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STABILITY AND CONTROL

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AGARD Conference Proceedings No.119

STABILITY AND CONTROL

I

Papers and discussion from the 40th Meeting of the Flight Mechanics Panel of AGARD held in Braunschweig, Germany on 10-13 April 1972.

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- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field.
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PREFACE

An aircraft's flight safety and handling qualities are directly related to its stability and control characteristics. It is for this reason that the AGARD Flight Mechanics Panel has periodic "Stability and Control" symposia. Previous Stability and Control symposia were held in Rhode St. Genèse, Belgium in 1961 and in Cambridge, England in 1966.

Progress in aircraft configuration, mechanical systems, hydraulic systems, electronic systems, and cockpit systems since the 1966 symposium have contributed an abundant supply of new information. This, along with refinements in criteria and requirements, provided a wealth of material for the third AGARD "Stability and Control" symposium which was held in Braunschweig, Federal Republic of Germany on 10-13 April, 1972.

The value of the papers and the subsequent discussions, which are included in these proceedings, are a credit to the authors and session organizers.

W.T.Hamilton
Member of the Flight Mechanics Panel
Program Chairman

J.Renaudie
Member of the Flight Mechanics Panel
Assistant Program Chairman

CONTENTS

SUMMARY OF AGARD MEETING ON "PROBLEMS OF THE COCKPIT ENVIRONMENT" NOVEMBER 1968 IN AMSTERDAM, NETHERLANDS

by

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ABSTRACT

The symposium on "Problems of the Cockpit Environment" held in Amsterdam in November 1968 was the first broad-scope conference on this subject.

Most problems dealt with are related to the process of man-machine communication, with emphasis on cockpit information-generation, -display, and -transfer.

Techniques for the evaluation of cockpit geometry, display systems and cockpit workload as presented during the symposium are summarized, together with the associated anthropometrical data and types of display systems.

INTRODUCTION

The symposium on "Problems of the Cockpit Environment" was the first broad-scope conference on this subject.

It was organized by the Avionics Panel and held in co-operation with the Aerospace Medical Panel, the Flight Mechanics Panel and the Guidance and Control Panel in Amsterdam, Netherlands, November 1968 (Ref.1).

In giving the survey the following remark has to be made in view of the relation with the present symposium on stability and control.

In a man-machine system there are the basic criteria of system efficiency at the system level and the individual's well-being at the subsystem level. Stability and control requirements are at the system level and because the cockpit is a subsystem, one should not be surprised that most of the attention was focused at the subsystem level, reason to do the same in this paper.

It must be disappointing for those who have not gone through the proceedings, that there is so little to be found about the aircraft designers's problems.

Roughly half of the papers were dealing with displays.

With several studies reported in the papers substantial progress was made in the meantime.

The Guidance and Control Panel, one of the participating panels, recently held a meeting on the subject of guidance and control displays (Ref.2), in which most of this work was reported and I will refer to that later.

Twentynine papers were presented during sessions on:
1. The problems of determining crew capability under stress,
2. Problems in analysis and measurement of information transfer requirements and effectiveness for various missions,
3. The problems of correlating crew training, crew size and composition, and automated assistance,
4. The problems of cockpit design including instrumentation computer/display/control systems and components,
5. The problems of cockpit information generation,
6. The problems of deriving in-cockpit and head-up information display configurations.

It is not my intention to follow the papers in the order given in the proceedings. The scope of problems of the cockpit environment, moreover, is too broad to summarize them easily. Typical aspects I selected are:

cockpit subsystem design and evaluation,
cockpit geometry,
cockpit displays,
pilot workload.

COCKPIT SUBSYSTEM DESIGN AND EVALUATION

Various approaches to the cockpit subsystem design and evaluation can be found in the papers.

As the development progresses, the types of simulation evolve from mathematical models with their associated analytical techniques, to mockups, fixed and moving base simulators and flight tests. A technique established at the Human Engineering Division at the RAE is, although very briefly, described in a paper on the "Optimisation of the cockpit environment and the crew cockpit interface".

The state-of-the-art in certain areas of Human Engineering can be described by quoting from the following statements in this paper:

"In thermal stress problems, as in many other cases, there is a lack of the basic understanding of and data about the man, which the engineer can understand and use".

"The subject of vibration effects is used to suggest that, while there is an enormous amount of work and data available, for various reasons most of it is very unhelpful to the engineer."

"In Human Engineering we can quantify very little and optimise next to nothing whilst in the other subjects (Aerodynamics, Structures, Engines, etc.) very considerable effort is devoted to project studies aimed at securing the optimum combination of airframe, engine, etc."

The Integrated Cockpit Research Procedure developed at Litton is applied to the target detection and acquisition problem, a trade-off comparison between human and computer systems (paper 11).

The results indicate that man does not execute "brain" type operations with a wide bandwidth as does computer technology. Man has, however, an immense mass and associative memory storage function which is well beyond computer technology. Only a limited conclusion is given and the problem needs further study.

The same procedure is applied in a later study aimed at identifying control and display requirements of aircraft in advanced time periods (Ref.2).

The first phase of a study to come to a method to evaluate the design and assess the lay-out of a cockpit is reported in the paper "Crew work-load sharing assessment in all-weather, low level strike aircraft". The method was further developed and is described in the conference pre-print on guidance and control displays (Ref.2).

Methods which cover only part of the cockpit subsystem were also presented. One of these is:

COCKPIT GEOMETRY

Three papers were devoted to this subject (Fig.1).

A research program was in progress at the Boeing Company, directed at the development of mathematical models of the geometry of the pilot, the cockpit and other crew members (paper 14).

The conclusion is that the application of computer techniques to cockpit geometry evaluation is feasible. Some early results were presented but new data, needed for an orderly progress of the project were specified.

A second paper deals with the analysis and evaluation of anthropometric data (paper 15). The state-of-the-art here does not seem to differ much from that mentioned earlier.

The conclusion is that new techniques of measurement and new methods of presentation of anthropometric data are required for meaningful progress to be made in the cockpit design and that more emphasis is required on dynamic measurements of the seated operator. New and in particular graphic methods of presentation for anthropometric data are required in order to aid the design engineer in the utilization of these data.

A third paper describes a method for evaluating and comparing aircraft in terms of the ground areas visible from the cockpit (paper 16). Use is being made of a binocular cockpit visibility camera. Ground visibility plots are deduced from the cockpit visibility photographs. The data are important for visual reconnaissance, strike, take-off and approach and landing.

COCKPIT DISPLAYS

Almost all of the papers in the second part of the proceedings deal with displays. Display design in total should follow the aircraft development program. This of course is not always necessary for individual instruments. Display design has many aspects. Some of the more important attributes are:

Mission (phase) The mission phases in which the pilot's performance is critical are e.g. approach and landing. Most of the attention was therefore focused on these flight phases (papers 7, 10, 18, 20, 24).

Type of aircraft The vertical take-off and landing aircraft pose problems of a different nature than conventional aircraft. Several papers are related to this type (papers 7, 18, 19, 26). The display formats indicate a significant influence of the type of aircraft on the displays.

Function For obvious reasons almost every electro-mechanical display e.g. engine instruments, navigation displays, primary flight instruments, is a mono-function display. Only the paper on the electronic display of primary flight data referred to multi-function displays (paper 28). More about the development of multi-function displays can be found in ref. 2, in which the approaches taken by Elliott Flight Automation and Thomson-CSF are presented.

Mode The mode of a display refers to the human sensor involved in the interaction between man and machine. This mode can be visual, audio, tactile, etc. Visual displays are definitely dominating.

Position of visual displays Three categories are normally recognized, head-down displays, head-up displays and helmet mounted displays. The head-up displays attracted far more attention (papers 20, 23, 24, 30) than helmet mounted displays (paper 26), although the development of the latter has progressed rapidly (ref.5).

Media for visual displays Although there are a large number of display media, examples are given only of electro-mechanical and cathode ray tube displays. New technology has evolved since then. In this resplect reference should be made to the session on new technology for guidance and control displays presented in ref. 2.

Head-up and helmet mounted displays are, except for simple sights and the CSF head-up display, cathode luminiscent devices.

Information transfer If there is a possibility that the information transfer from the display to the pilot does not occur, or if the information is misunderstood or misinterpreted, ways have to be found to improve this. The development of a central hot message and advice panel is subject of a study at Cornell Aeronautical Laboratory (paper 6 and ref. 4) (Fig.2).

The quantification of information transfer is still a problem.

A discussion of some other papers dealing with these attributes seems to be worthwhile.

The paper on the display of aeronautical charts outlines the status of airborne chart displays (paper 27). The same author gave a general review in ref. 2.

For those who are interested in map displays in more detail, reference is made to a symposium dealing with the Geographic Orientation in Air Operations (ref. 6). Four basic types of map displays are recognized: direct-view map displays, projected map displays, combined map/CRT displays and electronically generated map displays.

Some advantages and limitations are the following:

The direct-view display can be used with standard paper charts and the pilot has direct access to the chart, so that he can mark it, however, the storage capacity is rather limited.

The microfilm projection display has storage capacity for millions of square miles, however, the pilot cannot readily annotate the chart image.

The advantages of a combined map/CRT display originate from the versatility of the CRT as a display medium.

The limitations are that the systems are complex, large and heavy.

Maximum exploitation of the versatility is made in an electronically generated map display, the development of which has, however, hardly started.

A recommended engine instruments panel lay-out for current aircraft is presented in figure 3.
The development of computerised and motorised engine instrument displays, automonitoring and maintenance recorders is proposed to reduce the pilot's workload (paper 29) (Fig.4). It is further proposed that cockpit emergency controls should be standardised to a certain degree. This is quite logical in view of the accidents that have happened in relation to habit interference.
Advantages which have been stated for the head-up display (paper 30) are:

1. A better efficiency of the flight instruments,
2. Improved display sensitivity and accuracy in relation to the conventional instruments,
3. Less strain for the pilot,
4. Easy monitoring of the autopilot operation,
5. Accurate monitoring of the descent slope in VMC approach and landing,
6. Continuous external watch, accomodation at infinity,
7. The possibility for Category III roll-out, take-off and landing.

Only the last three advantages are proper to the head-up display.
Examples of possible display formats are shown in figs. 5, 6 and 7. A disadvantage of the head-up diplay is the relatively narrow field of view. The helmet mounted display is developed to overcome this problem and to save panel space (Fig.8). Little information is available in the human factors engineering literature as to the capability of the pilot to track with his head (paper 26). The development of helmet mounted displays has progressed rapidly (ref.5).

The electronic display of maps has been mentioned before. The electronic display of primary flight data was the subject of a paper by Walters (paper 28). Needed panel space can be limited by time sharing between various displays. This type of system will find increasing application. There is still a considerable amount of research needed in this area. Some typical formats are shown in fig.9.

For quasi static situations, numerical displays are considered to be good for presenting quantitative information and poor for providing qualitative information. The use of numerical displays in dynamic situations is treated in the paper on "Numerical displays for the presentation of dynamic information".
It is essential here to distinguish between open and closed loop tasks. It is concluded from the experiments that a numerical display can provide information, sufficient to allow subjects to perform continuous tracking tasks, but apparently at the expense of additional attentional cost.
Research is required to investigate the effectiveness of an operator when controlling a number of tasks each using numerical displays.

Two new analog displays were tested in a fixed-base simulator during approach and landing (paper 10) (Fig.10).
Some typical results indicated that an average pilot can perform consistently accurate landings using these displays even with unfavourable aircraft characteristics.

The results of a study on VTOL displays and controls for all-weather co-ordinated flight of helicopter formations has shown that the precision of control in the formation is a function of the quickened signals (paper 19). These signals are dynamically equivalent to those signals essential for a stability augmentation flight control system. Three formats were tested. The PPI format (fig.11) was the most satisfactory for a formation flight system.
A simulator program has been developed to design new engine displays and displays for hydraulic and electic systems for the Do 31 (Fig. 12 and 13). A theory for manual control display was applied to the instrument landing approach (paper 5). Improvements have been made and the theory has led to an analytic approach to display design (ref.2). The use of feedback control theory in display design still meets a lot of scepticism among human engineers. The verbal analytical models of pilot dynamics, which are used in the theory, do, however, lead to practical useful results.

PILOT WORKLOAD

All approaches both for the cockpit sub-system design and the display design lead to the critical points of allocation of functions and pilot workload. This probably is the most difficult aspect. Attempts are made to record eye movements (paper 3). This may give information about the foveal scanning pattern, but not about the parafoveal scanning pattern. Information on scanning patterns is used to determine the scanning workload (paper 5).
Head and hand movements are recorded, using cine-cameras with wide-angle lenses (papers 3, 12, 14).
Before measurements can be performed, the pilot's workload has to be estimated, based of course on suitable criteria.
In the Integrated Cockpit Procedure use is made of a so-called time-based load analysis to provide a quantitative index of operator load. Other criteria to assess man's performance and to define his task load can be found in the papers.
Stick movement and output error (paper 8), transinformation (paper 9), sinus arrhythmia and the dual task method for measuring mental load (paper 4), survey method (paper 6), questionnaires (paper 17), the well-known Cooper-Harper rating scale adapted to this application, semantic differentials, and questionnaire-guided interviews (ref. 2) and still many other techniques are known.
A year after the meeting on the cockpit environment, a symposium was held, also in Amsterdam, on "Measurement of man at work"; an appraisal of physiological and psychological criteria in man-machine systems (ref. 3), in which most of these techniques can be found.
The situation here does not seem to be very satisfactory either. I will not go further into this subject, however, because it is discussed later on during this symposium by Mr. Thorne.
It was stated by Mr. Fish that there had not been done enough work on man under stress in the cockpit. I like to close with the response of chairman Domeshek of the symposium on problems of the cockpit environment: "I can only say "amen" to that".

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5. Klass, P.J.
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Av. Week and Space Technology, 31 Jan. 1972.

6. Geographic orientation in air operations
Janair Symposium Washington, Nov. 1969.

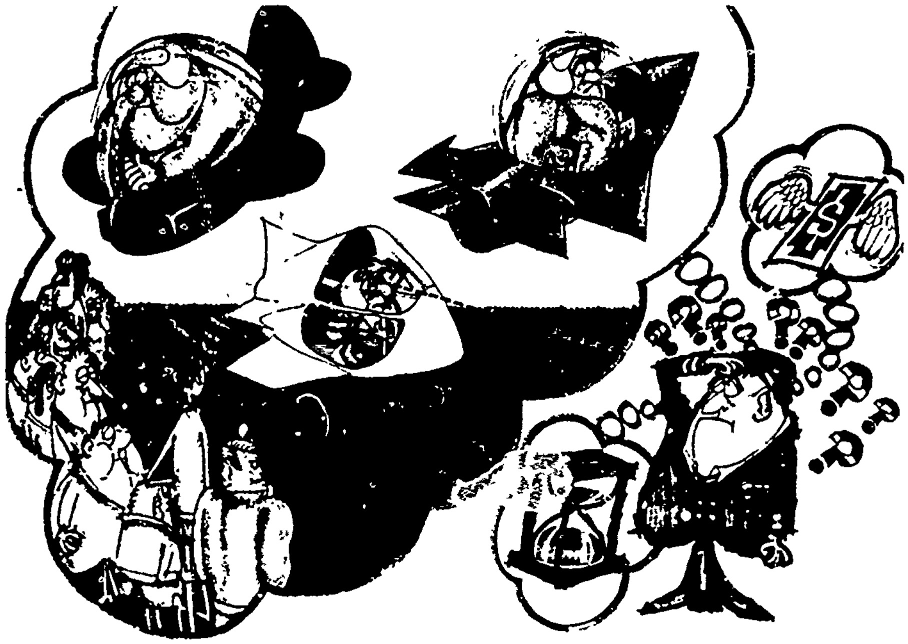


FIG. 1 COCKPIT GEOMETRY (A PROBLEM).

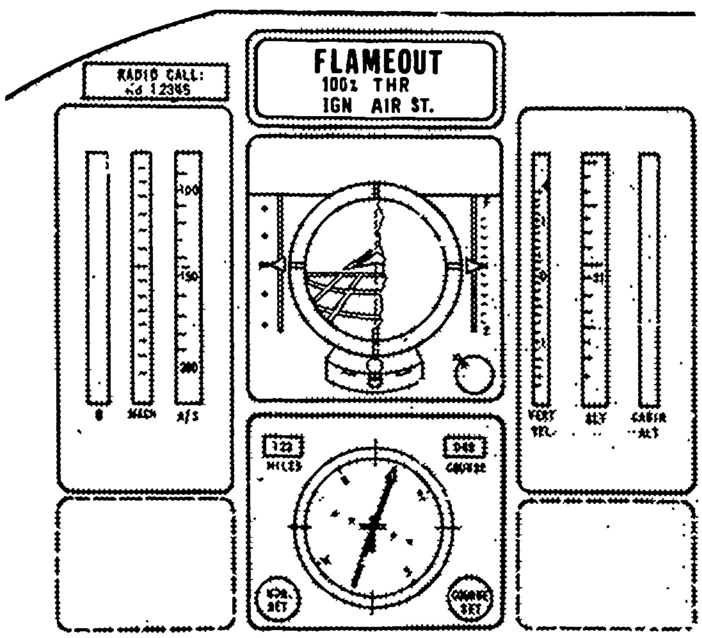


FIG. 2 HOT MESSAGE AND ADVICE PANEL.

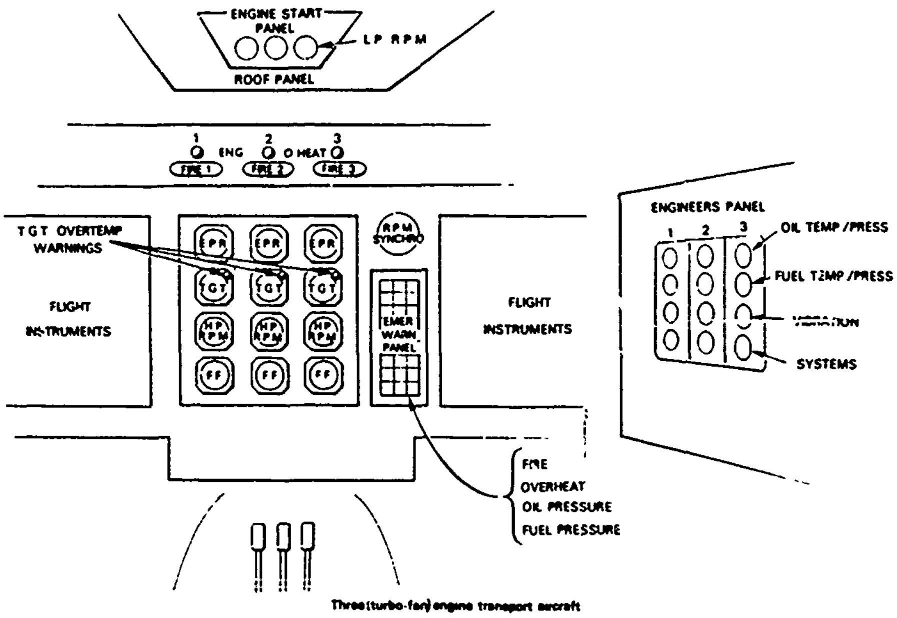


FIG. 3 RECOMMENDED ENGINE INSTRUMENTS PANELS LAYOUT FOR CURRENT AIRCRAFT.

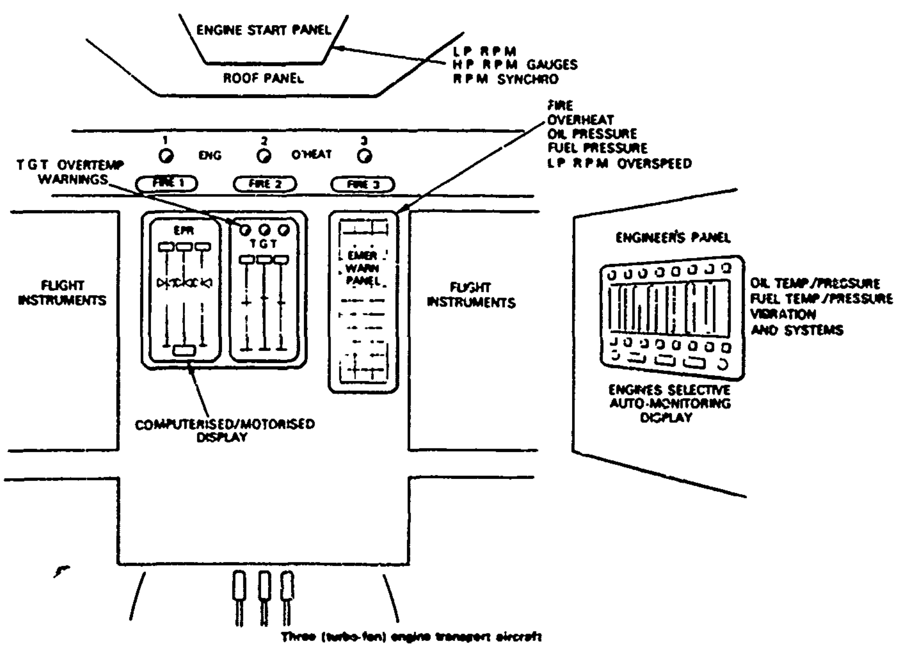


FIG. 4 PROPOSED ENGINE INSTRUMENTS PANELS LAYOUT FOR FUTURE AIRCRAFT.

