraft.

9.2 Disadvantages

The disadvantages of the RPRV approach are as follows. Higher program costs and time delays are often experienced as a result of the need to develop special miniature systems to fit into the limited space of the small aircraft. The limited space in small aircraft requires systems to be stacked, making work access during flight operations difficult. As program planning proceeds, very often extra redundancy or operating restrictions are imposed to ensure the safety of the people on the ground and in the launch aircraft. The up- and downlink communications are vulnerable to outside radio interference, which jeopardizes mission success. A large operational effort is required for crew training, the operation of tracking ranges and safety chase aircraft, and the preparation of the RPRV ground facility for each flight. In addition, line-of-sight range limitations restrict high-speed RPRV operations.

The successful operation of an RPRV requires a highly disciplined operational team and sometimes a very elaborate operational network. Many manhours must be expended in training exercises, dry runs, and briefings to ensure successful operation. As a result, the operational cost per flight of an RPRV can and very often does exceed that for an equivalent manned aircraft. However, if high data output per flight can be planned and if risk restrictions can be relaxed, RPRV operations can be cost effective.

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QUALITATIVE ARMAMENT SUBSYSTEM ASSESSMENT

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SUMMARY

Extensive and highly quantitative test and analysis is normally accomplished during the development of a weapon system. A qualitative system assessment is most appropriate for an Air Force contemplating purchase of a developed weapon system in providing first-hand knowledge of the system capabilities and evaluation of these capabilities against operational needs. This paper addresses a qualitative approach to armament subsystem test, the objectives which can be achieved and analysis methods for the major aircraft weapon subsystems. A substantial amount of information may be gathered to evaluate a weapon system without extensive or sophisticated instrumentation. Minimal aircraft and weapon range instrumentation requirements are detailed in the paper. Air-to-ground weapon delivery particularily lends itself to a quantitative statistical analysis for comparison to accuracy demonstrated in development and to that of other similar aircraft. Detail of this more quantitative analysis is included as appendix material.

INTRODUCTION

All major armament subsystems lend themselves to qualitative test and evaluation. This encompasses air-to-ground bombing in both visual and automatic modes, air-to-ground and air-to-air gunnery and missile subsystems. Aside from the major area of interest in weapon delivery accuracy and capability; supporting technical data, ground equipment, weapon loading techniques and aircraft turnaround for weapon sorties should be examined.

As mentioned earlier, a qualitative assessment of weapon delivery capability must be preceded by quantitative development test and evaluation. This phase of testing produces substantial amounts of data and documentation which should be thoroughly reviewed and understood prior to attempting qualitative assessment (references 1, 2, and 3). Specific documents for review include complete technical and operating handbooks for the aircraft and its subsystems, evaluation reports prepared by the airframe contractor and reports prepared by the original Air Force customer. These Air Force reports should include both implementing (or procuring) command (Air Force Systems Command in the U.S. Air Force) as well as operating command technical evaluations. The primary objectives of this review allow:

1. Understanding of how the weapon subsystems were designed to operate including primary modes, degraded (or backup) modes and any airspeed/altitude or g limiting envelopes.

2. Determination of problems encountered during development and corrections applied to resolve them.

3. Determination of specification compliance including any shortfalls. Previously encountered problems or specification shortfalls might prove to be valid indicators of where qualitative tests might be concentrated for optimum evaluation.

4. Selection of weapon delivery profiles from previous operating command evaluations and current operational procedures. These could form a baseline to which incountry operating profiles could be added for complete subsystem evaluation.

Two primary objectives which can be accomplished in a qualitative evaluation are pilots or aircrewmembers' analysis of system employment and numerical analysis and comparison of weapon delivery accuracy to a specification or to performance of another aircraft. Information on system maintainability, reliability and supporting equipment is also available to the evaluators as a byproduct of the primary test objectives.

General test procedures begin with selection of weapons and delivery profiles for the evaluation. Ground tests, including careful boresighting and alignment, are accomplished per the manufacturers' recommendations. Weapons may be delivered with samplings throughout the operating envelopes; however, concentration should be made of a select few conditions for statistical significance.

INSTRUMENTATION

Aircraft

Instrumentation requirements are few and may be kept simple. The primary requirement is for photorecording of the aircraft Head-up Display (HUD). This system is often included as a part of the operational equipment. If not, a suitable camera system should be installed by the airframe manufacturer prior to test. Weapon release/firing event should be

indicated on the HUD by a flag or light visible in the film frame. Care must be taken to insure that the difference in time between weapon release/firing and appearance of event on the HUD is known and is consistent. A known difference in time may be corrected for during data analysis. Beware of signal processing which is routed through an onboard computer. This can result in a variable, and therefore unknown, time delay depending upon computer workload. The best method is to provide a direct signal from the trigger/bomb button to the HUD electronics.

Time must be allowed for the HUD camera to attain operating speed before weapon release. This can be accomplished by pilot activation of a camera power switch during run-in. Some aircraft provide a two-detent trigger. Camera power is applied at the first detent and weapon firing occurs at the second detent as the trigger is pulled through. Camera frame rates may vary from 24 to 48 frames per second depending on magazine capacity, ease of changing magazines in flight, and mission duration. Higher frames rates are preferred for the greater accuracy in release/firing picture presented.

The following aircraft parameters should be recorded at weapon release: dive angle, airspeed, altitude, and load factor. This data may be available from the HUD and recorded on the HUD camera. It may be recorded by an over-the-shoulder camera photographing the cockpit instrument panel with a time reference to allow correlation with the weapon release event.

Weapon Range

A weapons delivery range should have level, cleared terrain for target areas. Bombing targets should have clearly visible concentric circles; one-hundred feet radii to four-hundred feet should be adequate. Two spotting towers with simple transits allow observers to identify bomb impacts in sequence for later miss-distance measurement. Impacts from the target center may be measured in x and y coordinates by survey transits or by chaining from the target center.

Strafe runs may be accomplished against 6 x 6 meters (20 x 20 feet) panel targets supported by telephone-type poles. Provisions should be made to position the panels perpendicular to the attacking flightpath. Angles of 10, 15, and 30 degrees are typical. Care must be taken to establish these targets in a soft soil or plowed area to minimize ricochet potential. The area should be maintained to remove spent projectiles or any hard objects which could result in ricochet occurance. Sufficient targets should be provided to permit one firing pass per panel. Scoring can be accomplished manually through measurement of x and y impact distance from the target center.

Wind direction and velocity must be measured as closely as possible to the range area and proximate to the time of the mission. Standard weather balloon with transit tracking is adequate to record wind direction and velocity from the target area through the altitude of the delivery profile. This data is provided to the pilot for preflight corrections in manual bombing and is used in error analysis discussed subsequently.

Tracking radar is a useful tool in analysis of the aircraft parameters at weapon release/firing. It is not; however, a requirement. Tracking radar, used with a beacon-augmented aircraft, can provide sufficiently accurate data for determination of gross positioning errors.

An extended treatment of range instrumentation may be found in AGARD-AG-219, "Range Instrumentation, Weapon System Testing and Related Techniques".

TEST AND EVALUATION

Ground Tests

Mass properties (weight, center of gravity, and moments of inertia) of weapons to be dropped must be measured. Moment of inertia in the yaw axis is most convenient to measure and may be assumed equal to pitch inertia for symmetrical weapons. Mass properties are compared to the weapon specification to assure that the units are within allowable limits. Visual inspection of fins should also be accomplished to assure that gross bending or misalignment is not present. Practice bombs do not normally require mass measurements since manufacturing tolerances will insure production within these limits. Fin alignment should be determined within specification by use of a template. Care in mass property/alignment checks will allow rejection of out-of-tolerance weapons which could mask true weapon system performance through introduction of large dispersion errors.

The loading of weapons onto the aircraft allows the opportunity to evaluate the adequacy of the loading checklists and compatibility of loading equipment. After load crews become proficient, timing required to reconfigure the aircraft with a load of weapons may be determined for rapid turnaround considerations. Ease of arming wire installation and preflight fuzing procedures may also be evaluated.

Careful alignment of rocket dispensers and HUD/gun harmonization is a vital precursor to any evaluation of these weapons. The airframe manufacturer should provide alignment fixtures, boards, and detailed technical manuals to accomplish these tasks. The entire gun system including feed mechanism should be carefully inspected prior to harmonization.

Most U.S. aircraft gun systems are designed such that the projectiles and the pipper image will be coincident at 686 meters (2,250 feet). The term "boresighting" refers to sighting through the gun barrel to establish the aiming point on the target. "Harmonization" is the process of correlating the gun aim point and the gunsight alignment such that the projectiles and the sight line will be coincident at 686 meters and the bullets will impact the target. Procedures and target layouts are contained in the aircraft technical order.

Dry (nonfiring) boresighting is most easily accomplished using a target (Appendix B, figure 1) located 25.4 meters (1,000 inches) in front of the aircraft. The fuselage reference line is aligned with the "Post" reference using sighting fixtures provided by the manufacturer. The target, mounted on a fixture moveable in azimuth and elevation, is positioned to achieve this alignment. This establishes the proper relationship between the target and the aircraft. The gun is then aligned so that the bore scope (inserted in the gun barrel) is aimed at the gun circle cross. For the Gatling type gun, each boresight barrel is rotated into firing position and sighting position noted as each will vary somewhat. All barrels should aim within the circle (size determined by the particular gun specification). The combining glass is removed from the cockpit mounted gunsight and an aiming fixture bolted into place. Crosshairs in this scope are aligned on the "sight o mil" mark for the zero mil depression setting. The combining glass is then replaced and checked for pipper alignment at the reference mark.

Dry harmonization will indicate a satisfactory gun/sight alignment; however, wet harmonization (firing) provides a higher degree of assurance that the system is, in fact, properly aligned and delivering bullets in the correct geometrical relationships to system aiming. This requires more elaborate support capability. A structure must be available to contain the spent projectiles and support the harmonization target (similar to the 1,000-inch target - but larger). The aircraft is jacked and tied down securely to provide a stable gun platform, normally at a distance of 304 meters (1,000 feet) from the target. The alignment procedure is accomplished as previously discussed. The gun is then fired using ground hydraulic/electrical power as required; bullets should impact at the "impact" reference with a dispersion not exceeding specifications (5 mils may be typical) for the gun under test. Wet harmonization will assure proper alignment and further reveal any errors due to barrel wear or gun dynamics. Harmonization should be accomplished periodically (i.e., beginning, midpoint, and end) during the test program to evaluate the capability of the system to maintain alignment under use. Alignment should also be verified whenever errors are encountered which cannot be attributed to other factors (i.e., delivery error, wind, etc).

It is also important at this time to boresight the gunsight/HUD camera. A bore scope is normally provided which is installed in the camera mounting to align the unit. A reference for camera boresight should be marked on the target. The camera is then run with the pipper visually fixed on a target reference. This will provide assurance that the camera photographs the pipper at the same location as viewed by the pilot; this may not always be the case. Any discrepancy between the viewed and photographed pipper must be corrected or the difference accounted for during the error analysis.

Aircrew Training

Sufficient flights must be planned to allow thorough familiarization of the pilots/crewmembers with aircraft handling qualities and operation of the weapon delivery subsystems in all modes of operation. Allowances should be made to include pilots with a variety of skill levels representative of the population to evaluate the system. It does little good to procure a weapon system which only a few of the highest skill level personnel can operate. When accomplishing delivery sorties; however, the number of pilots participating for data should be minimized to reduce scatter in data through variances in proficiency. Aircrew participants should have a variety of experience in other similar aircraft to allow comparison of the system under test with other tactical aircraft.

Air-to-Ground Bombing

A review of technical reports prepared by the original Air Force customer operational command will provide the best source of delivery profiles for the test program. Utilization of these profiles will result in a selection which has proven optimum for the system under test and will provide a baseline for comparison of incountry results. Any unique customer profiles may also be selected for evaluation. Weapons selected should also duplicate those previously tested to provide a basis for comparison. Use of "customer nation" weapons complicates an evaluation in that they must undergo extensive flutter, structural, separation, and ballistic quantitative test/analysis prior to use. It may generally be assumed that these weapons could be successfully integrated into the weapon system should it finally be procured. Practice weapons are suitable to demonstrate the proper functioning of a weapon system fire control subsystem or accuracy of a ballistic handbook. However, actual inert-ballasted operational weapons should also be used to allow verification of their ballistics, weapon separation, proper arming, and loading checklists. Live ordnance may be used; however, this introduces difficulty in scoring of impacts and additional safety considerations in weapon handling. It adds little value to the evaluation as the weapon blast characteristics and fragment lethality envelopes are usually well known.

Selection of bombing modes to be evaluated merits some close attention. Modern tactical aircraft normally offer a wide selection of weapon delivery modes ranging from manual

(ballistic table) to automatic modes such as Continuously Computed Impact Point, Continuously Computed Optimum Release, Offset Bombing and Beacon Bombing. Automatic modes are usually capable of degraded operation as primary sensors become inoperative and secondary (less accurate) sensors are used for aircraft state input. Evaluation of all of the potential modes with all available weapons would require an extensive test program. Selection of the modes/weapons to be evaluated should be limited based on a review of development test reports and the needs of the customer Air Force.

Conduct of the air-to-ground bombing tests consists of delivering the weapons selected from the profiles and delivery modes of primary interest. The weapon impacts are scored in x (range) and y (azimuth) coordinates from the target center. HUD film is analyzed to determine any aiming errors at weapon release. Onboard data is reviewed to determine any gross deviation from planned release conditions. Ground radar data may be used, if available, to aid in this error analysis. These procedures will allow deletion of data from impacts resulting from gross pilot delivery errors or fire control system malfunctions. The data may also be used to factor out delivery errors induced by aiming to compute statistical measures of system error (i.e., that error which excludes operator and wind effects). Most aircraft ballistic tables contain the data necessary to accomplish this type of error analysis. A statistical treatment of impact data forms the core of the evaluation.

This phase of weapons accuracy testing cannot be accomplished prior to geometry verification of a manual system or ballistic/functional verification of a weapon computing system. Any problems or errors inherent in the weapon delivery system must be solved; it is generally meaningless to expend weapons for a statistical analysis when a system has known deficiencies.

The following basic terms used in statistical analysis of weapons accuracy are defined:

Mean Impact Point (MIP): The geometric center of the weapon impact pattern as determined by the arithmetic mean of the x (range) and y (azimuth) weapon impacts from the target center. The MIP is useful primarily as a measure of weapon system bias. A "perfect" system would deliver weapons in a pattern in which the MIP and target center would be coincident.

Circular Error Probable (CEP): The radial distance from the origin describing a circle containing 50 percent of weapon impacts. The implication is that, of additional weapons dropped from like delivery conditions, 50 percent would fall within the CEP computed from the original tests. Confidence limits can be applied to more clearly define CEP. CEP about the MIP is a measure of system dispersion while MIP location with respect to the target describes system accuracy.

Circular Error Average (CEA): The arithmetic mean of radial distances of bombs from the target center. This statistic gives a feeling for "average" error of bombs actually dropped during a test. It therefore, has limited value in describing or predicting weapon system accuracy on a long term basis.

There are many other statistical definitions which may be useful in describing weapon accuracy, but these are the primary ones.

It is most important in statistical analysis to compute a statistic only on comparable items. Different delivery modes and profiles can produce widely varying accuracies. For example, high angle dive bombing is less accurate than lower angle dive delivery in the manual mode. Impacts from both profiles should not be grouped into a single CEP; the CEP should be computed for each profile. CEP from manual dive bombing and automatic delivery weapon impacts should not be computed as a group. In general, compute separate CEP's for separate modes, profiles and weapons. This will provide a numerical comparison for evaluation of effectiveness. A good general breakdown for dive bombing would be low angle (0 degrees - 30 degrees) and high angle (30 degrees - 60 degrees). The selection of comparative groups, will of course, depend on the individual system and the test objectives.

Sequential ratio techniques provide a means of determining if CEP of a weapon system under test meets a specified, or goal, CEP while limiting the number of weapon releases required to reach a decision point. Risks of accepting a bad system or of rejecting a good one must be assumed; however, the capability to evaluate a system with a relatively small number of samples makes this an attractive approach. The CEP statistic may be used to compare the system under test to a specification, performance of other similar aircraft, or related to predicted lethality for specific weapons. Detailed discussion of and sample calculations for CEP, MIP, normality of distribution, and sequential ratio techniques is included as Appendix A.

The pilot/aircrew can provide a substantial input to system evaluation for the air-to-ground delivery modes. Ease of tracking, clarity of weapon system controls and displays, accessability of switches and general cockpit layout are examples of interest for aircrew evaluation. A qualitative comparison with other tactical aircraft in use by the "customer" nation provides an excellent baseline for these judgements.

Air-to-Ground Gunnery

Profile selection again may best be made from review of evaluations conducted by the original Air Force customer operational command. Conduct of the flight tests and instrumentation usage is similar to that previously addressed under air-to-ground bombing. Profiles should be planned to allow tracking to the ground panel target after firing with gun camera operating. This will allow viewing of the impacts, assuming the range surface generates a visual impact dust cloud, to determine any gross misses of the target panel. Statistical presentation of bullet impacts may be computed as discussed in Appendix A for bombing. However, a simple computation of percent hits on a given pass normally provides sufficient data for comparison to previous results and a measure of system lethality/effect. Rounds fired per pass may be determined by a rounds counter, if available. Some gun systems are mechanized such that sections of spent brass may be counted during ammunition down load and correlated with firing passes to determine shots fired. Target practice or ball ammunition is normally used for test due to lower costs and handling safety over armor-piercing incendiary or high explosive rounds. Scoring results should be identical. Tracer rounds may be used for a visual aid in review of gun camera film; however, care must be taken with regard to potential range fires which they may ignite. Aircrew evaluation of tracking and time on the target forms a valuable portion of these tests.

Air-to-Air Gunnery

Selection of the tow target is of primary importance in conducting air-to-air gunnery evaluation. The target should have maneuver capability consistent with the test aircraft firing envelope to allow evaluation under demanding attack profiles. A scoring system is almost a "must" for reasonable recording of hits or near misses. Dart towed targets and towed banners may be used as nonscorable targets. The FIGAT fiberglass tow target provides a large profile available with a scoring system, but is costly with a demonstrated low rate of survival. The SECAPEM 901B tow target with SFENA MAE-15 acoustical scorer has been used with some success in gunnery evaluations. Targets may be flown in standard racetrack, figure 8, butterfly, or combat Dart patterns depending on desired shooter firing conditions. Tests should begin with benign low range, load factor, and angle off conditions and proceed to more demanding target/shooter ranges and geometeries to fully evaluate system capability. Analysis of HUD film will reveal target tracking capability and firing opportunity durations. Hits per pass or passes on which hits are made are recorded to provide comparisons with earlier results and some measure of system operational effectiveness. Pilot evaluation forms a very important part of this test phase covering aircraft handling and tracking characteristics, HUD display clarity, utility and overall comparison of gunsight effectiveness with other systems.

Air-to-Air Missiles

Qualitative evaluation of missile performance is a most difficult area of assessment. Costs can be substantial, both for the missiles themselves, and for the operation of the target drones required. If missiles are fired with live warheads, the probability of target destruction increases with attendant cost of replacement. If inert warheads are used to lower the probability of target destruction, costly range and telemetry instrumentation are required to evaluate missile/target proximity and warhead function. The most practical approach starts with a comprehensive review of Development Test and Evaluation reports of instrumented missile performance. Test missiles with active seekers, inert motors, and inert warheads may be carried captive against aircraft targets. This will allow a qualitative evaluation of missile/aircraft system interface up to actual launch and provide a look at missile capability to acquire and maintain sensor track with the target throughout the published envelope.

Heat-seeking missiles may be evaluated, to a limited degree, by firing at a flare-augmented target rocket. Five-inch high velocity aerial rockets may be used for this purpose, fired from the shooter or a test support aircraft. This allows a qualitative evaluation of the airframe/missile interface and allows the pilot to survey seeker side tone. However the augmented target bears no relation to actual aircraft infrared signature and thereby precludes realistic weapon evaluation. It does provide an opportunity for pilot familiarization and training.

Radar guided missile evaluation offers no low-cost means of live firing but requires the use of target drones and attendant telemetry and range capabilities.

CONCLUSIONS

A qualitative evaluation of the armament subsystem can provide the basis for determining the operational effectiveness and suitability of the system for procurement. Intensive investigation and review of Development Test and Evaluation technical data are required to augment these tests. This data also provide a comparative baseline on which judgements can be made of qualitative test results and system utility. Unique customer weapon delivery profiles and tactics may be accomplished to evaluate weapon system compatibility with intended usage.

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APPENDIX A

STATISTICAL ANALYSIS

There are many formulae for CEP computation. The following formula is the exact mathematical representation of CEP as defined and has been found to be most suitable for weapon accuracy analysis:

$$0.5 = \frac{1}{2\pi S_x S_y} \int_{-x}^{x} \int_{-\sqrt{CEP^2 - x^2}}^{\sqrt{CEP^2 - x^2}} \left[e^{-\frac{1}{2}} \quad \frac{(x_i - \bar{x})^2}{S_x^2} + \frac{(y_i - \bar{y})^2}{S_y^2} \right] \quad d_x d_y$$

Where: S_x = standard deviation in range

S_y = standard deviation in azimuth

$\bar{x}$ = MIP location in range

$\bar{y}$ = MIP location in azimuth

x_i = impact distance in range

y_i = impact distance in azimuth

This equation is based on the bivariate normal distribution and provides accurate results, but requires substantial computer time for solution. The following formulae are approximations of the double integral and have proven sufficiently accurate for practical use when computing CEP about the MIP:

IF: $S_x < S_y$ and $S_x/S_y \geqq 0.28$

CEP = $0.562\ S_y + 0.615\ S_x$

IF: $S_x > S_y$ and $S_y/S_x \geqq 0.28$

CEP = $0.615\ S_y + 0.562\ S_x$

IF NEITHER OF THE ABOVE APPLIES:

CEP = $0.5887\ (S_x + S_y)$

These formulae are for a bivariate normal distribution about the mean in range and azimuth. The bivariate normal means that we are dealing with two separate normal distributions; one in range and one in azimuth. This is the usual distribution encountered in weapon delivery where impacts are more widely distributed in range than in azimuth. A normal distribution cannot be assumed, but must be verified before these formulae are used. The verification process is described later in the Appendix.

To compute CEP about the MIP, the following applies:

$$S_x^2 = \frac{1}{n-1} \Sigma \quad (x_i - \bar{x})^2$$

$$S_y^2 = \frac{1}{n-1} \Sigma \quad (y_i - \bar{y})^2$$

and:

$$\bar{x} = \Sigma \frac{x_i}{n}$$

$$\bar{y} = \Sigma \frac{y_i}{n}$$

Where: n = number of weapons (impacts)

For example, see figure A-1 and the sample weapon impacts numbered 1 through 10. These impacts translate into the following range and azimuth errors:

Bomb No.	x_i (feet)	y_i (feet)	Sign Convention
1	-100	0	+ x long
2	-45	+20	- x short
3	-10	+40	+ y right of target
4	0	+30	- y left of target
5	+40	0	
6	+95	+20	
7	+100	-20	
8	0	-30	
9	0	-50	
10	-40	-20	
*11	+210	+121	
*12	-190	-75	

* For later reference

To compute the MIP coordinates:

$$\bar{x} = \frac{\Sigma x_i}{n} = - \frac{100-45-10+0+40+95+100+0+0-40}{10} = +4$$

and:

$$\bar{y} = \frac{\Sigma y_i}{n} = \frac{0+20+40+30+0+20-20-30-50-20}{10} = -1$$

These parameters describe the location of the MIP. To compute:

$$S_x^{\ 2} = \frac{1}{n-1} \ \Sigma \ (x_i - \bar{x})^2$$

$$= \frac{1}{9} \quad (-100-4)^2 + (-45-4)^2 + (-10-4)^2 + (0-4)^2 + (40-4)^2 + (95-4)^2$$

$$+ (100-4)^2 + (0-4)^2 + (0-4)^2 + (-40-4)^2$$

$$= \frac{1}{9} \quad (-104)^2 + (-49)^2 + (-14)^2 + (-4)^2 + (36)^2 + (91)^2 + (96)^2 + (-4)^2$$

$$+ (-4)^2 + (-44)^2$$

$$= \frac{1}{9} \quad 10816 + 2401 + 196 + 16 + 1296 + 8281 + 9216 + 16 + 16 + 1936$$

$$= \frac{1}{9} \ (34190)$$

$$S_x^{\ 2} = 3798.9$$

$$S_x = 61.6$$

In a similar manner:

$$S_y^2 = \frac{1}{n-1} \Sigma (y_i - \bar{y})$$

$$= \frac{1}{9} (7490)$$

$$S_y^2 = 832.2$$

$$S_y = 28.9$$

Since S_x (61.6) > S_y (28.9) and $\frac{S_y (28.9)}{S_x (61.6)} = 0.47$, which is > 0.28

$$CEP = 0.615\ S_y + 0.562\ S_x = 0.615\ (28.9) + 0.562\ (61.6)$$

$$= 17.8 + 34.6$$

CEP = 52.4 feet about the MIP (16 meters).

Let's look at the effect on CEP by doubling the number of bombs but maintaining the same distribution (i.e., 10 more bombs in the same holes as the first 10). This will give a concept on the quantity of bombs alone as it effects CEP value.

$$\Sigma (x_i-\bar{x})^2 = 34{,}190\ (2) = 68{,}380$$

$$S_x^2 = \frac{1}{19}\ (68{,}380) = 3599$$

$$S_x = 60$$

$$\Sigma (y_i-\bar{y})^2 = 7{,}490\ (2) = 14{,}980$$

$$S_y^2 = \frac{1}{19}\ (14{,}980) = 788.4$$

$$S_y = 28.08$$

Since $S_x > S_y$ and $S_y/S_x = 0.47$ which is > 0.28

$$CEP = 0.615\ S_y + 0.562\ S_x = 0.615\ (28.08) + 0.562\ (60)$$

CEP = 17.27 + 33.72 = 51.0 feet (15.5 meters)

By the same method with 100 bombs, CEP = 49.8 feet (15.2 meters) and with 1,000 bombs, CEP = 49.7 feet (15.1 meters).

This is an idealized situation but it illustrates the point that with a good system/good distribution, CEP will change little with larger sample size if the additional points are from the original distribution.