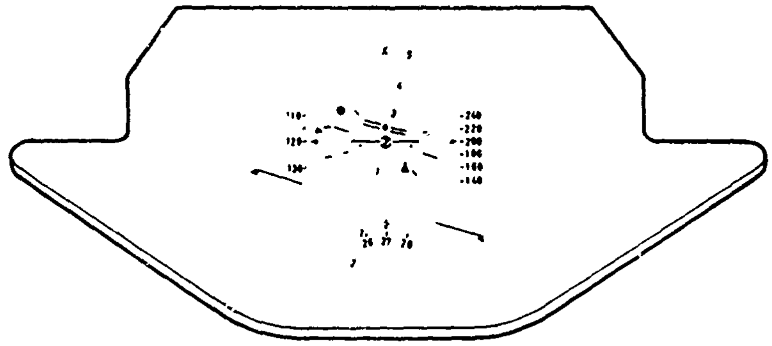


FIG. 5 IMAGE SEEN BY THE PILOT DURING LANDING (C S F).

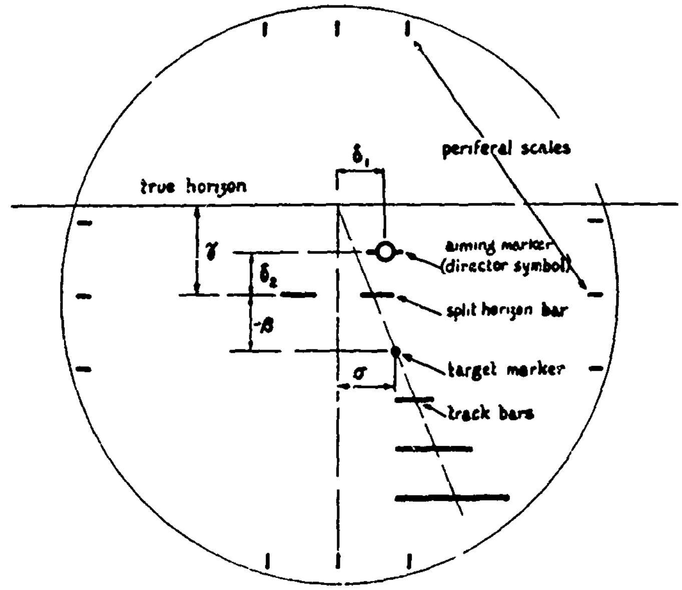


FIG. 6 IMAGE SEEN BY THE PILOT DURING LANDING (N A D).

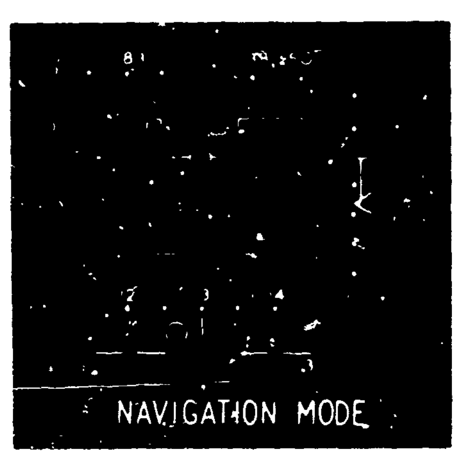
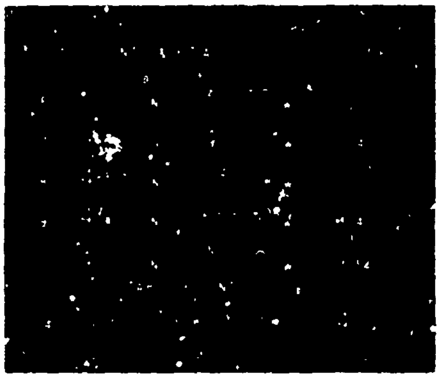


FIG. 7 IMAGE SEEN BY THE PILOT DURING NAVIGATION.

FIG. 8 HELMET MOUNTED D:SPLAY WITH ELECTROMECHANICAL HEADTRACKER ATTACHED TO THE HELMET.



'YPICAL TABULAR FORMAT FOR NAVIGATION DISPLAY.

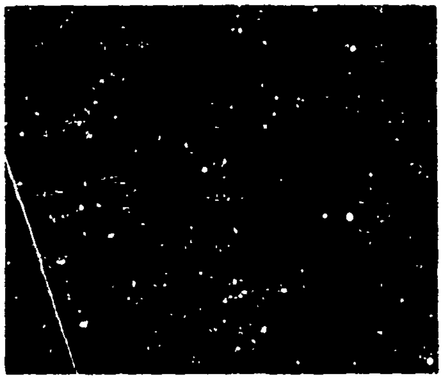
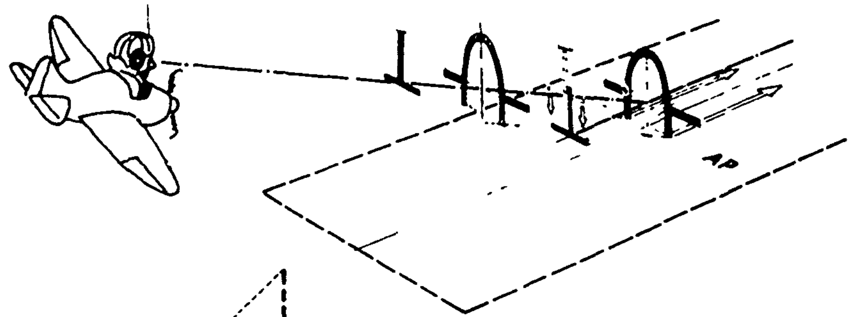
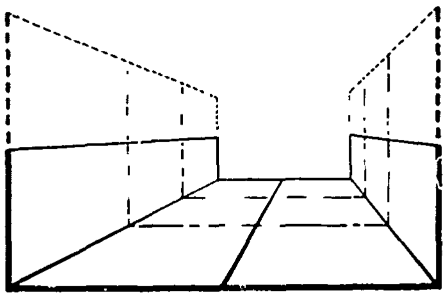


FIG. 9. TYPICAL FORMAT FOR DIVERSION DISPLAY.



b. FORMATION DISPLAY



a. BOULEVARD DISPLAY

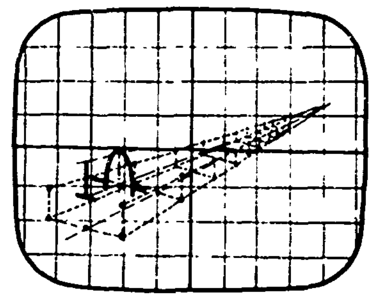


FIG. 10 RELATIONSHIP BETWEEN THE DISPLAYS.

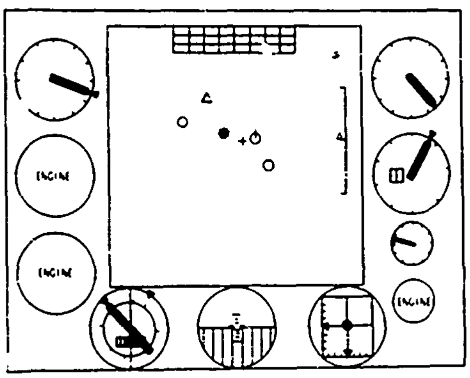


FIG. 11 PLAN POSITION INDICATOR (P P I) FORMAT.

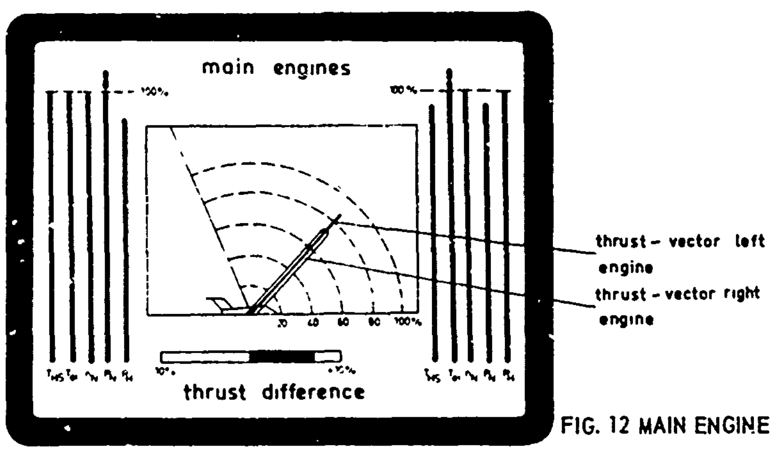


FIG. 12 MAIN ENGINE DISPLAY.

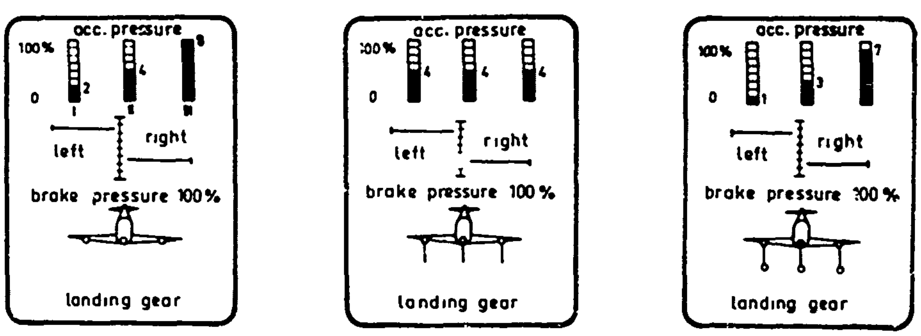


FIG. 13 EXAMPLE OF A DISPLAY FOR THE HYDRAULIC SYSTEM.

EFFETS AEROELASTIQUES DU POINT DE VUE DE LA MECANIQUE DU VOL

AEROELASTIC EFFECTS FROM A FLIGHT MECHANICS STAND POINT

Tentative pour résumer
la réunion du F.M.P. en
avril 1969 à MARSEILLE - France

par J. F. RENAUDIE
C E V - FRANCE

J'ai accepté, en mai 1971, à OTTAWA, de faire le résumé des travaux de la réunion du Groupe de Mécanique du Vol à MARSEILLE en avril 1969, en réponse à une suggestion de mon ami Pierre LECOMTE, alors président, dont le but était de présenter les travaux antérieurs du groupe ayant un rapport direct avec le Symposium sur la Stabilité et les commandes de vol.

Lorsqu'il y a quelque temps déjà je me suis mis à la tâche, j'ai tout de suite réalisé à quel point cette acceptation était imprudente. Le recueil des conférences, publié par AGARD contient plus de 300 pages grand format de petits caractères, d'explications mathématiques et symboles et de figures. Autant essayer de résumer une encyclopédie.

Pourtant, ayant assisté personnellement à la réunion de Marseille, j'en étais revenu avec une impression différente. C'est cette impression que je voudrais essayer de faire revivre.

Nous avons entendu à MARSEILLE vingt cinq exposés ; chacun de ces exposés était présenté par un ou plusieurs auteurs ; cinq sessions regroupaient des sujets voisins.

J'essaierai donc d'abord de résumer brièvement chacune de ces sessions. Ensuite je choisirai quelques uns des exposés les plus caractéristiques, dont certaines illustrations seront projetées à nouveau aujourd'hui.

Je voudrais à l'avance prier les auteurs des exposés choisis de me pardonner si l'image que je donnerai ainsi de leur travail leur parait inexacte, incomplète, ou même tendancieuse ... ils ne verront là que le procédé habituel des exploitants des salles de cinéma : afficher dans la rue les images les plus suggestives du film pour inciter le spectateur à en voir plus : je jugerai ma présentation satisfaisante si l'auditoire a l'envie de lire ou de relire ces exposés tels que présentés dans leur intégralité par leurs auteurs.

J'ai choisi six seulement des vingt cinq exposés présentés ; il me faut dire aussi que j'ai volontairement laissé de côté certains exposés pourtant très intéressants parce qu'ils me paraissaient moins liés au Symposium sur la Stabilité ; d'autres exposés n'ont pas été cités en dépit de leur rapport avec le sujet d'aujourd'hui simplement parce qu'il fallait bien faire un choix ; ce choix croyez-le bien a été difficile et je suis sûr qu'il est criticable ; là encore j'aurai réussi si l'auditoire a le désir de consulter le recueil des travaux présentés à MARSEILLE.

Comme je l'ai dit déjà, cinq sessions ont regroupé les exposés, de manière parfois assez arbitraire.

La lère session, dite préliminaire, était en principe consacrée aux règles de l'art ; en fait, mis à part un exposé de l' ONERA consacré à la mesure statistique de la turbulence, les exposés ont surtout été consacrés aux méthodes de calcul permettant de représenter les déformations de l'avion souple et leurs influences sur les forces aérodynamiques. Je citerai deux exposés : celui présenté par le NLR Hollandais et celui présenté par l' AEROSPATIALE (France).

La 2ème session était consacrée à l'aérodynamique des retors flexibles. En dépit de leur grand intérêt, je ne citerai aucun des exposés car le Symposium ne traite pas des appareils à voilure tournante.

La 3ème session était consacrée à la mécanique du vol proprement dite des avions flexibles, je citerai les exposés présentés par la NASA, la Société BOEING et le RAE de FARNBOROUGH.

La 4ème session était consacrée à la détermination expérimentale des effets de la flexibilité, je citerai un exposé de la N A (LANGLEY Field).

La 5ème session était consacrée aux systèmes d'augmentation de stabilité ; tous les exposés auraient pu être cités. En fait j'ai choisi de vous rappeler la présentation faite par le Canadien J. A. Mc KILLOP, ce qui est une façon d'honorer la mémoire de cet auteur, tué depuis lors d'un accident en vol d'essais.

L'impression dominante, à l'issue de cette réunion, est que l'aéroélasticité constitue, par ses nombreuses implications, l'un des facteurs qui préoccupe le plus le constructeur actuel d'avions. Lorsqu'avec J. T. HAMILTON de la Société BOEING, avec qui j'ai eu le plaisir d'organiser le programme du Symposium sur la Stabilité, nous avons cherché un thème, nous avions trouvé le suivant :

De bonnes caractéristiques de stabilité et de maniabilité, objectif fondamental de chaque phase de mise au point de l'avion, de la planche à dessin à la construction et aux essais en vol.

Et bien, on aurait aussi pu dire : Comment lutter contre l'aéroélasticité à tous les stades :

- conception et dessin initial,
- mise au point en soufflerie,
- fabrication du prototype,
- mise au point et fabrication de la série.

Mais revoyons quelques images sélectionnées ...

Voici 3 Figures extraites de l'exposé de MM. BERG et ZWAAN du N L R :

Fig. 1

MM. BERG et ZWAAN ont présenté la théorie des surfaces portantes pour calculer les coefficients aérodynamiques d'une voilure flexible.

Cette théorie repose entre autres sur le choix des fonctions permettant de représenter la forme de la répartition des charges de pression sur la surface portante, dites fonctions de KERNEL.

La figure montre le principe de la méthode utilisée et le texte donne un exemple d'application au mouvement harmonique.

Plusieurs hypothèses doivent être faites dans cette application :

- écoulement à potentiel sans choc ni décollement sur toute la surface qu'intéresse l'intégrale,

- mouvements assez petits pour autoriser la linéarisation,

- non influence de l'écoulement stationnaire à une incidence α donnée sur l'écoulement instationnaire.

D'autre part, par principe même, l'usage de fonctions de répartition de charges se heurte à des obstacles insolubles en transsonique. Par contre, en supersonique, l'hypothèse de charge constante simplifie les calculs.

Fig. 2

La figure 2 montre un exemple de découpage de l'aile en éléments.

Fig. 3

L'application de la théorie des surfaces portantes au mouvement oscillatoire de tangage d'une aile ogiva e donne les résultats que montre cette figure.

On constate un accord assez bon entre la théorie et les mesures pour les coefficients d'amortissement de tangage, mais par contre la raideur mesurée est double de la raideur calculée.

N'étant pas spécialiste de ces calculs théoriques, je ne me hasarderai pas à critiquer le détail de leur application. Mais je constate seulement que la plupart des théories ne peuvent par elles seules parvenir au résultat cherché. Chaque auteur s'efforce d'introduire au mieux un outil de correction empirique qui permettra, une fois réalisées des mesures sur un petit nombre de modes de déformation élastique d'en déduire les forces et coefficients aérodynamqiques correspondant aux autres modes. Les conclusions de MM. BERG et ZWAAN sont des conclusions de prudence : elles incitent à utiliser les calculs comme un outil d'interpolation ou d'extrapolation entre des données expérimentales.

Fig. 4

La figure suivante est extraite de l'exposé fait par Mr DAROVSKY, de l' AEROSPATIALE France. Mr DAROVSKY a tenté de répondre à la question : par quel ensemble d'équations peut-on représenter le vol de l'avion souple ?

L'auteur donne une réponse claire au problème statique : ce sont les mêmes équations que celles de l'avion rigide :

- il y a donc le même nombre d'équations différentielles que pour le solide indéformable ;

- chacune de ces équations a la même écriture qu'il s'agisse de l'avion flexible ou de l'avion indéformable. seuls les coefficients dits encore dérivées aérodynamiques diffèrent.

Seulement, à la différence des dérivées "rigides" les dérivées souples dépendent de plusieurs paramètres supplémentaires :

- répartition des masses et configuration initiale de vol avion rigide,

- répartition des pressions sur l'avion rigide.

L'auteur illustre par cette figure l'utilisation pour le cas instationnaire, de dérivées analogues aux dérivées statiques, à raison d'un jeu de dérivées par mode de déformation, dit encore dérivées quasi statiques. Si l'on considère un nombre de modes suffisant que l'auteur estime être de l'ordre de 6 ou 7 cette méthode beaucoup plus légère que la méthode générale harmonique, plus exacte mais laborieuse, donne cependant des résultats très suffisants dans la pratique. Sur cette figure présentant l'application à la réponse d'un avion souple à la turbulence atmosphérique, on voit comme il y a peu de différence entre le calcul complet instationnaire (lignes continues) et le calcul quasi statique (lignes pointillées).

Dans les conclusions de sa présentation l'auteur formule un voeu qui est aussi le mien : c'est qu'enfin aéroélasticiens et aérodynamiciens se mettent d'accord sur l'emploi de notations communes pour parler des forces aérodynamiques. Il y a là une suggestion pour briser cette barrière du langage qui pourrait fournir l'idée d'une initiative AGARD dans ce domaine.

Fig. 5

Cette figure est extraite de l'exposé de Mr CHEVALIER de la NASA (AMES) et de MM. DORNFELD et SCHWANZ de BOEING. Cet exposé donne l'exemple d'applications de méthodes de calcul des effets de l'aéroélasticité sur la stabilité statique et dynamique dans le cas de deux avions le BOEING 707 320 B et le projet de SST.

La méthode de calcul avait été précédemment exposée par Mr DUSTO (BOEING Co). Elle est assez voicine de celle présentée par Mr DAROVSKY (AEROSPATIALE), que je viens de rappeler, en ce sens que ce sont les coefficients des équations des petits mouvements qui sont modifiés pour introduire l'effet de la flexibilité.

Pour ce faire, l'avion est remplacé par un assemblage de panneaux élémentaires comme le représente cette figure. Pour chaque panneau la masse est supposée concentrée, et l'on calcule le déplacement d'un panneau déterminé, le n° 48 par exemple du BOEING 707, sous l'influence d'une force s'exerçant suivant une direction donnée sur le 25è panneau.

Répétant l'opération pour l'influence de chacun des 70 panneaux de l'avion sur le panneau n° 48, pour chacune des 3 directions de force pouvant être considérées, on obtient 3 x 70 = 210 coefficients d'influence par panneau soit pour tous les panneaux de l'avion une matrice de 210 lignes et 210 colonnes.

Le calcul du mouvement de chaque panneau est ainsi traité par l'emploi du calcul matriciel qui conduit finalement, compte tenu des diverses conditions de continuité et de limite à 6 équations sous forme matricielle.

Ces équations peuvent être simplifiées si l'on suppose que les forces d'inertie et d'amortissement associées à la flexion structurale sont négligeables et l'on est conduit ainsi à une représentation dite quasi statique.

Fig. 6

Appliquée à un modèle rigide, l'application de la théorie quasi statique au calcul des dérivées aérodynamiques avion donne les résultats suivants (stabilité statique) comparés avec les résultats de soufflerie sur le modèle rigide :

- bon accord sur le coefficient $C_{l\alpha}$ (gradient de portance avec l'incidence) ;

- accord moins bon pour le coefficient $C_{m\alpha}$ (gradient de moment de tangage avec l'incidence) ; seule l'évolution avec le nombre de MACH est bien représenté.

Fig. 7

Cette figure reprend en lignes continues les résultats "avion rigide" de la figure précédente ; on leur superpose les courbes pointillées qui représentent les résultats des calculs "avion flexible" et l'on compare avec les résultats d'essais en vol : incontestablement les calculs flexibles rendent mieux compte des faits.

Fig. 8

Si maintenant on considère la stabilité dynamique courte période, l'auteur retrouve la même conclusion que celle de Mr DAROVSKY : la théorie élastique et même la théorie rigide quasi statique rendent compte de manière très satisfaisante de l'amortissement du mouvement : il est inutile de procéder au calcul complet avec un grand nombre de mode pour cet avion. L'auteur souligne toutefois qu'une telle simplification ne serait pas valable si la fréquence de mode fondamental de structure n'était pas au moins quatre fois plus grande que celle de l'oscillation dynamique courte du mouvement de l'avion rigide.

Pour terminer je soulignerai certaines conclusions de MM. CHEVALIER, DORNFELD et SCHWANZ :

- les théories utilisées donnent une représentation utile des principales dérivées de stabilité ;

- les désaccords sont dus principalement aux théories aérodynamiques ;

- l'approximation quasi statique élastique est suffisante pour les modèles étudiés ;

- la plupart des effets de l'aéroélasticité sont adverses sur les caractéristiques statiques avion et gouvernes ;

- l'élasticité a peu d'effet sur la stabilité dynamique quand les fréquences de structure sont bien séparées des fréquences de la courte période.

Fig. 9

Cette figure est extraite de l'exposé de Mr ROSKAM, de l'Université de KANSAS. Elle montre l'importance d'une connaissance satisfaisante de la déformation de la structure afin de concevoir celle ci pour qu'en vol de croisière la forme de l'aile corresponde à l'optimum recherché pour les performances. Pour l'avion supersonique qui est pris comme exemple, cette recherche de la forme à donner au bâti de fabrication est essentielle. La méthode de calcul proposée à cet effet est celle déjà décrite lors des précédents exposés notamment celui de Mr DUSTO.

Un point toutefois mérite d'être souligné : c'est l'utilisation d'un modèle dit "équivalent élastique" pour décrire la déformation de l'avion satisfaisante : l'hypothèse faite est que charges et déformation de la surface extérieure de l'avion sont en phase les uns avec les autres et avec les mouvements des axes de stabilité.

Enfin, l'importance en valeur absolue des écarts entre la forme au bâti d'assemblage et celle en vol de croisière doit être notée : plus de 4 % de la corde de référence de la section d'aile considérée : 40 cm pour une corde de 10 mètres !

Fig. 10

Un autre effet très important de l'aérodistorsion et probablement l'un des plus difficiles à prédire est le déplacement du foyer aérodynamique. Cette figure extraite de l'exposé de Mr ROSKAM en fournit l'exemple.

Mais, et c'est un point très important que souligne Mr ROSKAM, la définition même du foyer aérodynamique dépend des manoeuvres envisagées, de la même façon que les dérivées partielles d'une fonction de multiples variables dépendent du choix de ces variables. Ainsi on est amené à définir plusieurs foyers supplémentaires ; on avait déjà pour l'avion rigide un foyer aérodynamique manche bloqué, un foyer aérodynamique manche libre, un point de manoeuvre.

Il faut maintenant distinguer aussi :

- le foyer aérodynamique à facteur de charge constant pour les transitions quasi statiques entre les différentes incidences ;

- le foyer aérodynamique à facteur de charge croissant pour les ressources et les virages.

Fig. 11

Avec la figure suivante, extraite de l'exposé de Mr BURNHAM du RAE BEDFORD, c'est un sujet différent des précédents qui est traité : celui de l'influence des rafales atmosphériques sur l'avion flexible.

L'exemple présenté met en évidence l'amplification des rafales atmosphériques lorsque les fréquences qu'elles contiennent sont voisines des fréquences de résonnance des différents modes de déformation structurale. L'auteur souligne également la nécessité de tenir compte de la grande sensibilité de l'homme aux fréquences voisines de 4 Hertz.

Fig. 12

La question qui se pose alors est la suivante : dans quelle mesure peut-on à l'aide de dispositifs automatiques d'autostabilisation ou autres, réduire les effets combinés des rafales atmosphériques et des modes de structure ? Pour obtenir ce résultat l'auteur préconise l'emploi du contrôle direct de la portance, dont cette figure montre les effets, comparés à ceux obtenus par l'emploi de gouvernes classiques. Le gain obtenu dans la lutte contre les effets d'une rafale tient essentiellement à l'instantanéité de l'action, sur le facteur de charge du contrôle direct de portance.

Fig. 13

Cette figure est extraite de l'exposé de MM. RAYNEY et ABEL de la NASA LANGLEY, présentant des méthodes expérimentales pour déterminer au stade des essais en soufflerie la réponse d'un avion aux rafales.

Les techniques expérimentales ont été assez peu traitées lors de la réunion de Marseille, mis à part un exposé de l' ONERA (FRANCE) sur la détermination en vol de la fonction de transfert de la réponse d'un avion existant, à la turbulence.

Le thème du Symposium sur la Stabilité est, répétons-le : de bonnes qualités de vol, objectif essentiel aux différents stades de la genèse d'un avion, de sa conception à sa mise en opération. Et bien l'exposé de MM. RAYNEY et ABEL montre comment déterminer dès le stade de la soufflerie les coefficients aérodynamiques de l'avion flexible.

Le modèle flexible est suspendu dans la chambre d'expérience du tunnel aérodynamique transsonique de LANGLEY. La suspension est étudiée de manière à avoir une fréquence propre très basse comparée à celle des oscillations aérodynamiques et structurales du modèle et une masse mobile négligeable par rapport à celle du modèle ; elle fournit les entraves de sécurité.

Un dispositif de volets oscillants (vannes) engendre des rafales sinusoïdales qui excitent le modèle.

Ces modèles flexibles sont extrêmement coûteux (50 000 à 500 000 $) et afin d'éviter leur détérioration lors de la mise au point de l'expérience, on les remplace généralement par des modèles rigides ayant la même forme, la même masse et la même inertie.

Fig. 14

La qualité du système de suspension doit être telle que le modèle vole d'une manière pratiquement libre dans le tunnel. La non influence de la suspension est montrée dans cette figure qui représente une expérience faite avec un modèle rigide dont les coefficients aérodynamiques sont bien connus, et par suite la réponse aux rafales exactement calculable. Dans ces conditions on voit que la réponse mesurée par le système est très voisine de la réponse calculée aux alentours du pic à 2 Hertz qui représente l'oscillation courte période. Aux plus basses fréquences la suspension perturbe les mesures.

Fig. 15

Extraite du même document de MM. RAYNEY et ABEL voici une figure qui montre l'utilisation de la soufflerie pour déterminer la vitesse à laquelle s'inverse l'efficacité d'ailerons.

On voit l'excellente concordance des mesures en vol réel et celles effectuées avec le modèle en soufflerie.

Fig. 16

Pour terminer ce survol des exposés faits à MARSEILLE j'ai choisi l'exposé fait par Mr Mc KILLOP qui a trouvé la mort lors d'un essai en vol.

Cet exposé a montré comment un système permettant de réduire l'influence des rafales atmosphériques peut faire partie de la conception même de l'avion.

Avion bien extraordinaire vous en conviendrez ... Il s'agit d'une poutre volante longue de 378 ft, capable de véhiculer à 50 MPH des inscriptions publicitaires et comportant une paire d'ailes à l'avant, une paire d'ailes à l'arrière ; un pilote à l'avant, un pilote à l'arrière.

Fig. 17

Pour rendre acceptable un tel avion pour des missions d'environ 4 heures, il était essentiel de réduire l'influence des rafales sur son mouvement.

La solution trouvée dans ce but est de rendre chaque paire d'ailes libre de tourner autour d'un axe perpendiculaire au plan de symétrie afin de se placer constamment à la même incidence.

Il en résulte quelques caractéristiques inusuelles, telles que la disparition des modes naturels d'oscillation, extrêmement amorties (plus de phugoide).

- - -

Avant de terminer cette revue de la réunion de MARSEILLE du Groupe de Mécanique du Vol, je voudrais à nouveau m'excuser auprès des auteurs d'avoir emprunté les figures qu'ils ont présentées et aussi de n'avoir pu citer tous ceux qui ont activement participé à cette réunion.

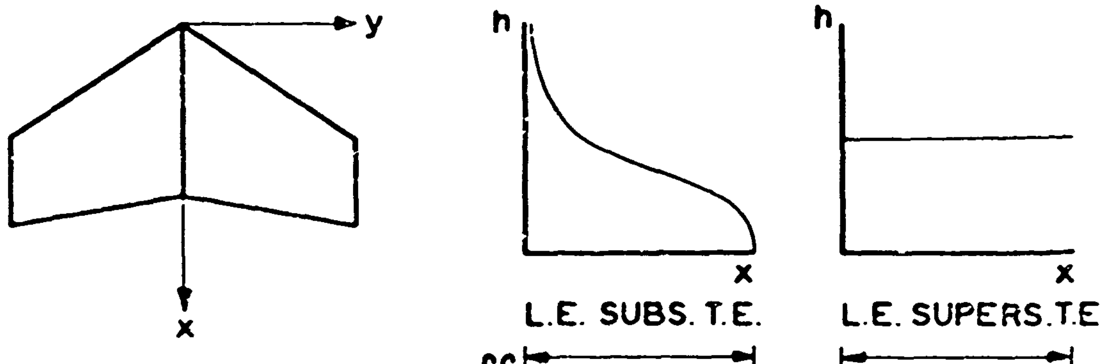
Plus que jamais l'aéroélasticité apparaît désormais comme une science fondamentale dont il ne faut jamais négliger les effets à quelque stade que ce soit de la naissance d'un nouvel avion.

Bien des questions restent cependant posées qui probablement n'auront pas de réponse dans un avenir immédiat, par exemple comment prévoir l'évolution de certaines dérivées aérodynamiques lorsque sont combinés les effets de la flexibilité et les bouleversements qui affectent les écoulements aérodynamiques du passage du subsonique au supersonique.

ANNEXE

PROGRAMME OF THE FMP MEETING

MARSEILLE - APRIL 1969

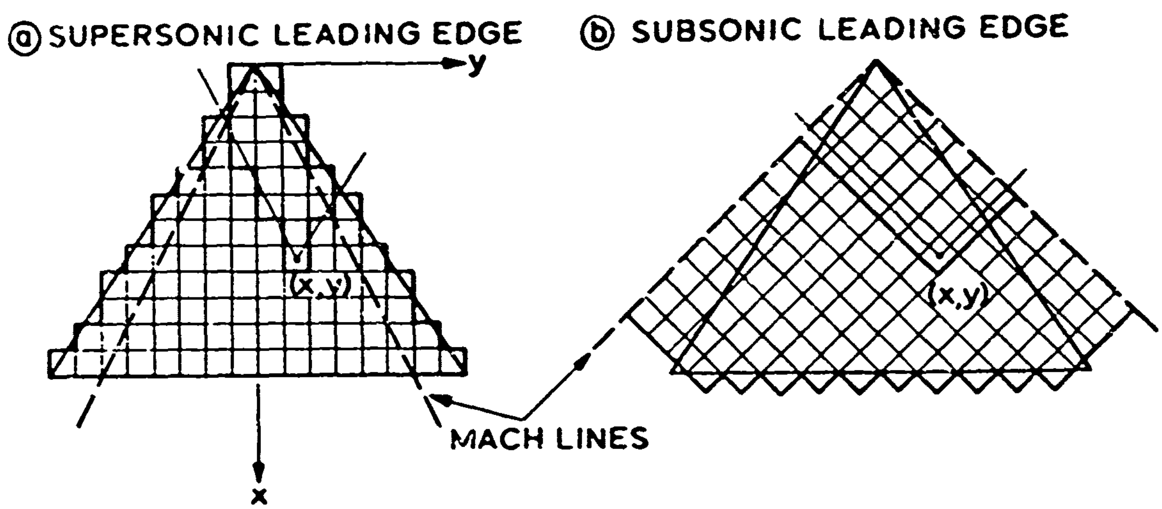


INTEGRAL EQ. : $\dfrac{W\langle x,y\rangle}{U} = \iint_S \Delta C_p \langle x',y'\rangle K_1 \langle x-x'/y-y'\rangle \, dx'dy'$

PRESSURE DISTR. : $\Delta C_p \langle x,y\rangle = \sum_m \sum_n a_{mn} \, h\langle x\rangle g\langle y\rangle$

SOLUTION : COLLOCATION, LEAST SQUARES, VARIATIONAL PRINCIPLE.

FIG.1 KERNEL - FUNCTION METHOD, M ≷ 1



$$\Delta C_p \langle x,y\rangle = \left(\frac{1}{U}\frac{\partial}{\partial x} + ik\right) \sum_n \frac{W}{U}\langle x_n, y_n\rangle \iint_{S_n} K_2 \langle x',y/x',y'\rangle \, dx'dy'$$

FIG.2 BOX METHOD, M > 1

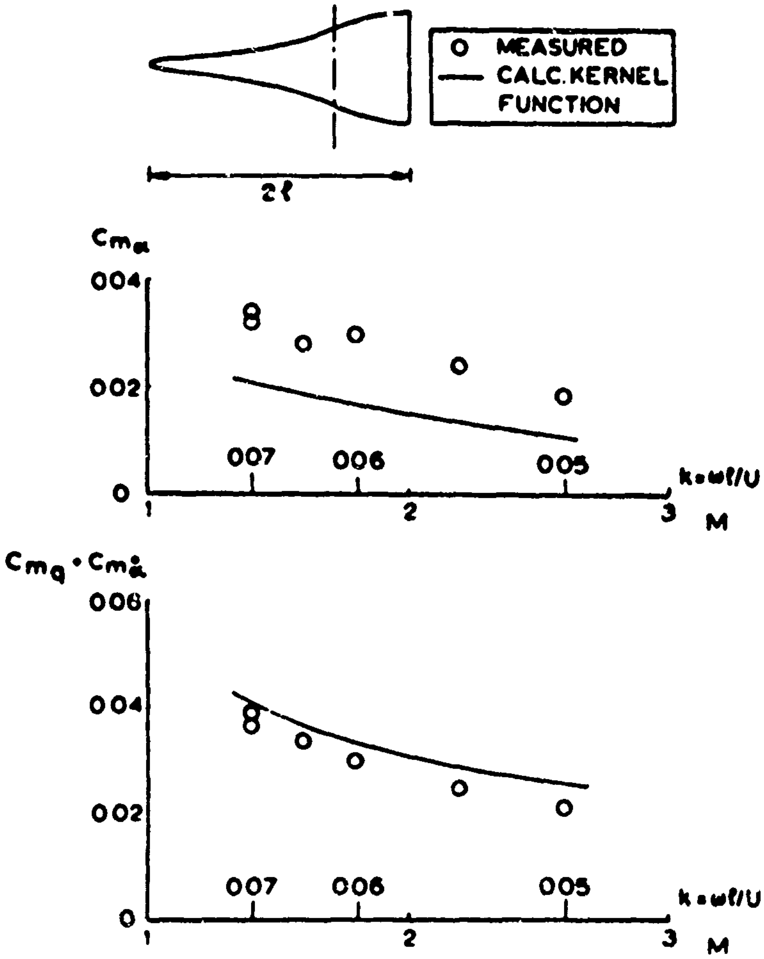


FIG. 3 PITCHING MOMENT ON OGEE WING DUE TO OSCILLATION ABOUT AXIS AT 71% OF ROOT CHORD (REF. 27).

REPONSE DE L AVION SOUPLE EN ACCELERATION

AU DROIT DU PILOTE

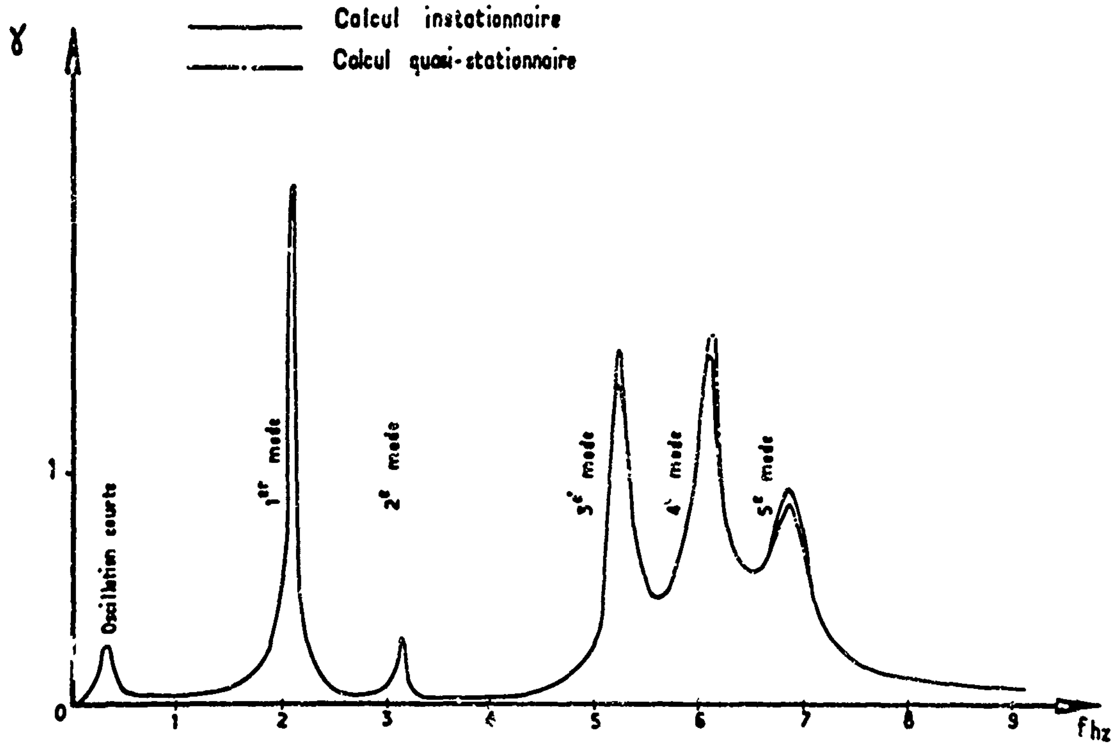
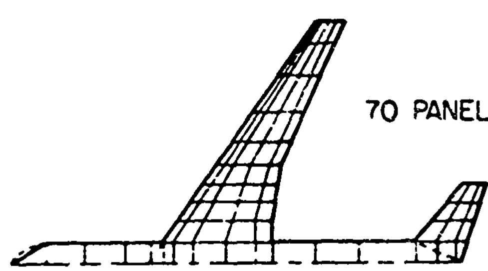
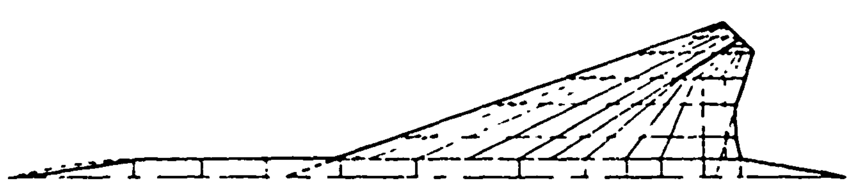


FIG 4



a) BOEING 707 - 320B



b) REPRESENTATIVE SUPERSONIC TRANSPORT (SST)

Fig. 5 Airplane paneling.

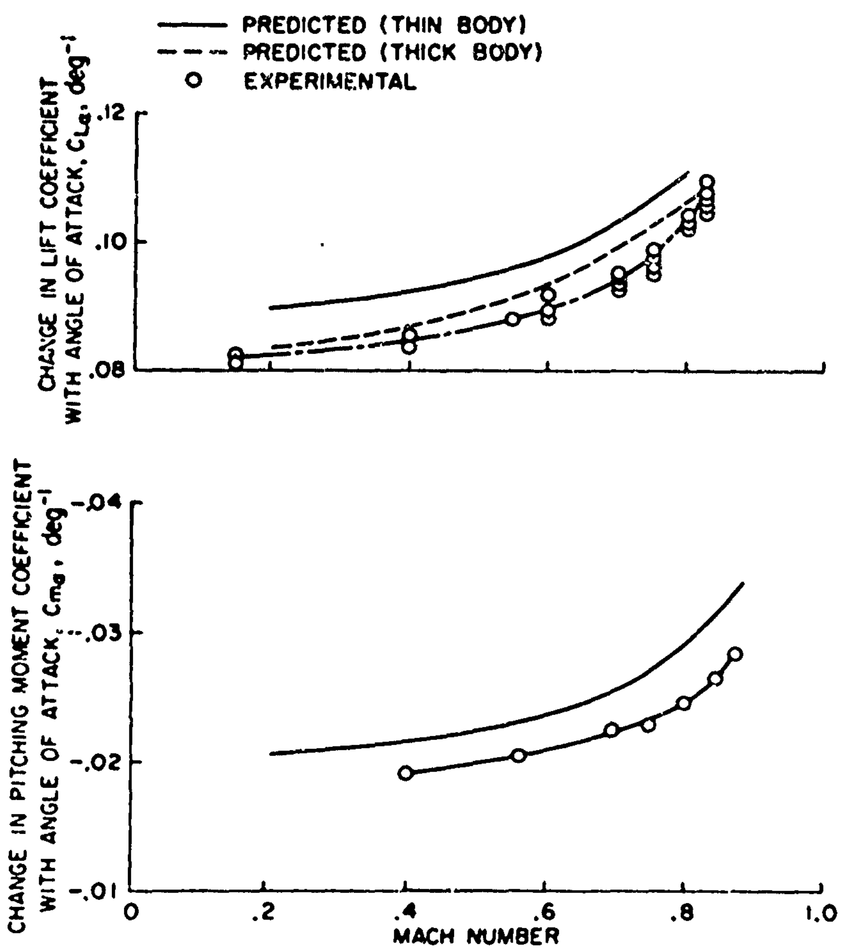


Fig. 6 Calculated and experimental angle-of-attack derivatives, rigid body, Boeing 707-320B.