Now let's look at the effect on CEP produced by two outlying impacts. In addition to the original 10 bombs, consider bombs 11 and 12 (figure A-1). Computing CEP for the 12 bombs yields:

CEP = 88.1 feet (26.9 meters)

This compares to a CEP of 52.4 feet (16 meters) for the original and illustrates the profound effect the outlyers have on a small sample. Looking at the effect of adding these two outlyers to the original 100 bomb sample yields:

CEP = 55.2 feet (16.8 meters)

This compares to original CEP of 49.8 feet (15.2 meters). This illustrates the value of a large sample in minimizing the influence of a few "wild bomb" data points.

CEP about the MIP may or may not bear a significant relationship to the primary point of interest, namely the target. For example, weapons may be delivered in a very tight pattern (low CEP about the MIP) but impact 304 meters (1,000 feet) beyond the target due to inertial groundspeed error. In this case, weapon system accuracy could be described by computing CEP about the MIP and the location of the MIP in relation to the target. If the inertial error were subsequently corrected, allowing delivery about the target, the original biased impacts could be computed about the corrected MIP for an expanded data base. In our example, where target center and MIP were nearly coincident, CEP about both target and MIP would be identical for practical purposes.

In the preceding computations for the maximum likelihood estimates of CEP about the MIP, it was assumed that a bivariate normal distribution existed for the weapon impacts. The distribution must, in fact, be normal for the CEP computations illustrated to be valid. The Kolmogorov-Smirnov (referred to as K-S) test may be used to evaluate the data for normality (tables A-1 and A-2). Normality must be determined for both x (range) and y (azimuth). Table A-1 illustrates the method for the x variable about the MIP. The ten impacts are rank-ordered (i) from largest negative (short) to largest positive (long) and values (x_i). The $x_i - \bar{x}$ and $\frac{x_i - \bar{x}}{S_x}$ columns are computed using the previously calculated values, $\bar{x} = +4$ and $S_x = 61.6$. Succeeding columns are computed as illustrated. The K-S test is implemented by selection of a significance level at the sample level (n) and comparing the largest value of $\frac{i}{n} - F\left(\frac{x_i - \bar{x}}{S_x}\right)$. Table A-1 shows this value to be 0.22 for our example. The following critical values were extracted from reference 4, page 426.

Sample Size (n)	Significance Level 0.20	0.15	0.10	0.05	0.01
10	0.322	0.342	0.368	0.409	0.468

Since 0.22 is <u>less than</u> any of the critical values, our distribution has a 20 percent or greater probability of being normal. If the largest value of $\frac{i}{n} - F\left(\frac{x_i - \bar{x}}{S_x}\right)$ had been 0.368 we would have been "10 percent sure" that our sample distribution were normal. The selection of significance level in this process is judgemental. The same procedure for the azimuth distribution (table A-2) shows a maximum value of 0.15 which is also less than the critical values, verifying normality in the y-direction. The CEP computation about the MIP is therefore, valid for our sample.

Failure to meet normality in either range or azimuth invalidates the CEP computation as accomplished. CEP can be calculated for other than normally distributed impacts using nonparametric or unassumed common form of probability distribution function. However, the lack of normal weapon impacts about the target usually results from system error or bias. This should be corrected, since computing CEP for a malfunctioning system has little or no meaning.

Table A-1

K-S TEST RANGE IMPACTS

No.	i	x_i	$x_i - \bar{x}$	$\frac{x_i - \bar{x}}{S_x}$	$F\left(\frac{x_i - \bar{x}}{S_x}\right)$	$\frac{i}{n}$	$\frac{i}{n} - F\left(\frac{x_i - \bar{x}}{S_x}\right)$
1	1	-100	-96	-1.56	0.0594	0.10	0.04
2	2	-45	-41	-0.67	0.2514	0.20	-0.05
10	3	-40	-36	-0.58	0.2810	0.30	0.02
3	4	-10	-6	-0.10	0.4602	0.40	-0.06
4	5	0	-4	-0.06	0.4761	0.50	0.02
8	6	0	-4	-0.06	0.4761	0.60	0.12
9	7	0	-4	-0.06	0.4761	0.70	0.22
5	8	+40	+36	+0.58	0.7190	0.80	0.08
6	9	+95	+91	+1.48	0.9306	0.90	-0.03
7	10	+100	+96	+1.56	0.0406	1.00	0.06

Where $F\left(\frac{x_i - \bar{x}}{S_x}\right) = F(x)$ as found in normal distribution tables starting on Page 127 in reference 4. (Use $1 - F(x)$ for negative $\frac{x_i - \bar{x}}{S_x}$ values).

$\bar{x} = +4$

$S_x = 61.6$

Table A-2

K-S TEST AZIMUTH IMPACTS

No.	i	y_i	$y_i - \bar{y}$	$\frac{y_i - \bar{y}}{S_y}$	$F\left(\frac{y_i - \bar{y}}{S_y}\right)$	$\frac{i}{n}$	$\frac{i}{n} - F\left(\frac{y_i - \bar{y}}{S_y}\right)$
9	1	-50	-49	-1.70	0.0446	0.10	0.06
8	2	-30	-29	-1.00	0.1587	0.20	0.04
10	3	-20	-19	-0.66	0.2546	0.30	0.05
7	4	-20	-19	-0.66	0.2546	0.40	0.15
5	5	0	+1	+0.03	0.5120	0.50	-0.01
1	6	0	+1	+0.03	0.5120	0.60	0.09
2	7	+20	+21	+0.73	0.7673	0.70	-0.07
6	8	+20	+21	+0.73	0.7673	0.80	0.03
4	9	+30	+31	+1.07	0.8577	0.90	0.04
3	10	+40	+41	+1.42	0.9222	1.00	0.08

1. Note on table A-1 applies

$\bar{y} = -1$

$S_y = 28.9$

The sequential ratio test provides a method for determining if a bombing system meets, or fails to meet, a predetermined specification while limiting the number of weapons to be dropped. There's a price to pay in terms of "risk". The risks involve accepting a bad system or rejecting a good one. Figure A-2 illustrates a typical sequential ratio test which we have adopted for use when none is specified or contracted. It represents a reasonable balance between risk and sample size. The seller's (or contractor's) risk has been chosen at 10 percent and is termed alpha. This means that the seller has agreed to a 10 percent chance that the test will reject a system which meets specification. The buyer's (Air Force) risk (beta) has also been identified as 10 percent. The buyer has thereby assumed a 10 percent chance of the test accepting a system which does not meet specification. The discrimination ratio (lambda) describes a bad system in terms of a good one. In our example, a lambda of 1.4 means that if the specified CEP were 30.5 meters (100 feet), we would accept a 42.7 meter (140 feet) CEP as having met acceptance criteria. These risks are, of course, variables and must be agreed to prior to accomplishing the test. Buyer and seller may agree on unequal risk levels and discrimination ratio may be chosen as desired. The effects of varying risks will be presented later.

The boundaries of figure A-2 are computed as follows:

$$\lambda \sqrt{\frac{\chi^2_{\alpha,\ n-1}}{n-1}}$$

Solution of this expression with variable sample size describes the upper curve.

$$\sqrt{\frac{\chi^2_{1-\beta,\ n-1}}{n-1}}$$

describes the lower curve where:

$$\chi^2_{1-\beta,\ n-1}$$

is extracted from the Chi-square tables for selected β and n values

$$\chi^2_{\alpha,\ n-1}$$

is extracted from the Chi-square tables for selected α and n values.

λ is the discrimination ratio of 1.4

α is the seller's risk of 0.10

β is the buyer's risk of 0.10

n is the sample size (number of bombs used in the CEP computation).

These boundaries define "accept", "reject", and "continue testing" regions.

Figure A-3 shows the effects of variation in α, β, λ and curves 2 and 4 illustrate what happens with a change in buyer and seller risks while the discrimination ratio is held constant. The maximum decision point moves from 35 samples when five percent risks are assumed to 22 sample with 10 percent risk assumption. This shows a substantial change in the number of tests required for decision with not a really big change in risk. While not shown on the figure, a differential buyer/seller risk would close down the appropriate decision area while increasing the maximum sample required. For example, if the seller's risk on curve 1 were dropped five percent the upper curve would move up, closing down the "accept" area, opening the "continue testing" area and increasing the maximum decision sample. Comparison of curves 1, 2, and 3 illustrates the effect of varying discrimination ratio while buyer/seller risk are held constant. As discrimination ratio increases the accept area enlarges and maximum decision sample drops. This is as expected, since by increasing discrimination ratio, we are willing to call a larger CEP "good" with respect to the specified CEP.

The test is accomplished by conducting a weapon delivery mission in the desired profile/mode. One weapon is delivered on target on each pass. The number of weapons per mission will depend on the type of weapon and suspension equipment. For example, if the aircraft carries one SUU-20 training dispenser (six bomb loading), it would be logical to expend six BDU-33 practice bombs on the first mission. It takes at least four weapons to reach any decision for the test as we have set it up.

The measured impacts for the first four bombs would be used to compute CEP as previously discussed. This CEP would be divided by the specified CEP and the result plotted on the sequential ratio graph. If the CEP ratio fell within the "continue testing" region, a CEP for the first five bombs would be computed and plotted. The process continues one bomb at a time in the sequence of delivery until the "accept" or "reject" region is entered. The test stops then. If, for example, the CEP ratio enters the "reject" region

on the 19th bomb and you have already dropped 24 bombs - stop. Do not keep computing to the 24th bomb to see if you can get out of the "reject" zone. This would invalidate the test. The risks are assumed for this purpose: to limit the number of samples. A good system may have been rejected, but that's the risk involved in the sample-limiting approach.

Bombs may be eliminated from the computation for large errors if there is a valid cause external to the system under test. Some valid reasons might be:

The pilot inadvertently aimed at the wrong target.

The bomb momentarily hung on the rack and then released.

An input subsystem for automatic delivery failed.

These are just some examples; each instance would have to be judged on the merits. The important thing is not to eliminate impacts because "they're too far out". This same selection process is equally applicable to the CEP computations previously discussed.

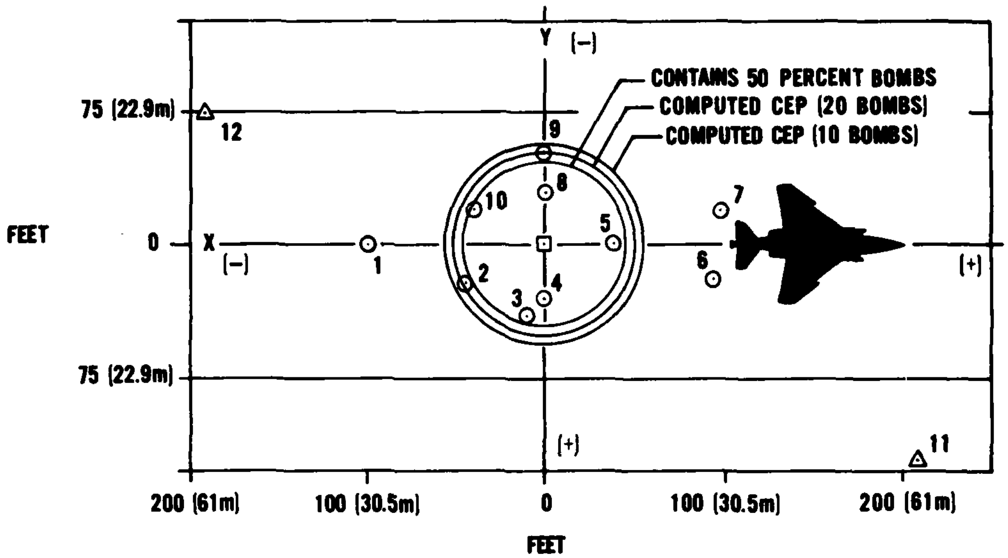

Figure A-1 Weapon Impact Distribution

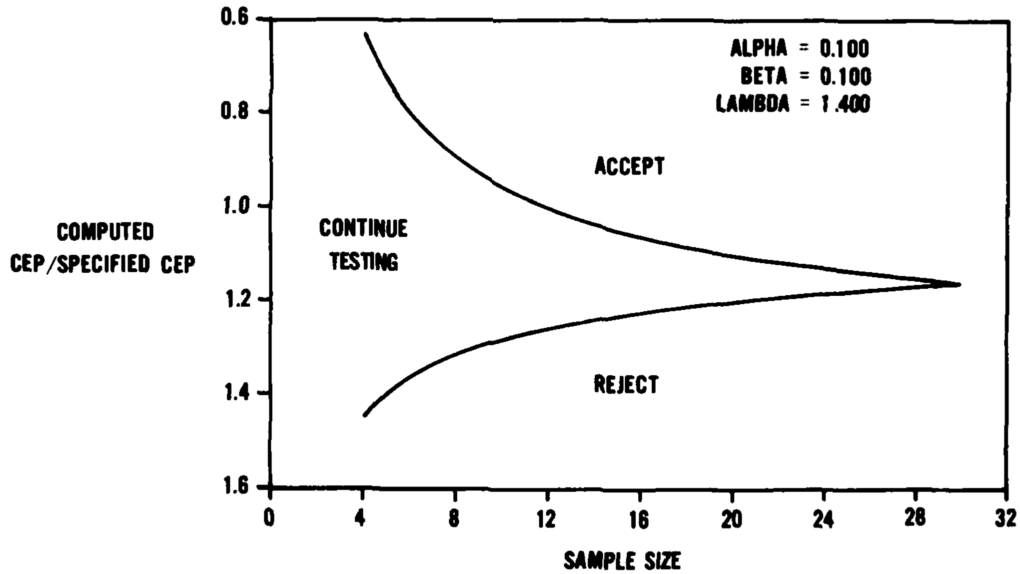

Figure A-2 Sequential Ratio Test

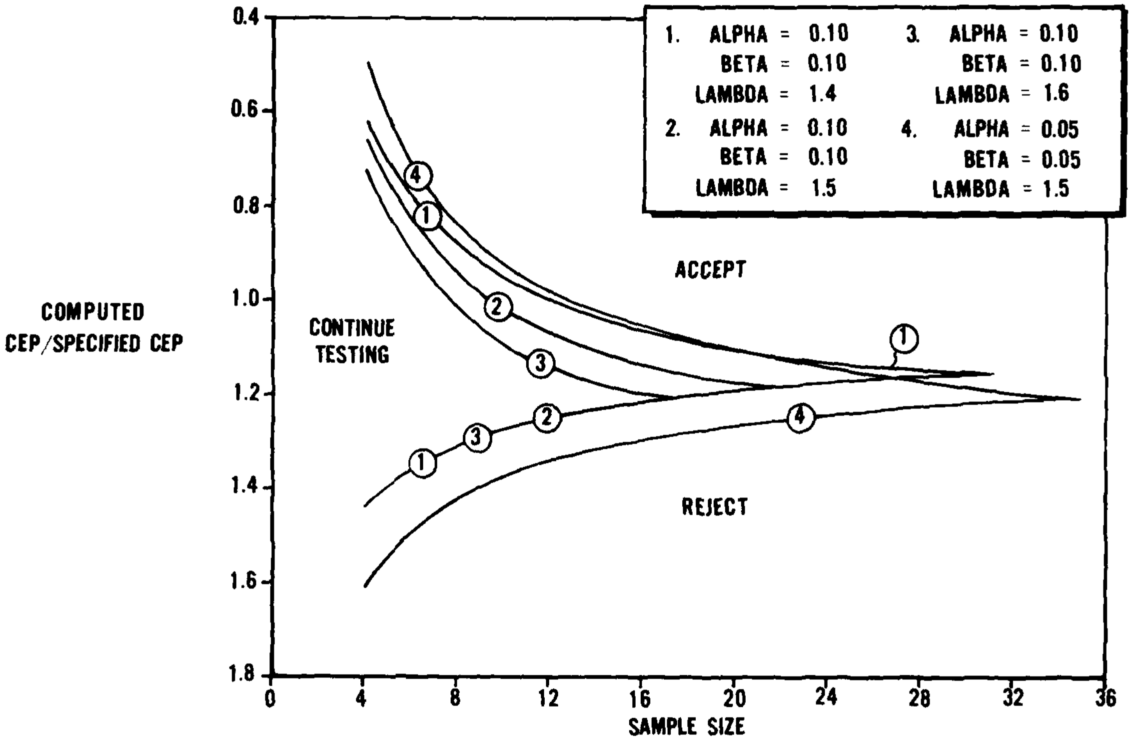

Figure A-3 Parameter Variation in Sequential Ratio

APPENDIX B

FIGURES

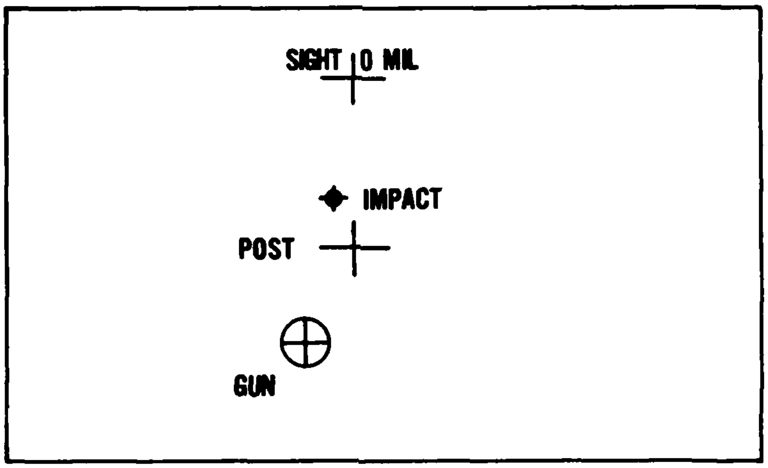

Figure 1 Typical Boresight Target

ELECTRO-MAGNETIC COMPATIBILITY

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SUMMARY

The wider use of electronic circuits in modern aircraft and their extension into critical areas of engine and flight control systems, have increased the dangers of malfunction due to electromagnetic interference, and highlighted the importance of tests that demonstrate the adequacy of the EMC standard achieved.

This Paper reviews the historical background to the growth in problems of EMC in UK Military Aircraft, and discusses the present approach for minimising these problems during development. The importance of using representative aircraft for final EMC assessments is stressed, and the methods of approach in planning and executing such tests are also outlined. Some of the limitations of present test techniques are also discussed.

INTRODUCTION

Twenty years ago, interference problems were largely confined to radio and radar systems, and received little attention from aircraft manufacturers beyond the fairly elementary requirements for bonding, suppression, and isolation of armament circuits. The introduction of semi-conductor technology encouraged the use of electronic circuits in many more aircraft systems, and the effects of interference became more widespread and serious, particularly in critical electronic control systems affecting flight safety, and it became apparent that the contractual responsibilities and design requirements and guides for EMC were inadequate. A state of EMC in aircraft is still far from assured at the design stage, in spite of improvements in the various design specifications and guides available, and the importance of following through good EMC design practise by thorough development tests of the complete aircraft has been recognised for the past decade.

Most systems in modern aircraft now employ electronic control loops, in analogue or digital form, that are potentially susceptible to Radio Frequency Interference (RFI). Long signal lines interconnecting sensors and control amplifiers are the most common means of coupling unwanted interference fields into these systems, and a variety of effects have been observed in electrical generation, engine and fuel, flight control, inertial and gyro platforms and weapon systems. Some of these effects can have serious consequences, particularly in flight and engine control behaviour, weapons and explosives. Spurious deflections or corruption of readings can occur on various flight instruments now that most flight data displays are electronically driven. Head-up and Head-down multi-function CRT displays are particularly prone to interference through their inter-connections to various sub-systems and sensors. By their nature, intercom systems and RF receivers are orders more sensitive to certain frequencies than most other electronic circuits, and the rejection of unwanted interference has always presented special problems in the congested aircraft environment. Interference with communications or radars have usually been self-evident to the user, often no more than a transient nuisance factor. However, some intereferences can have a greater consequence when automatic signal processing is used, especially where the intereference source is continuous; resulting in reduced range performance, and spurious or corrupted signals that are not always self-evident.

The effects of interference generated by on-board systems are generally the most significant, but externally generated fields cannot be neglected, where the susceptibility of electrically detonated explosive devices have always posed particular problems (1), and more recently, flight control systems. For on-board sources, HF communications transmitters cause most general interference, followed by UHF, where the trend towards higher power transmitters have aggravated the problems. The effects of other on-board sources, such as electrical generation systems, motors, actuators, de-icers, and circuit breakers are usually most evident in RF receivers, but large inductive transients can cause nuisance trips or shut down of other systems. The existence of high power RF installations on ships, airfields or elsewhere, can affect performance up to take-off and aircraft flying at low altitude, and during various aircraft development programmes there have been examples of flight control and other disturbances when flying near ship and land based transmitters. In the UK, at A&AEE, we have a facility for simulating the external environment, which at present is capable of generating the fields shown in Table 1.

FREQUENCY BAND MHz	MAX: FIELD STRENGTH	TRANSMITTER CHARACTERISTICS				
		TYPE	POLARISATION	AVERAGE POWER KW	PEAK POWER MW	PRF (pps)
0.2-0.525	300 V/m	CW	Linear	0.4	-	-
0.525-32	300 V/m	CW	Linear	30	-	MOD^n
32-200	10 W/m^2	CW	Linear	1.0	-	MOD^n
200-225	40 W/m^2	Pulse	Linear	0.9	0.45	250
225-400	20 W/m^2	CW	Linear	0.1	-	MOD^n
400-430	6 W/m^2	CW	Linear	0.015	-	MOD^n
1120 to 5850	300 W/m^2	Pulse	Linear	1.6	1.4	800/267
	800 W/m^2		Circular	3.38	2.25	300
	200 W/m^2	Pulse	Linear	1.5	1.0	1500
	300 W/m^2	Pulse	Circular	1.5	1.0	300
5850 to 14000	1000 W/m^2	CW	Circular	1.4	-	-
	200 W/m^2	Pulse	Circular	0.56	0.75	1500

TABLE 1 A&AEE RADIO ENVIRONMENT GENERATOR FACILITIES

DEVELOPMENT

UK aircraft contracts require EMC to be considered at the design stage and followed up by thorough development testing of systems in the laboratory, and on the aircraft. This is achieved in a manner similar to that followed in the USA (2). Each project generates its own EMC Control Document defining the administrative and design practises, the specifications for the emission and susceptibility characteristics of equipment, and the test methods and procedures to be followed.

Laboratory or Test House measurements are performed on all equipments in accordance with the requirements specified in the EMC Control Document. The current trend for critical equipment, is to introduce more requirements to test susceptibility of signal lines and to raise field intensity levels beyond those generally specified (3, 4). In cases where safety is of prime concern, the trend is also to raise test levels until the thresholds for faulty operation are determined. Laboratory results give confidence regarding the potential EMC integrity of the design and provide data that assists in the derivation of test schedules for the complete aircraft.

Avionics Systems development rigs are usually of limited value for EMC assessments, because they do not normally have the representative lay-out and screening properties of the aircraft. More recently they are being used to provide development insight into potential EMC problems, by obtaining spectral signatures of radiated fields, and revealing additional potential susceptibilities of the inter-connected systems, when irradiated.

Prototype aircraft used during development usually differ in significant detail from production aircraft in wiring and additions for instrumentation, and in the use of development standard equipments; they are nevertheless more representative than rigs. Ground test programmes are necessary for safety clearance for flight, and to provide further development opportunity for enhancing the EMC status of the design.

ASSESSMENT

In a fully developed aircraft the logical choice for a full EMC assessment is a production aircraft of the same standard as that delivered to the user. For new aircraft being introduced in Service, this often means that the assessment takes place some time after the initial deliveries, when aircraft have been upgraded to a definitive standard. The development programme is relied on for evidence regarding the EMC integrity in the interim period. Where doubts exist in areas of safety, limited, but thorough tests are performed prior to the delivery schedule. In this case any deviations or concessions to the definitive standard need to be carefully examined, and are acceptable only if their relevance to EMC can be confidently predicted. Subsequent changes to in-Service aircraft in the form of modifications or improvements are considered for their effect on EMC, and the requirement for reassessment reviewed periodically, in the manner shown in Table 2.

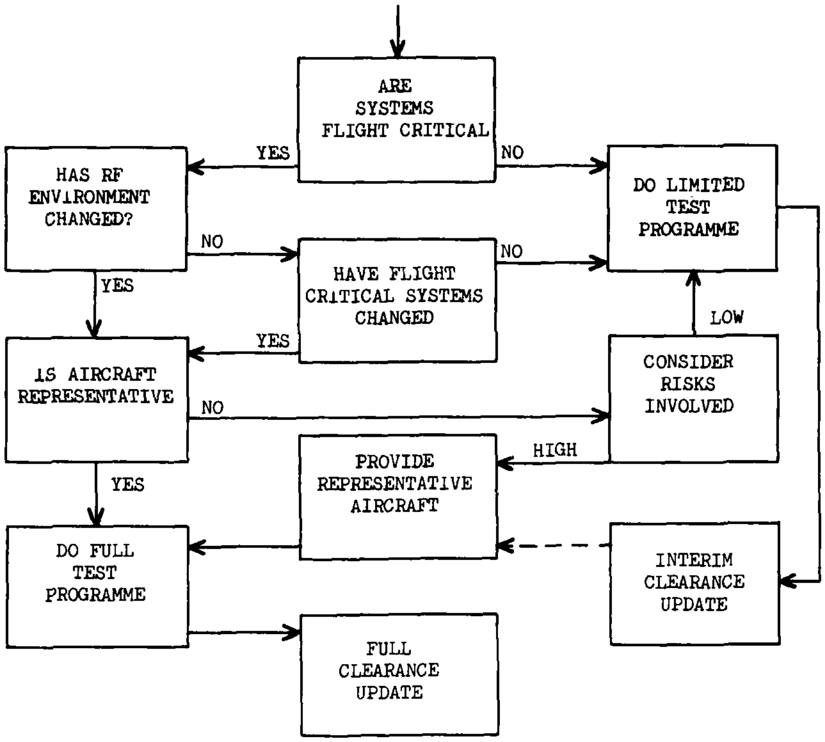

TABLE 2 EMC ASSESSMENT PROGRAMME DECISION CHART

CATEGORISATION

The depth of testing depends on the EMC categorisation of systems according to the expected consequences of interference, recognising that this can depend on the severity of the interference. Categorisation in the UK is broadly similar to that used in the US (2) and is:-

a Category i - Degradation of performance that could lead directly to an accident or loss of operational effectiveness.

b " ii - Degradation of performance that could contribute to an accident or reduce operational effectiveness.

c " iii - Degradation of performance that results only in annoyance or minor discomfort, or in minor loss of effectiveness.

Systems are grouped into the areas relating to armament, flight control and mission avionics, and categorised according to the potential seriousness of their malfunction.