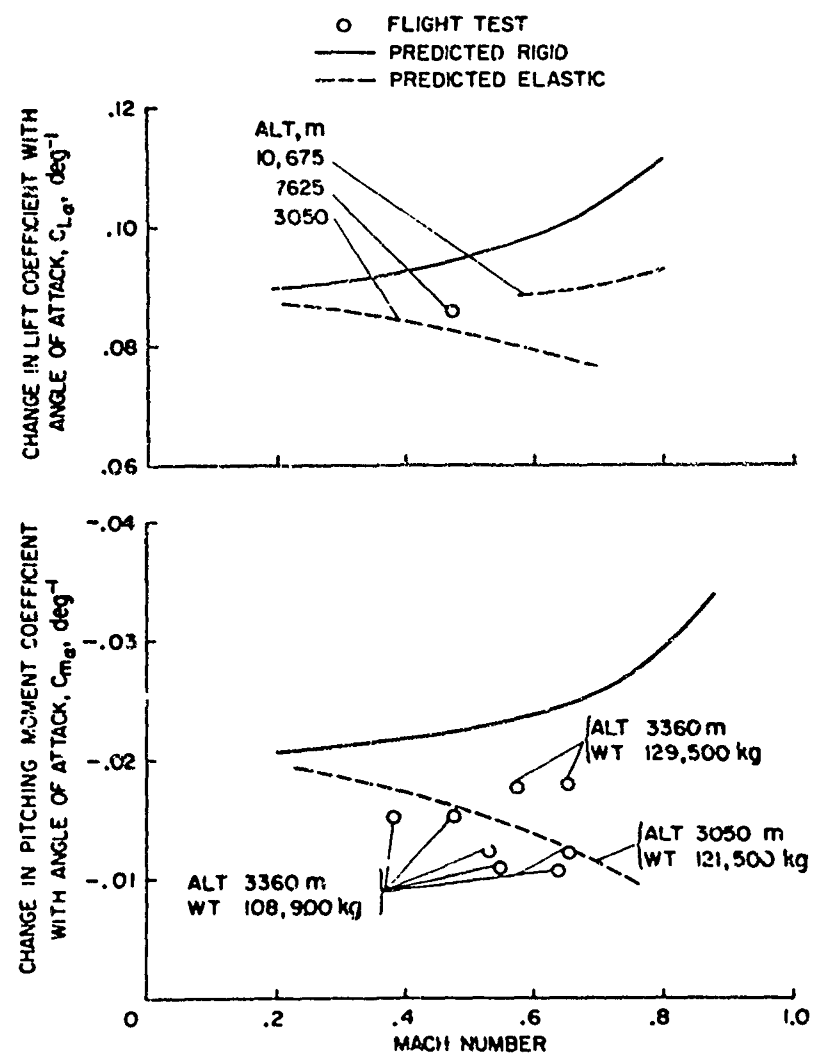


Fig. 7 Calculated and experimental angle-of-attack derivatives, elastic body, Boeing 707-320B.

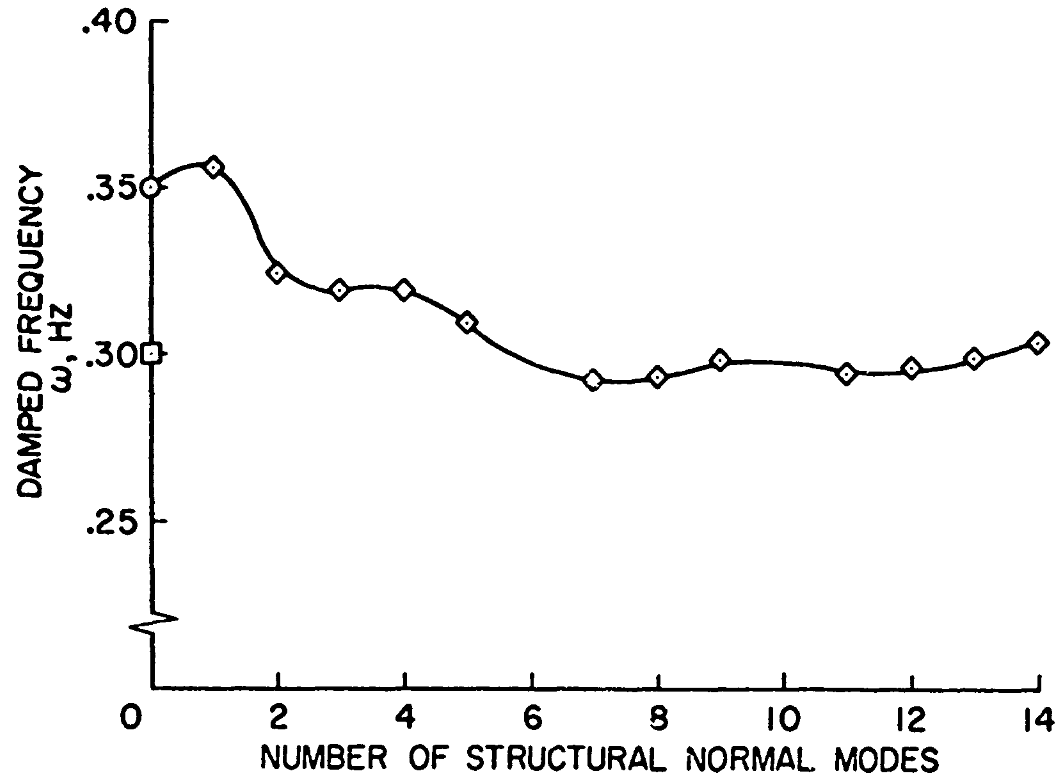


Fig. 8 Effect of number of normal modes on short-period damping and frequency, Boeing 707-320B.

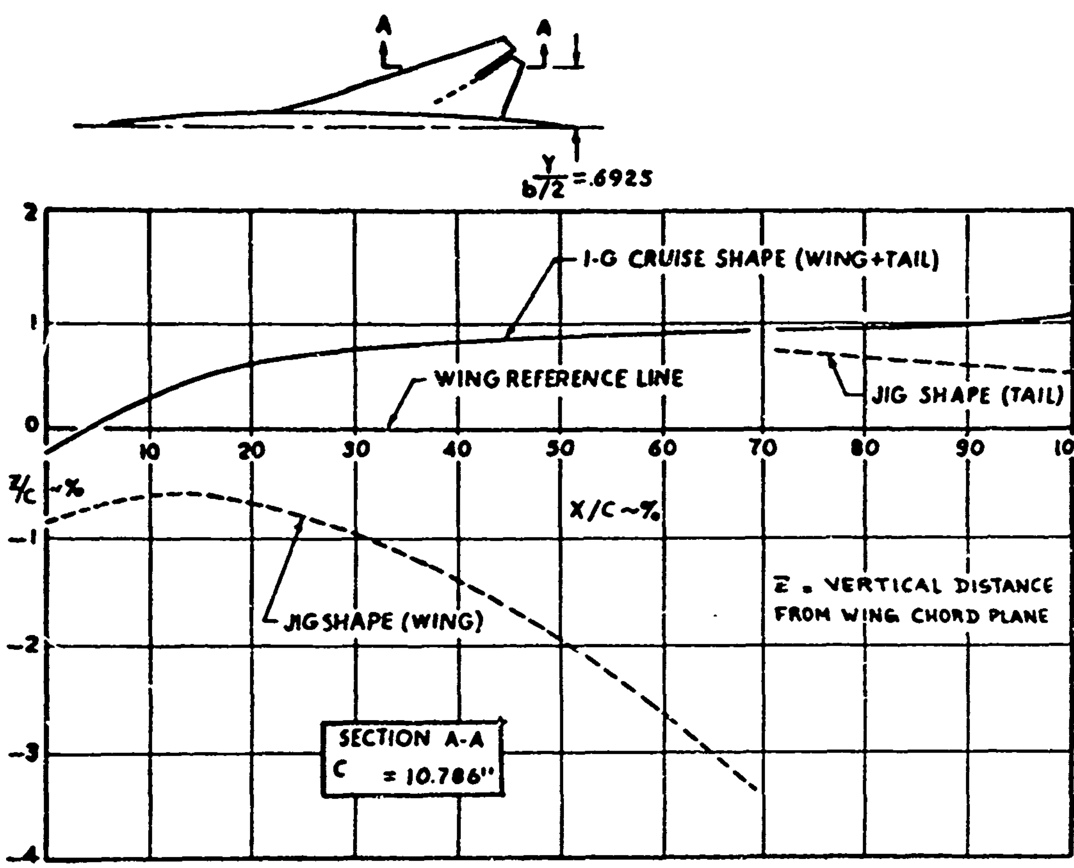


FIGURE 9 EXAMPLE OF JIG SHAPE AND CRUISE SHAPE

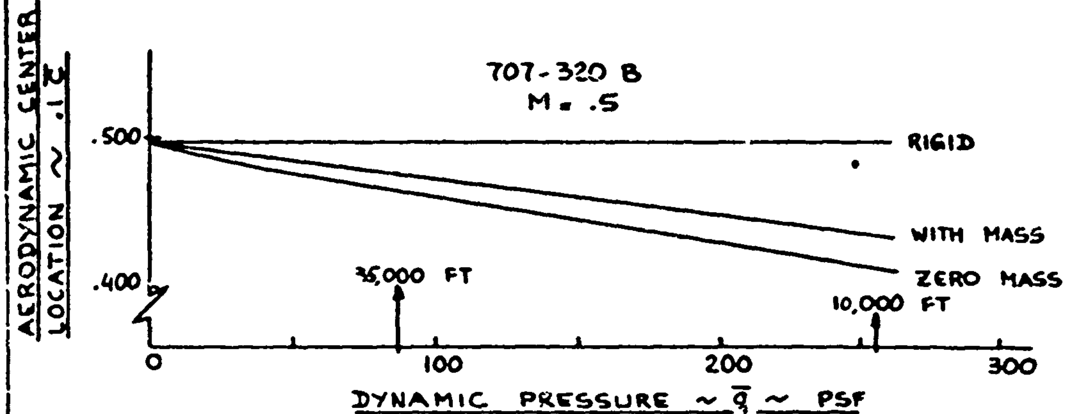


FIGURE 10 - EFFECT OF MASS DISTRIBUTION AND ELASTICITY ON AERODYNAMIC CENTER LOCATION

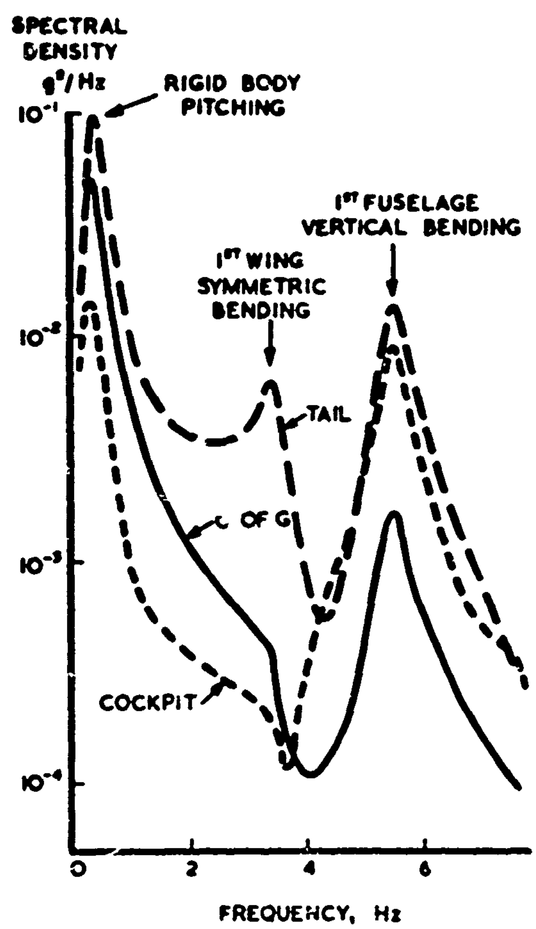


FIG 11

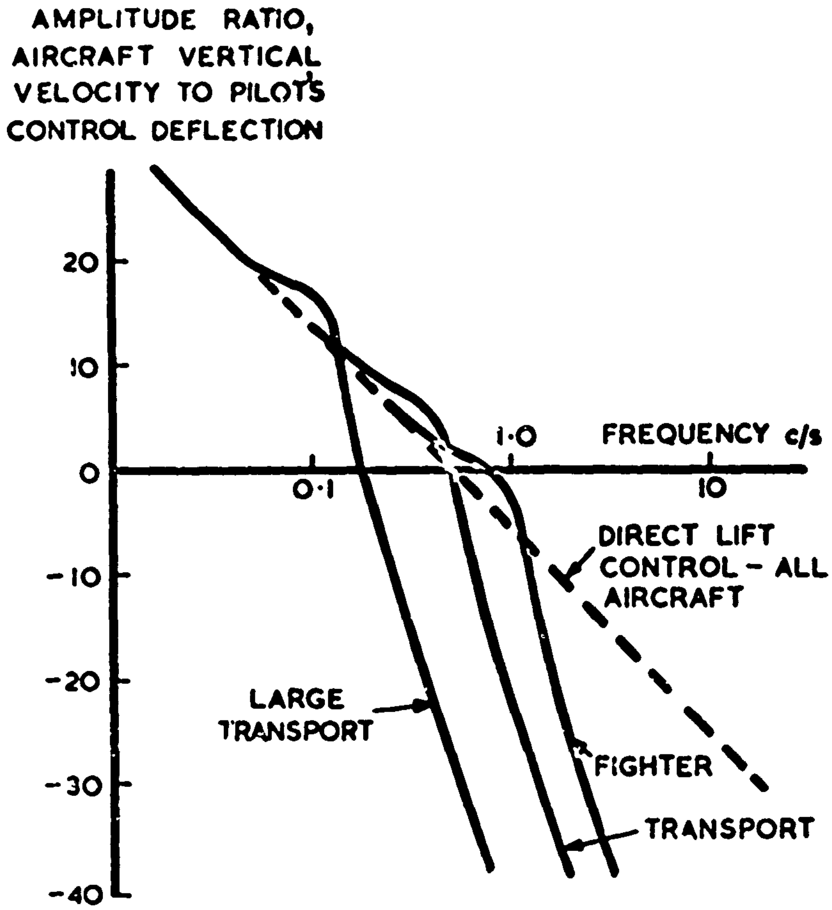


FIG 12

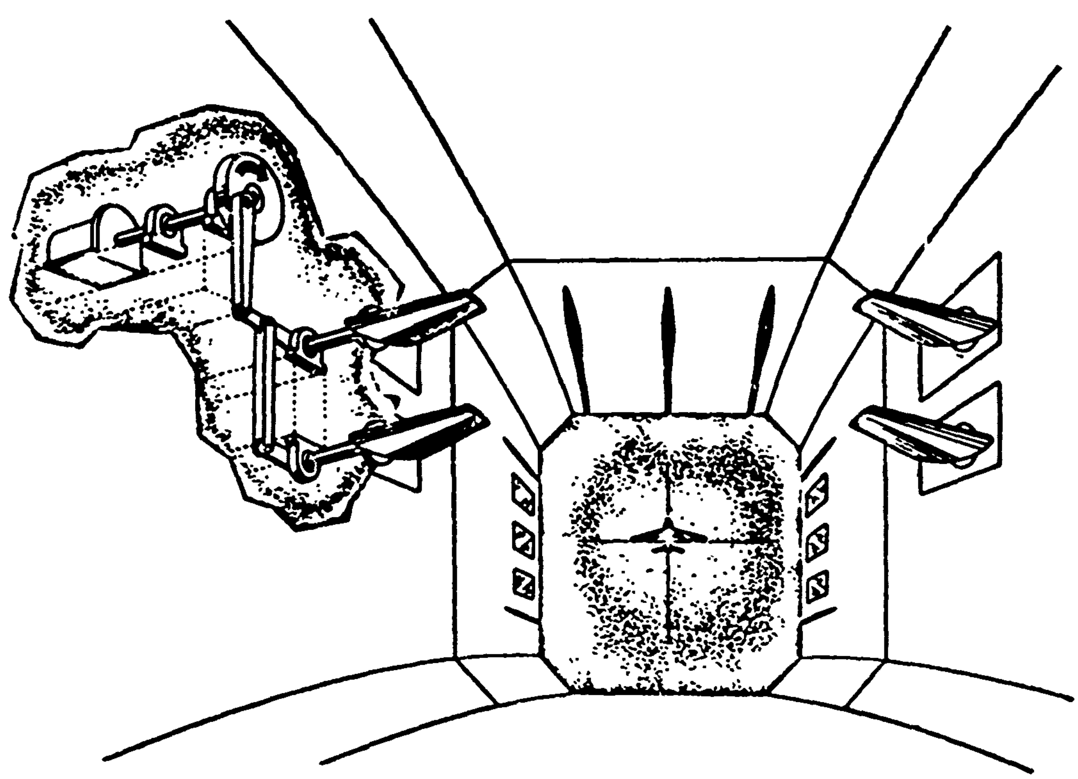


Figure 13 Langley transonic dy. amics tunnel airstream oscillator. View looking downstream toward model with cutaway showing schematic of mechanism.

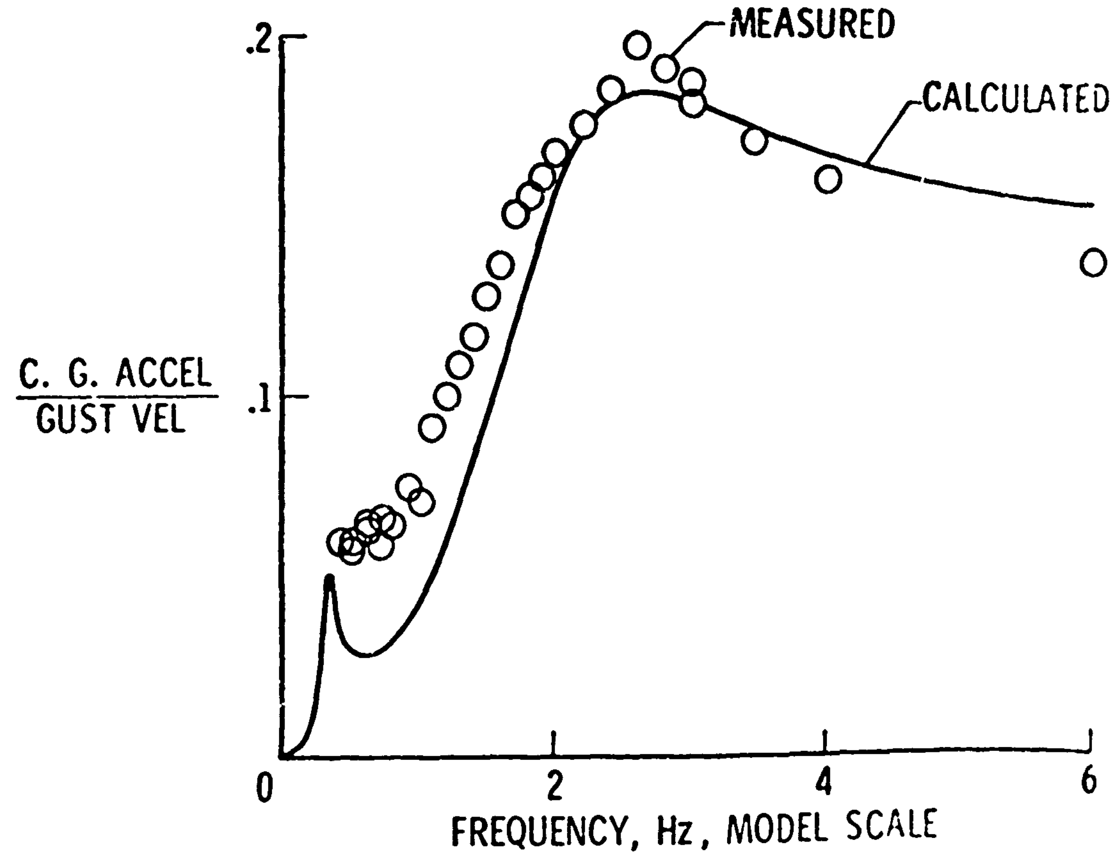


FIG 14

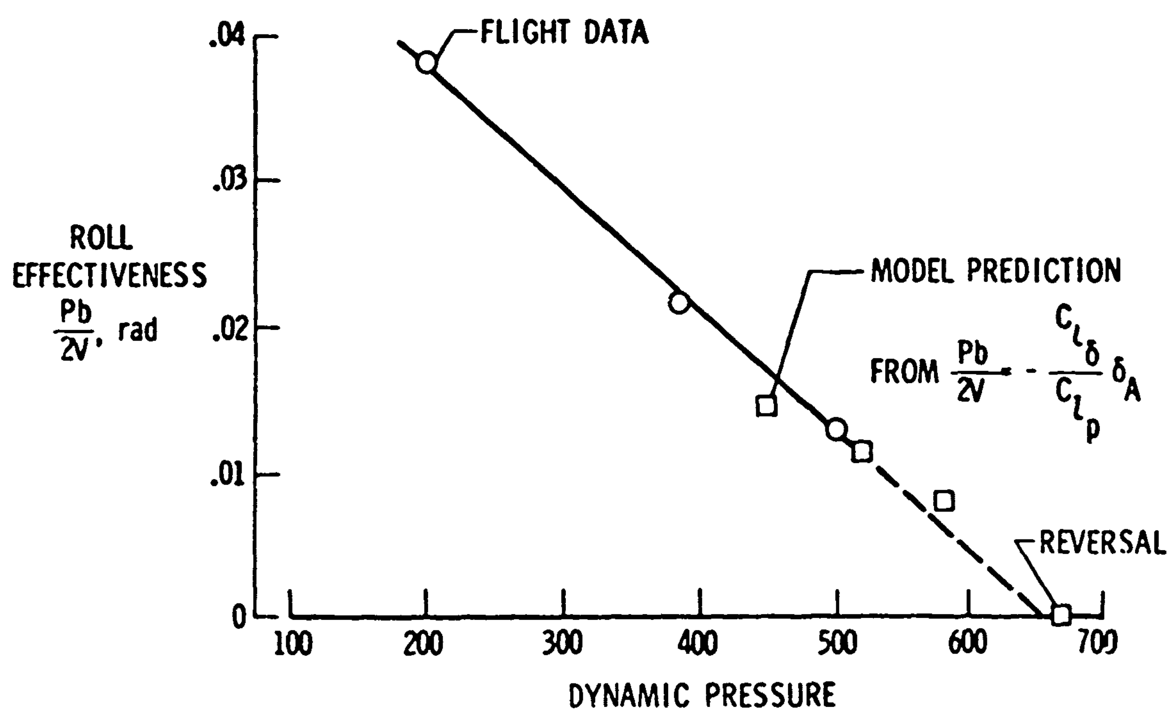


Figure 15 Comparison of flight measurement and model predicted aileron effectiveness.