"Armament", covers all weapons and explosive devices, detonators, ejectors and initiators, and all control systems for release and jettison. Proven methods are available for determining the safety margins in detonator circuits, and techniques for establishing similar criteria for semi-conductor control logic, now used in some armament circuits, are evolving. "Flight control" covers systems such as auto-stabiliser and auto-pilot, engine, fuel, electrical generation and flight instrument systems, where certain tests need to determine adequate margins of safety; as for armament systems. "Mission avionics" covers broadly the remaining systems of this nature in the aircraft. Some, such as communications and navigation sub-systems may be vital to mission success, and performance margins may need to be determined, in others the effects of interference may be less serious.

A typical categorisation may appear as shown in the Table.

	ARMAMENT				FLIGHT					MISSION					MISC		
SYSTEMS / CATEGORY	AIMING	FUSING	RELEASE	JETTISON	INSTRUMENT	CONTROL	ENGINE	FUEL	ELEC GEN	COMMUNICATION	NAVIGATION	TARGETTING	IDENTIFICATION	COUNTERMEASURE	ENVIRONMENTAL	REVERSIONARY	RECORDING
I	•	•	•	•	•	•	•	•	•								
II										•	•	•		•			
III													•		•	•	•

TABLE 3 AIRCRAFT SYSTEMS CATEGORISATION

The table shows the highest level that applies to a system; some systems have functional modes that lie in more than one area, or carry lower categorisation, and these are identified by considering the functional modes during the preparation of the test schedules.

TEST SCHEDULES

EMC test schedules are based on the operation of all potential interference sources and the observation of their effect on the various functional modes of the aircraft systems. For on-board sources this could be achieved by permutating all modes and channels of all systems, each item acting in turn as a potential emitter and susceptor. However, to produce a manageable programme of tests, a selective approach is needed. Interaction matrices assist in identifying the test requirements. Table 4 illustrates the first step in reducing the number of tests.

SYSTEM	SUSCEPTOR \ EMITTER	MISSILES	ELECTRICS	ENGINE	FUEL	STAB: AUG	AUTOPILOT	HUD	HDD	INSTS:	ILS	TF	R ALT	MC	IN	DOPPLER	MAP R	LASER	MFD	HF	UHF	INTERCOM	WNG: RX	TACAN	IFF
ARMAMENT	WEAPON SELECTION											*	*	*		*	*	*		*	*			*	*
	BOMBS											*	*			*	*	*		*	*			*	*
	MISSILES											*	*	*		*	*	*		*	*			*	*
	EJECTORS											*	*			*	*	*		*	*			*	*
	INITIATORS											*	*			*	*	*		*	*			*	*
CONTROL	ELECTRICAL GENERATION					*	*	*	*	*	*	*	*	*	*	*	*	*	*	*	*			*	*
	ENGINE CONTROL & PROTECTION											*	*			*	*	*		*	*			*	*
	FUEL CONTROL											*	*			*	*	*		*	*			*	*
	STABILITY AUGMENTATION											*	*			*	*	*		*	*			*	*
	AUTOPILOT											*	*	*	*	*	*	*	*	*	*			*	*
	HEAD UP DISPLAY		*	*						*										*					
	HEAD DOWN DISPLAY		*	*						*										*					
	STALL WARNING											*	*			*	*	*		*	*			*	*
	CENTRAL WARNING				*	*						*	*			*	*	*		*	*			*	*
	NOSEWHEEL STEER											*	*			*	*	*		*	*			*	*
	FLIGHT INSTRUMENTS		*									*	*	*		*	*	*		*	*			*	*
	ILS		*	*								*	*	*		*	*			*	*			*	*
	TERRAIN FOLLOW	*	*	*									*	*	*	*	*	*		*	*			*	*
	RADAR ALTIMETER	*	*											*		*	*	*		*	*			*	*
MISSION	MISSION COMPUTER	*	*	*			*	*	*			*	*		*	*	*	*	*	*	*			*	*
	INERTIAL NAV		*									*	*			*	*	*		*	*			*	*
	DOPPLER		*									*	*				*	*		*	*			*	*
	MAPPING RADAR	*	*									*	*	*				*							
	LASER RANGE FINDER	*	*									*	*	*		*	*			*	*			*	*
	MULTIFUNCTION DISPLAY		*									*	*	*	*	*	*	*		*	*			*	*
	HF COMMS	*	*	*	*	*	*	*	*	*	*	*	*	*	*	*	*	*	*		*		*	*	*
	UHF COMMS	*	*								*	*	*	*		*							*	*	*
	INTERCOM	*	*	*	*	*	*	*	*	*	*	*	*	*		*	*	*	*	*	*	*	*	*	*
	WARNING RECEIVER	*	*	*	*	*		*		*	*	*	*	*	*	*	*	*	*	*	*			*	*
	TACAN	*	*								*	*	*	*		*	*	*	*	*	*				*
	IFF	*	*								*	*	*	*		*	*	*	*	*	*			*	
	MISSION RECORDER		*									*				*	*			*	*				

TABLE 4 EMC INTERACTION MATRIX

For each interaction identified, further matrices can be used to specify the particular modes of each sub-system and depth of test required, according to its categorisation. The intersections on this matrix contain the reference to the appropriate section of the schedule that would detail the tests to be performed. Knowledge gained from the general development flight test programme is used to modify the content of the schedules, according to the accumulated experience of compatibility between regularly used sub-systems and their susceptibilities. During flight development, many observations of system malfunctions not of immediate interest at the time may have relevance to EMC, and need to be considered by EMC specialists. Some mission systems, such as HF communications and counter-measure receivers are used infrequently in the development flight programme, and rarely in association with other mission systems as in an operational context, so these aspects need particular attention during EMC tests. Most transmitter receivers have multi-channel capability, and predictive methods (5) can be used to determine potential interference frequencies, and Test House and rig results of emission and susceptibility spectra can be of assistance in selecting the more probable conditions for tests. Currently, predictive methods for determining interference paths cannot be relied upon, because of the uncertainty that surrounds the coupling mechanisms that occur in practice. An example of the prediction procedure is given in the Appendix.

The operating conditions of each functional mode require study in order to determine the appropriate conditions for observing the effect of potential interference. For some receiving systems it is necessary to provide test signals in order to check for de-sensitisation or corruption of output. In control systems, checks are made for the existence of undemanded outputs, eg control surface deflections, or loss of control functions. In digital systems, test programs are loaded to monitor voltage levels on signal inputs and parity error rates, and operational programs are checked for corruption. Critical systems require methods that provide for a quantitative measure of the margins of safety or performance available

(6) to provide for uncertainties regarding in-Service degradation. Qualitative judgements of aural, or displayed interference are usually adequate for systems that have no immediate safety or critical mission consequences. The importance of any interference observed is related to its severity and duration, as well as its consequence. The results of test may require reassessment of the original categorisation, eg if a source of interference can override UHF communications, this may be considered critical to flight safety; whereas transient interference may be judged as nuisance factor only.

AIRCRAFT FOR ASSESSMENT

It is known that interference problems vary from aircraft to aircraft of the same type, and are influenced by variations in performance of individual equipments. In the UK, importance is attached to the need for careful preparation of the aircraft used for EMC assessment, so that the results are related to a known standard of performance. This can help in identifying the reason for any EMC differences between aircraft.

In preparing an aircraft for assessment, systematic checks of aircraft equipment and antenna bonding are performed, and antenna system VSWR and loss are measured. The serviceability of equipments are carefully checked before installing, using standard schedules. Transmitters are measured for spectral content and output power, and where possible, upper tolerance samples are selected. Receivers are checked for sensitivity, and local oscillator harmonic radiation.

TEST PROGRAMME

The main EMC test programme is performed on an open site and is usually divided into three phases. The first phase is conducted using external supplies to operate the aircraft systems, and detailed tests performed. The second phase comprises tests of the engine systems, and other critical systems operating from normal flight power sources. The third phase consists of tests in the simulated external environment at A&AEE. There are shortcomings in the use of an open site in tests of on-board transmitters due to ground reflection paths. Experience has shown that this aggravates interference problems, but does not mask them. A limited flight test programme follows ground tests in order to confirm the presence of critical interference mechanisms observed during the preceding ground tests. In some systems, it is not practical to stimulate the required dynamic operating conditions on the ground, and tests can only be performed in flight.

The results of EMC tests are judged by A&AEE on the basis of the safety and performance margins available, or severity of malfunctions observed, when recommending the need for modifications to eliminate serious defects, or the requirement to avoid incompatible transmissions or mode selections, if these have little operational significance.

CONCLUSIONS

The present equipment qualification procedures are based on assumptions regarding the electro-magnetic fields present within the airframe, and the nature of the coupling mechanisms. These cannot be measured with any certainty in representative aircraft. Thus EMC assessments rely on practical tests. Avionics systems critical to flight safety, and systems vital to mission effectiveness require test methods that provide a measure of the safety and performance margins available; to account for variations that occur in production and Service use. Some proven methods are available, notably for detonator circuits, but in most other areas further work is required. Encouraging progress has been made in the use of current probes for the measurement of interfering signals on critical signal lines, in conjunction with complementary Test House procedures, as a means for obtaining the safety margins required in flight and engine control systems. Performance margins for mission systems using digital techniques are difficult to determine, and there is a need for improved test techniques.

The present EMC qualification tests for equipment in the laboratory do not guarantee freedom from interference when installed, and the results are limited in value for correlating with aircraft tests. Thus there is a need to evolve equipment design and qualification procedures that pay more attention to the effect of interference on signal lines, and the use of test conditions that can be related to easily performed measurements in the aircraft.

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APPENDIX

FREQUENCY PREDICTION

These are based on the relationship between a transmitter frequency and its harmonics, with the susceptibility of a tuned receiver due to its spurious responses; as derived from the following relationships:

$$f_{tx} = \pm n.LO_1 \pm m.LO_2 \pm IF_2 \quad (1)$$

$$= \frac{f_{rx}}{p}, \; p.f_{rx} \quad (2)$$

$$= \frac{IF}{p} \quad (3)$$

where:

f_{tx} = Transmitter fundamental frequency.

f_{rx} = Receiver tuned frequency.

LO_1 = Receiver 1st local oscillator frequency.

LO_2 = Receiver 2nd local oscillator frequency.

IF_2 = 2nd intermediate frequency.

n,m,p, = harmonic integers.

An example of this prediction procedure is given for the case of an airborne TACAN transmitter interfering with a typical double superhet UHF receiver, and the 37 relationships that may exist at four tuned frequencies in the band are shown in the Table.

f_{rx}	LO_1	LO_2	IF_2	f_{tx}	TACAN CHANNEL	$\pm nLO_1$	$\pm mLO_2$	IF_2	TEST NO
236.1	210.0	27.95	1.85	1048.15	24	5	0	-1	15
				1051.85	28	5	0	1	19
				1076.1	52	5	1	-1	3
				1079.8	56	5	1	1	9
				1104.05	80	5	2	-1	31
				1107.75	84	5	2	1	33
				1132.0	108	5	3	-1	35
				1135.7	112	5	3	1	37
243.0	206.7	34.45		1031.65	08	5	0	-1	14
				1035.35	11	5	0	1	18
				1066.1	42	5	1	-1	4
				1069.8	46	5	1	1	8
				1100.55	77	5	2	-1	30
				1104.25	80	5	2	1	32
				1135.0	111	5	3	-1	34
				1135.0	111	6	-3	-1	26
				1138.7	115	5	3	1	36
				1138.7	115	6	-3	1	29
309.1	280.0	30.95		1025.3	01	4	-3	-1	25
				1029.0	05	4	-3	1	28
				1056.25	32	4	-2	-1	21
				1059.95	36	4	-2	1	23
				1087.2	63	4	-1	-1	6
				1090.9	67	4	-1	1	11
				1118.15	94	4	0	-1	13
				1121.85	98	4	0	1	17
				1149.1	125	4	1	-1	2
395.8	370.0	27.65		1025.2	01	3	-3	-1	24
				1028.9	05	3	-3	1	27
				1052.85	29	3	-2	-1	20
				1056.55	33	3	-2	1	22
				1080.5	57	3	-1	-1	5
				1084.2	60	3	-1	1	10
				1108.15	84	3	0	-1	12
				1111.85	88	3	0	1	16
				1135.8	112	3	1	-1	1
				1139.5	116	3	1	1	7

TABLE A1 INTERFERENCE PREDICTIONS, TACAN TO UHF

The last column indicates the most probable order of likelihood of an interference. A practical schedule of tests would consist of a test sequence 1,5,7,10,12 and 20, for the highest UHF frequency chosen. From the results of these initial tests, the harmonic and image (IF) response characteristics shown to be susceptible, can be checked at the other receiver frequencies, and in this way the number of individual observations are reduced from 37 to typically 10 or 12. Note that there are no relationships for equations 2 and 3 in the example given.

CHECKING OF COMMUNICATIONS AND RADIO NAVIGATION SYSTEMS

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SUMMARY

Limited assessments need to take account of variations in ground station performance, and uncertainty regarding radio wave propagation conditions. Antenna performance can vary markedly in different aircraft to affect the radial coverage of all systems, or the accuracy achieved by Direction Finding Equipment. Aircraft transmitting antenna characteristics can be quantified for a relatively modest outlay on ground-station equipment, and subjective communications quality assessments can be enhanced by using suitable yet inexpensive voice recorders that are also useful for noting observed navigation data for subsequent analysis.

An accompanying aircraft of known performance can provide a comparative basis for the assessment of range performance and provide the means for checking air to air modes. Test objectives should be related to the intended operational use of the aircraft, but need to be interpreted into schedules for use over standard test routes, that have the widest application for all classes of aircraft. An example is given of a possible schedule of tests and the form of analysis that might apply.

INTRODUCTION

All communications and Radio-Navigation Aids rely on the characteristics of radio wave propagation and the ground station for the way in which they perform at any time from one station to another, and the extent by which a flight test Engineer can account for these variables, determines the credibility of his assessment. The subject of radio wave propagation is complex (1), but I will attempt to summarise the characteristics for the frequency band 10 KHz to 1 GHz, within which most of the commonly used military Comms and Nav systems operate, because an appreciation is essential for anyone considering flight testing of these systems.

At low frequencies, up to 2 MHz, ground wave mode of propogation is predominant, strongly influenced by the conductivity of the soil or water. From 2 to 30 MHz 'sky-wave' is the major mode, propagating via the ionosphere. 'Sky-wave' mode is very dependent on frequency, according to diurnal and seasonal influences and sunspot activity, that affects the ionosphere. Above 30 MHz, direct wave, or 'line of sight' mode predominates, which is horizon limited, but less dependent on diurnal or seasonal effects, although subject to anomalous propagation due primarily to variations of refractive index of the atmosphere, associated with temperature inversions, commonly referred to as "ducting" which can cause a loss or increase in range according to the height flown. From about 100 KHz to 10 MHz, ground and sky wave paths can interfere to cause severe fading, whereas above this frequency, and sometimes up to as high as 100 MHz, sky wave does not overlap, but can propagate for long distances. The losses in the 'diffraction' mode of propagation beyond line of sight, increase with frequency and vary according to terrain profile. Associated with propagation losses are the sources of noise that determine the minimum usable signal. Below about 100 MHz atmospheric sources of noise, or man-made noise generated within the aircraft, or resulting from overflying industrial areas may degrade receiver performance. Above this frequency the usable signal is normally dependent on the receiver noise threshold alone.

COMMUNICATIONS

Most military communications systems operate either in the HF/SSB/AM (2-30 MHz), the Tactical VHF/FM (30-76 MHz), or the V-UHF/AM bands (118-136, 225-400 MHz). HF is used in long range aircraft for communications up to about 2,500 nm using horizontally polarised antennas and depends primarily on sky-wave mode of propagation, which require the use of frequency prediction charts (2) to determine the optimum working frequency. HF is also used for close support aircraft to exploit the short range ground-wave mode of propagation, which is less affected by beyond line of sight losses in rugged terrain. Satisfactory performance for this mode of propagation requires vertically polarised antennas of good efficiency at low frequencies, and few aircraft antennas meet these requirements (3). Tactical VHF equipment is frequently used for the same role for communications in relatively flat or gently rolling terrain. The performance beyond line of sight is very dependent on terrain characteristics and antenna and equipment performance on the ground and in the aircraft. Most military aircraft are fitted with VHF for communication with civil air traffic authorities, and UHF for military use. Limited propagation beyond line of sight is feasible (4) but is not normally exploited in military applications.

RADIO-NAV AIDS

The most commonly used Nav Aids are summarised in Table 1:

Type	Designation	Frequency	Coverage	Accuracy	Error Sources
Bearing Only	MF/DF	100 KHz-2 MHz	30-500 nm	± 2° to ± 5°	Aircraft Antenna Location, Propagation
	UHF/DF	225-400 MHz	Line of Sight	± 5°	Aircraft Antenna Location
	HOMERS	30-400 MHz	Line of Sight	± 5° to ± 15°	Aircraft Antenna and Ground Station Location
Range and Bearing	TACAN	960-1215 MHz	200 nm or Line of Sight	± 1°, 0.2 nm	Ground Beacon Calibration
	VOR/DME	118-136 MHz (VOR) 960-1215 (DME)	200 nm or Line of Sight	± 1°, 0.2 nm	Ground Beacon Calibration
Hyperbolic Position	OMEGA	10-13 KHz	Worldwide	± 1 to ± 3 nm	User Interpretation. Propagation. Antenna Noise
	LORAN-C	100 KHz	550-1500 nm	± 0.5 to ± 10 nm	User Interpretation. Sky wave Propagation
	DECCA	70-130 KHz	250 nm	± 0.25 to ± 5 nm	User Interpretation. Sky wave Propagation

TABLE 1 RADIO NAVIGATION AIDS - TYPICAL PARAMETERS

MF/DF 'Radio Compass' equipments are generally used in areas not served by other relatively inexpensive aids. Their accuracies are very dependent on aircraft antenna location, and the care exercised in determining errors and inserting the corrections applicable to each installation. Individual MF/DF beacon transmissions are subject to propagation anomalies that can cause large errors in apparent direction over land/sea paths or complex geological strata. UHF/DF accuracy is also very dependent on the aircraft D/F antenna location and aircraft attitude with respect to the sight-line, such that air-to-air D/F errors may differ from the air-to-ground case. Tactical VHF homers are used for various applications in close support helicopters, and UHF homers in search and rescue, air-to-air rendezvous, or supply dropping in other aircraft. Homers are also very dependent on antenna location for their performance.

TACAN is the most widely fitted position aid in military aircraft, and Figure 1 shows the disposition of the principal European beacons, derived from the RAF en-route supplements (5). The rated coverage of beacons is dependent on geographic location, co-channel interference, and aircraft height. The accuracy of modern airborne equipment is typically better than 0.2 nm in range. Bearing accuracy is very dependent on beacon calibration, which has been known to be in error by as much as 3° (6). Range measurements from two beacons are capable of providing one of the most accurate Nav fixing techniques available. VOR/DME has very similar characteristics to TACAN, primarily for use on civil air routes. The hyperbolic aids provide better low altitude coverage than TACAN, but are more susceptible to propagation path errors. They are very dependent on operator interpretation and receiver ambiguity resolving characteristics for the accuracies attainable in practice, particularly in the sky wave interference region. Decca is capable of high accuracy when used with care (7), and provides coverage in northern and western European areas. Loran-C has a wider coverage as shown in Figure 2, but is primarily intended for over sea routes, and appears to be less well documented in terms of accuracies attainable for aircraft use in the areas concerned (8). Omega is the least influenced by propagation, except during the hours of sunrise and sunset, by virtue of the redundancy available from the overlapping and worldwide coverage of the ground stations.

PROBLEM AREAS

Aircraft communications installations frequently fail to achieve optimum performance, because of mismatch between headsets and transmitter/receivers. Correct matching of impedance and sensitivity of mictels are usually achieved in a customised manner within the Communications Control System (CCS) of each aircraft. The choice of headset plays an important role in communications performance according to its acoustic noise attenuation or rejection properties. Microphone signals can overdrive CCS amplifiers or transmitters, introducing distortion if incorrectly matched, or under conditions of high acoustic noise. Voice operated switches and various automatic modulation control devices vary in their ability to compensate for the shortcomings and are often incapable of functioning satisfactorily over the flight envelope range of acoustic noise, or user speech levels. Failure to optimise CCS interfaces can also lead to loss of communications range due to under modulation of transmitters, and excessive sidetone levels can produce a similar effect. Changes in volume between services, or distracting crosstalk, and electrical interference from other aircraft systems are also commonly met shortcomings in aircraft installation.

I have already mentioned some of the problems associated with the siting of antenna systems, and it is rarely possible to satisfy the often competing requirements of all systems, given the constraints imposed by aircraft shape, size and aerodynamic requirements. However, the selection of antenna location does not always match operational priorities for system usage. The choice of antenna can influence the efficiency attainable in terms of signal loss, and hence range, particularly below 10 MHz. Loop antennas are often more effective than capacitance antennas for frequencies below about 1 MHz, as they are less susceptible to aircraft static noise in cases where wick dischargers are not fitted, or are ineffective. Incorrect mounting or bonding of antennas, or excessive attenuation in feeder systems are frequent sources of performance loss and a contributory factor in electromagnetic interference; these problems often arise through inadequate design consideration of the installation requirements, or the result of subsequent in-Service

degradation due to corrosion, fluid contamination or chafing, - easily overlooked if adequate means of access for inspection and testing has not been provided. Poor frequency accuracy or drift; long and short term stability of transmitter power, and variation in receiver sensitivity across the frequency band, are also common sources of performance variation, which are sometime aggravated by the thermal environment in flight.

TRIALS FACILITIES AND REQUIREMENTS

Before I describe the minimum facilities required for checking aircraft Comms and Nav systems, I will briefly outline those used in the UK for this purpose. Most systems are evaluated in Laboratory aircraft such as the Comet, Figure 3 and using Groundstation equipment as shown in Figure 4. Advantage is taken of the relative ease by which comparative or absolute tests of different systems can be made in the Comet, using on-line computing and recording facilities, and on-board and external datum aids. The on-board navigation datum aids include Decca, two-range Tacan, twin Loran-C and Inertial systems, high and low altitude Radio Altimeters, a Doppler DR system and vertical cameras; in conjunction with precision ground tracking radars, synchronised with airborne records, by air-ground UHF links. In this manner, the intrinsic equipment characteristics can be determined in more representative conditions than on the ground, leaving the in-flight assessment of operational aircraft to concentrate on the installation dependent aspects of systems, with the aid of the ground station facilities.

The A&AEE ground station antenna farm is shown in Figure 4a. The V/UHF tower is of interest because the disposition of the dipole antennas at various heights is according to the test frequencies used, so that each antenna is approximately 10 wavelengths above mean ground level. The coincidence of the vertical lobe patterns provide a common basis for assessing signal strength during range assessments, and allow the use of one height for measurement of several polar diagrams at different frequencies. The predictions of the theoretical performance shown in Figure 5 are based on simple ray theory modification of the free space propagation law taken from Williams (9). These are subject to error, and in-flight calibrations are performed using the Comet Laboratory aircraft as shown in Figures 5 and 6. The ground station is equipped (Figure 4b) for measurement of signal strength and is capable of recording signal strength from up to six antennas simultaneously, covering HF, VHF and UHF, with facilities for monitoring modulation levels, transmitter powers, and recording and assessing speech quality. The measuring instruments are IEEE bus compatible and interconnected with a computer data acquisition and processing system.

For limited assessments, much can be done on a purely subjective basis, given that a small dedicated team is available, consisting of flight test Observers or Pilots with a wide experience in the use of the systems considered, and complemented by a test Engineer having a good appreciation of radio wave propagation, antenna and transmitter/receiver design characteristics, and with practical experience of maintenance and calibration of installations. The Engineer can be responsible for any quantitative measurements and their analysis, and the monitoring or operation of the ground station facilities associated with the assessments.

A ground station site should be selected on a geographic basis so that it lies on a relatively low-lying flat open site for a radius of at least 1 nm and provides a suitable track as free as possible of air space restrictions or mountainous terrain up to 200 nm. If possible the site should be co-located with a TACAN beacon, and the track should feature regular landmarks for use by aircraft not equipped with suitable fixing aids. Desirably the ground station should be equipped with means for monitoring signal strength. The simplest reliable method is to employ an antenna change-over switch, so that a signal generator may be used to provide a calibration signal, which is adjusted to the level of the signal observed on a meter monitoring the receiver automatic gain control line, after each reading. One of the more competitive methods for monitoring signal strength, is to use one of the current wide frequency range spectrum analysers (10) which can provide direct reading of signal amplitude to an accuracy better than 1 dB, and frequency to an accuracy in the order of 1 part in 10^6. If equipped with data bus facilities, automatic recording or data processing can be added at a later date. However, useful assessments can be made on a subjective or comparative basis, provided that certain requirements are met:

1 Standard test frequencies are allocated for trials, so that all assessments can be correlated with frequency, and interference from other traffic is minimised.

2 Test altitudes are maintained for all results, eg one low and one high altitude condition chosen to suit the widest range of aircraft.

3 Outbound and inbound legs always made on a common track for all aircraft.

4 Orbits always performed in the same area, associated with a given altitude as determined in 2 above.

5 Tape recordings are made of all communications, using a good quality recorder.

6 All incoming signals should be monitored on an oscilloscope, and modulation meter to assist in diagnosis or reasons for poor intelligibility.

7 Qualitative assessments of signal strength and readability should be noted by ground and air observers according to a standard code such as that given in Appendix A.

8 The ground station performance characteristics are monitored as indicated in Table 2 below.

ITEM	MONITOR	TEST FACILITIES
Antenna and Feeders	Physical condition waterproofing, corrosion of connections. Antenna VSWR, feeder attenuation.	Thru-line wattmeter or network analyser
Transmitter	Power output. Frequency accuracy. Modulation depth.	RF wattmeter Frequency meter Modulation meter
Receiver	Signal to noise ratio. Squelch level. Frequency accuracy.	Signal generator Audio wattmeter Frequency meter

TABLE 2 GROUND STATION MONITORING

The ground station antenna radiation patterns, such as the example shown in Figure 6 can only really be satisfactorily determined by controlled calibration flights. Nevertheless, by flying a variety of aircraft on common heights and tracks, it is possible to arrive at an estimate of the expected performance to maximum range, as indicated by the aircraft results plotted in Figure 5. The range at which there is little variation of signal strength with distance, ie between 95 and 120 nm in Figure 5, should be explored by flying at 90° to the chosen track in order to determine the cross-track width of the area for constant signal ± 1 dB, as in Figure 6. This then represents the optimum area within which polar diagrams may be measured. Preliminary calibrations of this nature, even if only done subjectively, using a variety of aircraft, are essential if assessments of any new aircraft are to avoid erroneous conclusions due to adverse ground station characteristics.