CAMCO V-LINER

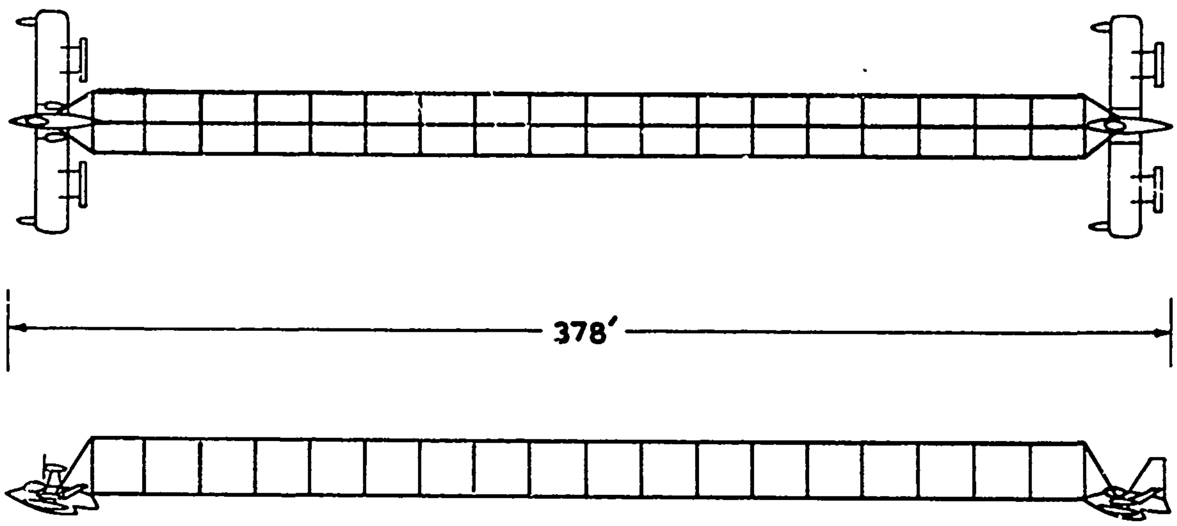


FIG 16

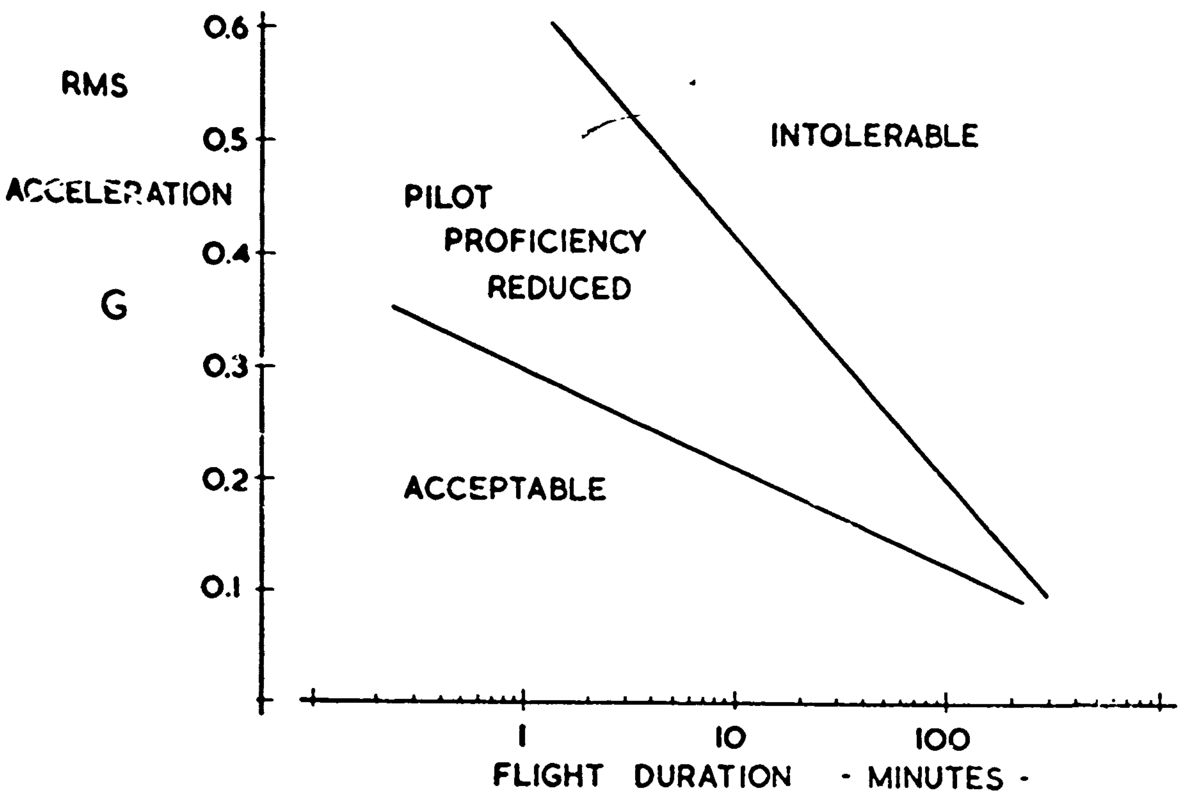


FIG 17

SUMMARY PAPER ON SIMULATION MEETING, SPRING 1970 AT NASA AMES RESEARCH CENTER

A. G. Barnes,
British Aircraft Corporation Limited,
Military Aircraft Division,
Warton Aerodrome,
Preston,
Lancs.,
PR4 1AX.

1. INTRODUCTION

This paper will attempt to distil the contents of the meeting on Simulation at Ames in 1970. The meeting was a lively affair, and it is doubtful if an abridged version can convey the flavour of the meeting - of enthusiasm, of controversy, of commitment to a discipline which is as much an art as a science. The definition of "Simulation" was retricted to cover only those problems which include a man in the loop, and also excluded Space Vehicles. Even so, a formidable gathering of 58 experts, including engineers, pilots and psychologists were present, and represented 8 NATO countries.

The form of the meeting was a slight departure from earlier practice, in that each paper was followed by one or two lead discussion papers, which were intended to stimulate contributions from the floor. The success of the formula is reflected in the fact that more than half of the people present made formal contributions, and that almost everyone at the meeting made comment during the discussion periods.

The Scope of the meeting was large. It ranged from the philosophical - Dr. Bruning had consulted 5 dictionaries and still had not found a definition of "simulation" - to the practical - one mathematically-minded pilot remarked that "a bad six degree of freedom motion system is likely to be twice as bad as a bad three degree of freedom motion system".

The conference consisted of four sessions, as follows.

1. Simulation Objectives
2. Simulator Characteristics
3. Design of Experiments
4. Simulation Results and Analysis

As is usual on these occasions, the topics contained considerable overlap. This, together with the spirited discussion, meant that the later in the programme you appeared, the more likely it was that someone had pre-empted your unique and illuminating contribution. The following paragraphs will try to pick out the points made by the various contributors, both from the podium and from the floor.

2. SESSION 1 OBJECTIVES OF SIMULATION

Dr. G. Bruning, of DFVLR Germany set the scene with "Simulation - an Introduction and Survey". He gave an overall coverage with emphasis on In-Flight Simulation (Variable Stability Aircraft, VSA). He presented a hybrid technique - conditional feedback model control - to apply to VSA. The ensuing discussion revealed that a similar technique is successful at Cornell (TIFS), Nasa Langley (VS helicopter) and NRC (VS helicopter). Northrop had reservations about its success at frequencies around and higher than 1 hertz.

Dr. Bruning then ran briefly through several aspects of flight simulation - motion cues, visual cues, psychological factors, and work-load - with quotations from his extensive bibliography. He concluded that care is needed in choosing the right simulator for a given task, and gave a danger warning. "In all technical areas there is an inherent tendency to develop towards more and more sophisticated and complex systems. In our field, this is not only true for the flight-vehicle itself, but even to a higher degree for the devices built for their simulation. .... The more intricate a facility becomes, the more personnel are needed to run it, and suddenly it starts to live its own individual existence, detached from the original idea behind it." I am sorry to report that no-one was brave enough to stand up and confess to having such a facility!

The next paper, "Objectives of Simulation" by Mr. Barnes of British Aircraft Corporation, continued on this theme. The intention of the paper was to illustrate how in practice, the worthy objectives of simulation can be distorted. Because of the expensive and yet indispensable role that simulation now plays in aircraft design and development, an open examination of the use and mis-use of simulation should be made. Four objects of simulation were defined:

1. To derive statements about the properties of a system which may be read across to the real situation.

2. To provide a framework for the interpretation of experiments.
3. To improve the model.
4. To suggest further experiments.

Examples were then given of the pitfalls that arise in trying to achieve these objectives. They include the use of a simulator for purposes outside its range, the dilemma of conducting impartial experiments in a charged or biased environment, the growth factor (Dr. Bruning's point) and the snowball effect of one experiment leading to another.

In his lead discussion paper, Dr. Gould of NRC, Canada, said that simulation should improve the engineer's detailed understanding of a system. The use of big computers leads to a loss in flexibility. One object should be to account for the environment to which the test results will apply - for example, turbulence or terrain. He was also concerned with the ability to read across to the real situation - does limited motion in a simulator do more harm than good? Tests were needed solely to observe the effects of motion cues.

Mr. Westbrook, of AFFDL, firmly stated that one objective of simulation is to save money. This lead to a spirited discussion on costs. He also puts his faith in results from a single calibrated pilot, than take the mean opinion from several pilots. "It has been my experience", he added, "that test pilots have almost universally been honest and willing to take a stand, something that cannot always be said for engineers." On the subject of costs, M. Pinet, of S.N.I.A.S., gave figures relating to Concorde. The total expenditure on simulation to date has been less than 85% of the cost of flying hours saved. Mr. Aitken of NASA reminded us that if simulation is the only way to solve certain problems, then cost-effectiveness has little meaning.

3. SESSION 2 SIMULATOR CHARACTERISTICS

The opening paper was "Flight Simulator Mathematical Models in Aircraft Design", by Mr. Alan H. Lee, of Boeing. He gave a comprehensive account of the requirements for mathematical models, covering equations, method of solution, aerodynamic representation, flight control system representation, propulsion system and turbulence. Comments on motion system requirements and training simulator requirements were also made. A study of the complete paper, in reference 1, is recommended.

Mr. Vermeulen, of N.A.L., Netherlands, reminded us of the importance of choice of axis system and axis transformation method in saving computing capacity; also that integration on a digital computer is not plain sailing. In the discussion, Mr. Hass, of AFFDL, said that we are prone to make errors on big digital models. Mr. Gallagher, of Northrop, gave a plug for old fashioned analog computers, particularly for high order models.

"Motion, Visual, and Aural Cues in Piloted Flight Simulation" was the title of the paper by Mr. Staples, R.A.E., Bedford. Again, this is a paper which must be read to be fully appreciated. The author takes a long and thoughtful look at the whole simulation scenario, without trying to reach conclusions. He wants pilots ("highly adaptable animals") with "the power to suspend disbelief". A discussion is made of various cues available to the pilot. Motion is considered axis by axis. The point is made that the effects of inter-axis coupling make the mechanisation of a motion system a complex procedure. Practical difficulties of travel and frequency response also limit the success of a motion system in simulating the sensations of flight. The situation with respect to visual cues is no less complex. 15 factors are listed which influence perception of the visual scene.

Two rather ominous suggestions come out of it all, i) that sub-threshold motions may influence pilot behaviour, and ii) that unperceived distortions in visual displays may influence pilot behaviour.

In leading the discussion M. Deque, of S.N.I.A.S., said that p.i.o.'s are difficult to reproduce with a limited heave motion system. Prof. Gerlach, of Delft University, Netherlands, acknowledged the work of Young (M.I.T.) and Peters (S.T.I.) in relating man's physiological make-up into engineering terms (lags, filters), and saw great possibilities for the analysis of motion requirements.

Mr. Bray (NASA) suggested that motion cues are unnecessary for problems whose objectives are not related to the short period dynamics of the control system - for example problems of navigation, or operational procedures. They are needed, however, for circumstances where the control characteristics are marginally acceptable. He then detailed NASA Ames experience. In roll, they find an attenuation to 25% of true value is used for landing simulation. His comments on sway motion were also significant, since Ames now operate the FSAA (Flight Simulator for Advanced Aircraft) which has ± 50 feet of sway travel. "In our experience", he said, "no other single motion cue has contributed as much to the sense of realism in a simulation as has sway motion." Cold comfort to those of us who can only get translation from an interpreter.

The next paper, by M. Pinet, was entitled "Cockpit Environment". His comments, based on Concorde flight experience compared to a v.s. Mirage simulation, the Toulouse simulator, and the Bristol simulator were invaluable. He found that the aircraft, visor down, was less susceptible to overcontrol in roll, probably because of peripheral vision from the side windows. The visual system at Toulouse gives a worrying impression due to apparent yaw motions at high angle of attack. Landing approaches are "calmer" (smaller bank angles) in flight than in the simulator. Poor resolution and lack of perspective in the TV display may account for hard landings in the simulator. There is a need for good representation in a simulator of the cockpit layout and feel system - no "smell" of artificial. With respect to motion cues, neither the Mirage nor the Toulouse simulator feels like the aircraft. The simulator is adjusted to give "a minimum of false perceptions", and "an impression of going in the right direction".

Dr. Strother, Bell Helicopter Co., argued that M. Pinet was wrong in his insistence on no smell of artificiality ("face validity" she called it). Transfer of training has been shown by Muckler to be insensitive to pilot acceptance of the simulator. Mr. Hendels, H.S.A., asked what value the Bristol 221 aircraft had been to the Concorde programme. Mr. Lean, R.A.E., replied that its most valuable contribution was to validate ground based simulations, and thus give credibility to Concorde simulations.

Several speakers referred to the fact that successful landings have been made in aircraft with the pilot using a TV monitor as visual reference - thus indicating that the trouble in simulators is not confined to the display alone.

4. SESSION 3 DESIGN OF EXPERIMENTS

Mr. McGregor of NRC Canada gave the first paper: "Some Factors Influencing the Choice of a Simulator". He expressed a liking for "lots of motion with a real world visual display". He emphasised the link between visual and motion cues, and the fact that the stabilisation mechanism of the eye provides small amplitude rate cues. He suggested that motion is essential for simulator results to be applied directly to manoeuvring flight, even though qualitative data may be obtained from a fixed base simulator. Motion cues should be provided in any study of stability augmentation or engine failure, handling in turbulence, and cases with marginal stability. He concluded with three areas where further research is required. These are (i) the best use of "wash out" in ground-based simulators, (ii) methods of measurement to support pilot subjective assessment, and (iii) visual resolution requirements.

The lead discusser, Mr. Gallagher, discussed Northrop experience and agreed with these conclusions. Predictably, however, he argued that the variable stability aircraft does not always provide the best way to conduct an investigation into handling qualities for some fighter missions. The inability of the v.s. aircraft to match flight condition, cockpit layout, and certain failure states may lead to the use of a ground-based simulator with motion. This comment echoed an earlier remark by M. Pinet, that the Toulouse simulator gave a better representation of the Concorde than the v.s. Mirage.

Mr. Breuhaus remarked that v.s. aircraft and ground based simulators are complementary pieces of equipment. M. Deque quietly commented that simulators sometimes create their own problems - for example, the limited cues in ground based simulators can give the pilot false impressions.

The second paper was "The Selection of Tasks and Subjects of Flight Simulation Experiments", by Mr. Breuhaus and Mr. Harper of Cornell. By definition, the task in a simulator differs from the real one, and so the pilot's psychological situation is also different - a different type of stress. An extrapolation of results to the real situation is needed and is best done by the pilot. On the topic of rating, they believe that inter-subject and intra-subject rating should be about the same - if not, perhaps inadequate briefing is indicated. The selection of subjects is difficult. The use of a small sample from the pilot population has advantages, because data manipulation is easier, and so is control of the experiment. How many subjects should be used? One can produce useful answers for many applications, and three has proved to be a reasonable compromise. The personal qualities of the subject are then listed - motivation, objectivity, experience, availability, confidence and communication. Opening the discussion, Sig. Filisetti, of Fiat, liked to see flight tests where possible concurrent with simulator tests, for validation purposes. Ensuring that a pilot is completely familiar with the simulator is also important. Mr. Brown of R.A.E., U.K. said that in his experience, variations of results between pilots is always large, but that individual pilots are consistent in terms of dynamic performance and decision making. For some problems the elimination of learning effects in the simulator is not desirable - learning occurs in the air also and can seriously distort tests on system failures.

In the discussion, Prof. Doetsch, DFVLR, Germany wondered if the pilots we use in simulators are so skilful as to be unrepresentative; Dr. Beyer, DFVLR, Germany asked if Mr. Breuhaus tested his pilots for intraverted or extraverted tendencies. Mercifully, Mr. Breuhaus said no.

5. SESSION 4 SIMULATOR RESULTS AND ANALYSIS

"Engineering Analysis", by M. Montfort, of C.E.V., France opened the session. The analysis of physical systems is not too difficult, he said. The criterion may differ from case to case - performance or pilot comment on the stability. Much more difficult is the analysis of pilot behaviour, for example, workload. Engineers like to quantify the results of experiments, and to do so even with respect to psychological reactions is desirable. Performance measures are insufficient - simulator tests of ILS approaches showed the same performance as the aircraft stability was reduced, until the pilot lost control completely.

To measure workload, all pilot inputs and outputs must be measured. In particular, the pilot's scan pattern may correlate with workload.

Mr. Madill, of D.H., Canada, said that they had found the application of statistical and response surface techniques to both pilot ratings and pilot comments to be rewarding.

Mr. Ashkenas of Systems Technology Inc., felt that closed loop analysis is the lead to an understanding of pilot behaviour. Scan pattern measurement is only partially successful to measure workload, because the eye derives peripheral information, and so eye fixation or movement is not a unique measure of input. Experiments confirm the complexity of the relationship of eye position to workload. A better measure of workload is the degree of adaptation in an adaptive secondary task.

The last word came from the pilots. "Pilot Assessment Aspects of Simulation" by Mr. G. Cooper and Mr. Drinkwater of NASA Ames started by saying that the pilot's primary concern is with the fidelity of the simulator, in other words, the degree of extrapolation that is called for. And yet the usefulness of a simulator is not necessarily related to its sophistication.

The pilot must participate in programme definition. The role of the pilot was then discussed. If the pilot is treated as a subject (or performer), then workload measurement methods are needed. If the pilot is regarded as an assessor, then the value of his extrapolation to the real situation is obtained. Perhaps the best place to measure workload is in the training simulator, because of its high fidelity. This is a new role for such devices.

The first lead discusser was M. Pinet. He emphasised the need for pilots and engineers to work together for assessments. He then discussed a new rating scale, which had been formulated because he found the Cooper-Harper scale difficult to apply. The new scale takes into account three factors - skill, attention, safety, and the pilot is asked to give to each of these factors a numerical rating of 1, 2 or 3.

Lt. Wheal of R.A.E., gave the second lead discussion paper. He was unhappy about being asked to extrapolate to the flight situation, and questioned the value of such judgements. On the other hand, he had found that if a handling problem occurs in flight as predicted by a simulator, then the chances are that the solution found on the simulator is successful in flight also. On the subject of workload, he reminded us that R/T transmissions make up a significant proportion of the total workload. He had flown the FSAA at Ames, and concluded that with such a good motion system, the visual display is the weak link. He wondered, on the basis of a pilot's ability to perform deck landings on a black night, whether simple, accurate contact analogue displays should receive attention.

6. CLOSING DISCUSSION

The last session was devoted to a discussion of the recommendations for further research which had emerged from the meeting. They related to the simulation and influence of visual and motion cues, pilot workload, and the modelling of turbulence.

Finally, M. Lecoste summarised the important conclusions which emerged from the meeting. They may be paraphrased as

1. "Sit down and think" before you simulate. Then cross-check with theory and other simulations.
2. The two most delicate problems of simulation are the visual and motion cues. Much remains to be done both to improve our methods of simulating these cues, and to utilise these methods to best advantage.
3. The pilot is the final judge, and we must study the pilot himself in the physiological, psychological, and servo-mechanism sense.
4. The overlap between Research Simulators and Training Simulators is becoming more pronounced.