For the aircraft, means for recording speech at the headset telephone are required, and are sometimes available as part of the communications installation, but where not, it is usually relatively easy to install a good quality commercial battery operated cassette recorder, connected by means of an adaptor plug inserted in series with the mic-tel socket. The recorder should not be of the type using automatic record level circuits. The record gain needs to be set so that the recorder itself does not overload in the worst flight condition for acoustic noise with speech, and this is best determined experimentally before the scheduled test programme. The final setting should be checked, and a 1 KHz test tone recorded to provide an amplitude calibration at the beginning of each cassette. A locally manufactured battery powered oscillator small enough to be carried by the aircrew will enable this to be done in flight on some aircraft. It is also desirable to be able to monitor received signal strength of communications or navigation receivers on a meter display, but this is usually difficult to achieve in a simple manner, unless an appropriate signal line not susceptible to calibration drift is readily accessible. For navigation systems, independent position references are required, and in the UK, certain instrumentation and position datum facilities such as I have already mentioned when describing the Comet Laboratory aircraft are sometimes fitted, but they require extensive modification to the aircraft installation and are not practicable for limited checks. Visual 'on-top' fixes of surveyed landmarks are an adequate means at low altitude for checking most positional navigation systems for gross errors. Homing systems often employ uncalibrated null indicators, and paper scales graduated in arbitrary units, pasted to the face of the instrument enable readings of deflection to be noted against compass heading, from which the response of the homing system may be judged.

TEST OBJECTIVES

I do not propose to describe individual tests for every system in the catalogue of Comms - Nav Equipment, as the general principles may I hope be deduced from what I am about to discuss. What can be achieved in any limited programme depends on the particular aircraft, its endurance, crew complement and range of equipment requiring assessment. The objectives for a limited assessment should take account of the prime operational role for the aircraft, particularly in terms of the choice of heights and speeds for the determination of system performance, or the importance of maintaining system operation in turning flight. Nav systems require good all round coverage in level flight but can usually accept degradation in turns, because position fixing is normally done in level flight. Communications systems also require good all round coverage in level flight, and with the possible exception of highly manoeuvrable aircraft such as interceptors, some degradation in turns may be acceptable provided that this only occurs on the beam aspects. If two antennas are required for a given system, they should provide complementary azimuth coverage. Antenna vertical coverage at VHF and UHF is strongly influenced by flight attitude; and aircraft whose incidence angle varies considerably over the height/speed range should be flown at representative speeds for the heights chosen to determine the range performance.

High cockpit noise levels may occur at different points of the speed/height envelope, according to the source of noise, be it aerodynamic, engine or air-conditioning in origin, so tests of the effect on communications intelligibility should be made in the flight regime that is operationally most severe. The flexibility of facilities of multi-crew aircraft are closely related to the operational requirements in terms of the need for independent operation of communication systems and intercomm networks; which should be considered in an assessment along with the ergonomic considerations of placement of controls and displays for ease of use and viewing. Accessibility of equipment for maintenance, and the extent and value of Built-In-Test (BIT), or need for external test equipment are important considerations in determining the degree of front line readiness that might be achieved.

GROUND AND FLIGHT TEST PROGRAMMES

The extent to which antenna systems can be usefully checked on the ground are limited unless an RF network analyser is available for antenna Voltage Standing Wave Ratio (VSWR) and co-axial feeder loss measurements. Most passive antennas may be checked in this way, but these measurements are usually imprac-

tical for tuned antennas. For transmitting antennas a thru-line wattmeter provides an easily used method for determining transmitter power and, if measurements are possible at the antenna as well as the transmitter connections, the feeder loss and antenna VSWR may be determined. These measurements should be made at the test frequencies to be employed which should be evenly spaced across the band. Methods do exist for checking polar diagrams of antenna systems on the ground. Full sized and scale models of aircraft are used as a design aid for optimising antenna location on most UK aircraft; but they are subject to limitations in measurement technique or modelling accuracy, so that actual installed performance is always confirmed in the UK by in-flight polar diagram measurements.

Intercomm systems should be checked on the ground in order to assess the noise levels under quiet conditions and whilst the various electrical and avionic systems are switched on, and the dynamic range for speech. Ground station transmissions should be monitored on the aircraft receivers as part of the intercomm tests, to judge speech quality and determine the balance of volume between services and crosstalk characteristics. Transmitter sidetone levels should be assessed at the volume previously set for satisfactory reception. The ground station should record the aircraft transmissions and monitor speech quality, modulation depth and if possible, transmitter frequency. Voice recordings should also be made at the aircraft.

Nav position aid accuracies may be checked where signals are receivable on the ground, with respect to the airfield position, and to confirm that their behaviour is consistent with experience of other aircraft installations. Ground 'swings' of MF/DF are a suitable method for determining D/F accuracy (11), provided that the site chosen is calibrated by a competent agency. Class I magnetic compass bases are often wrongly assumed to be satisfactory for this purpose; however, a site suitable for MF/DF use is invariably satisfactory for magnetic compass use. It is usual to check the accuracy of the aircraft compass heading system at the same time.

The simplest approach for limited flight tests is to make comparative assessments, using an aircraft of similar capabilities to the aircraft undergoing tests. This guards against the possibility of erroneous conclusions due to propagation anomalies or ground station defects, and provides a relative basis for performance assessments. The choice of chase aircraft should be restricted to those whose performance characteristics are already well known from the ground station calibration tests mentioned earlier. Two aircraft flying in company will also enable communications and Homer, or TACAN air-air modes to be checked.

V/UHF Communications, TACAN or DME range performance tests should be performed at the pre-determined heights and radial tracks from the ground station. On the outbound run at optimum cruise speed, two-way air-to-ground communications should be checked at 10 nm intervals at high altitude, or 5 nm intervals at low altitude, using TACAN or other means for relaying information on distance, and noting readability and/or signal strength. These transmissions should continue until two-way contact is lost. The aircraft should continue outbound for a further distance approximately 20% of that already flown, before flying on a reciprocal track. The ground station and aircraft should make regular calls once contact is lost, these should be brief call-sign exchanges to minimise the possibility of overlapping transmissions. Once contact is regained the procedure continues until the area for constant signal level (figure 6) is reached, where the aircraft executes an orbit, transmitting at 10° heading intervals, and listening to ground station replies. The procedure is then repeated for opposite bank. Continuous voice recordings are made of all transmissions in the aircraft and at the ground station. For TACAN, VOR/DME systems the ranges (or headings during orbit) at which loss of lock occurs are noted, unless facilities are available for signal strength monitoring. Desirably results should be obtained for several frequencies in the band. When assessment time is limited and only one frequency is possible, this should be in the upper part of the band for VHF and UHF, where the antennas are usually, though not invariably less efficient. Range performance of HF systems can only be assessed at frequencies that are dictated by the propagation conditions for the time of day and path over which the tests are contemplated, so that comparative tests between two aircraft are the only feasible approach in a limited assessment. Hyperbolic aids, which are receive only systems, depend on the means available for observing signal strength and it is usually more pertinent to observe the consistency of position information near the limits of ground wave cover, by overlaying position plots from an accompanying aircraft and observing their characteristics during a low bank angle orbit in the same region. Such tests should be performed well outside sunrise and sunset periods. MF/DF systems range performance is somewhat academic, provided that stable unambiguous bearing information is attainable beyond the rated range of beacons.

Antenna polar diagrams (PDs) for the level flight condition are obtained by one of two methods in the UK. For aircraft with high altitude capability, polygonal orbits are performed whereby the aircraft levels the wings at increments of 30° for 5 seconds, until 36 headings have been accomplished. For aircraft of low altitude only capability, this method is unsuitable and a 'cloverleaf' pattern is used, whereby the aircraft flies over a pin-point on successive headings. In each case the aircraft transmits the heading being flown, and for the 'cloverleaf' pattern, counts down to the overhead position. Thirty-six heading azimuth polar diagrams for each antenna and several frequencies in each band are very time consuming, and the compromise shown in Figure 7 is one method of reducing the time taken without serious loss of information. Switched antennas can be measured during one pattern, by selecting in sequence at each heading. HF system polar diagrams can only be performed at short range and low altitude, because of the problems of sky-wave interference and it is debatable whether such PDs have much value in assessments. TACAN or DME should be selected to a channel not employed by beacons in the region, so that the ground station can monitor an otherwise unoccupied channel for the aircraft PD transmissions. Homing system PDs are useful for determining the locations of false homing nulls which may have pseudo-stable homing characteristics. The false nulls in Figure 8 at 70° and 80° for instance, would merit examination in flight to see if this were so. Because of the strong dependence on frequency, even limited assessments should attempt homing tests on more than one frequency. The tail aspect null must exhibit clear reverse sense indications, and on the head aspect it must be sharp enough to give positive directional information, but at the same time be tolerant of reasonable aircraft manoeuvre, ie ± 20° bank angle, without a tendency to overshoot the homing course. When the homing system is fitted with a sensitivity control these characteristics will be affected by the setting. The Homing characteristics will also be affected by the type of transmitting source and the aircraft distance from it.

A subjective assessment of communications quality (Appendix A) can be augmented by tests that assist in determining the underlying reasons for any degradation observed. Recordings should be made of aircraft test transmissions with the microphone circuits OFF, microphone ON with no speech, and normal speech transmission. This sequence should be performed at normal cruise altitude conditions and repeated for any condition for high acoustic noise, in an area of good signal strength at the ground station. The ground station should also record these transmissions, and, if possible monitor the modulation depth for each case, and inspect the audio signal on an oscilloscope for indications of excessive waveform clipping.

Navigation accuracy should be checked for gross errors against visual fixes, and by cross-reference to other navigation sensors in the aircraft. MF/DF Radio Compass air swings should be performed during daytime at heights not less than 4,000 ft and at a distance that is within the rated range of the beacon. Beacons need to be chosen on the basis of local knowledge regarding their accuracy and freedom from interfering stations. A downwind homing run should be made, passing over a suitable landmark to determine the DF error with respect to the aircraft heading reference. Further passes should be made over the landmark on successive headings in order to determine an eight-point air calibration. A range run should be performed at a similar height until DF indications begin to wander significantly, say ± 5° and at this range a low bank angle orbit should be performed; at several points around the orbit, the aircraft should momentarily bank steeply to check that the DF indications restore correctly and without ambiguity. The beacon should be over-flown in order to check that the cone angle of uncertainty is symmetric relative to the overhead position, and not excessively large such that determination of overhead position is difficult.

TYPICAL TEST SCHEDULE

I have drafted a schedule for a limited assessment of an imaginary tactical strike aircraft, the 'Gremlin Mk 40' (Appendix B) that sets out to determine the range performance, antenna polar patterns, communications quality and general navigation performance of the aircraft systems, in a manner that could be accomplished in two sorties by an experienced test crew; although in practice it would be prudent to allow a further sortie in order to complete any unfulfilled objectives. The systems I have chosen for assessment are listed in Table 3 and the antenna locations are shown in Figure 9.

EQUIPMENT	ANTENNA
UHF (Main) 225-400 MHz Transmitter 20 watts Squelch lift 2μV 10 dB Signal/Noise 5μV	(1) Fin Cap, Suppressed. (2) Undernose Blade.
UHF (Standby) 238-247 MHz Two channel Transmitter 5 watts Squelch lift 5μV 10 dB Signal/Noise 8μV	a Shares main UHF antennas. b Automatic selection of undernose blade when generators fail.
UHF Homer, (associated with main UHF receiver).	Twin blades behind canopy.
TACAN. 126 channel Transmitter 1.5 KW peak Range & Brg lock threshold - 90 dBm	Single blade under rear fuselage.
ADF Radio Compass MF/DF 150-2000 KHz	Ventrally mounted suppressed loop. Suppressed sense antenna in wing leading edge.

TABLE 3 COMMS-NAV INSTALLATION

I have assumed that the aircraft is fitted with a DR navigation system comprising a twin-gyro platform and doppler velocity and drift sensor, with error characteristics in the order of 1.0°/hour in heading and 10 nm/hour in radial position. The general aircraft performance assumptions are given in Table 4.

ALTITUDE (FT)	SPEED kn (TAS)	ENDURANCE
30,000	480	2 hrs
3,000	420	1 hr (includes 5 min at 540)
300	540	

TABLE 4 AIRCRAFT PERFORMANCE ASSUMPTIONS

The schedules are based on the use of a co-operating chase aircraft equipped with UHF comms and TACAN, whose performance characteristics have already been established over the routes chosen. I have also assumed that both aircraft have voice recording facilities installed; that the tests are being performed from an airfield equipped with ground station facilities for monitoring UHF and TACAN received signal strength, TACAN and NDB MF/DF beacons and that the routes used have been calibrated in the manner I have already dis-

cussed. A summary of flight tests are given in Table 5 below, and the sortie patterns are given in Figure 10.

Sortie	Flt: Condition	Test	Time/Mins
1	30,000 ft 480 kn	Outbound Range Assessment.	25
	30,000 ft 480 kn	Short Range Air-Air Comms and Homing.	10
	30,000 ft 480 kn	Inbound Range Assessment.	15
	30,000 ft 480 kn	Polar Diagrams.	30
	30,000 ft 480 kn	Long Range Air-Air Comms and Homing.	10
	5,000 ft 280 kn	Homing Polar Diagram.	15
	5,000 ft 420 kn	Air-ground Homing.	10
		TOTAL	2 hrs
2	300 ft 540 kn	Comms Intelligibility.	5
	3,000 ft 420 kn	Outbound Range Assessment.	10
	5,000 ft 420 kn	MF/DF Air Swing.	15
	3,000 ft 420 kn	Inbound Range Assessment.	7
	3,000 ft 420 kn	Hi-low, Lo-Lo, Air-Air Comms.	7
		TOTAL	47 min

TABLE 5 SUMMARY OF FLIGHT TESTS

ANALYSIS

In analysing the results from the limited tests I have described, it should be possible to identify any important weaknesses in the systems and obtain a comparative assessment of whether the aircraft was better or worse than the chase aircraft employed. If the trial were performed on a day when anomalous propagation conditions prevailed, meteorological data on the presence of temperature inversions, or unusual gradients in refractive index should corroborate the chase aircraft evidence of abnormal performance over the established routes. The radio horizon is more usually taken as the limit, based on a 4/3 earth radius, or $1.23\sqrt{h_{ft}}$ (nm). However this value varies significantly, particularly in the lower latitudes (4). It is safer to use the optical horizon ie $1.06\sqrt{h_{ft}}$ (nm) as the criterion for predicting the expected range for satisfactory operation of UHF and TACAN equipment. This would represent 180 nm at 30,000 ft or 65 nm at 3,000 ft over relatively flat terrain.

I have invented some antenna PDs from the sortie profiles as shown in Figures 11 and 12. These suggest that the UHF fin-cap antenna has a very lobed pattern when compared with the lower antenna. Furthermore, both UHF antennas have poor coverage to the rear of the aircraft and are unlikely to be adequately complementary, which would have been apparent during the outbound range runs. The vertical pattern of the fin-cap is probably multi-lobed and this would cause difficulty in achieving consistent air-to-air communications at the longer ranges, and different relative altitudes for the sortie 2 case; for routes over ground of high surface reflectivity it could aggravate rapid fading effects due to multipath wave propagation.

The standby UHF, having lower power than the main UHF will not provide the range performance of the latter at high altitudes. The standby UHF range assessment was chosen for the low altitude case because this reduction in power is more than compensated for by the expected decrease in propagation loss at the reduced horizon range, provided that the antennas perform satisfactorily. The polar diagram for the lower antenna in level flight shown in Figure 11 has significant loss on the tail aspect, but in an emergency situation the aircraft would most probably be heading towards base and performance might be regarded as satisfactory.

The TACAN antenna would most probably be adversely affected in the area of the head aspect, due to screening by the under-fuselage missiles. Some screening may also occur on the beam aspects due to the wing stores which would be particularly noticeable in banked turns; this could be acceptable, whereas loss of performance on the head aspect may be sufficiently serious to merit the addition of a complementary antenna. As the schedule did not call for measurements of TACAN signal strength except during the polar diagram, the range performance will have to be judged on a comparative basis using the loss of lock indications and their frequency of occurrence, in conjunction with the polar diagram and the chase aircraft results.

The Radio Compass air swing may show significant errors due to the presence of external stores which could merit further investigation on the ground calibration site. Lack of symmetry in the overhead indications might be evident as a result of the non-ideal location of the loop antenna, and may be an acceptable operational limitation, but a failure to resolve bearing ambiguities in some directions during the 'air-swing' manoeuvres, could be sufficiently serious to justify relocating the sense antenna.

The tests of the Homer installation will enable the pilot to judge whether the flight path is controllable, and the homing squint can be deduced by plotting the track flown from the recorded TACAN data, and the pilot's own observations from the air-to-air cases. The homer azimuth response in Figure 8 would

indicate whether or not there are likely to be ambiguous homing bearings. It should be remembered that this type of Homer may have other frequencies at which the response characteristics could be significantly different, and that completely satisfactory performance with respect to range, sensitivity and controllability, squint error or overhead performance, at all frequencies is unlikely to be achieved.

The communications quality may be judged from the ground station observers and pilot's reports, but particular conditions such as the low level high speed tests may need further examination of the recordings in order to establish whether the causes for loss of intelligibility are due to CCS overload or excessive background noise, perhaps due to inadequate microphone or headset acoustic attenuation. Care needs to be exercised in interpreting the recordings of the pilot's telephone signal, as these do not reproduce the acoustic noise to signal condition actually present at the ear. A comparison of chase aircraft recordings with those of the aircraft under test may show significant differences in the way in which the airborne receivers respond to rapid signal fluctuation, or very strong signals during the air-to-air tests.

CONCLUSIONS

Worthwhile limited assessments of Comms-Nav systems can be made by a small but experienced and dedicated team with relatively moderate resources, provided that the test objectives are carefully considered beforehand and conducted over well rehearsed routes, so that the performance expectations are predictable. The requirements for these routes, and the profiles flown typically do not represent the more interesting operational situations, but they do provide a more objective basis for comparing various aircraft and systems performance. The use of a known chase aircraft, not only provides a control check on ground station performance and propagation conditions of the day, but also allows the scope of the test to be extended to cover air-to-air requirements.

Navigation system tests need to concentrate on the range and antenna dependent characteristics, and the consistency of performance and ease of use rather than accuracy in absolute terms, as the latter requires a more elaborate programme of trials.

Subjective assessments of communications quality are more appropriate to limited programmes, particularly if use is made of speech recordings to help in identifying the features that may be the cause of loss of intelligibility. The ability to monitor signal strength on the ground enables useful quantitative data to be obtained on antenna performance and does not require extensive facilities.

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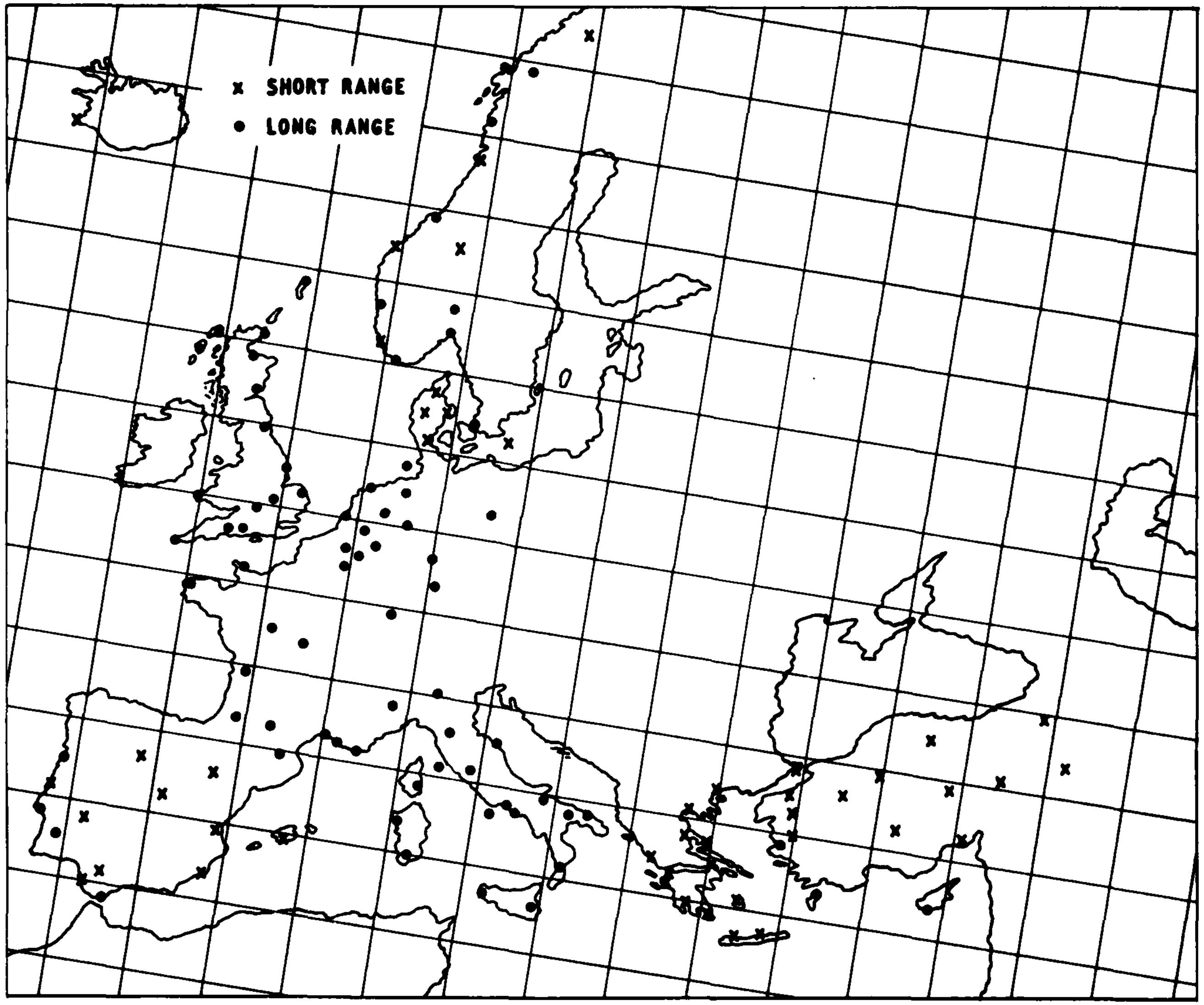


FIG. 1 PRINCIPAL EUROPEAN TACAN BEACONS.

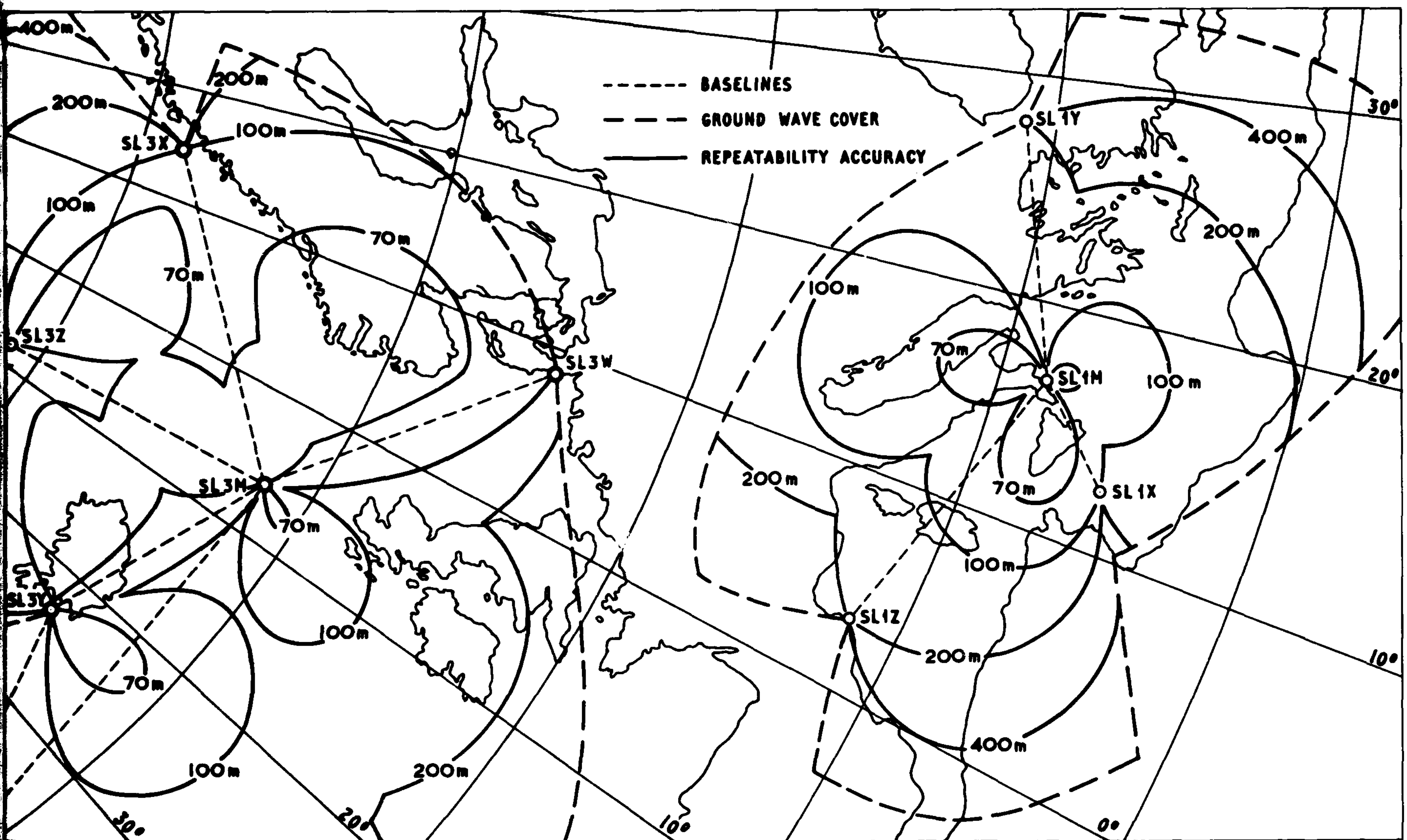


FIG.2 EUROPEAN LORAN-C GROUNDWAVE COVER.

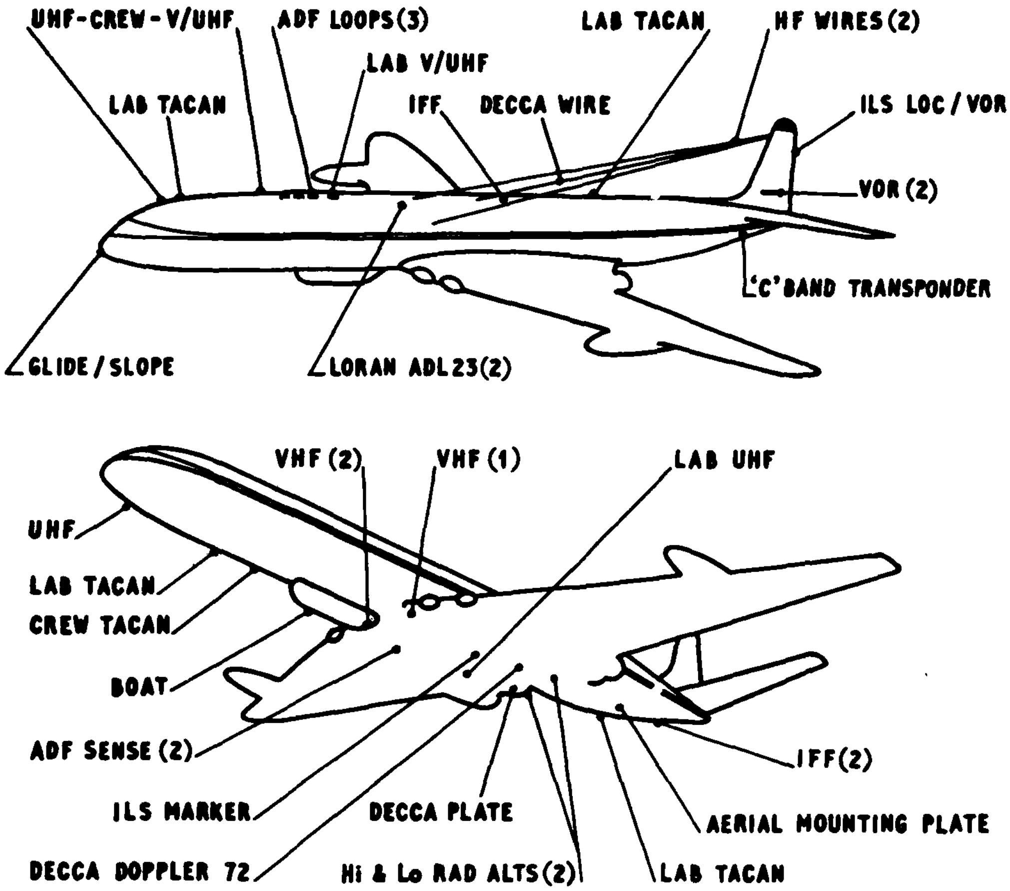


FIG. 3a COMET 235 - ANTENNA SYSTEMS.

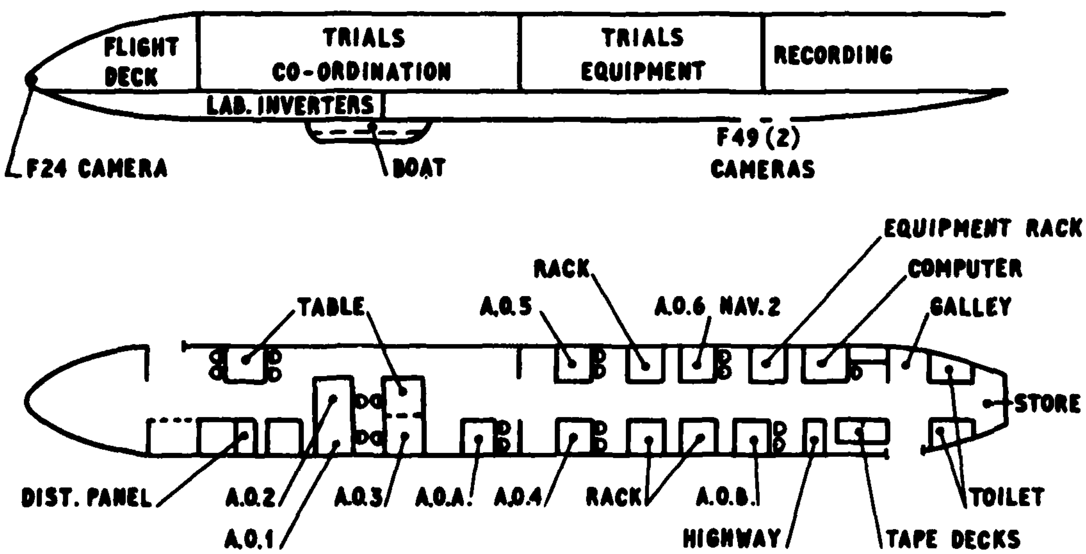


FIG. 3b COMET 235 - INTERIOR LAYOUT.

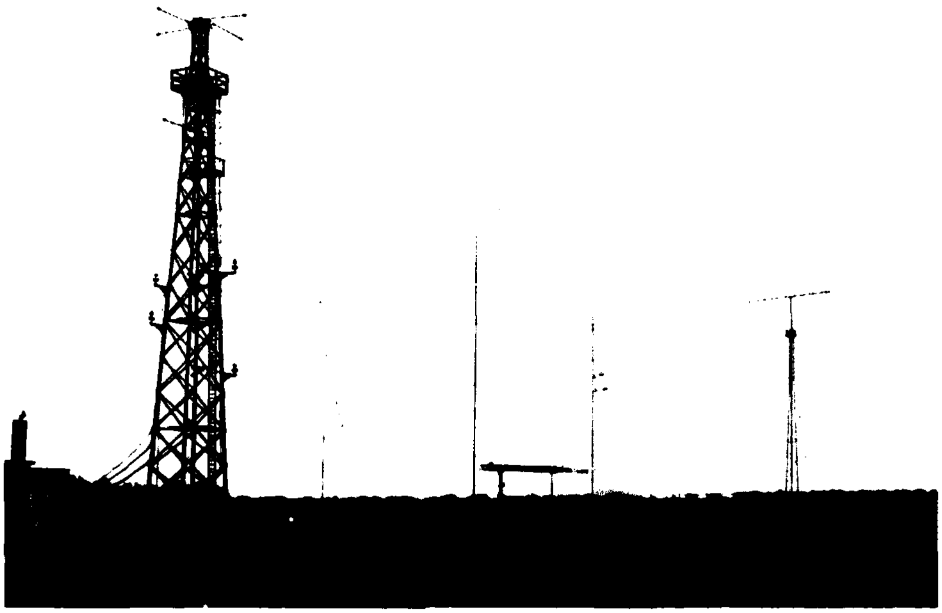


FIG. 4a Antenna Farm

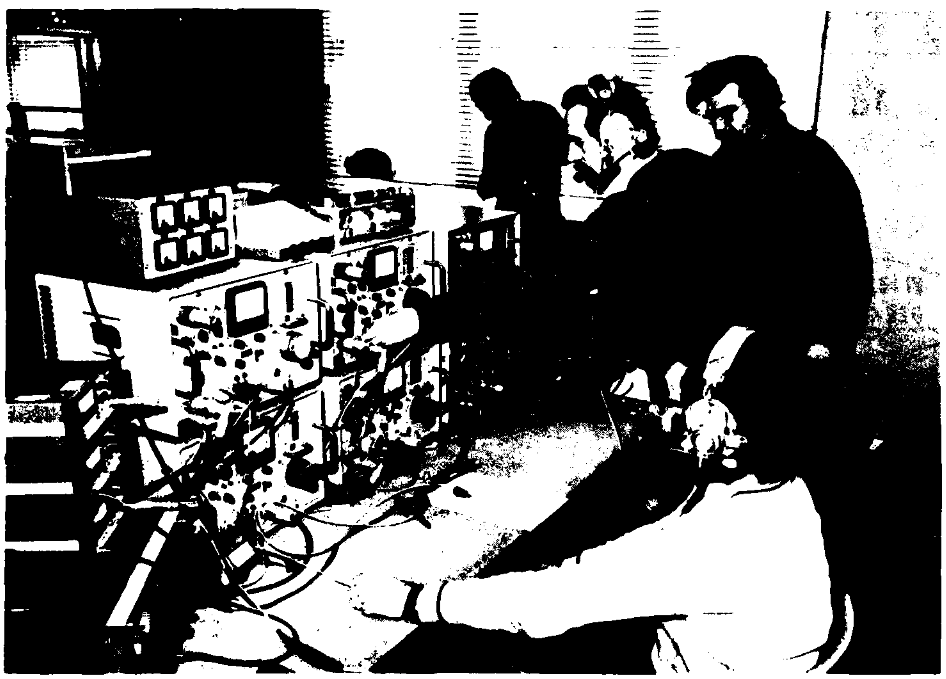


FIG 4b Laboratory

FIG. 4 GROUNDSTATION FACILITIES

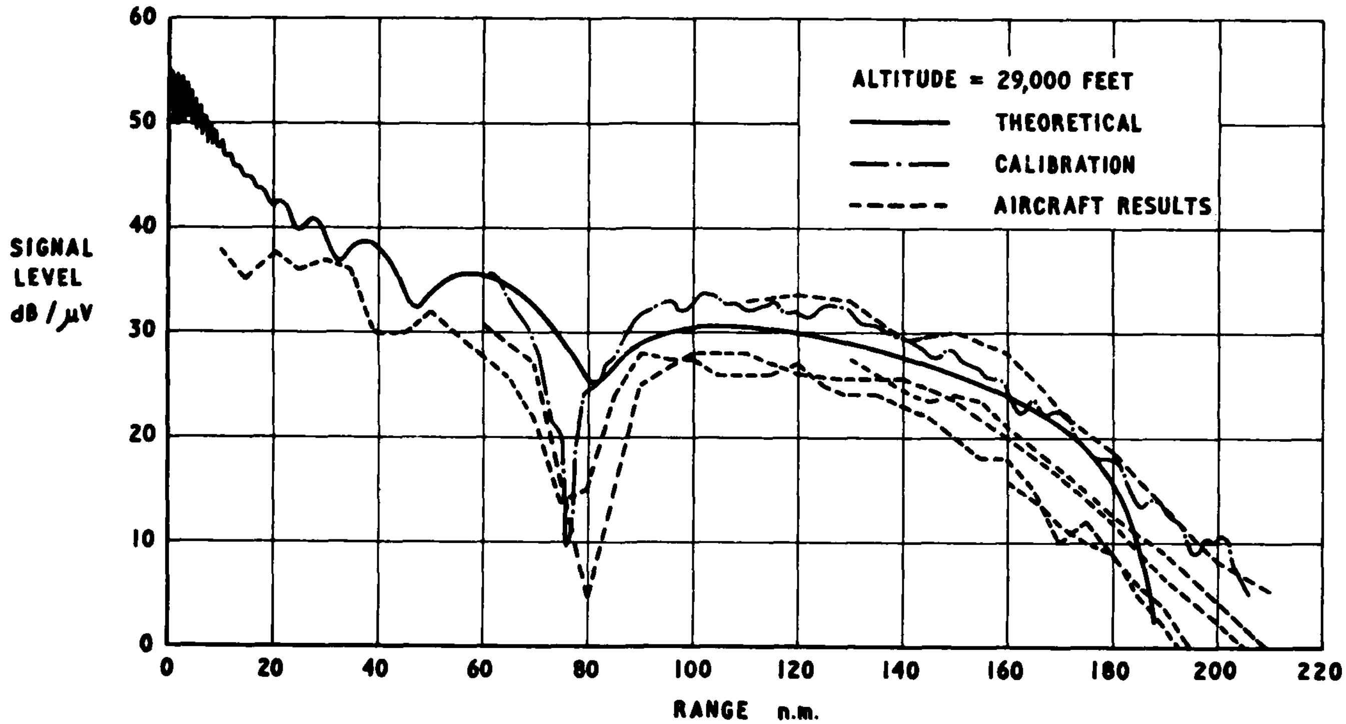


FIG. 5 V.H.F. GROUNDSTATION PERFORMANCE.

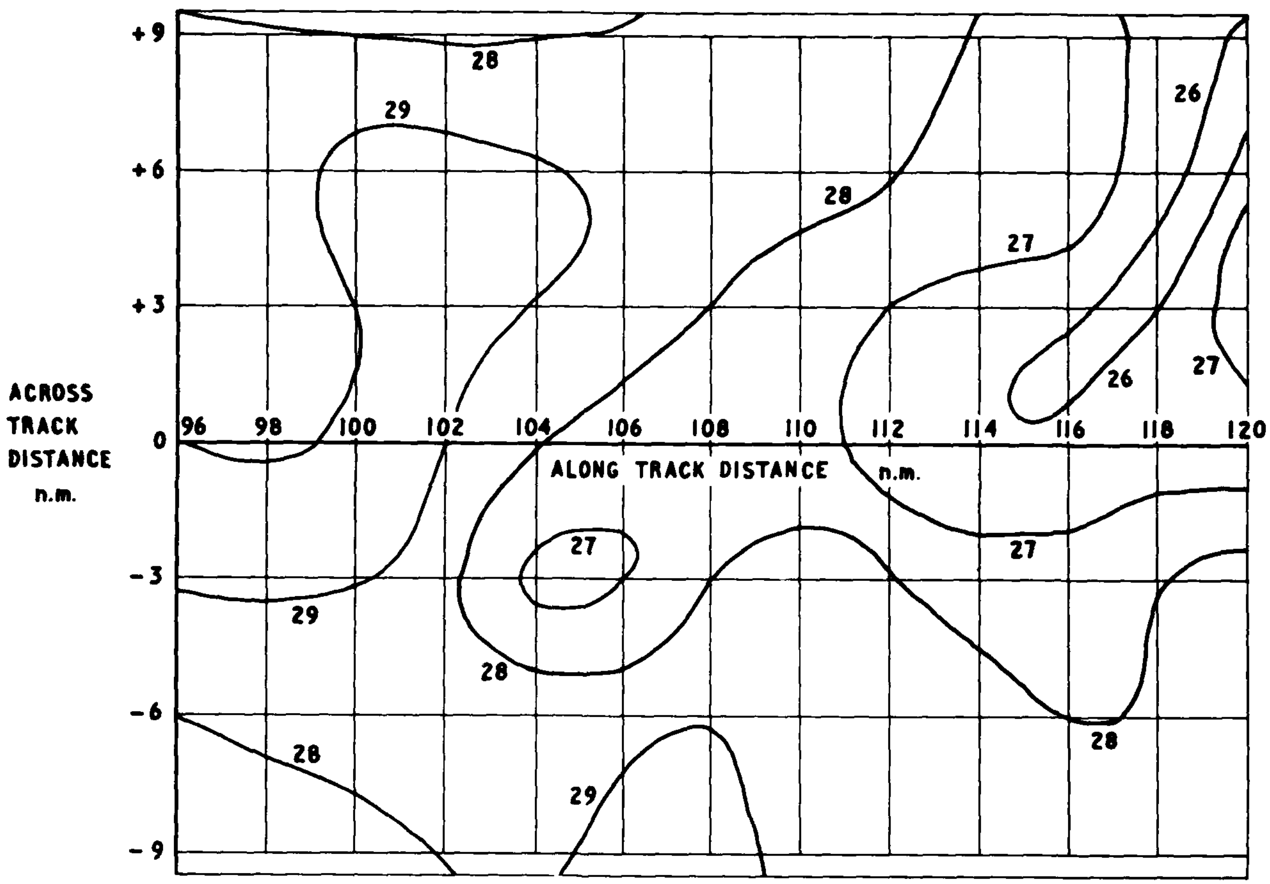
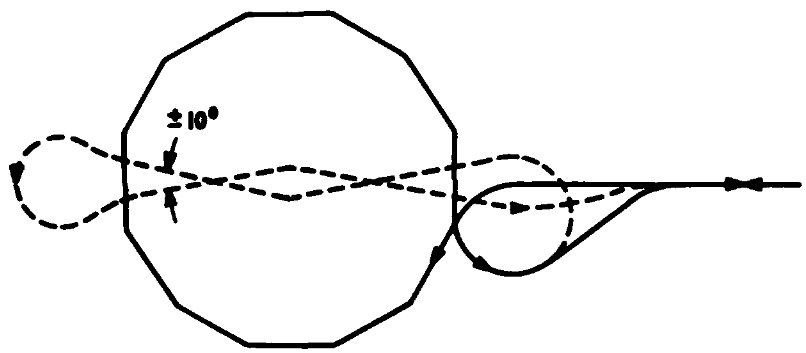
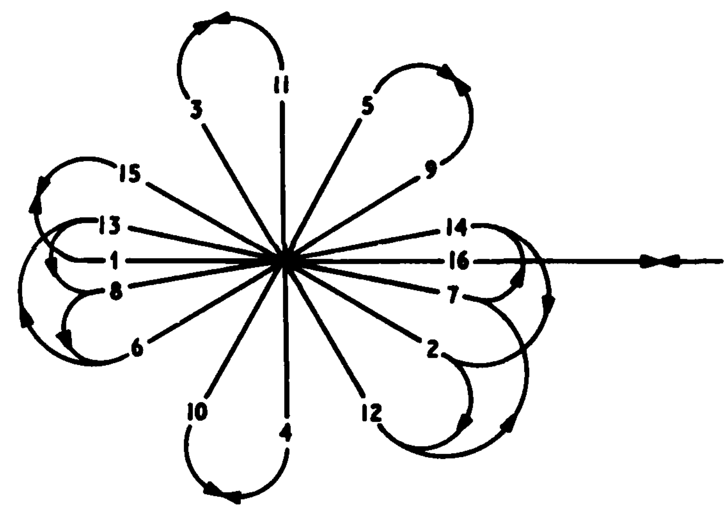


FIG. 6 V.H.F. GROUNDSTATION CONTOURS dB/µV



a POLYGONAL ORBIT FOR CALIBRATED AREA AT HIGH ALTITUDE.



b CLOVER-LEAF FOR OVERFLYING PIN POINT AT LOW ALTITUDE.

FIG.7 POLAR DIAGRAM FLIGHT PATTERNS.

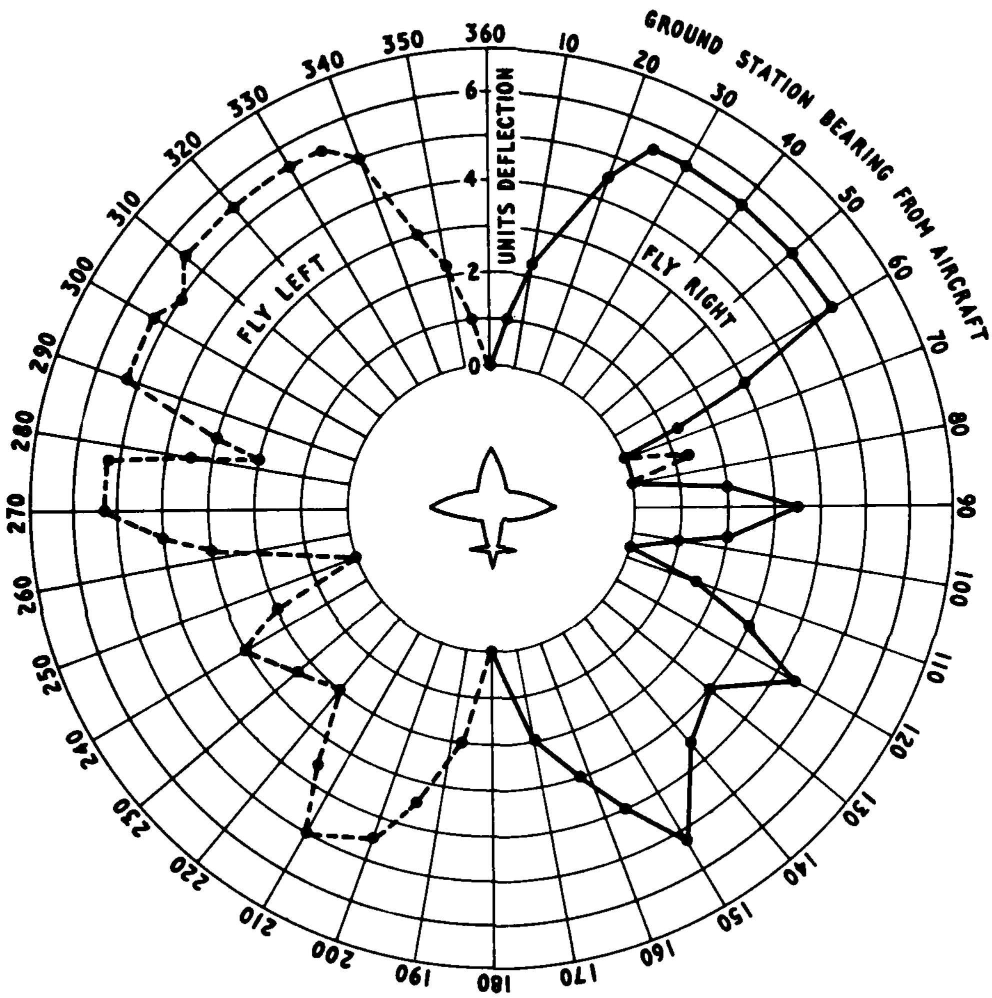


FIG. 8 HOMING INDICATOR AZIMUTH RESPONSE.

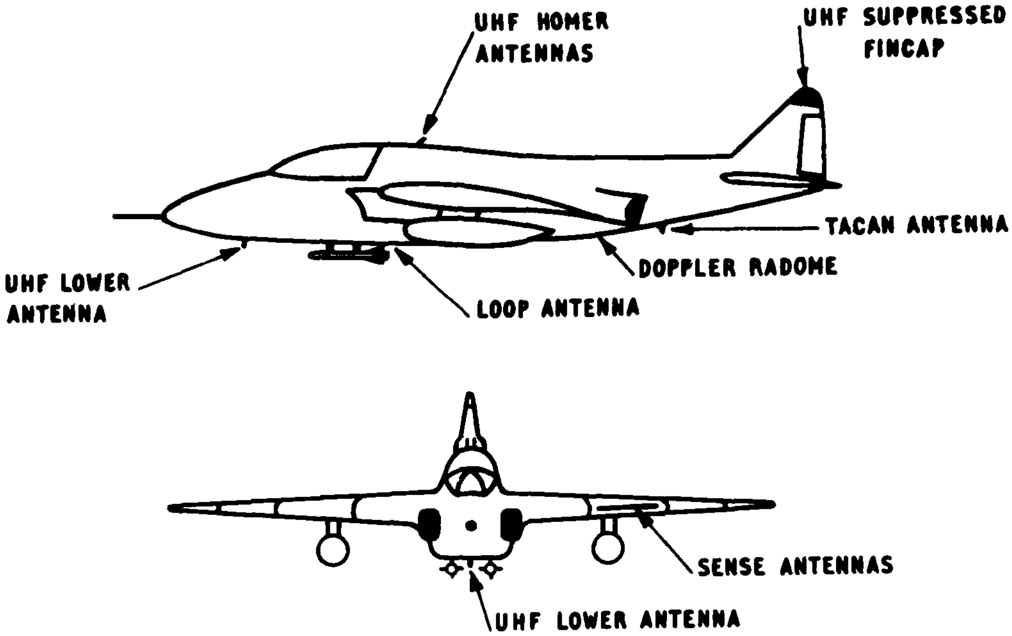


FIG. 9 GREMLIN MK 40 ANTENNA LOCATION.

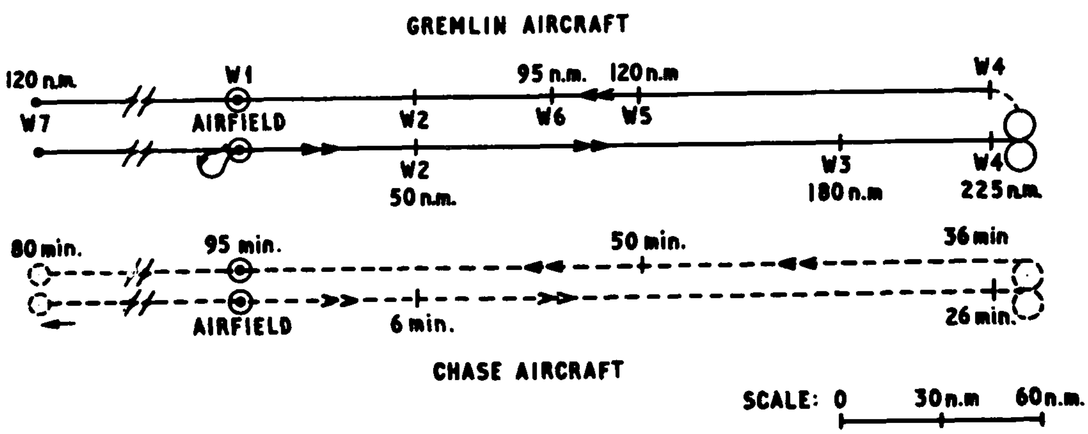


FIG. 10a FLIGHT PROFILE SORTIE 1

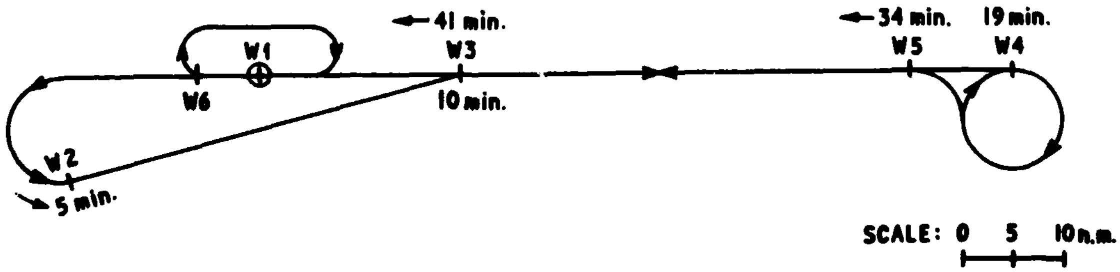


FIG. 10b FLIGHT PROFILE SORTIE 2

DATE 1·1·80
AIRCRAFT GREMLIN MK 40
ANTENNA UHF UNDER NOSE
POLARISATION VERTICAL
ALTITUDE 30,000 FT.
RANGE 110 n.m.
BEARING 213° M.
PATTERN ORBIT / POLYGON 30°

390 MHz ——— 30° PORT BANK
390 MHz - - - - 30° STBD. BANK
245 MHz —·— LEVEL FLIGHT (NORMALISED TO 20 WATTS)

SCALE dB/μV

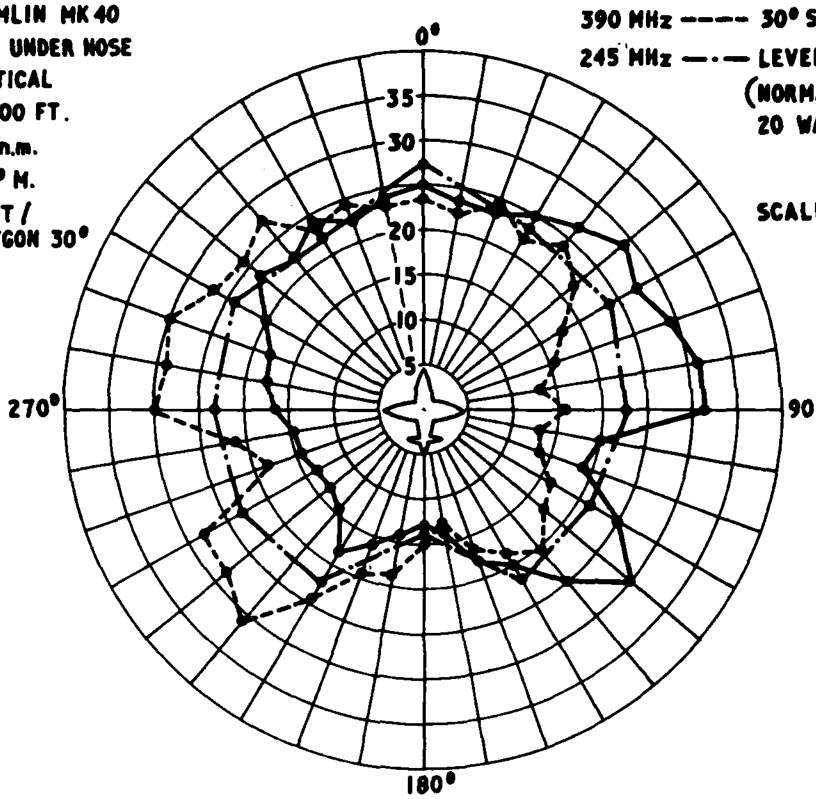


FIG. 11 POLAR DIAGRAMS, SORTIE 1, RUNS 1, 2 & 5.