7. REFERENCES

1. AGARD-CP-79-70 "Conference Proceedings No. 79 - Simulation" January 1971

OPEN DISCUSSION

H.H.B.M.Thomas, UK: On the question of supplying motion cues on simulators it seems to me that there has been too ready acceptance of the need to move towards realism in providing actual motion rather than trying to find out what features of motion are essential to the pilot. Would the author care to comment?

A.G.Barnes, UK: This question raises two problems: first, that the true motions are difficult to produce; and second, that it is difficult to isolate the features that the pilot uses. Most of us, because of the first difficulty, accept severe limits in authority and degrees of freedom in our motion systems. In consequence, we must use ad hoc methods such as wash out or gain reduction before pilots even accept our simulators. As M. Pinet reported, the simulator is adjusted to give "a minimum of false perceptions." However, at NASA Ames Research Center the FSAA allows greater realism in the representation of motion, and because they can start from that point and reduce the fidelity, they can begin to isolate those features which are essential to the pilot. You may find reassurance in this work and complementary work on physiological models of the pilot.

HANDLING QUALITIES CRITERIA AND REQUIREMENTS

by

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SUMMARY

This paper was prepared to summarize the AGARD Flight Mechanics Panel Specialists Meeting on "Handling Qualities Criteria" held in Ottawa, Canada, 28 September to 1 October 1971, and to discuss the current status, problems, activities and issues in the development and application of handling qualities criteria as reflected by this meeting.

An AGARD report containing the papers, discussions and questions of this meeting has been compiled and edited, and will soon be available for those who wish to pursue in more detail the activities and issues covered therein. With only a few exceptions, each paper presented at the meeting is summarized by the author's abstract and summary, plus a brief lead discussor's paper. Thus, there is no need for this paper to cover the same ground by presenting a detailed paper-by-paper summary.

Our approach will be to provide an overview of the meeting and each session, discuss the state of the art and specific items and activities of interest, and briefly review problems and issues. A discussion of basic definitions and the historical evolution of flying qualities precedes the review of current status, problems and techniques used in the development of flying qualities. The concepts of TSS-5 and MIL-F-8785B receive emphasis in recognition of their considerable impact on thinking and the frequent discussions devoted to them throughout the meeting. Special problems and research activities are summarized in much the same order as in the meeting. The paper is concluded with an overview of current problems, issues and future actions needed, as highlighted by the round-table discussion of "Where do we go from here?" and supplemented by screening of the discussions within each session.

MEETING OVERVIEW

SCOPE

The few words on Figure 1 provide a quick perspective of the size and activity of the meeting.

SCOPE

- 3 1/2 DAYS
- 6 TECHNICAL SESSIONS + "ROUND TABLE - WHERE DO WE GO FROM HERE"
- 22 PAPERS
- 21 DISCUSSIONS
- 120 RECORDED COMMENTS
- TOUR OF NAE FACILITIES
- 103 ATTENDEES (INCL 2 NATO STAFF)
  - 7 NATIONS

FIGURE 1

The six technical sessions plus the panel discussion are listed below:

Session I	Status of Flying Qualities Requirements for Conventional Aircraft
Session II	Status of Flying Qualities Criteria for V/STOL Aircraft
Session III	Establishment of Criteria
Session IV	Special Problems and Interfaces
Session V	Man-Machine Research
Session VI	Additional Research
Panel Discussion	Where Do We Go From Here?

INTERRELATIONSHIP AND LIMITATION OF SESSIONS

Figure 2 depicts how these sessions tended to overlap because of the very nature of the problems in developing handling qualities criteria. For example, there are some problems which are unique to V/STOL handling qualities; however, there are also some basic V/STOL handling qualities problems which are common to conventional aircraft; e.g., handling of display effects, development of good turbulence models, effect of control systems, impact of the pilot, etc. As a result of such unavoidable interactions of coverage in the sessions, it was not uncommon to have discussions in one session which were applicable to one or more other sessions.

SESSION RELATIONSHIPS

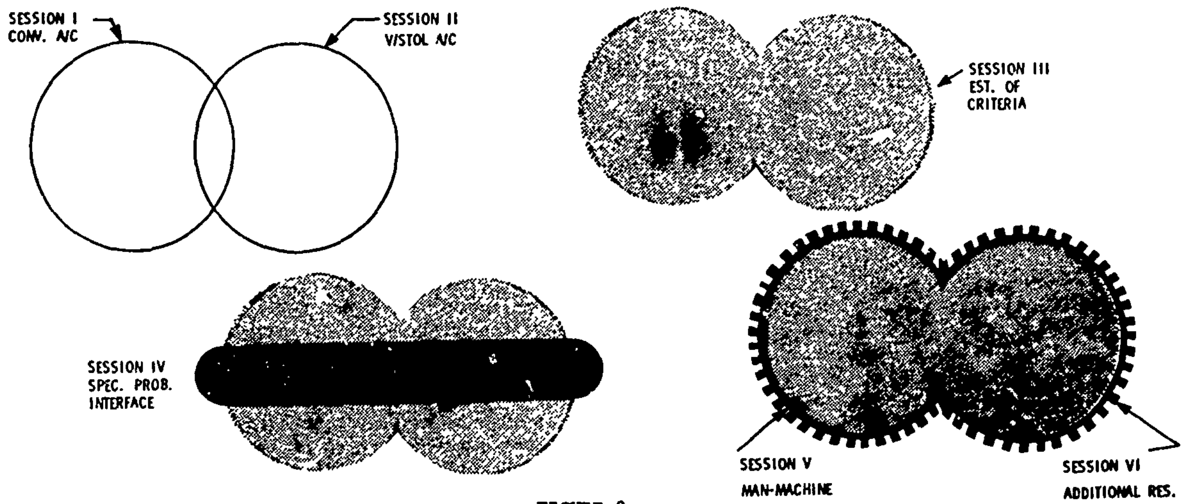


FIGURE 2

While coverage of the subjects during the sessions was as complete as practical in the time available, and served its purpose of providing a forum for very lively discussions, it is clear that much more remains for future review and discussion. Conventional aircraft handling requirements and criteria review, for example, covered the United States military specification, the French E.S.A.U. (Etude de la Sécurite' des Aéronefs en Utilisation) philosophy of the Anglo-French SST specification, U.S. civil aircraft philosophy, and very briefly the British AVP970 criteria. Discussions in equivalent detail and comparisons of similar information from the other NATO countries, validation of specification and criteria by aircraft test results as discussed in the V/STOL session, and further discussions of this type would also be enlightening. Validation of the U.S. military specification by analysis of F-4 and F-5 test results was noted, but time did not permit inclusion.

The effect of flying qualities on accomplishment of such military functions as air-to-air refueling, weapon delivery and fighter combat, were referenced several times, but not included. Impact of major new control system developments and changes underway in pilot displays,were also reserved for discussion at future meetings.

INDIVIDUAL SESSION DISCUSSIONS

Now, let's briefly look at the individual sessions.

SESSION I, STATUS OF FLYING QUALITIES REQUIREMENTS FOR CONVENTIONAL AIRCRAFT

Papers 1&2 - Comparison of French and US Flying Qualities Requirements

Authors: J-C. Wanner and J.W. Carlson
Discussor: A.G. Barnes

Paper #3 - The Nature and Use of the Rules for Judging the Acceptability of the Flying Qualities of Fixed Wing Aircraft

Author: S.J. Andrews
Discussor: H. Eisenlohr

Paper #4 - FAA Flying Qualities Requirements

Authors: R. S. Sliff and R.F. LeSuer
Discussor: J. Renaudie

The first session addressed the status of flying qualities requirements for conventional aircraft. It began with Messrs. Wanner and Carlson finding general agreement in their comparison with the French E.S.A.U. philosophy and approach of Anglo-French TSS-5 and U.S. MIL-F-8785B flying qualities requirements with regard to the SST and military aircraft. A.G. Barnes, lead discussor for these papers summed it up well by saying "let me congratulate Mr. Wanner and Mr. Carlson on the skillful way they led us through the maze of this complex subject," a comment that we second.

This session included and led to numerous later discussions on the pros and cons of having criteria versus requirements. It is clear that the viewpoint was quite frequently dependent on the relationships of the user to the aircraft development. Mr. Andrews briefly discussed several aspects of the British Military Specification AVP970 and his views on flight test acceptability rules, and made a plea for simple criteria and avoidance of incorporating handling quality criteria into aircraft specifications. He pointed to a need to accumulate data on specialized roles and concentrate flight testing on mission effectiveness and operational reliability. He finished with an excellent film of the Harrier operating in the Swiss mountains. Sliff and LeSuer's paper presented a discussion of the philosophy of FAR (Federal Air Regulations), the task of keeping them up to date, and some of the current and anticipated problems in the determination of compliance of civil aircraft with the existing airworthiness rules.

SESSION II, STATUS OF FLYING QUALITIES CRITERIA FOR V/STOL AIRCRAFT

Paper #5 - Revisions to V/STOL Handling Qualities Criteria of AGARD Report No. 408

Authors: S.B. Anderson and L.G. Schroers
Open Discussion

Paper #6 - US Military V/STOL Requirements

Authors: C.B. Westbrook and C.R. Chalk
Discussor: D.G. Gould

Paper #7 - Application of V/STOL Handling Qualities Criteria to the CL-84 Aircraft

Author: O. Michaelsen
Discussor: A. Filisetti

Paper #8 - V/STOL Handling Qualities Criteria Compared with Flight Test Results of the V/STOL Supersonic Fighter VJ 101C and the V/STOL Transport Aircraft DO-31E

Authors: G.K. Kissel and H. Wünnenberg
Discussor: J. Teplitz

The second session was a discussion of the status of flying qualities criteria for V/STOL aircraft. Mr. S.B. Anderson started the session with a discussion of AGARD 408A V/STOL handling qualities criteria (Reference 6). He was followed by Messrs. Westbrook and Chalk's discussion of the U.S. military V/STOL requirements contained in MIL-F-83300. The need for operational data was stressed. The next two papers (References 7 and 8) compared the CL-84, VJ 101C and DO-31E aircraft with existing V/STOL handling qualities criteria. In addition to the papers presented, Wünnenberg included an informative film showing the pilot's activity in the DO-31E during powered lift flight.

SESSION III, ESTABLISHMENT OF CRITERIA

Paper #9 - Criteria Trends Obtained from Analysis of Current Aircraft

Author: C.E. Adolph
Open Discussion

Paper #10 - Role of Simulation and Analysis in Criteria

Author: J.T. Gallagher
Discussor: P.L. Bisgood

Paper #11 - Criteria for Supersonic Transport Certification

Author: W. Kehrer
Open Discussion

Paper #12 - The Role of Pilot Opinion Ratings

Author: R.P. Harper, Jr.
Discussor: J-C. Wanner

Session III discussed the techniques involved in the establishment of criteria. C.E. Adolph's paper (Reference 9) was concerned with the role and limitations of existing criteria in the flight test evaluation of aircraft. The need for correlating criteria with mission tasks was emphasized and additional needs were pointed out. The role of simulation and analysis as a foundation for developing handling qualities requirements was addressed by J.T. Gallagher (Reference 10). This paper illustrated, through the use of examples, the capabilities and limitations of both ground-based and inflight simulators, analysis and flight testing. Kehrer discussed the influence of handling qualities criteria on aircraft design, especially as they applied to the Boeing SST configuration. The role of MIL-F-8785B in this development was discussed as well as the reliance on past Boeing experience in design and certification of large commercial jet transport aircraft. Harper's paper highlighted problems encountered in obtaining pilot ratings, an important aspect of criteria development, and emphasized the need to supplement ratings with correlated comments (Reference 12).

SESSION IV, SPECIAL PROBLEMS AND INTERFACES

Paper #13 - Criteria for Stall and Post Stall Gyrations

Author: G.J. Hancock
Discussor: W. Bihrle

Paper #14 - Turbulence Models for Handling Qualities During Take-Off and Landing

Author: J.G. Jones
Discussor: J-C. Wanner

Paper #15 - Flying Qualities Interaction with Elastic Airframes

Author: T.H. Wykes
Discussor: H.A. Mooij

Paper #16 - Flight Control System Interface

Author: R. Deque
Discussor: W. Sobotta

The first paper (Reference 13) discussed problems involved with the interpretation of B.C.A.R. handling requirements for commercial aircraft during approaches to and excursions beyond limits related to either stall, minimum flight speed, or high angle of attack characteristics. Unique aspects of different aircraft types (e.g., slender wing, V/STOL, and STOL) were addressed and special attention was given to dynamic stalls and to adequate stall warning, either natural or artificial.

The next topic was turbulence modeling development (Reference 14) by J.C. Jones. One of the major problems is developing a model which is a suitable representation of the properties of atmospheric turbulence. This paper investigated these properties with emphasis on the aspects relevant to an aircraft on a landing approach or during take-off.

The Wykes paper (Reference 15) of this session approached the problem of ride control with a flexible airplane and the interaction of handling qualities with elastic airframes. As noted by Mr. Wykes: "It is possible that future vehicles will have increasing difficulty in demonstrating satisfactory compliance with handling qualities criteria during flight testing unless increased attention and time are permitted to be given to flexibility effects analyses during preliminary and early development."

Reference 16 by R. Deque was based on Concorde experience and examined the close interdependency which exists between handling qualities and the flight control systems. The interdependency is quite influential and cannot nor should not be separated when establishing criteria or developing an aircraft.

SESSION V, MAN-MACHINE RESEARCH

Paper #17 - Parameters Affecting Lateral-Directional Handling Qualities at Low Speeds

Author: K-H. Doetsch, Jr.
Discussor: R.J. Woodcock

Paper #18 - Pilot Vehicle Analysis

Author: R.J.A.W. Hosman
Discussor: I.L. Ashkenas

Paper #19 - Pilot Workload

Authors: R.K. Bernotat and J-C. Wanner
Discussor: Same

Paper #20 - Theoretical Pilot Rating Predictors

Author: R.O. Anderson
Discussor: D.M. McGregor

The fifth session was on Man-Machine Research. The session opening paper by K-H. Doetsch, Jr. (Reference 17) discussed the added significance of the side force equation in establishing the lateral-directional oscillatory mode.

Two of the papers, References 18 and 20, presented departures from the more traditional approach to specifying handling qualities, both using human response theory.

An impromptu paper by Bernotat and Wanner on Pilot Workload presented some of the considerations and difficulties encountered in measuring pilot workload. The main problem is pinpointed by the authors in their closing remarks; i.e., ". . .there is up to now no inflight-method for continuous precise measurement of mental load, which could help us to adapt the machine to the human pilot."

SESSION VI, ADDITIONAL RESEARCH

Paper #21 - Recent NASA Handling Qualities Research

Author: R.J. Wasicko
Discussor: D. Covelli

Paper #22 - Rec - U.S. Navy Flying Qualities Research

Author: R.F. Siewert
Discussor: D. Lean and P.L. Bisgood

The sixth session represents the discussions of recent NASA (Reference 21) and Navy (Reference 22) research programs. The importance of keeping information regarding research programs available to other agencies cannot be underestimated. Wasicko showed an interesting film of a number of research aircraft used by NASA to acquire handling qualities data. His paper covers a wide range of research plus current NASA activities oriented to solve problems for many type aircraft, ranging from general aviation types, subsonic and supersonic transports, tactical military aircraft to STOL and VTOL aircraft.

Siewert concentrated on naval research to solve problems peculiar to naval aviation, such as those associated with carrier operation. He noted that use of the NADC centrifuge for spin simulation led to development of an excellent high fidelity tool for further research.

ROUND TABLE DISCUSSION, "WHERE DO WE GO FROM HERE?"

Moderator	P. Lecomte	France
Panelist	K-H. Doetsch	Germany
Panelist	O.H. Gerlach	Netherlands
Panelist	W.T. Hamilton	USA
Panelist	D.M. McGregor	Canada
Panelist	J.B. Scott-Wilson	UK
Panelist	J-C. Wanner	France

The round-table discussion by AGARD Flight Mechanics Panel members provided an overall summary and projection of "Where Do We Go From Here?" The panelists summarized key issues brought out in the meeting and highlighted the useful role that AGARD can play in standardizing many of the important models used in the analysis and simulation of handling qualities, sharing results of mutual interest, identifying important issues, and validating criteria by means of flight tests. Many of their viewpoints will be reflected in the conclusions to this paper.

DEFINITION OF HANDLING QUALITIES CRITERIA AND REQUIREMENTS

One of the first steps in problem solving is to define the problem, i.e., to be sure what is really of concern. This discussion begins by asking, "What is Handling Qualities Criteria?" While definition of basic terms may appear to be quite simple, it is complicated by differences in language usage. A surprising number of different viewpoints regarding meaning and application of criteria and specifications was found to exist throughout the meeting. For this reason, it is necessary to provide our definitions!

Handling qualities is defined by NASA TN-D-5153 (Reference 23) as, "those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role." Webster's Third New International Dictionary defines criteria as, "a standard on which a decision or judgment may be based," and requirements as, "something that is wanted or needed."

DIFFERENCES BETWEEN CRITERIA AND REQUIREMENTS

The use of the word criteria as opposed to requirements is not an insignificant difference. During this meeting, there were some who used the two interchangeably and some who used one or the other to denote increased stringency. From the definitions given above, it can be seen that requirements are more appropriate for specifications where the procurer is stating what he wants from an aircraft, and criteria are more in line for use in design guides where the designer is searching for design assistance.

It might be appropriate to discuss some of the considerations which are involved in distinguishing between criteria and requirements. The most obvious consideration, which was mentioned previously, is the intended use of the handling qualities characteristics. For example, the military procuring activities specify the handling qualities characteristics that are necessary to perform a mission; thus, they would use requirements. However, where the contractor initiates design of the aircraft, handling qualities criteria provides a useful guide. MIL-F-8785, a requirements specification, is also designed for use in the development of new aircraft.

Another consideration in distinguishing between criteria and requirements is the data base from which it is derived. Requirements should be based on a "good" data base. Criteria, because of their more flexible nature, can be based on a lesser data base. This particular aspect of criteria and requirements is especially important when dealing with V/STOL or reentry vehicles. The lack of good V/STOL or reentry handling qualities data, on which to base requirements, presents a problem for anyone attempting to establish requirements in those areas.

Another aspect of the differences between criteria and specification was pointed out by Teplitz in his comment, "The differences in criteria and intended application make detailed comparison of the civil and military requirements not always feasible. This is only one facet of the FAA problem in applying the criteria derived from MILSPEC - related handling qualities research to the establishment of civil airworthiness regulations. We have made a start on this, however, and we hope soon to begin to investigate the problem of multiple degraded characteristics on minimum acceptable level of safety, under carefully controlled-conditions, which is possible with the use of available ground-based and in-flight simulators."

FACTORS AFFECTING HANDLING QUALITIES

As stated before, "handling qualities" is defined as, "those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role." From this definition it can be seen that handling qualities involve those factors which affect the pilot workload (ease) and performance (precision) of the task. The pilot workload and performance are affected by surprise, fear, excitement, etc. (all of those items causing stress), by the visual, audio and kinesthetic information he receives, and by the aircraft characteristics. More specifically, handling qualities are affected by the aircraft stability and control characteristics, the cockpit interface (e.g., displays, controls), the aircraft environment (e.g., weather conditions, visibility, turbulence, and pilot stress level). One major problem confronting the handling qualities engineer is that the effects of these factors cannot easily be isolated. The relationship of these factors, as shown by Cooper and Harper, is shown in Figure 3. For example, when performing an investigation on the effects of the stability and control characteristics, the investigator must be careful to account for the remaining factors such as aircraft environment in such a way as not to obscure the effects being studied.

HANDLING QUALITIES FACTORS

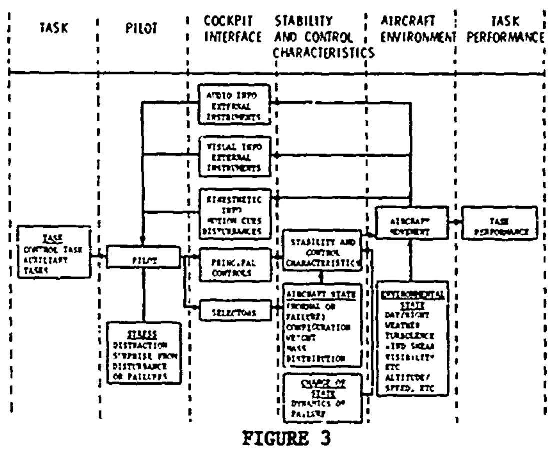


FIGURE 3

CRITERIA AND SPECIFICATIONS

EVOLUTION

Although the history of the criteria was not discussed in any systematic fashion at the Specialists Meeting, it is relevant to provide a brief perspective of handling qualities criteria and specification developments. Figure 4 provides a perspective of the issuance of flying qualities criteria and specifications over past years.

FLYING QUALITY CRITERIA AND REQUIREMENTS

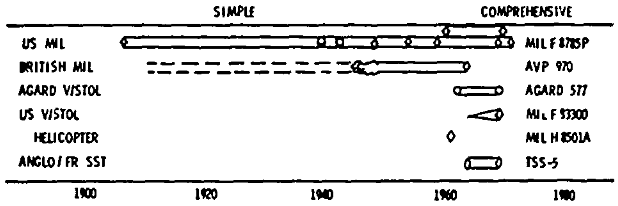


FIGURE 4

In December, 1907, the first United States heavier-than-air flying machine specification included a requirement for what we now call either handling qualities or flying qualities (for piloted vehicles). It stated, "During this trial flight of one hour it must be steered in all directions without difficulty and at all times be under perfect control and equilibrium." By the early 1940's, the equivalent requirement in the Army Air Corps Designer's Handbook had been simplified to read, "The stability and control characteristics should be satisfactory."