DATE 1·1·80
AIRCRAFT GREMLIN MK 40
ANTENNA UHF FIN CAP
POLARISATION VERTICAL
ALTITUDE 30,000 FT.
RANGE 110 n.m.
BEARING 213° M.
PATTERN ORBIT

390 MHz ——— 30° PORT BANK
390 MHz - - - - 30° STBD. BANK

SCALE dB/μV

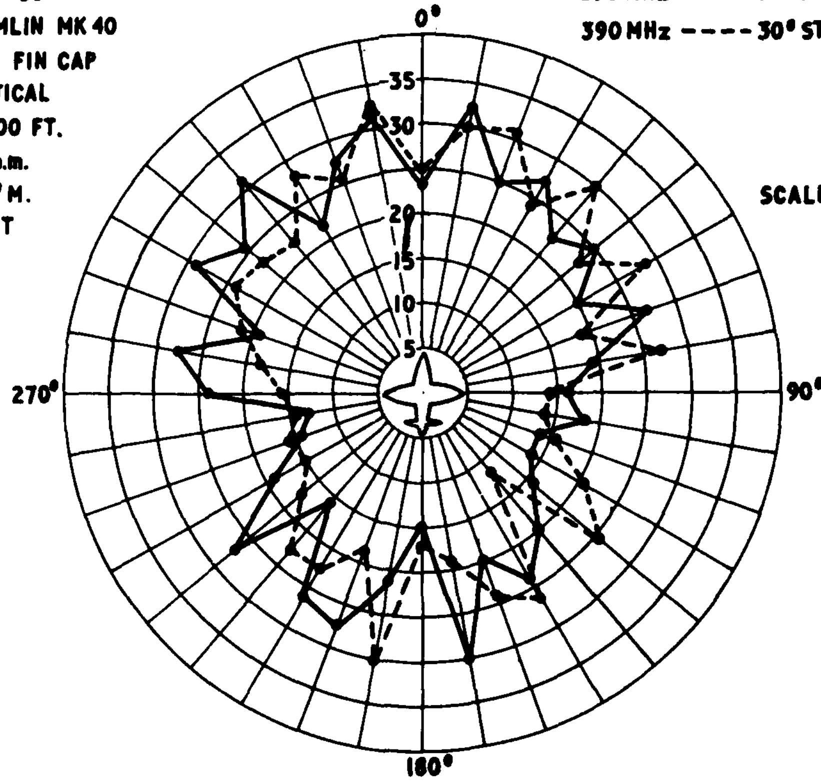


FIG. 12 POLAR DIAGRAMS, SORTIE 1, RUNS 3 & 4.

APPENDIX A

COMMUNICATIONS QUALITY ASSESSMENT

COMMUNICATIONS READABILITY

When carrying out communication trials the message heard in a telephone earpiece has to be described as clearly as possible. If readability trials are carried out in conjunction with other tasks, an operator should be allocated exclusively to this task. When subjectively assessing the quality of communication the following scale may be used:

1. Unreadable.
2. Readable intermittently.
3. Readable with difficulty.
4. Readable.
5. Perfectly readable.

It is evident that readings less than perfect must be so for a reason. A list of possible reasons is given below, to be used in conjunction with the readability scale.

a. With slight/heavy distortion.

b. With interference present.

c. With intermittency.

d. With background noise.

e. Weak.

These may be defined as follows:

Distortion is present when the speech waveform has been so modified as to affect the degree of intelligibility.

Interference is present when spurious signals can be heard in addition to the wanted signal. Its source should be established if possible, either as coming from a local transmitter, vehicle ignition, or adjacent channel traffic or that the spurious signal is being internally generated.

Intermittency is present when the received signal is arriving with breaks in the continuity of the carrier or its modulation.

Background noise is present when noise is superimposed on the transmitted signal or is superimposed on the signal the operator hears.

The audio signal is weak when it is low with respect to the level the operator normally expects.

KEYWORD INTELLIGIBILITY TESTS

These consist of random selections of 5 figure number and letter groups, followed by unfamiliar sentences containing 5 keywords that are essential for a comprehension of what has been transmitted. The listener either writes down what he thinks has been said, re-transmits it or records it on a tape recorder for later analysis.

Intelligibility scoring is performed separately for each group, so as to arrive at an index of successful comprehension for numbers, letters, and sentence keywords out of a maximum of five, in order of increasing difficulty. Sufficient message groups are prepared so as not to use the same message more than once. Examples of these are shown in the Table below:

Message No	Number Group	Letter Group	Keyword Sentences
1	71542	QNAJM	A crowd listened to the radio during the night.
2	66845	VVOCF	He is fond of the sea and has his own dinghy.
3	61402	GABBK	The increase is due to the rise in price.
4	46357	FTRCK	There are many hard problems to solve.
5	32034	NRMID	He has no thought for their comfort.
6	81943	BWHCD	The passage was lit by a dim lamp.
7	89429	UVAHK	The crew stayed aboard the ship all night.
8	30605	XDPLH	They did not make many runs in the first innings.
9	40332	RPVBE	A new sect is being set up in the country.
10	00618	YIMAK	Ten of the crew were being returned home.

TABLE A-1 TYPICAL KEYWORD GROUPS

There appears to be no entirely satisfactory objective method for use in a limited assessment of communication intelligibility. There are variants of the keyword technique (12), and also the "Articulation Index" method (13) which requires audio spectrum analysis facilities; but none of these methods are likely to differ significantly from the subjective assessment of an experienced communications operator unless a statistically significant programme of tests are performed.

APPENDIX B

TYPICAL SCHEDULE OF TESTS

SCHEDULE OF GROUND TESTS

Qualification or Control Checks

a UHF Communications

At the test frequencies
Measure transmitter power at the equipment.
Measure receiver threshold for unsquelch.
Measure transmitter forward and reverse power at each antenna (or determine antenna and feeder parameters with Network analyser).

b Tacan

At the test frequencies
Measure transmitter power at the equipment.
Measure receiver threshold for range and bearing lock, at test set frequency only, if suitable equipment not available.
Measure antenna and feeder parameters as for a.

c Establish initial record level setting for voice recorder with test tone in lieu of microphone.

d Fit homing indicator scale.

Assessment Checks

a Perform 24 point heading swing of MF/DF installation on calibrated base. Observe correct procedures (11). Check aircraft heading reference system, with aircraft sighting rods and landing compass.

b With external aircraft power.
Record test transmissions in the aircraft and at the ground station, with microphone OFF, microphone ON, under acoustically quiet conditions, and with normal speech. At the aircraft check the balance of audio between UHF main, UHF standby, Tacan ident, MF/DF, and other selectable services, check range of volume control available, and level of cross-talk from activated but deselected systems. At the ground station, check modulation depth and *frequency of aircraft UHF transmissions on required flight* test frequencies.

c With engine on full ground power, transmit to ground station at one UHF frequency. Record on cockpit voice recorder. At ground station record transmission and monitor modulation depth.

d Examine aircraft installation for ease of access for maintenance and test.

e Check aircraft voice recordings at b and c and reset recorder level, as required.

SCHEDULE OF FLIGHT TESTS

The following schedules have been prepared for the aircraft under test, designated Gremlin. The legend for the codes used are:

A = Gremlin Aircraft
C = Chase Aircraft
G = Ground Station

H = Homer
L = Lower Antenna
M = Main UHF
U = Upper Antenna

R = Radio Compass
S = Standby UHF
T = Tacan

SORTIE 1

EVENT	TIME MINS	DISTANCE FROM BASE nm	WAY-POINT	FREQ:MHz ANT:	HEIGHT ft TAS	ACTION
START-UP	-0	0	1	390MU 245SL Base Channels T & R	0	(A) Enter Waypoints. Check transmissions on M and S to G and C. Check R and T with base beacons. (G) Check transmissions to A and C. (C) Check transmissions to A and G. Check T with base beacon.
TAKE-OFF	0	0	1	"		(A) Overfly base and initialise DR Nav.
CLIMB	0 to +6	0-50	2	"	30,000 480	(A) Voice recorder ON. Contact G on M. Transmit Mic OFF/ON sequence. Repeat on S, at cruise height. Contact C for rendezvous at W2. (G) Voice recorder ON. Monitor modulation on transmissions from A. (C) Join A at W2. Voice recorder ON.

EVENT	TIME MINS	DISTANCE FROM BASE nm	WAY-POINT	FREQ:MHz ANT:	HEIGHT ft TAS	ACTION
OUT-BOUND RANGE	+6 to +20	50-180	3	390MU 245SL Base Channels T & R	30,000 480	(A) Transmit on M, pass T range and brg, heading, height, speed. Repeat on S. Repeat sequence at 10 nm intervals until readability degrades, then repeat back readability of G at 5 nm intervals. Record loss of T lock vs distance. Record R bearings vs T range and brg on opportunity. Note general R performance. (G) Monitor signal strength and readability on M. Note readability only on S. (C) Fly in loose formation abeam. Monitor A and G on M and compare T position, record errors. Note readability of G vs distance.
MAX RANGE	+20 to +26	180-225	4	"	"	(A) Note distance of last contact on M & S. Abandon S but continue with regular M transmissions pass T range or DR range. Note maximum T range. (G) Make regular short transmissions on M after loss of contact. (C) Note distance for last contact heard on M from G to A. Note maximum T range.
AIR-AIR COMMS & HOMING	+26 to +36	225-250	4	390MU 245SL 390ML 245SU	" or as required (A)+500	(A) Turn to allow C to achieve 5 nm separation. Maintain 2 way contact on S. When C turns change to M, Homer mode. Homing run to close formation. Note homing response to manoeuvre, and visual error on sight. Deselect Homer and check M in close formation, change antenna selection and repeat. Change Cassette. (G) Stand by. (C) At 5 nm separation turn 90° and reduce speed. Change UHF channel as requested by A. Note readability of all A transmissions. Change Cassette.
INBOUND RANGE	+36 to + 50	225-120	5	ML390 245SU	"	(A) Regular short transmissions on M until two way contact achieved, then at 5 nm intervals until good readability; thence 10 nm intervals. Actions as for outbound leg. Near W5 update DR Nav with T fix. (G) Commence regular short transmissions on M. (C) Regain loose formation and continue monitoring as for outbound run.
POLAR DIAGRAM ORBITS	+50 to +80	120-95	5, 6	(1)390ML (2)390ML (3)390MU (4)390MU (5)245SL (6)390ML T ch 60	30,000 480	(A) In orbit area. 360°+ turn 30° port bank. (G) Record signal strength vs heading. (C) Continue on track to Waypoint 7. (A) Repeat (1) starboard bank. (A) Repeat (1). (A) Repeat (2). In 1 thru 4 transmit on M calling headings every 10°. Reset orbit to correct for wind. (A) Fly polygon, wings level every 30° thru 360° calling actual heading when wings level. (G) Request changeover to ML245 if signal strength inadequate. (A) Correct position then repeat (5) with T selected to Ch 60. Transmit on M calling actual heading. (G) Record T signal strength vs heading. (C) Loiter at W7. Listen on 245 MHz.

EVENT	TIME MINS	DISTANCE FROM BASE nm	WAY-POINT	FREQ:MHz ANT:	HEIGHT ft TAS	ACTION
AIR-AIR COMMS & HOMING	+80 to +91	95-50	2	245ML	30,000 480	(A) Overfly W6, select T base channel. Reset DR Nav. Change Cassette. Contact C (thru G if necessary) on M. (G) Monitor 245 MHz. Assist in A to C contact if required. (C) Fly on reciprocal track for base TAS 480 kn. Change Cassette. Change frequency on request from A.
				390ML		(A) Repeat contact and note readability. Select Homer on best comms frequency. Note heading for Homer null. Check response to manoeuvre and record Homer Comms readability. Check Homer on worst frequency at closer range. Spiral descent to W2. (G) Prepare low power transmitter for frequency requested by A. (C) Return to Base.
HOMER POLAR DIAGRAM	+95 to +109	50	2	See Action	5,000 300	(A) On worst Homer frequency. Commence 15° orbit record heading for every unit change of homer indicator deflection left or right thru 360°. (G) Switch on low power transmitter. Listen on other frequency. (A) Repeat orbit procedure on best frequency. (G) Change transmitter frequency.
HOMER SQUINT	+109 to +120	50-0	1	See Action	5,000 420	(A) Overfly W2. Perform homing run against continuous ground station transmission. Record T range and Brg every 2 nm to overhead. Estimate cone of uncertainty and for reversed sense indications on overshoot. Return to Base.

SORTIE 2

EVENT	TIME MINS	DISTANCE FLOWN	WAY-POINT	FREQ:MHz ANT:	HEIGHT ft TAS	ACTION
START-UP	-0	0	1	390ML 245SU Base Channels T & R	0	(A) Enter Waypoints. Check transmissions on M & S to G. (G) Check A transmissions.
TAKE-OFF	0	0	1	"		(A) Overfly on reciprocal track. Initialise DR Nav. (G) Voice recorder ON.
ACCEL & TURN	0 to +5	0-37	2	"	300 420-540	(A) Voice recorder ON. Check transmission to G prior to acceleration using Mic OFF/ON sequence. (G) Check modulation depths. (C) Check transmission to G prior to take off.
HIGH SPEED COMMS INTEL-LIGI-BILITY	+5 to +10	37-77	3	"	300 540	(A) Repeat Mic ON/OFF sequence. Repeat back keyword groups as heard. Continue until W3 reached. (G) Check modulation depth. Transmit keyword groups until terminated by A. (C) Proceed to Waypoint 3 to join up in loose formation abeam of A at W3.

EVENT	TIME MINS	DISTANCE FLOWN	WAY-POINT	FREQ:MHz ANT:	HEIGHT ft TAS	ACTION
OUT-BOUND RANGE	+10 to +19	77-132	4	245SU 390ML	3,000 420	(A) Decelerate and climb. Contact C to formate on track. Transmit on S pass T range and brg heading height and speed. Repeat on M every 5 nm until W4. Note T performance and distance for unlock as for Sortie 1. (G) Record signal strength of S and monitor read-ability of M transmissions as for Sortie 1. (C) Fly in loose formation abeam. Monitor A & G on 245 MHz, compare T position, record errors. Note readability of G vs distance.
RADIO COMPASS SWING	+19 to +34	132-164	4	390ML	5,000 420	(A) Turn and climb to height and perform octagon orbit, initially on reciprocal heading for base. Record R bearing, aircraft heading and T range and bearing at each wings level condition. (G) Standby. (C) Climb to 20,000 ft return and loiter at W3.
INBOUND RANGE	+34 to +41	164-209	3	390MU 245SL	3,000 420	(A) Descend and overfly W5. Fix update DR Nav. Change cassette. Recommence S transmissions until contact with G then continue at 5 nm intervals as for outbound run. (G) Commence short transmissions on 245 MHz until contact established. Continue as for outbound run. Prepare low power transmitters for next stage. (C) Change cassette.
RADIO COMPASS AND HOMER CHECK	+41 to +48	209-229	1	See Action	"	(A) Home on to G transmitter, using H. Record T brg and range, R bearing and heading every 2 nm. Overfly Base and note overhead and overshoot response of H and R. (G) Transmit on worst Homer frequency from Sortie 1. (C) Select 390 MHz.
AIR-AIR COMMS	+48 to +55	229-279	1	390MU	300 420	(A) Descend to 300 ft. Check two way comms. Repeat on ML. If time permits repeat on parallel track 2 nm offset from C (at 300 ft). (G) Monitor air-air comms. (C) Commence race track at 90° to Base track. Descend to 300 ft when requested by A.

PERFORMANCE OF NAVIGATION SYSTEMS

by

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SUMMARY

Aircraft of today are navigated mainly by the radio navigation systems of VOR/DME and TACAN, as well as Doppler navigation and inertial navigation (INS). These systems will be described to show in which way and how far they can be used. Their error behaviour will be described especially and in detail.

By combining various navigation systems, navigation accuracy can be greatly increased. One integrated system will be dealt with and will be explained with the aid of flight tests carried out by the DFVLR in Braunschweig.

1. INTRODUCTION

As far as any navigation system is concerned, it is very important to know how accurate it is. Before describing the most outstanding navigation systems, certain basic facts on the manner of representation of navigation errors will be given here:

A navigation system generally consists of the following components:

- sensors for measuring values of interest
- data conversion and transmission systems
- navigation computer
- indicators.

All these components are inaccurate to a certain degree which means that the navigation information is also inaccurate.

Deterministic and Random Errors

One has to distinguish between deterministic and random errors. The same deterministic error always occurs, when the navigation equipment is turned on. Random errors vary and cannot be anticipated. In most systems, random errors can be described best by a Gaussian distribution:

$$p(x) = \frac{1}{\sqrt{2\pi}\,\sigma} \exp\left(-(x-\bar{x})^2/2\sigma^2\right) .$$

Here

x is the random error

p is the distribution density

$\bar{x}$ is the mean error

σ is the standard deviation.

The Gaussian distribution is best for most errors, because these errors are composed of many individual errors, especially in the case of navigation systems.

Normally, the error behaviour of a single component is of no interest, e. g. the error behaviour of a VOR-receiver showing a certain bias error. When taking into consideration all components, or a certain navigation system, respectively, a systematic error may be disregarded, since positive and negative systematic errors may cancel each other out.

Determination of Navigation Errors

It would be unrealistic to quote the maximum errors as far as position errors are concerned, because maximum errors seldom occur. Therefore, it is better to use certain statistic parameters.

A position error is the difference between the true position and the indicated position:

$$\Delta x = x_t - x_d$$
$$\Delta y = y_t - y_d \; .$$

Fig. 1 shows the position errors which have been measured in connection with a certain navigation system. The error behaviour of this navigation system can be described by two standard deviations σ_x and σ_y, as well as the correlation coefficient k_{xy}:

$$\sigma_x^2 = \frac{\Sigma \Delta x_i^2}{N}$$

$$\sigma_y^2 = \frac{\Sigma \Delta y_i^2}{N}$$

$$k_{xy} = \frac{\Sigma\, x_i\, y_i}{N} \; \frac{1}{\sigma_x \sigma_y} \; .$$

N is the number of measurements.

If the one-dimensional errors Δx and Δy belong to normal distributions, the following statistic facts are valid:

50	percent of the errors are within	$\pm$ 0.675	σ
68.3	percent of the errors are within	$\pm$ 1	σ
95.4	percent of the errors are within	$\pm$ 2	σ
99.7	percent of the errors are within	$\pm$ 3	σ

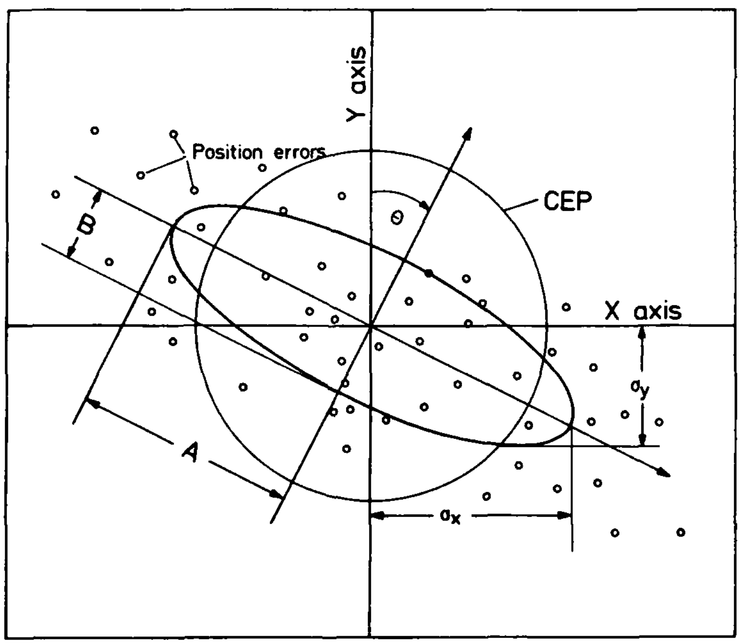

Fig. 1: Position Errors.

Error Ellipsis

The navigation errors shown in Fig. 1 lead to a correlation coefficient of about - 0.73. This negative correlation coefficient means that it is very likely that a negative value Δy corresponds to a positive error Δx, and vice versa. Therefore, position errors can best be shown by means of the so-called error ellipsis. The individual parameters of an error ellipsis can be calculated with the aid of the following equations:

$$\Theta = \frac{1}{2} \arctan \; 2k_{xy} \frac{\sigma_x \sigma_y}{\sigma_y^2 - \sigma_x^2}$$

$$\rho = \sqrt{(\sigma_y^2 - \sigma_x^2)^2 + (2k_{xy}\sigma_x\sigma_y)^2}$$

$$A^2 = \frac{\sigma_x^2 + \sigma_y^2 + \rho}{2}$$

$$B^2 = \frac{\sigma_x^2 + \sigma_y^2 - \rho}{2}$$

If there is a normal distribution, 39.4 % of all values can be found within an error ellipsis. An ellipsis having the ellipsis axes doubled (2σ), comprises 86.5 % of all values.

Circular Error Probability (CEP)

The two-dimensional error behaviour of a navigation system is often described as so-called Circular Error Probability (CEP). The CEP is considered to be the radius of a circle that encloses 50 % of the measurements. This circle is also shown in Fig. 1. An approximation for calculating the CEP would be:

$$\mathrm{CEP} = 0.59\ (\sigma_x + \sigma_y) \pm 3\ \% \qquad \text{if} \qquad \frac{\sigma_y}{3} < \sigma_x < 3\ \sigma_y\ .$$

In the special case where σ_x is σ_y, CEP will be 1.18 σ.

Covariance Matrix

The so-called covariance matrix is frequently used in order to describe the error behaviour of complicated systems, such as inertial navigation systems, for example:

$$Q = \begin{vmatrix} Q_{11} & Q_{21} & \cdots \\ Q_{12} & Q_{22} & \cdots \\ \cdots & \cdots & \cdots \\ \cdots & \cdots & Q_{NN} \end{vmatrix}$$

It normally depends on time and position. The diagonal elements (Q_{11}, Q_{22} ... Q_{NN}) are the squares of the standard deviations. Thus these elements always have to be positive. The off-diagonal elements are the cross-covariances which describe the correlation between the individual errors. The correlation coefficient results from:

$$k_{mn} = \frac{Q_{mn}}{\sqrt{Q_{mm}\ Q_{nn}}}\ .$$

A covariance has to be set up such as to satisfy the condition that k is always:

$$-1 < k < +1\ .$$

The covariance matrix of the two-dimensional case shown in Fig. 1 is

$$\begin{vmatrix} \sigma_x^2 & k_{xy}\sigma_x\sigma_y \\ k_{xy}\sigma_x\sigma_y & \sigma_y^2 \end{vmatrix}$$

Error Propagation

Normally, the covariance matrix is not directly available. It has to be calculated from errors of the various elements of a system. This will be demonstrated here in the case of the VOR/DME-system which will serve as an example. VOR/DME is the civil standard navigation system measuring the azimuth and distance from a ground station to an aircraft. As far as this system is concerned, a certain angle error as well as a distance error have to be taken into account. Both errors are uncorrelated. The covariance matrix of the VOR/DME-system is therefore,

$$P = \begin{vmatrix} \sigma_E^2 & 0 \\ 0 & \sigma_\Theta^2 \end{vmatrix}\ ,$$

σ_E or σ_Θ are the standard deviations of measurement errors of azimuth and of distance, respectively. The conversion of the covariance matrix P into an x-y-coordinate system has to be made by using the following equation:

$$Q = C P C^T .$$

C stands for the transition matrix. For the VOR/DME-system, it reads as follows:

$$T = \begin{vmatrix} \frac{\partial x}{\partial E} & \frac{\partial x}{\partial \Theta} \\ \frac{\partial y}{\partial E} & \frac{\partial y}{\partial \Theta} \end{vmatrix} = \begin{vmatrix} \sin\Theta & E\cos\Theta \\ \cos\Theta & -E\sin\Theta \end{vmatrix}$$

Θ is the azimuth and E is the distance which were measured. The index T incicates a matrix transposition. The various elements of the matrix Q of the VOR/DME-system are calculated as follows:

$$\sigma_x^2 = Q_{11} = \sigma_E^2 \sin^2\Theta + \sigma_\Theta^2 E^2 \cos^2\Theta$$

$$\sigma_y^2 = Q_{22} = \sigma_E^2 \cos^2\Theta + \sigma_\Theta^2 E^2 \sin^2\Theta$$

$$Q_{12} = Q_{21} = (\sigma_E^2 - \sigma_\Theta^2 E^2)\cos\Theta \sin\Theta .$$

Element Q_{21} is normally not zero. The position errors Δx and Δy are thus mostly correlated in this example.

Time Behaviour of Navigation Errors

Fig. 2 shows the distance error of DME-measurements as a function of time. It is clearly visible that this error mainly consists of a relatively high-frequency noise. This error behaviour is typical of most radio navigation systems.

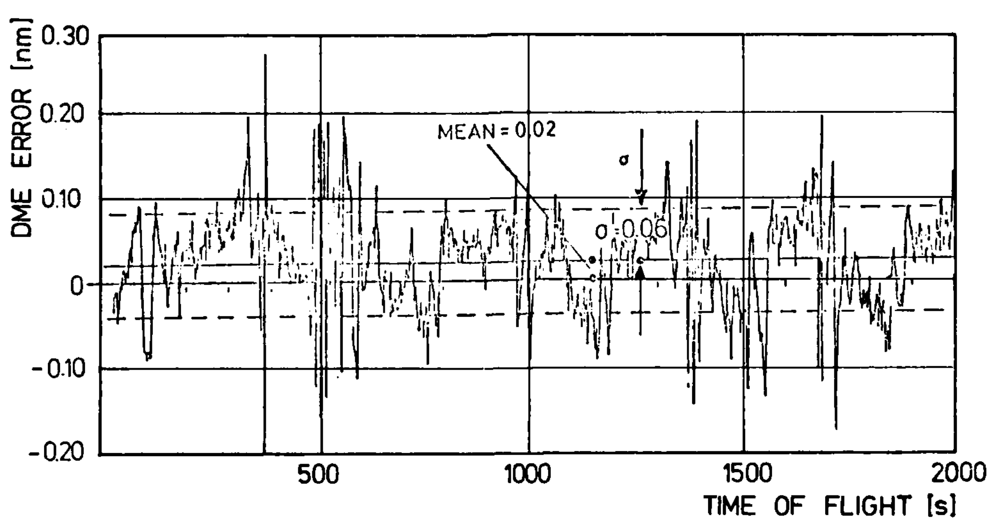

Fig. 2: DME-Errors.

The dead-reckoning systems, such as doppler navigation systems or inertial navigation systems, show a completely different error behaviour. In these systems, the navigation errors constantly increase. This is caused by random or systematic errors of the sensors such as errors of the accelerometers in inertial navigation systems, for example. Yet the shorttime accuracy of these systems is extremely high.

By combining radio navigation systems with inertial navigation systems or doppler navigation systems, it is now possible to determine the sensor errors such as gyrodrifts, for example, and thus to consider them for the navigation calculations. Such a combined system can, therefore, reach an extremely high degree of navigation accuracy. Modern systems use Kalman filters for this purpose. The principle of Kalman filters will not be explained here.

Measurements of Navigation Errors

In order to determine navigation errors during flight tests, it is necessary to measure a reference flight path. Stabilized cameras in the test aircraft, cinetheodolites, as well as tracking-radars are used today in order to reach this aim. With the help of these instruments it is possible to obtain position accuracies which are greater than that of any navigation system used today.

Fig. 3 shows the tracking-radar DIR made by RCA. This radar can measure automatically distance, azimuth, and elevation between the radar and the test aircraft. The accuracies (1σ) which are obtained are as follows:

angle : 0.03°
distance: 8 m.

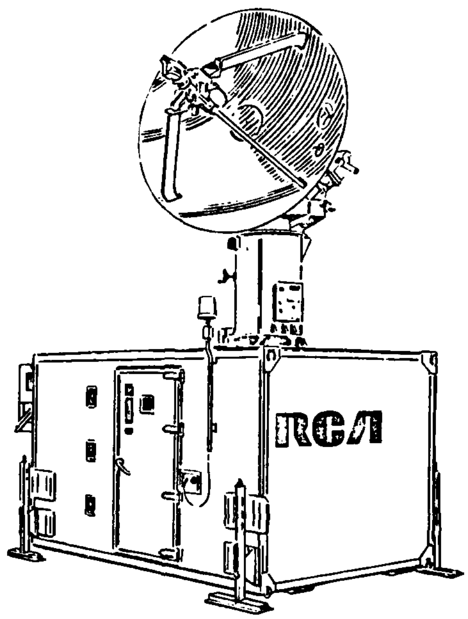

Fig. 3: Digital Instrumentation Radar.

2. VOR/DME AND TACAN

At present, the standard navigation systems in many countries are VOR (VHF omnidirectional range), DME (distance measuring equipment), and TACAN (tactical air navigation system). VOR is the civil bearing measuring system, DME is the civil distance measuring system, and TACAN is a military system measuring bearing and distance. The civil DME and the distance measuring part of TACAN are essentially identical.

VOR-System

The VOR-system operates in the VHF-band from 108 to 118 MHz. The band from 108 to 112 MHz is shared with the localizer portion of the instrument landing system. The space between two adjacent VOR-channels or localizer channels is 50 kHz which means that there are 200 channels altogether. The method of operation of the VOR-system as well as the angle measurement portion of the TACAN-system is similar in principle. Therefore, this method will be explained briefly by means of an optical system:

This system consists of a rotating beacon and a flash light. Both lights are installed on a tower. The rotating beacon is always rotating continously in azimuth. An observer who is onboard a ship, for instance, always sees the rotating beacon shining brightly, whenever it faces the ship. The period of time for one rotation of the beacon is 360 seconds. Whenever the beacon points to North, the flash light is lit. The observer is then able to determine the bearing of his ship from Northern direction relative to the tower with the aid of a simple time measurement. All he needs to do is to take the time elapsing between the flash of the lamp and the rotating beacon pointing towards him which he can do with an ordinary stop-watch. The number of seconds thus stopped corresponds to the bearing of his ship, measured in degrees.

As far as the VOR-system is concerned, the rotating beacon is replaced by a rotating antenna. It rotates at a rate of 30 rotations per second. This produces a 30 Hz-tone in the VOR-receiver. Its phase relationship depends on the azimuth between North and the air-

craft in relation to the ground station. The flash light is replaced by an omni-directional reference signal which is also a 30 Hz-tone and is radiated by an auxiliary subcarrier. The phase between the two 30 Hz-tones corresponds to the bearing of the aircraft from magnetic North relative to the ground station.

Airborne VOR-Equipment

Since the VOR-signal is horizontally polarized, an appropriate antenna has to be fitted onboard the aircraft for this purpose. The function of the VOR-receiver is mainly to receive the VOR-signal as well as to produce the two 30 Hz-tones. Two procedures are used in order to show the 30 Hz-phase relationship: when using the first procedure, the phase comparison is made manually by the so-called omni-bearing selector. With the aid of this selector, it is possible to select a certain bearing or 'radial'; the deviation indicator which is part of this, shows a 'zero' deviation, when the aircraft is on the selected radial. A full scale deflection is approximately 10^{o} off the selected bearing. The so-called 'TO/FROM'-flag is part of the deviation indicator. This flag shows whether the phase between the two VOR-signals is near 0^{o} (FROM) or 180^{o} (TO). This arrangement enables the pilot to fly towards a VOR or away from it, along certain radials. As far as the second procedure is concerned, the bearing is automatically indicated, normally on the radio-magnetic indicator.

VOR-Coverage

There are three catagories of VOR-facilities:

Class		
H (high altitude)	18000 - 45000 ft	130 nm
L (low altitude)	up to 18000 ft	40 nm
T (terminal)	up to 12000 ft	25 nm

The maximum range is limited by signal power limitations or frequency protection. In the USA and in the west european countries, there is VOR-coverage everywhere. Since the VOR-signals are basically line-of-sight-limited, there is sometimes no coverage between individual stations in extremely low altitudes. Therefore, problems arise especially in mountainous areas.

VOR-Accuracy

The VOR-system error is usually divided into

ground station errors,
airborne equipment errors,
flight-technical errors.

Errors of ground stations are mainly caused by multi-path effects and reflections which cause bearing errors.

Each reflected VOR-signal being received by the VOR-onboard-receiver, produces a bearing error, because the phase of the two 30 Hz-tones is changed. Most reflections can be found in the vicinity of VOR-sites. This means that great care has to be taken when selecting these sites.

Errors of modern VOR-receivers are generally smaller than those of ground stations. Possible origins or causes of errors related to the receiver will not be dealt with here.

Fig. 4 shows the typical error behaviour of the VOR-system. It represents the sum of ground errors and onboard errors as a function of time. A conventional VOR-station in Northern Germany was used for this purpose. The measured data were collected by a test aircraft of the type HFB 320. A tracking-radar and an inertial navigation system were used as reference. This Figure also shows a relatively high-frequency noise which is typical of most radio-navigation systems. The standard deviation here is about 2.4^{o} (1σ); the mean is very low.

Use of the VOR-System

The modern Air Traffic Control System (ATC) above continents is based on rigid airways which are determined with the aid of ground stations. One airway segment is determined by the line between two ground stations. Today these ground stations are almost exclusively VOR-stations. During the flight, the pilots have to try to fly as precisely as possible along the centrelines of the airways. How this is done in practice is shown in Fig. 5. When the aircraft has passed the VOR-ground station, the TO/FROM-flag changes to 'FROM'. Then the pilot has to select 70^{o} on the omni-bearing selector, according to the new radial of 70^{o}. If the heading of 50^{o} is maintained, the aircraft will gradually reach the radial of 70^{o}.

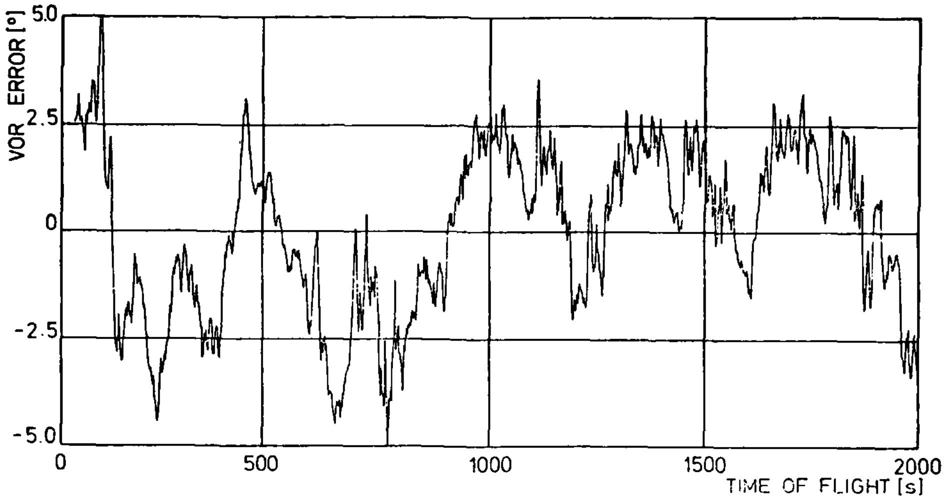

Fig. 4: VOR-Errors.

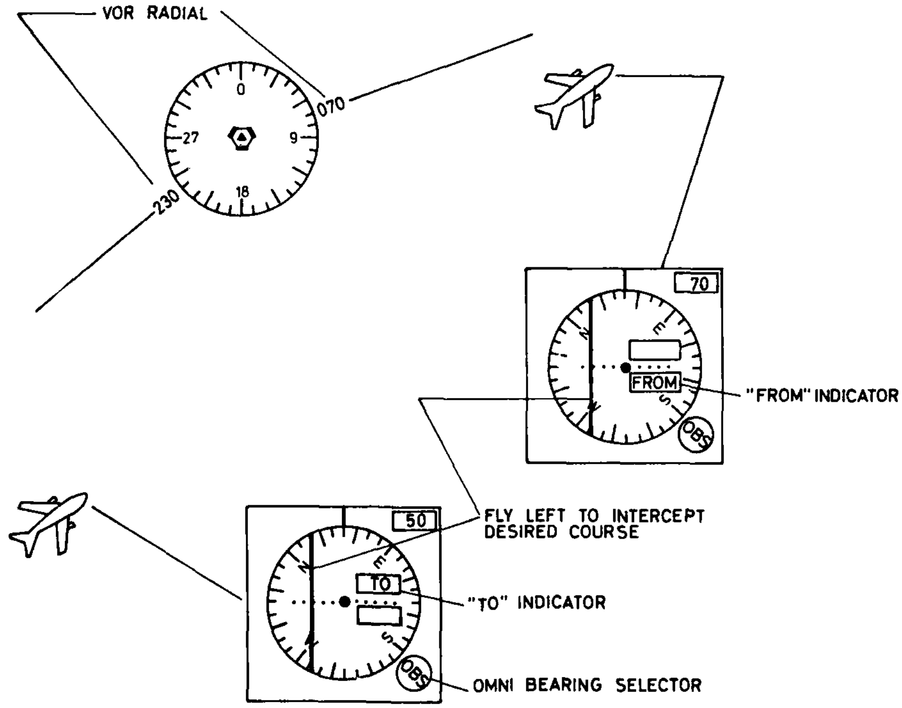

Fig. 5: Use of the VOR-System.

shows how accurately aircraft can follow the centre-line of airways. It shows tracks out of a total of 1000 tracks. The standard deviation of the aircraft from the centre-line is approximately 1.6 nm.

now, the VOR-system has been improved by the so-called Doppler-VOR. will not be explained here.

aperture of the Doppler-VOR is about 13 meters. Therefore, it is greater than that of a conventional VOR. Since the site error is the antenna aperture, the error of a Doppler-VOR should decrease used at sites where conventional VORs are too inaccurate.

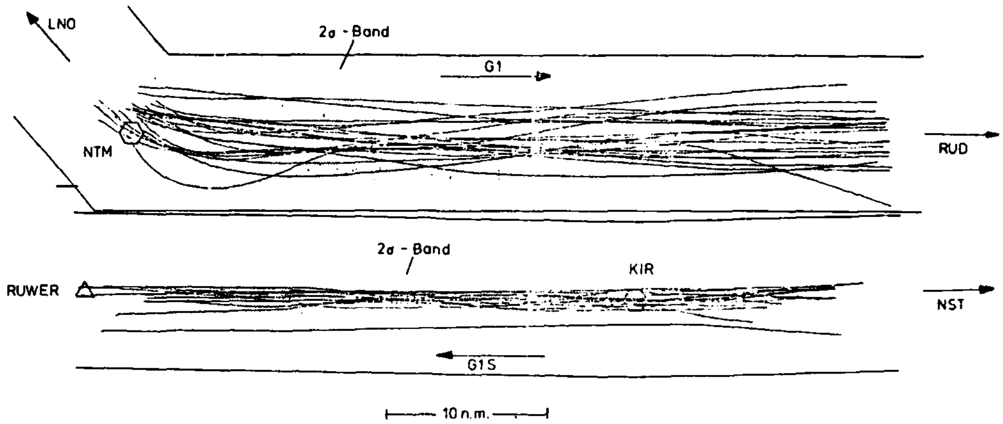

Fig. 6: Aircraft Trajectories.

Distance Measuring Equipment (DME)

Since 1959 DME, together with VOR, have been the standard short-range navigation system. It is an active system measuring the distance between an aircraft and a certain ground station. In order to measure such a distance, the round trip propagation time, from the aircraft to the ground station, and back again, is used. DME uses the same signals as that part of TACAN which measures the distance. For this reason, VOR and TACAN stations are normally combined in a so-called VORTAC. The DME-frequency band covers 960 MHz to 1215 MHz. This band is divided into 126 channels. One hundred of these are provided for civil DME, i. e. mainly for VORTAC-stations, whereas 26 are reserved for military TACAN-stations only. As there are altogether 200 VOR channels and localizer channels, the number of channels for DME and TACAN is doubled by the so-called x-channels and y-channels.

Twin pulses are used in order to measure the round trip propagation time. Each twin pulse received by the ground station causes the ground station itself to retransmit a twin pulse to the interrogator after a fixed time delay (50 μs) on a frequency shifted by 63 MHz upwards or downwards. The intervals between pulses of the twin pulse are

aircraft - ground	x-channel	12 μs
	y-channel	36 μs
ground - aircraft	x-channel	12 μs
	y-channel	30 μs

Ground Transponder Unit

For several reasons, the DME ground stations also radiate twin pulses, even if they receive no interrogation pulses. The radiation of these 'squitter pulses' happens on a frequency which is varying in a random manner. The average pulse repetition frequency is 3000 pulse pairs per second. This frequency is kept also, if interrogation pulses are received. The squitter pulses are reduced accordingly under such circumstances.

Since the upper limit of the pulse repetition frequency is approximately 3000 pulse pairs per second, a ground station can only be used by a restricted number of aircraft; normally about 100. If too many aircraft want to use the same ground station, this ground station can become saturated. In such a case, only those aircraft having a stronger signal will obtain the necessary information from the ground station. Consequently, those aircraft being closer to the ground station or having a more powerful interrogator transmitter, are favoured.

As far as the VORTAC-ground stations are concerned, each VOR-frequency is paired with a certain TACAN-channel. DME-ground stations can also be used together with the Instrument Landing System (ILS). Therefore, each localizer channel is also paired with a certain DME-channel.

DME-Airborne Equipment

Before a DME-onboard unit can start measuring the distance continously, i. e. before it is in the so-called 'track-mode', the distance itself has to be determined by an acquisition process. Older types of onboard units used a moving gate for this purpose. Those units needed acquisition times of 30 to 60 seconds. After changing a channel, 30 to 60 seconds ellapsed until the receiver is geared to track-mode. Modern units use digital techniques for the acquisition process and need less than one second for this.

During the acquisition process, the pulse pair repetition rate is approximately 75 to 150 pulse pairs per second. This rate is reduced to 15 to 30 pulse pairs per second in the track-mode. In the case of localizer channels, a pulse pair repetition rate of 75 to 100 pulse pairs per second is maintained in the track-mode. This is done in order to obtain greater accuracies. Owing to the fact that VOR-frequencies or localizers are allotted to the DME-frequencies, the right DME-channel is automatically selected by the VOR-receiver.

DME-Coverage

Here, the same applies as in the case of VOR.

DME-Accuracy

The ICAO limit for DME-accuracy is either 0.5 nm or 3 % of the DME-range. Today most of the DME-systems are much more accurate than that ICAO-standard. Modern equipment has errors of less than 0.1 nm. Fig. 2 which has already been mentioned earlier, shows the error behaviour of a modern DME-equipment. It has the typical error noise again. The standard deviation is 0.06 nm (1σ) here; the mean is 0.02 nm. The plot in this Figure is obtained from the same test flight as that mentioned in connection with the VOR-error in Fig. 4.

TACAN-System

The principle of distance measuring has already been described. The basic principle of azimuth measuring is the same as in the VOR-system. In the TACAN-system, an antenna is rotating at a rate of 15 rotations per second. This causes a 15 Hz-amplitude modulation of the twin pulses which are radiated. When the antenna points to North, a certain group of pulses is radiated as a North reference signal. In addition, an amplitude modulation of 135 Hz is produced by means of a certain construction of the rotating antenna. Further groups of pulses are used by the ground station to indicate a North reference for a so generated multilobe signal of 135 Hz. The 135 Hz-signal is used as a fine measurement system of the bearing.

TACAN Bearing Receiver

In the bearing receiver, the reference pulse groups of 15 and 135 Hz, as well as the 15 Hz and 135 Hz-tones have to be decoded first of all. The reference pulse groups are used in order to determine the phase relation of the two tones. The phase of the 15 Hz-signal is equal to the bearing in degrees. The 135 Hz-tone helps to improve the bearing accuracy. A phase angle of 360^o corresponds to a bearing angle of 40^o. As far as the receiver is concerned, the accuracy is thus theoretically improved by a factor of 9, if this fine system of 135 Hz is applied.

TACAN Bearing Accuracy

As in the case of the VOR-system, the bearing accuracy of the TACAN-system largely depends on site conditions. If site conditions are favourable, and if modern receivers are used, accuracies of up to 0.2^o or 0.5^o can be achieved. If the conditions are less favourable, the accuracies may decrease to 2^o or 3^o. In case of both poor signal and poor site conditions, care must be taken to ensure that large bearing errors in the 15 Hz-tone do not cause a false acquisition of the 135 Hz fine signal. This causes a bearing error of $\pm 40^o$.

3. DOPPLER NAVIGATION

A doppler navigation system is a dead-reckoning system. A doppler radar measures primarily the ground velocity of an aircraft. This velocity is measured in a coordinate system which is fixed to the aircraft. The aircraft velocity related to a navigation coordinate system has to be determined by means of heading and attitude angle of the aircraft. A navigation computer is used to determine the position of an aircraft with the aid of the measured velocities.

The advantages of doppler navigation are as follows:

1. It is a self-contained system onboard the aircraft. No ground stations are required.
2. Velocity measuring is very accurate.
3. Doppler navigation can be applied everywhere, i. e. even in underdeveloped areas, where there are only very few radio navigation aids.

4. Doppler navigation is possible under almost all weather conditions; only extremely heavy precipitation may cause disturbance.

5. As no ground stations are used, no international agreements are necessary.

6. A doppler navigation system needs no preflight alignment procedure.

The disadvantages of doppler navigation are:

1. The position error increases according to the distance which is covered.

2. The short-time accuracy of the velocity measurement is not as great as the long-time accuracy.

3. A heading and attitude reference system is required in order to convert the velocities into a ground fixed coordinate system.

A doppler navigation system comprises the following components:

doppler radar
navigation computer
heading and attitude reference system
display unit
steering indicator unit.

Fig. 7 shows a typical physical configuration of a doppler navigation system. Many navigation computers are able to carry out the necessary calculation for area navigation. The control display unit (CDU) can be used for the following tasks:

input of take-off position of an aircraft into the navigation computer,

input of way-points into the navigation computer,

presentation of position of an aircraft,

presentation of the distance of an aircraft to the next way-point, and similar tasks.

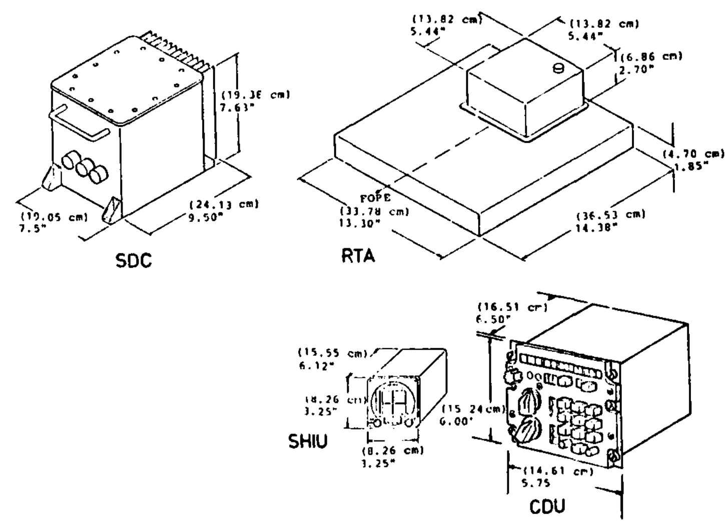

Fig. 7: Typical Doppler Radar Configuration.

CDU : Computer Display Unit
RTA : Receiver Transmitter Antenna
SDC : Signal Data Converter
SHIU: Steering Hover Indicator Unit

Similar to the deviation indicator in the VOR-system the steering control unit can be used as indicator of the deviation of an aircraft from a certain track.

Principle of Doppler Radar

The principle of a doppler radar will be described briefly by explaining Fig. 8. The transmitter of the doppler radar sends micro wave energy to the surface of the earth. This energy is reflected from the ground. Part of the reflected energy is received by the doppler radar and evaluated by it. Owing to the fact that the aircraft is moving, the frequency of the micro wave energy which is received, is shifted due to the doppler effect. The velocity of aircraft is calculated from this shift.

In order to determine the direction of velocity, it is necessary to radiate at least three different beams to the surface of the earth. Most doppler radars direct four beams; 2 forwards and 2 backwards.

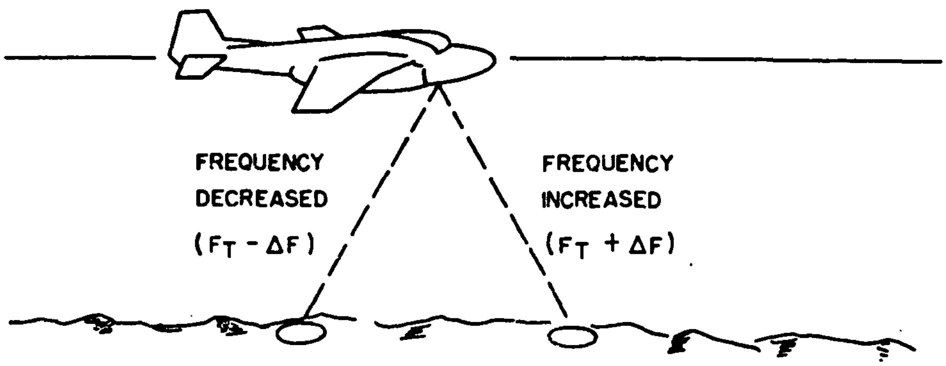

Fig. 8: Principle of a Doppler Radar.