The first substantive handling qualities requirements were published by the U.S. Army Air Corps, Spec C 1815, in 1943, as a result of joint efforts by the Army Air Corps, Navy and NASA. Several updatings and outgrowth of this specification can be noted; however, in the late 1940's the introduction of jet and rocket powered vehicles, expanded operational flight regimes and exponentially increasing technological capabilities led to a major effort to improve the criteria and specifications. Time relationships of more recent criteria and specifications in the 1960-1972 time period are shown on Figure 5.

FLYING QUALITY CRITERIA AND REQUIREMENTS

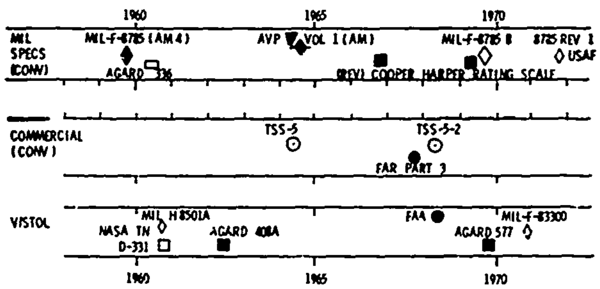


FIGURE 5

It is apparent that the 1969 issue of MIL-F-8785B, "MIL Spec - Flying Qualities of Piloted Airplanes," with its 89 pages, supplemented by a detailed and highly useful 715 page Background Information and User's Guide (BIUG), is far more complex than the 1940 requirements. In part this is due to the fact that in the early years, analytical methods were meager and design of the simple aircraft of that era for adequate stability and control and handling qualities was based on broad criteria, judgment and a cut-and-try approach. Final reliance for judging adequacy of flying qualities depended on the pilot.

Today, aircraft are highly complex. We have the benefit of experience with past and existing aircraft, sophisticated prediction and analysis techniques for both the aircraft and its environment, and greatly improved aerodynamic and dynamic wind tunnel test capabilities. Further, we have harnessed the computer to process vast quantities of data and handle complex higher order differential equations, developed an array of fixed base, moving base, and inflight simulators, developed a wide variety of specialized engineering and scientific skills, formed interdisciplinary teams to solve the problems, and supplemented the engineering skills with physiologists and psychologists to more fully understand the complex relationships between the machine and the pilot. Despite all this, the final judgment of the adequacy of the flying qualities still lies with the pilot!

So it is that we still depend on the pilot to judge the adequacy of flying qualities, a situation which was the source of much discussion throughout the Handling Qualities Criteria Specialists Meeting. As a result of this reliance on the pilots for the final acceptance of an aircraft, the question frequently was posed during the meeting as to why do we need all of these sophisticated criteria and specifications, when all that is necessary is to have some general simplified requirements -- supplemented by broad criteria -- oriented around the mission. Further, the importance of relating handling qualities more directly to the mission capabilities was strongly voiced by Andrews, Adolph and others. (See Westbrook's paper to be given later in this meeting.)

In addition, the aircraft designer, stability and control engineers, and cockpit display and controller developer, need quantitative relationships between what constitutes good flying qualities to numbers of different pilots and the design parameters and characteristics which they have to provide. The high cost, complexity and interacting disciplines of modern aircraft, which operate in many modes over broad flight regimes in both favorable and hostile environments, greatly limits the old cut-and-try approach. The need to design and build new types of aircraft with confidence that they will be completely satisfactory to the pilot in performing the military missions for which they are designed demands continued progress in the development of prediction and analysis methods, simulation techniques, and dependable criteria for design.

The customer who is making irreversible partial payments during the aircraft development needs assurance that all is going well during development. The manufacturer who is dependent on fulfilling some acceptance or certification criteria before he can deliver his aircraft and collect final payments needs a clear understanding of what capabilities he must meet for acceptance.

The needs of the designer, manufacturer, customer, acceptance or certifying authorities alike thus provide reasons for less subjective and more specific statements of acceptable flying qualities criteria and requirements.

One of the most important considerations of a specification, brought out by Andrews, Sliff and many others, is to keep up to date with the data base and technology. More specifically, there were also questions regarding the means used to keep MIL-F-8785B up to date. The mechanism for keeping the specification current is built in and has been used quite extensively throughout its utilization. The procurement specification for a military aircraft either includes MIL-F-8785 by reference, with or without deviations, or uses it as a guide to write a detailed requirement specification for the specific aircraft being procured. During the negotiation of the specification, the contractor and procuring activity have the opportunity to introduce modifications or revisions to any of the requirements of MIL-F-8785B. In addition, as the need arises, MIL-F-8785B may be amended or changed. These changes, however, must be substantiated by a sufficient data base to insure a specification that will aid attainment of the aircraft's mission goals.

Realistically, there are occasions when the lack of good handling qualities data has necessitated writing requirements which are not well substantiated. It is for that reason that a Background Information and User's Guide (BIUG) was especially important to each of the new handling qualities specifications (MIL-F-8785B and MIL-F-83300). The BIUG discusses each requirement and the data base for that requirement. Thus, the contractor knows how well founded any requirement may be. And, as a result, a contractor may take exception with any requirement (especially those with poor data bases) if he has a reliable set of data which indicates that the characteristics of his aircraft enable the pilot to <u>satisfactorily</u> perform the aircraft's design mission.

Where do we stand in resolving the basic questions and needs in this area so important to aircraft design and operation? What progress has been made, how did we do it, and what new research is underway? The next section of the paper will address these questions in more depth.

HANDLING QUALITIES STATUS - CONVENTIONAL AIRCRAFT

STATUS

The current content of flying qualities criteria and specifications for conventional piloted aircraft not only varies between NATO countries but also between military and commercial aircraft applications. While the standardization of military specifications between the AGARD-involved nations thus appears to be somewhat in question, there is one distinct exception. The 7 August 1969 issue of U. S. MIL-F-8785B and the French E.S.A.U., from which the July 1969 Anglo-French Supersonic Transport Aircraft Flying Qualities TSS-5, Issue 2, is derived, are markedly similar in both philosophy and approach. This, of course, was not just coincidence, but the result of an effective interchange between French and U.S. personnel involved in the development of the E.S.A.U. philosophy and MIL-F-8785B. A further step in utilization of common requirements was foretold by J-C. Wanner, when, in response to a question regarding the specification for French military airplanes, he said, "For the military purpose, we intend to apply the philosophy of TSS-5, but I think now it is not necessary, because you have done the job. So I think that our military specification shall be the translation of the 8785B."

While the philosophy of the U.S. Military Specification and the Anglo-French TSS-5 is similar as noted, the U.S. Federal Air Regulations (FAR), used for commercial aircraft, are different in both approach and intent. MIL-F-8785B is a quantitative specification to be used in the procurement of military aircraft and is intended to be used for design requirements, and as a criteria during development, with all its requirements demonstratable by flight test.

On the other hand, as Sliff said, FAR 25 (for flying qualities) is written in a qualitative and general sense to provide the flying quality requirements to assure commercial aircraft meet minimum standards for safety.

The British AVP970 "Design Requirements for Service Aircraft," issued in three books, contains chapters on handling qualities with both basic requirements and a large number of recommendations to the designer, many of which are operational in nature and qualitative rather than quantitative. Andrews noted that the average date of the elements which make up the chapter on flying qualities is 1960, but updating is now under consideration. This is in contrast to the two books of British Civil Airworthiness Requirements, "BCAR's," which are updated frequently.

And, so it is that current requirements cover a spectrum of different concepts and features, different degrees of qualitative versus quantitative requirements, and are of different vintages.

Since MIL-F-8785B and TSS-5 (and now TSS-3) are the newest and most comprehensive specifications now available, further review of their objectives, philosophy and approach will provide a better insight into the current status of flying quality requirements.

MIL-F-8785B AND TSS-5 (Subsequently redesignated TSS Standard No. 3)

Figure 6 compares objectives of the French and U.S. specifications. The goal of TSS-5, which was prepared for commercial supersonic transports, specifically Concorde, is to assure that there will be no limitations on flight safety due to deficiencies in flying qualities.

COMPARISON OF FLYING QUALITIES REQUIREMENTS

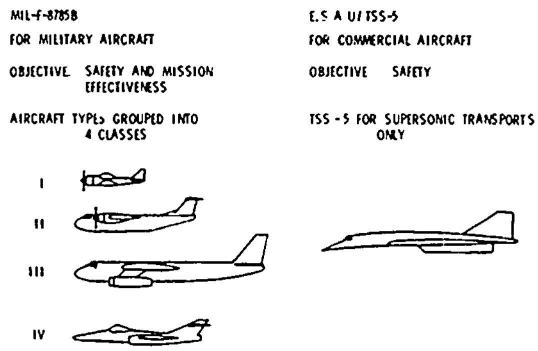


FIGURE 6

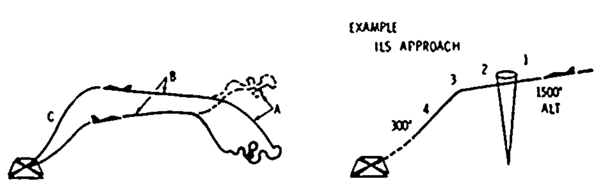
The objective of MIL-F-8785B, which is intended to cover all conventional military aircraft, is significantly different as reflected by the specification statement governing its application, which states that, "This specification shall be applied to assure that no limitations on flight safety or on the capability to perform intended missions will result from deficiencies in flying qualities." The requirement for mission success led to the grouping of different types of aircraft into four different classes, defined on the basis of intended mission, size, weight and maneuverability, as noted.

At this point, it is desirable to consider a basic philosophy of TSS-5 and MIL-F-8785B. In brief, it is recognized that despite what one ideally wants, component failures or excursions from the intended flight regime will occur, with an attendant degradation of flying qualities and possible compromise of mission effectiveness and safety. It is further recognized that the critical cases will vary with aircraft configuration, mission use, flight regime and reliability of systems and components. If the effect of the failure is to degrade the level of handling qualities below that required for mission success or safety, the designer has several options to resolve the problem. He can increase component reliability or modify the aircraft configuration or design to provide adequate flying qualities with degraded or failed components.

Many similarities and also a number of differences exist in the philosophies and applications of MIL-F-8785B and TSS-5. As an example, both documents subdivide the mission into various phases, but the phases are quite different, and TSS-5 is further subdivided into Sub-phases. Since the specific requirements for safety and mission effectiveness for MIL-F-8785B and safety for TSS-5 will vary in different parts of the mission, it is necessary to look at these mission parts in more detail. As shown in Figure 7, the MIL-F-8785B missions are subdivided into Flight Phases. To keep the job of writing and applying the requirements within reason, the Flight Phases are grouped into three mission segments or Categories, according to the similarity of the type of task to be accomplished and the ability of pilots to rate the task. Category A, nonterminal flight phases, require rapid maneuvering and either precision tracking or precise flight path control. Category B is also for nonterminal flight phases, but normally requires only gradual maneuvers without precision tracking. Category C phases are in the vicinity of the airport or base, and, while usually requiring precise flight path control, only require gradual small amplitude maneuvers.

MIL F-8785 B
FLIGHT PHASE CATEGORIES

ESAU TSS5 PHASES



	PHASE	MANEUVER	TRACKING
TERMINAL	C	GRADUAL	PRECISE
NON TERMINAL	B	GRADUAL	NON PRECISE
	A	RAPID	PRECISE

SUB PHASES

1. SEARCH FOR LOCALIZER @ 1500' ALT
2. WAIT GLIDE (REACH GLIDE PLANE @ 1500')
3. PUSH OVER
4. FINAL DESCENT (TO 300' ON HEADING FOR LOC)

FIGURE 7

TSS-5 also divides the flight into parts called Phases, such as "ILS Approach" shown on the figure. The phases are, in turn, subdivided into Sub-phases, each of which has one elementary purpose, as illustrated by the four Sub-phases under the "ILS Approach" Phase.

Application of the specifications to aircraft require numerous additional considerations. Examples of the numerous terms used in the two specifications are shown on Figure 8. These terms, each of which requires careful review before the full implications of the two specifications can be appreciated, are discussed in more detail in the excellent comparison of the French and U.S. specifications by Wanner and Carlson. While such a depth is not possible within the scope of this paper, the unique philosophy and approach of these specifications warrant further attention, especially since the same philosophy is also used in the U.S. Specification for Flying Qualities of Piloted V/STOL Aircraft, MIL-F-83300, and may well influence the thinking in other future specifications. MIL-F-8785B will be used as the basis for the discussion to follow.

FLYING QUALITY SPECS

MIL-F-8785B	LEVELS	E.S A U / TSS - 5
CLASS I, II, III, IV		SUPERSONIC TRANSPORT
FLIGHT PHASE CATEGORY		PHASES / SUB PHASES
SELECTED CONFIGURATION		SELECTED CONFIGURATION CONTROLS / SELECTOR
		TRUE CONFIGURATION CREW TASK
STATE OF THE AIRPLANE		STATE OF THE AIRPLANE
STATE OF ATMOSPHERE		STATE OF THE ATMOSPHERE FLIGHT TECHNIQUES / SECONDARY WORK
FLIGHT ENVELOPES OPERATIONAL SERVICE PERMISSIBLE		FLIGHT ENVELOPES AUTHORIZED CIRCUMSCRIBING
		FLIGHT CASE EXCURSION AROUND THE TASK
PROBABILITY OF A STATE		PROBABILITY

FIGURE 8

The concept of "Levels of Flying Qualities" is basic to the philosophy of MIL-F-8785B. A Level is a relative value or amount of goodness of a stability and control or flying qualities parameter. Levels are a measure of how well the job must be done and can be linked with pilot ratings obtained from flight tests as shown on Figure 9. Here the revised Cooper-Harper scale of pilot ratings is used. Levels are used directly in determining compliance with quantitative specification requirements and, as shown on Figure 10, are linked with a number of the other concepts and parameters used in MIL-F-8785B. In addition to the Levels/mission accomplishment definitions and pilot ratings previously shown, Levels are directly involved in determining the values of the numerous MIL-F-8785B handling quality parameters which are required for adequate mission effectiveness. Levels are related to airplane normal states and failure states which take into account the probability of component failures and with flight envelopes bounded by values of speed-altitude and speed-load factor at which the airplane may be operated during each flight phase.

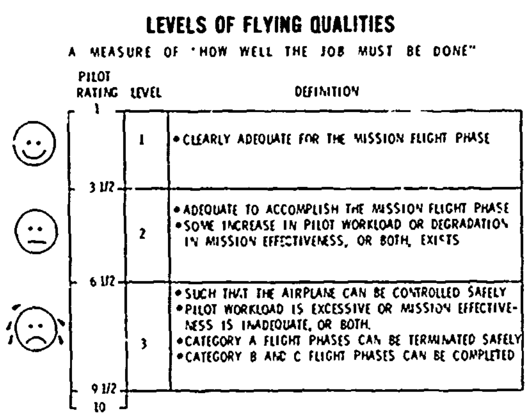


FIGURE 9

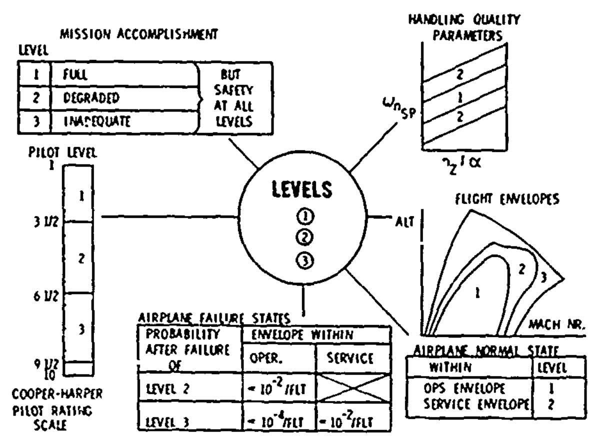


FIGURE 10

Although, as Carlson pointed out, one airplane could have had as many as 367,427 envelopes, the number has been limited in actual practice to some 20 to 40. Flight envelopes are used to specify the flight regimes for which the precise requirements of MIL-F-8785B are to be applied. Such requirements should only be applied where they are needed, and other values should be used for other conditions in order to avoid overdesign and excessive complexity or costs. The boundaries of these envelopes are determined by how the airplane is required to be used, not flying quality limitations. Three different sets of envelopes, Operational, Service and Permissible, are required for each flight phase used by the airplane. The Operational Flight Envelope encloses the region necessary to perform the design mission. The larger Service Flight Envelope provides for the occasional necessity of the airplane flying outside the Operational Flight Envelope, at some reduced level of mission effectiveness, either inadvertently or because of new mission needs. The Permissible Flight Envelope includes all the regions where flight is both possible and permissible.

An example of the relationship between levels, flight envelopes and failure states is depicted by Figure 11. Typical altitude/Mach No. flight envelopes for the Category A combat phase of a Class IV airplane are shown, with the airplane normal state flying qualities levels depicted in the rectangular boxes. As can be seen, Level 1 flying qualities are normally required within the Operational Envelope, Level 2 within the Service Envelope and no lower in flying qualities than Level 3 in the Permissible Envelope. The degraded levels allowed after failures, on a probability basis, are shown in the circles within the Operational and Service Envelopes. No degradation below Level 3 flying qualities is allowed in the Permissible Envelope, except for special failure states. As an example, in the Operational Envelope, the probability of encountering Level 2 shall not occur on an average of more than once each 100 flights, and Level 3 shall not be encountered on an average of more than once each 10,000 flights. So it is that the level concept coupled with probability analyses of failure states and other concepts of MIL-F-8785B provides a technique to help assure:

1. a high probability of good flying qualities where they are most needed for mission success,
2. acceptable flying qualities under occasional conditions, and
3. a safe flyable airplane under all conditions.

CONCEPT OF FLIGHT LEVELS

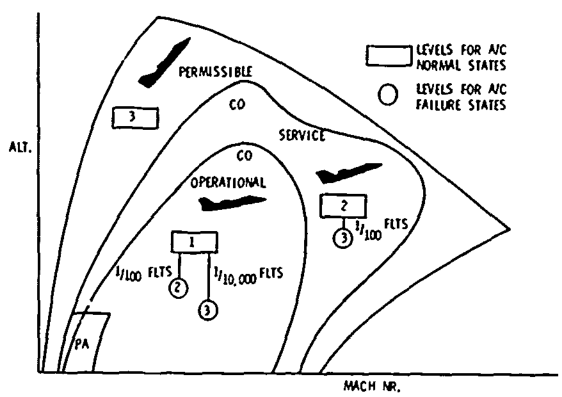


FIGURE 11

It is evident that use of MIL-F-8785B requires extensive analyses of component failures and determination of their impact on flying qualities. Although the work may be extensive, reliability analyses should be accomplished as a matter of course to determine adequacy for mission success and to avoid operational failures and excessive costs. Further, prevention of one aircraft loss will often more than pay for the cost involved.

One implication of this philosophy noted in the meeting is the need to improve the acquisition of component failure data from both developmental and operational experiences.

STATUS OF V/STOL FLYING QUALITIES CRITERIA

When faced with the task of writing criteria on the handling qualities of V/STOL aircraft, it becomes readily apparent that there is some speed, which we shall define as V_{con}, above which the V/STOL aircraft will have to meet the same requirements as a conventional aircraft with the same design mission. This V_{con} speed may be based on "entering the aerodynamic flight regimes" as done in AGARD-577 or it may be based on the manner in which the vehicle is controlled as done in MIL-F-83300. In either case, above the V_{con} speed, the "conventional" handling qualities criteria will apply. As was noted, the two U.S. military specifications (MIL-F-8785B and MIL-F-83300) were written specifically with the same philosophy (classes, levels, failure states, etc.), thus making the conversion at V_{con} from the application of V/STOL to conventional requirements an easier process.

In the V/STOL handling qualities area, AGARD-R-577-70 (revised version of 408A) and MIL-F-83300 are good indicators of the status of development in this field. The major point of distinction between these two documents is that AGARD-R-577-70 is a document of criteria, while MIL-F-83300 is a specification. This distinction is not an insignificant one. In this particular case, AGARD-577 has elected to change "its emphasis to reflect criteria rather than specifications," because of a lack of operational experience. The lack of good information on display effects, V/STOL turbulence models, and general operational usage of V/STOL aircraft has severely handicapped the development of these documents and is reflected in the criteria (or requirements).

For example, the authors of AGARD-577 elected not to distinguish between the various classes of V/STOL aircraft, and not to incorporate the concept of "Levels." The reasons for not including these concepts are discussed in the Introduction to AGARD-577. Again the principal reason is the lack of "operational experience with V/STOL aircraft." While data available from experimental aircraft has been helpful in establishing and validating criteria, as will be seen later, the implications of full operational use can only be determined by extrapolation of research data to anticipated missions. Kissel and Wünnenberg's paper (Reference 8), which compared VJ-101 and DO-31E with AGARD-577, suggests the use in AGARD-577 of the definition of "certain 'Levels' similar to the USAF-MIL Spec. [MIL-F-83300] for Handling of V/STOL-Aircrafts. For instance: Level 1 for mission tasks, Level 2 for normal flight and Level 3 for emergency like engine or system failure." The concept of Levels and Classes is a desirable format for criteria. However, the unfortunate aspect is that the data base is so minimal that it prevents establishment of criteria which can adequately distinguish between various classes of aircraft or various levels of operation.