Doppler Noise

Since each doppler radar antenna beam has a finite beam width, the received signal therefore, represents a frequency spectrum, and not a single frequency. This frequency spectrum causes a noise which is typical of all doppler radars. The velocity error mainly consists of this noise. The standard deviation is approximately

$$\sigma \quad \sqrt{0.05\ v(m/s)} \quad [m/s]$$

for all usual types of doppler radars, and thus it depends on the velocity. Fig 9. shows typical doppler velocities as a function of time. The along and across velocity of a jet aircraft during take-off is plotted. The smoother curves represent the velocities by an

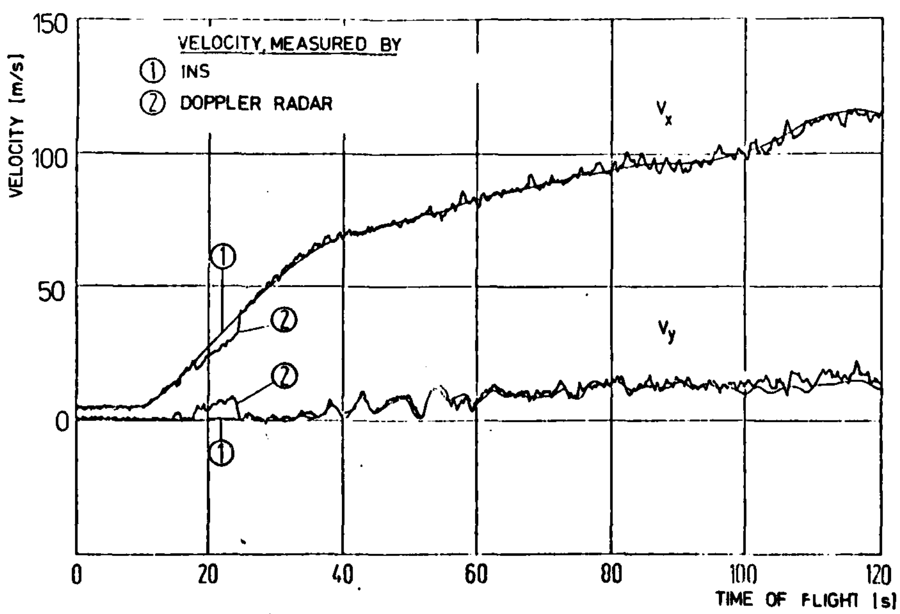

Fig. 9: Velocities measured by a Doppler Radar ② and INS ①.

inertial navigation system. At first, the doppler noise is small due to the low velocity, but it increases gradually while the aircraft is accelerating, and finally the doppler noise reaches about 2 m/s (1σ). The doppler noise also leads to a limitation of navigation accuracy. As far as typical doppler sensors are concerned, it is approximately

$$\Delta s \approx \sqrt{0.05\ d(m)} \quad [m]$$

This equation shows a basic behaviour of doppler navigation systems: The position error depends on the travelled distance (d). For this reason, the error given by the manufacturer, is always related to a certain distance, 10 nautical miles, for example.

Land Sea-Effect

The back-scattering behaviour of normal land differs from that of sea. As a consequence, there is a different doppler shift over sea. Many doppler sensors have a special so-called 'land-sea' switch which takes care of this. Some doppler sensors are able to eliminate automatically the land-sea effect. Above water, micro-waves can be reflected only, if there are irregularities as sea-waves. A completely calm sea is like a mirror, so that no reflected signal will be returned to the doppler radar. This may happen overhead lakes, when there is absolutely no wind.

Further Errors of Doppler Sensors

The following table comprises further errors of doppler sensors:

error source	type
beam direction	scale factor
transmission frequency	scale factor
frequency tracker	bias
installation	bias, scale factor
calibration	scale factor
read-out	bias, scale factor

The total error of a typical doppler radar is approximately

$$0.3\ \% + 0.25\ \text{knot} \quad (1\sigma)\ .$$

This error is related to a distance of about 10 nm.

Accuracy of a Doppler Navigation

The accuracy of a doppler navigation system largely depends on the performance of the attitude and heading reference system. The heading error is of special significance. An example will show the navigation accuracy of three types of attitude and heading reference systems. These systems are:

1. a simple gyro-compass having an alignment error of about 2° in azimuth,
2. a high-performance attitude and heading reference system having a random drift of $0.1^{\circ}/h$ and an initial alignment error of 10'.
3. an inertial navigation system having a random gyro-drift of $0.01^{\circ}/h$, and an azimuth alignment error of approximately 3'.

The following table shows the navigation errors of those three above mentioned navigation systems; i. e. for a flight in western direction, as well as for various distances: 10, 50, 100, and 200 km. This table shows that the position error caused by the doppler noise is relatively small in each case. With regard to a reference of the quality of an inertial system, the total error is essentially caused by the scale factor and bias error (0.3 % + 0.25 knot). In the case of system 2, the error in East-West direction is caused by the scale factor error, whereas the error in North-South direction is mainly due to a bad initial alignment of the heading reference. As far as a normal gyro-compass is concerned (system 1), the error in East-West direction is mainly caused by the scale factor error of the doppler sensor, whereas the position error in North-South direction is mainly caused by the heading reference.

System	Flight Distance	Errors caused by: Doppler Noise and Gyro-Drift		Scale Factor and Bias 0,3 % + 0,25 kn		Alignment Error in Azimuth		Total Error	
	km	East-West	North-South	East-West	North-South	East-West	North-South	East-West	North-South
1	10	17 m	56 m	30 m	19 m	0 m	349 m	34 m	354 m
	50	38	605	150	93	0	1745	154	1849
	100	53	1715	300	187	0	3490	305	3893
	200	76	4855	600	374	0	6981	605	8511
2	10	17	17	30	19	0	29	34	39
	50	38	44	150	93	0	145	154	188
	100	53	84	300	187	0	290	305	355
	200	76	198	600	374	0	581	605	719
3	10	17	17	30	19	0	9	34	27
	50	38	38	150	93	0	44	154	110
	100	53	54	300	187	0	87	305	213
	200	76	78	600	374	0	174	605	420

Table 1: Errors of Doppler Navigation Systems.

Limits of Application of Doppler Radars

Doppler radars can be used up to a certain altitude above ground, depending on the purpose they are to serve. The maximum altitude depends on the performance of the transmitter as well as on the principle of operation. The maximum usable altitude is about 40 000 ft.

There are some types of doppler sensors (e. g. pulse doppler radars) which cannot measure velocities below a certain altitude, i. e. 40 ft above ground. Nor can many types of doppler radars measure velocities below a certain velocity, for instance below 80 knots.

Lately, doppler radars for helicopters have been developed, directly operating on the ground and having a velocity range of negative as well as positive velocities. Velocities obtained with such a doppler radar are plotted in Fig. 9.

4. INERTIAL NAVIGATION

Similar to doppler navigation inertial navigation is a dead-reckoning system. Position and velocity of an aircraft are determined by measuring their acceleration and processing the acceleration data in a computer. An inertial navigation system has the following advantages:

1. The velocity is determined instantaneously and continuously.
2. An inertial navigation system is completely self-contained, since accelerations are measured inside the aircraft. Therefore, it is non-radiating and cannot be jammed.
3. Navigation information is nearly independent of the manoeuvres of aircraft.
4. Inertial navigation currently determines azimuth-, pitch- and roll-angle. It is the most accurate attitude reference system.
5. Inertial navigation is possible everywhere and under all weather conditions. No ground stations are needed.

Inertial navigation has the following disadvantages:

1. The position error increases gradually, even if the aircraft does not move.
2. The equipment is very expensive (approximately $ 110,000 in 1980).
3. An initial alignment is necessary.

A typical inertial navigation system comprises the following units:

navigation unit,
control display unit,
mode selector.

The navigation unit contains the major system elements and the computer. The control display unit has the same function as explained with the doppler navigation system. The mode selector is required in order to turn on the system. Moreover, it is used for selecting the basic operating modes, such as stand-by, align, navigate.

Error Behaviour

No detailed description of the way, in which an inertial navigation system operates, will be given here. Only the error behaviour will be explained briefly. The errors are caused by errors of the gyros and of the accelerometers. As far as the gyros are concerned, there are constant drift and irregular drift errors, whereas bias errors are the important ones of accelerometers. Fig. 10 shows, what effect such errors will have in a platform system. The left hand side shows the effect, which an accelerometer bias has on vecocity and position. It can be seen in Fig. 10 that the velocity error is a sine oscillating with the so-called 'Schuler-Period'. The position error is a cosine and its magnitude is limited. The effect of gyro-drift can be seen on the right hand side of Fig. 10. Here the velocity error is a shifted cosine also with the 'Schuler-Period'. Owing to this shift, the position error increases nearly in a linear manner. There is an important rule of thumb for inertial navigation:

navigation error in kilometers per hour
corresponds to a one hundredth degree per
hour gyro-drift.

Inertial navigation systems being used today in civil aviation operate with an accuracy of about 1 nm/h. A MTBF of approximately 3.000 hours is achieved.

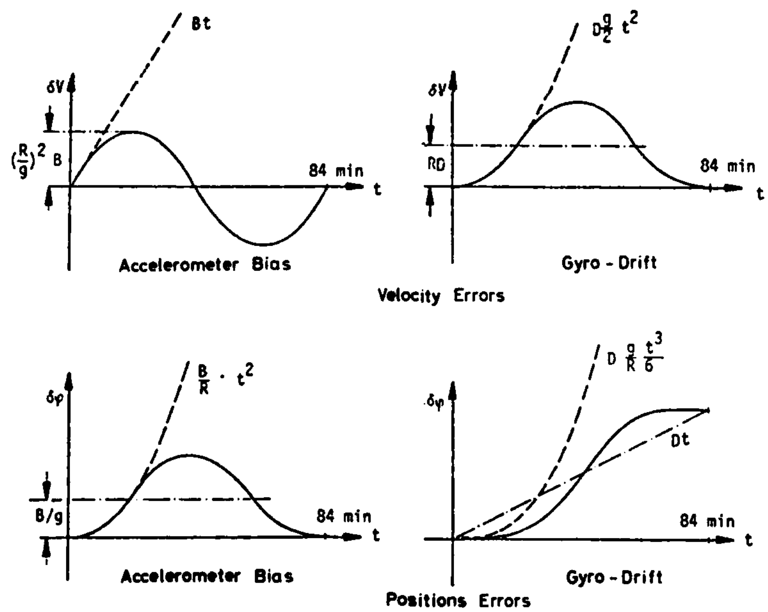

Fig. 10: Error Behaviour of INS.

5. INTEGRATED NAVIGATION SYSTEM

It has already been mentioned earlier that it is possible to improve navigation accuracy substantially by combining various navigation systems of different error behaviour. The combination of radio-navigation systems with dead-reckoning systems is particularly effective. A simple integrated system will be described here as an example. This system consists of a simple dead-reckoning system aided by DME distance measurements. Both systems are integrated by means of a Kalman filter. The dead-reckoning system uses heading, true air speed (TAS), and an estimated velocity of the wind. The DME-interrogator is constantly switched over to frequencies of different ground stations (multiple DME).

The position errors of a dead-reckoning system using true air speed, heading, and wind, are caused by the following sources:

1. inaccurate determination of true air speed,
2. inaccurate heading and side slip angle,
3. unknown wind conditions.

Taking into account the above errors, and in addition systematic errors of the DME-ground stations, the Kalman filter has to estimate the following elements:

1. position errors in eastern direction or northern direction, respectively,
2. scale factor error when determining true air speed,
3. heading error,
4. wind velocity in eastern direction or northern direction, respectively,
5. vector comprising the systematic errors of the DME-ground stations.

The cycle time of the Kalman filter was set to 2 seconds. The DME-station was also changed every 2 seconds.

The mechanization of the Kalman filter is of the closed loop type: The results are immediately used by the dead-reckoning system. The advantage of this mechanization is that no great position error will occur due to a strong wind, for example.

During intervals when there are no DME measurements, the navigation procedure continues using the wind and all the other elements which were estimated by the Kalman filter.

Flight Test Results

Some test flights were made by a HFB 320 (Hansa jet) of the DFVLR in Braunschweig in order to check this simple multiple DME-system. Fig. 11 shows two of these test flights. They were made across North Germany. Altogether 10 different ground stations were used during these flights. Figure 12 shows the degree of accuracy which can be achieved. The cross track error as well as the along-track error are plotted in this figure. When the aircraft had reached an altitude of 4500 ft, after 10 minutes flight, five different DME stations could be used. Afterwards the navigation error was always smaller than 100 metres.

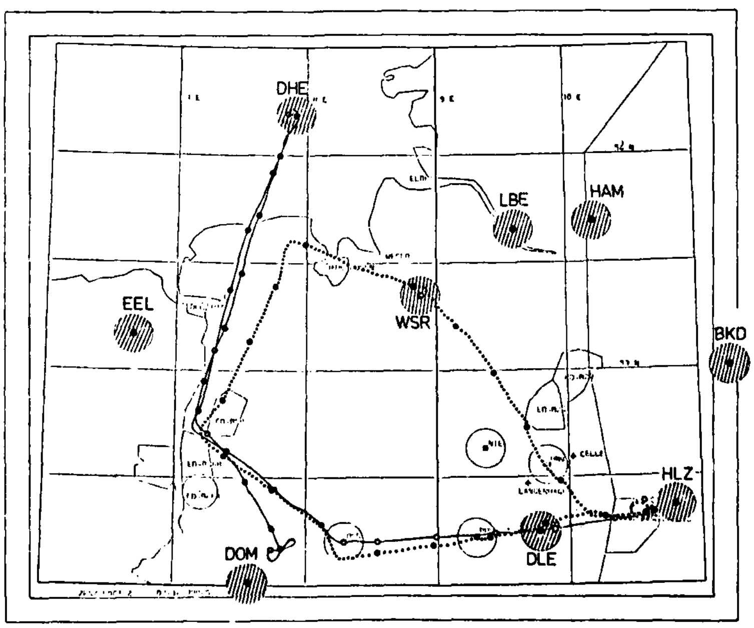

Fig. 11: Map with Test Flight Paths and DME Stations.

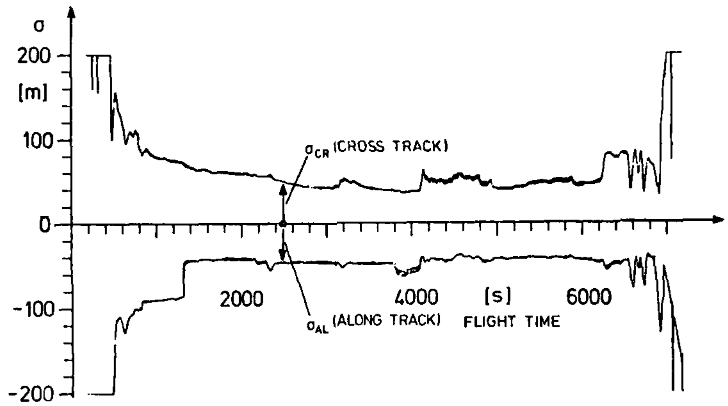

Fig. 12: Cross and Along Track Error.

The advantages of this integrated system are:

1. The accuracy of the VOR/DME navigation is improved.
2. The navigation is not interrupted in case of loss of DME signals.
3. The wind is continuously estimated and thus available onboard the aircraft.
4. The system is suitable for precise area navigation.

The disadvantages are:

1. A digital computer and the necessary interfaces are required in addition to the standard equipment.
2. The site coordinates of the VOR/DME ground stations have to be entered into the computer. This means more work load for the pilot, if there is no device for automatic entering of these data.

With regard to the necessary digital computer, the following has to be stated: during the last few years, the computer development has made great progress, i. e. the computers have become smaller, less expensive and more efficient. There is a possibility that all aircraft will be equipped with a computer within the next few years, which is able to make all calculations for both the dead-reckoning and the Kalman filter.

Even better results can be obtained, if an inertial navigation system or a doppler navigation system is used instead of the simple system using heading and true air speed. The navigation accuracies thus achieved are of a very high degree so that such an integrated system can be used as a reference system for calibrating radio-navigation systems.

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A great number of documents have been published concerning flight test techniques – the following list is a selection of AGARD documents and other publications.

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CP 85 – Flight Test Techniques 1971

CP 160 – Take-off and Landing 1974

CP 187 – Flight/Ground Testing Facilities Correlation 1975

CP 223 – Flight Test Techniques 1976

CP 260 – Stability and Control 1978

CP 242 – Performance Prediction Methods

AG 160 – Volume 1 – Basic Principles of Flight Test Instrumentation

AG 160 – Volume 2 – In-flight Temperature Measurements

AG 160 – Volume 3 – The Measurement of Fuel Flow

AG 160 – Volume 4 – The Measurement of Engine Rotation Speed

AG 160 – Volume 5 – Magnetic Recording of Flight Test Data

AG 160 – Volume 6 – Open and Closed Loop Accelerometers

Additional volumes of this series are in progress. A new series on flight test techniques will be produced very soon.

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CORP: Calspan Corp., Buffalo, N. Y.
A two-phase study was performed using the USAF NI-33A variable stability airplane. In Phase 1 of the study the effect of the evaluation technique used in the assessment of airplane flying qualities for the air combat flight phase was investigated. In-flight evaluations with and without a target airplane were performed for selected configurations from AFFDL-TR-70-74. The results from the evaluations with a target airplane were compared to results obtained with a target airplane and to the results in AFFDL-TR-70-74. In all cases, the evaluations performed without a target airplane were adequate to reveal potential flying qualities problems in the detailed pilot comments. There was, however, larger variability in the rating differences between the evaluations performed with and without a target. There were also significant rating differences between the evaluations performed with and without a target for about 15% of the configurations. (Modified author abstract)
AD-782218 CALSPAN-AK-5280-F-2 AFFDL-TR-74-9 74/04/00
75N22315

UTTL: Inflight engine performance evaluations
A/BECK, W. E., JR. PAA: A/(Lockheed-California Co., Burbank, Calif.) American Society of Mechanical Engineers, Gas Turbine Conference and Products Show, New Orleans, La., Mar. 21-25, 1976. 11 p.
The measurement and determination of installed in-flight engine performance on an aircraft are complex tasks. To be successful requires that wind-tunnel-model and full-scale in-flight test programs be fully planned early in the development phase of the aircraft design. In particular, methodology and instrumentation requirements must be coordinated in these programs. This paper discusses the requirements and methodology for in-flight engine performance evaluations as applied to a subsonic antisubmarine-warfare aircraft. Data are presented which show that two independent methods of calculating steady-state engine net thrust levels agree to within plus or minus 1.5 percent of each other over the full operating regime of the aircraft. Finally, comparisons of calculated performance with that predicted from the engine manufacturer's cycle deck are presented.
ASME PAPER 76-GT-70 76/03/00 76A25813

UTTL: Operational experience on Concorde
A/MCKINLAY, R. M.; B/HEATON, G. R. I.; C/FRANCHI, J.
PAA: A/(British Aircraft Corp., Ltd., Filton, Bristol, England); B/(British Aircraft Corp., Ltd., Fairford, Glos., England); C/(Societe Nationale Industrielle Aerospatiale, Toulouse, France) International Council of the Aeronautical Sciences, Congress, 10th, Ottawa, Canada, Oct. 3-8, 1976. 10 p.
The paper summarizes the 7-year (over 7,000 hours of flying) operational experience of the supersonic Concorde aircraft in service with Air France and British Airways. Statistics on the Certification Program development flights, including endurance flights intended to assess aircraft airworthiness, are followed by information on Concorde's airline service. Main features of the aircraft operation are analyzed.

AD-A088 530 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
AIRCRAFT ASSESSMENT AND ACCEPTANCE TESTING.(U)
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taking a flight phase by phase. The discussion covers aircraft loading, thrust checking, takeoff characteristics, subsonic climb, acceleration, supersonic climb, cruise, supersonic operations over the sea and land, descent, approach, and landing. Information is given on the influence of atmospheric conditions on aircraft operation, including radiation, turbulence, temperature and wind shear effects at cruise altitude, as well as on some problems of ATC and communications. An operational assessment of the aircraft from the pilot's point of view is given, along with some information on training program for Concorde's pilots.
ICAS PAPER 76-44   76/10/00   76A47388

UTTL: Instrument flight evaluation AH-1G helicopter
A/SKINNER, G. L.; B/SMITH, G. A.; C/BONIN, P. R.; D/WADDELL, R. W.; E/HEPLER, L. J.; F/MERRILL, R. K.; G/REID, J. S.; H/LARSEN, M. E.   CORP: Army Aviation Engineering Flight Activity, Edwards AFB, Calif.
A limited evaluation of handling qualities and operational performance was conducted on the AH-1G helicopter to determine its suitability for operation under instrument flight conditions with the stability and control augmentation system ON and OFF are discussed. Tests were conducted intermittently from February 1973 to December 1974. Ten shortcomings were identified, and it is recommended that two of these should be corrected prior to release for flight in instrument conditions.
AD-A026633 USAAEFA-72-29   75/07/00   77N13053

UTTL: Examples of laser utilization in civil aircraft certification tests
A/LAPCHINE, N.   CORP: Centre d'Essais en Vol, Istres (France).   In AGARD Flight Test Tech.   20 p (SEE N77-24107 15-05)
The STRADA system installed at the Brittany Flight Test Center was used in flight certification tests of the Concorde and Mercure 100 aircraft. The system uses optical radar (LIDAR) mounted on a turret, consisting of a yttrium-aluminum yag laser transmitting in the infrared at a frequency of 3200 impulses per second. The luminous energy received by passive retroreflectors mounted on the aircraft is received in an optical reception system linked to two receivers which measure distance and angle variations. A computer is used for real time calculation of the trajectory of the reflectors mounted on the aircraft. Tests for both the Concorde and the Mercure flights are described.   77/04/00   77N24127

UTTL: The role of simulation methods in the aircraft certification process
A/ARCHIBALD, D. M.   CORP: Lockheed-California Co., Burbank.
The extent to which the use of simulation may be increased during aircraft certification in the future was assessed by surveying recent industry experience and obtaining recommendations relative to the objectives of Advisory Circular 21-14. In addition to the industry assessment, a review of FAR-25 requirements from the economic and technical standpoint was made. Additional simulation in the certification process appears to be both economically attractive and technically feasible, particularly in the area of aircraft systems. The performance demonstrations, currently accomplished by flight testing, is an additional area where simulation may be applied.
AD-A039637 FAA-RD-77-17   77/03/00   77N27097

UTTL: The many disciplines of flight test: Proceedings of the Seventh Annual Symposium, Eastsound Orcas Island, Wash., August 4-6, 1976   Symposium sponsored by the Society of Flight Test Engineers. Lancaster, Calif., Society of Flight Test Engineers, 1976. 518 p (For individual items see A77-38004 to A77-38030)
The major sessions of the symposium dealt with: military flight testing operations and concepts; development and certification of test programs; techniques for data analysis and data acquisition; and special test requirements. Aeroservoelasticity, spin susceptibility testing, and failure detection of MLS position errors are among the special test requirements addressed. Use of minicomputers, and in some cases programmable pocket calculators, in flight test data analysis is discussed. Data acquisiton hardware and techniques are covered. Systems flight-tested include: air-launched Pershing II missile (from a Lockheed C-5), area navigation (RNAV) systems, an electronic integrated propulsion control system, and a quiet STOL concept.   76/00/00
77A38003

UTTL: An electronic method for measuring takeoff and landing distances
A/SCHICK, F. D.   PAA: A/(Gates Learjet Corp., Wichita, Kan.)   In: The many disciplines of flight test; Proceedings of the Seventh Annual Symposium, Eastsound Orcas Island, Wash., August 4-6, 1976. (A77-38003 17-05) Lancaster, Calif., Society of Flight Test Engineers, 1976, p. 22-1 to 22-21.
A system which consists of a 'trisponder' for

measuring horizontal distance and a radio altimeter for measuring height has been developed to replace the phototheodolite as a method for measuring takeoff and landing distances. The outputs of the two instruments are recorded on a magnetic tape data acquisition system and are computer reduced and plotted more rapidly than a photographic reduction can be processed. The system is designed to obtain information required for federal certification of takeoff and landing fields for jet aircraft.
76/00/00 77A38023

UTTL: Flight test technology: Proceedings of the Eighth Annual Symposium, Washington, D.C., August 10-12, 1977 Symposium sponsored by the Society of Flight Test Engineers. Lancaster, Calif., Society of Flight Test Engineers, 1977. 287 p. (For individual items see A78-19427 to A78-19445)
Recent contributions to flight testing from many engineering and allied fields are discussed. Attention is directed at requirements of major flight-test programs, human factors engineering problems, V/STOL testing and evaluation, weapon/mission system evaluation, and aircraft testing for determination of FAA stall speeds. Featured topics include YC-15 STOL performance flight-test methods, development of an on-board minicomputer system capable of real-time display of large amounts of flight-test data in instrument-corrected units, drawbacks in cockpit geometry design, and B-1 terrain following. Various analysis techniques and correction procedures are outlined. 77/00/00 78A19426

UTTL: Aircraft noise certification procedures
A/CALLAWAY, V. E.; B/LORETTE, M. L.; C/POTTER, R. L.
PAA: C/(Boeing Commercial Airplane Co., Seattle, Wash.) In: Inter-noise 78: Designing for noise control; Proceedings of the International Conference, San Francisco, Calif., May 8-10, 1978. (A79-15551 04-71) Poughkeepsie, N.Y., Noise Control Foundation, 1978, p. 675-684.
The paper outlines Boeing noise certification experience, with special emphasis on constructive suggestions. Nine years of experience with certification has identified ways to improve the methods and procedures involved. Suggestions for improvement include: (1) using level flyovers to establish noise levels of the first model of an airplane family, (2) establishing noise levels of derivative models using equivalent procedures, (3) requiring all testing to be within a new weather window and noise data to be adjusted to reference weather conditions using 10-m weather, (4) disregarding the use of the layered atmosphere weather adjustment procedure with its associated potential errors, and (5) specifying criteria in terms of noise rather than airplane weight for adjusting noise data in accordance to changes in aerodynamic performance parameters. These suggestions would decrease costs to the public, the FAA and industry. 78/00/00
79A15552

UTTL: Principles of helicopter performance
A/RICHARDS, R. B. CORP: Naval Test Pilot School, Patuxent River, Md.
This textbook is used as the primary reference for the Helicopter Performance Course at the U.S. Naval Test Pilot School. The Helicopter Performance Course is an integral part of the School curriculum, the particular requirements of which influence the manner and degree of development of these notes. The course is intended to provide a background for the helicopter performance flight projects conducted by the students. The helicopter, with emphasis on the main rotor, is analyzed in various flight conditions to determine the major factors which influence the performance. Simplified analysis is first applied to hover and then extended to include translational conditions after which consideration is given to the effect of some of the more significant simplifications.
AD-A061671 USNTPS-T-1 68/03/08 79N18970

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14. Abstract

The Lecture Series reviews the present state of the art of aircraft assessment and acceptance testing of production aircraft. This particular kind of testing is needed to select, from a variety of offers, the best aircraft type for a mission and to check that any aircraft on the production line is identical to the type, with acceptable tolerances.

Emphasis is placed on the practical aspects of this technique in order to help the flight crews and organisations dealing with this activity. Flight test instrumentation methods are set forth that do not require the use of sophisticated ground and airborne instrumentation for data acquisition or large computers for data processing. Flight test techniques described illustrate ways to acquire acceptable results utilizing a minimum of instrumentation or no instrumentation.

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