Let's briefly look at these documents to get a feel for the differences and status of the V/STOL handling qualities area. AGARD-408A, the forerunner of 577, was discussed in Reference 5 and the development of MIL-F-83300 was discussed in Reference 6. Reference 6 also contains a brief comparison of the criteria contained in AGARD-577 and MIL-F-83300. To illustrate some of the additional differences, Figures 12 and 13 present the roll control power criteria from 577 and 83300, respectively. The keypoints to note from the AGARD-577 Table is: (1) the different parameters specified, (2) the breakdown of the requirements into that needed for maneuvering, trim and upsets (due to gusts, recirculation, etc.), and (3) breakdown into type of control system. Now, note that the Table from MIL-F-83300 has a breakdown into: (1) class of vehicle, (2) level, and is an extension of MIL-F-8785B format. For detailed discussion of these specific criteria, consult the discussion in AGARD-577 and AFFDL-TR-70-88 (the Background and User's Guide for MIL-F-83300).

AGARD-577 ROLL CONTROL POWER

PARAMETER TO BE MEASURED	CONTROL POWER REQUIRED FOR:	TYPE OF CONTROL SYSTEM	MINIMUM LEVELS FOR SATISFACTORY OPERATION	
			HOVER	STOL
ROLL ANGULAR ACCELERATION, RAD/SEC^2	MANEUVERING	ATTITUDE COMMAND	0.2 - 0.4	0.1 - 0.6
		RATE	0.2 - 0.4	
		ACCELERATION	0.3 - 0.4	
BANK ANGLE AFTER 1 SEC, DEG	MANEUVERING	ATTITUDE COMMAND		
		RATE	2 - 4	2 - 6
		ACCELERATION	2 - 4	2 - 6
ROLL CONTROL DEFLECTION AT ZERO ROLLING VELOCITY, IN.	TRIM	ALL	SUFFICIENT CONTROL IN EXCESS OF MANEUVERING REQUIREMENTS TO TRIM OVER DESIGNATED SPEED AND C.G. RANGE AND FOR MOST CRITICAL ENGINE FAILURE	
TIME TO RECOVER TO INITIAL ATTITUDE OR CONTROL DEFLECTION, SEC	UPSET (DUE TO GUSTS, RECIRCULATION, GROUND EFFECT, ETC.)	ALL	SUFFICIENT CONTROL IN EXCESS OF MANEUVERING AND TRIM REQUIREMENTS TO BALANCE MOMENT DUE TO A SPECIFIC GUST, FOR EXAMPLE, 30 FT/SEC GUST	
			BUILDING UP IN 1 SEC	BUILDING OVER A 100 FT DISTANCE
ROLL ANGULAR ACCELERATION, RAD/SEC^2	TYPICAL RANGE OF VALUES USED BY V/STOL AIRCRAFT FOR MANEUVERING, TRIM, AND UPSET	ATTITUDE COMMAND	0.4 - 1.5	0.2 - 2.0
		RATE	0.8 - 2.0	0.3 - 2.5
		ACCELERATION	0.8 - 2.0	-

FIGURE 12

MIL-F-83300 - ROLL CONTROL POWER

CLASS	$t_{30°}$, SECONDS		
	LEVEL 1	LEVEL 2	LEVEL 3
I	1.3	1.8	2.6
II	1.8	2.5	3.6
III	2.5	3.2	4.0
IV	1.0	1.3	2.0

FIGURE 13

Conspicuously absent from these documents are effects of displays, effects of turbulence, unconventional controllers (side-arm, etc.), but these are missing because of the state of the art of V/STOL handling qualities. If asked what one item was needed for the V/STOL handling qualities, it would be operational data. Such data is required not only to provide data directly to analyses, but to guide and validate ground and air simulations.

Questions were posed as to why there are two documents on V/STOL handling qualities and why they appear to be so different. The answer is partially explained by Westbrook, who noted that 577 is a criteria prepared for NATO nations and 83300 is a specification for design and procurement of U.S. military aircraft. It is likely that differences which exist will be minimized as more and better data become available from V/STOL programs. While coordination was maintained between S. Anderson, NASA Ames, and C.B. Westbrook, AFFDL, who were involved in the development of AGARD-577 and MIL-F-83300, respectively, each has expressed interest in further coordination and resolution of differences. It is obvious that with the limited resources available, and the vastness of the problem to be tackled, there is an urgent need to maximize cooperative efforts and take full advantage of all the data being generated.

PROBLEMS IN DEVELOPING OF CRITERIA

There are several factors, which make the establishment of handling qualities criteria difficult. Among these are: (1) the inability to quantify the various factors affecting handling qualities, (2) new mission requirements requiring extrapolation of experience, and (3) advancements in controls and displays.

It is difficult to quantify such items as the pilot stress level, or level of cockpit displays. To study their effects of the various handling qualities factors and, in turn, establish criteria, some means of quantifying these items would be desirable. Another factor which makes establishment of criteria difficult is the quantification of handling qualities goodness. The approach used presently is the employment of pilot ratings and comments. An alternate approach, which has received attention recently, is the pilot-vehicle ("paper" pilot, pilot modeling, etc.) which will be discussed a little later. Both approaches consider the pilot workload and performance and associate a number to indicate the relative ease and precision with which a task can be performed. For more specific consideration of each approach, consult the papers presented (References 12, 18 and 20) at this meeting.

The difficulty of developing handling qualities criteria, which will assure adequate mission success and safety for missions for which we have little or no experience, has and will continue to cause extreme consternation to handling qualities criteria developers. V/STOL and reentry missions are two examples of new mission vistas that have necessitated extrapolation of experience to develop adequate handling qualities criteria.

The impact of displays and automatic control systems on handling qualities criteria further complicates the problem of establishing criteria. These two effects tend to add a new dimension to the development of the handling qualities criteria, because with proper displayed information and automatic controls, it is possible for a pilot to do a task easier and with better precision. Thus, the new criteria must take into consideration the effects of the displays and the increased order of complexity of the total system. For example, when the pitch response is not classical second order, the frequency and damping ratio criteria cannot be applied, and some alternate means of specifying criteria is needed. A completely satisfactory alternative has not been developed yet.

ROLE OF ANALYSIS, SIMULATION AND FLIGHT TESTING

The three principal sources of data from which the handling qualities criteria are derived, refined and substantiated, are analysis, simulation, and flight testing. Their role in the evaluation of handling qualities criteria is presented in Figure 14.

EVOLUTIONARY CYCLE
FOR HANDLING QUALITIES CRITERIA

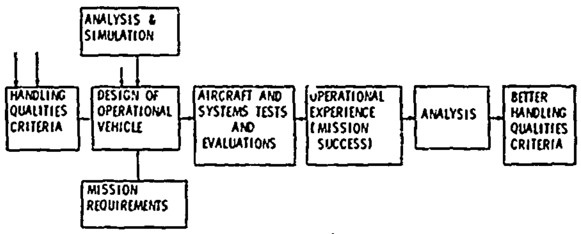


FIGURE 14

This figure depicts the cyclic nature of the development refinement and substantiation of criteria. As each new aircraft is developed, the criteria used for that aircraft are evaluated as to its capability of insuring mission success and safety. If meeting the criteria has not resulted in the desired mission capability, the criteria is modified to develop better handling qualities criteria. Unfortunately, as Westbrook noted, much of the flight test data is received without correlated pilot ratings, and its value in improving criteria is minimal.

Ideally, each simulation program or flight test program has associated with it an analysis phase. However, recently the analysis portions have taken on a new dimension. Through the use of pilot modeling approaches, predictions of flight test and simulation results are possible. Figure 15 from Gallagher's paper (Reference 10) is typical of the accuracy which is achieved. This figure is a comparison of computed and measured performance during the tracking-in-gust task and illustrates the accuracy of the prediction techniques. The pilot describing function is of the form

$$Y_\phi = K_\phi (T_{L_\phi} s+1) e^{-\tau s}$$

where Y_ϕ - Pilot describing function

K_ϕ - Pilot gain in roll closure

T_{L_ϕ} - Pilot lead

τ - Pilot reaction time delay in roll

s - Laplace transform variable

PREDICTIONS AND SIMULATION

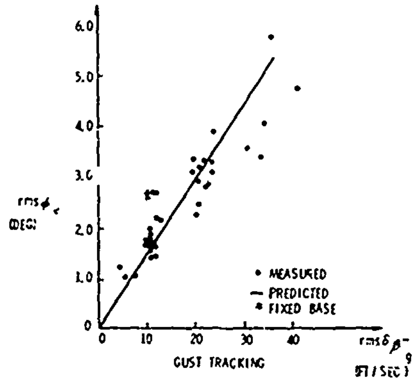


FIGURE 15

This same form of pilot describing function was used by R.O. Anderson in Theoretical Pilot Rating Prediction (Reference 20). A procedure is described whereby a theoretical rating prediction called "paper pilot" has had some success for hover tasks, but only limited success so far for other tasks. An example of the ability of the "paper pilot" to predict pilot ratings for a hover task and pitch task with the effects of the addition of a first-order lag representation of actuator dynamics (or "effective" control system) is shown in Figure 16. However, as D.M. McGregor, Lead Discussor for Mr. Anderson's paper, pointed out, ". . .a pilot predictor producing positive postulations presents possibilities and should be pursued."

"PAPER PILOT PREDICTIONS"

TASK	CASE	ACTUATOR TIME CONST	PAPER PILOT	HUMAN PILOTS
HOVER	PH 3	0.10 SEC.	4.14	4.0
	PH 3	0.50 SEC.	5.93	6.0
PITCH	2D	0.50 SEC.	3.50	5.6 & 5.5
	"	2.0 SEC.	6.08	6 & 6

FIGURE 16

One of the most productive sources of data for handling qualities research is simulation. Whether it is inflight, moving base or fixed base simulation, it is the source of much of the data used for establishing handling qualities criteria. K-H. Doetsch of National Research Council of Canada presented a paper (Reference 17), which investigated the ranges of various lateral-directional characteristics required to provide adequate flying qualities for turning maneuvers at low speed, using an inflight V/STOL simulator. This study varied damping ratio, frequency, and the ratio of the roll-angle to sideslip-angle in the Dutch roll mode, together with the damping ratio and frequency of the numerator quadratic of the roll-angle to aileron-control input transfer function. Much of the data presented was used to establish requirements for MIL-F-83300.

There is a discussion by C.E. Adolph in his paper (Reference 9) on the present procedure of testing aircraft for compliance with criteria. Mr. Adolph's main criticism was lack of a more mission-oriented evaluation of the weapon system and the need for developing additional criteria specifically for evaluation purposes.

Along this same line of thought are two papers (References 7 and 8) which were presented at this meeting, and compared the V/STOL handling qualities criteria (principally AGARD-577) with flight test results of the CL-84, VJ-101C and DO-31E. As mentioned before, this process is an integral and necessary part of the evaluation of handling qualities criteria. For example, Michaelsen's paper (Reference 7) presents comparisons of criteria and the handling qualities of the CL-84. A representative and informative comparison is the Vertical Thrust Margins. The comparison between the criteria and flight test values is shown on Figures 17A and B (Figure 4.1 and Table 4.2 of Reference 7). It should be noted that the margin used for takeoff is less than that called for by the criteria. As stated by Michaelsen, "While the CL-84 operates successfully in take-off with a vertical thrust margin less than that of the criteria, the values in the criteria (AGARD-577) are considered reasonable." This reflects an interesting aspect of the comparison between flight test and criteria; that is, it is equally important to the development of a criteria that the data substantiate as well as cause refinements in criteria. Michaelson's paper also made an assessment to determine:

1. whether the CL-84 needs improvement, or whether the criteria are too demanding or not applicable in those cases where the CL-84 does not meet the criteria, and
2. where the criteria appear to be too lenient in light of the CL-84 flight test experience.

VERTICAL HEIGHT CONTROL CHARACTERISTICS

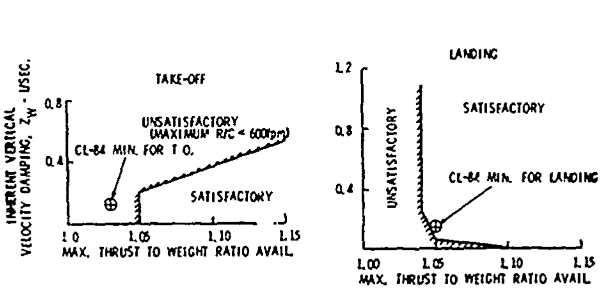


FIGURE 17A

VERTICAL VELOCITY AND THRUST RESPONSE

VERTICAL VELOCITY AND THRUST RESPONSE CHARACTERISTICS			
PARAMETER	AGARD CRITERIA		CL-84-1
	MINIMUM	MAXIMUM	
HEIGHT CONTROL SENSITIVITY "g"/IN	.1	.4	.25
VERTICAL VELOCITY RESPONSE R/C (AFTER 1 SEC.), fpm	150	775	450
THRUST RESPONSE, FIRST ORDER TIME CONSTANT, SEC.	NOT GREATER THAN .5		.15

FIGURE 17B

For example, Michaelsen indicated that "the minimum levels of pitch control power and damping for satisfactory operation given by the Criteria [AGARD-577] are certainly too low for the CL-84 in and near hover. On the other hand, these levels are probably quite satisfactory for large V/STOL aircraft or for aircraft with high wing or disk loadings, such as jet lift aircraft. It is appreciated that it is virtually impossible to specify general requirements that will prove satisfactory for all concepts and sizes of aircraft. The discussion of the criteria in AGARD-577 makes this point, but it is questioned if the point is emphasized strong enough." This discussion is particularly interesting to those people who are involved in establishing criteria or developing STOL aircraft.

Reference 8, by Kissel and Wünnenberg, compared V/STOL handling qualities criteria with flight test results of the VJ-101C, V/STOL supersonic fighter, and DO-31E, V/STOL transport. One of the items pointed out by this paper was that, "From flight tests with the VJ-101 -- and the results were quite similar for the DO-31 -- it was found that the natural frequency should be lower and the damping ratio should be higher than found by the simulator tests. This tendency is even stronger for the pitch axis." This and several enlightening aspects regarding the criteria contained in AGARD-577 were discussed. In addition, some interesting data is presented on the VJ-101C and DO-31E. One set of data of particular interest is the control usage in hover and transition flights shown in Figure 18 (Figure 9 from Reference 8). The authors concluded from this figure that, "The lower limit of the recommended control accelerations of AGARD-577 corresponds good to larger aircraft and the upper limit good to small aircraft. The exception is the yaw control power of the VJ-101; these values are smaller than expected due to dynamic structure problems of the heavy swivelling engine pods."

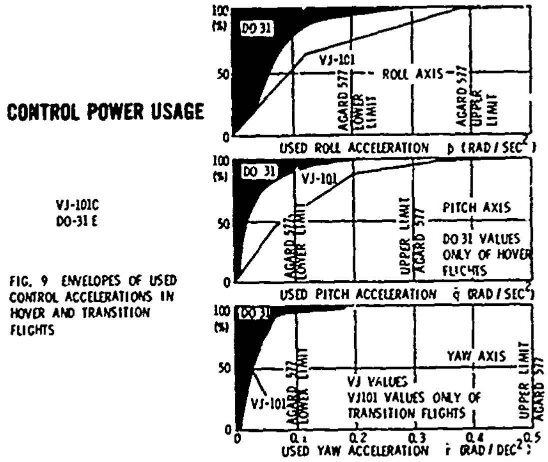


FIGURE 18

SPECIAL PROBLEMS

STALL/SPIN

The stall/spin problem is of sufficient magnitude and complexity and, as Adolph noted, cannot be solved by legislating qualitative requirements with design criteria, such as "neither post stall gyrations nor spins shall be readily attainable for (a variety of entry conditions) except by prolonged gross misapplication of controls." Hancock points out that the problem should be attacked at the design stage by acquiring all the information necessary to predict aircraft behavior at high angle of attack (α). The AGARD Fluid Dynamics Panel Meeting is Lisbon, 26-28 April, on Fluid Dynamics of Aircraft Stalling, will be a step in this direction. Bihrle seconded the importance of early design work, and noted that we can now grossly predict in the conceptual design phase that a fighter will have inherent ability to enter different post-stall gyrations and spin modes, as well as which controls are necessary for recovery. The problem remains, however, of identifying the fast, flat spin mode from which recovery will be difficult.

The traditional way of avoiding entering potential stall departure conditions is to provide the pilot with adequate stall warning in the form of aerodynamic buffeting. Unfortunately, as Woodcock noted, buffeting may occur at angles of attack considerably below stall and as a result a fighter pilot in combat will penetrate past buffet onset to use this additional margin for maneuvering. The result may be a sudden stall departure with little warning. This is especially true, as Adolph notes, if the inherent aerodynamic response characteristics are masked by use of a stability augmentation system (SAS) in order to maintain good flying qualities at high angles of attack. Integrity of the SAS is also a worry.

Use of artificial stall warning devices, such as lights, rudder pedal or stick shakers, are beset with problems. One commentator noted that the pilot could not feel the rudder pedal shakers because they were masked by high intensity buffet. Hancock and Bihrle recommended use of angle-of-attack indicators in the cockpit as the most logical indicator to tell the pilot of a potential stall problem. Siewert noted all U.S. Navy carrier aircraft are so equipped. However, for fighter aircraft, Adolph warned that cockpit indicators are of little value since the pilot's attention is outside the cockpit. Davis also noted it had been tried on Concorde but never used in normal operation. Pilots have resisted automatic devices such as stick pushers because of concern over possible unwanted actuation; however, we may find newer automatic angle-of-attack limiters to be far more acceptable than inadvertant spins.

Hancock noted that spin tunnel testing starts in the design stage and could provide very useful data. Bihrle agreed, but noted the prediction of full scale spin modes from such data requires much "agonizing interpretation of the experimental results, hopefully made under devine guidance." The real problem has occurred since the introduction of high wing loading, highly swept low aspect ratio configurations with low roll inertia.

Research by the U.S. Navy on use of a ground based centrifuge as a spin simulator for pilot proficiency training was reported by Siewert. Excellent fidelity was reported when running in the fully dynamic mode. Initial success has been so encouraging that additional investigations in the post stall and incipient spin areas are planned on the F-14A fighter. Wasicko reported on NASA research, which indicated success in use of wind tunnel tests, analyses and fixed base simulator for study of stall and spin characteristics of fighter aircraft, with special attention to the directional divergence or "nose slice" response with a swept wing fighter. Bihrle earlier noted, however, that fixed base simulators do not supply angular acceleration, a basic anticipatory cue used by pilots, and that other physiological limitations of man limit value of a fixed base simulation.

U.S. spin flight test requirements have recently been updated by issuance of Military Specification, Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes, dated 31 March 1971. In addition, an amendment to MIL-F-8785B has been recently issued to expand coverage in this area. Representative spin test requirements, Tables I and II from Reference 9 and MIL-S-83691, are shown in Figure 19 and Figure 20.

FLIGHT TEST DEMONSTRATION MANEUVERS

TEST PHASE	CONTROL APPLICATION	STALL / DEPARTURE ENTRY CONDITIONS
A STALLS	PITCH / AOA RATE ROLL / NEUTRAL YAW / NEUTRAL RECOVERY INITIATED AFTER DEFINITE g - BREAK RAPID, UNCOMMANDED ANGULAR MOTION STICK FULL AFT & AOA AT LIMIT INTOLERABLE BUFFET	(1) SLOW AOA RATE (2) TACTICAL (ATTITUDE AND RATE RELEVANT TO SIMULATION)
B STALL WITH AGGRAVATED CONTROL	PITCH TO AOA RATE ROLL AND YAW AS REQUIRED WHEN ACHIEVE A, B, C MISAPPLY CONTROLS BRIEFLY	
C B + SUSTAINED CONTROL		
D SPIN ATTEMPTS		(1) ABRUPT AOA RATE (2) TACTICAL

FIGURE 19

SUSCEPTIBILITY/RESISTANCE CLASSIFICATION

(DETERMINED BY TEST PHASE IN WHICH DEPARTURES / SPINS FIRST OCCUR)

	CLASSIFICATION	
	DEPARTURES	SPINS
A - STALLS	EXTREMELY SUSCEPTIBLE	EXTREMELY SUSCEPTIBLE
B - STALLS WITH AGGRAVATED CONTROL INPUTS	SUSCEPTIBLE	SUSCEPTIBLE
C - STALLS WITH AGGRAVATED AND SUSTAINED CONTROL INPUTS	RESISTANT	RESISTANT
D - SPIN ATTEMPTS	EXTREMELY RESISTANT	EXTREMELY RESISTANT

FIGURE 20

TURBULENCE

The development of good turbulence models continues to be a problem to the handling qualities investigator. If an aircraft stability, damping and control power are chosen to optimize maneuvering in still air, the characteristics may result in unsatisfactory response in turbulence. There is little question that consideration of the effects of turbulence is a significant aspect in the development of good handling qualities criteria. This is a particularly difficult problem for the development of V/STOL handling qualities criteria because of the added complication of local projections such as buildings, terrain, etc.). Jones, in his discussion of the development of turbulence models, indicates that it is not only the power spectrum which is important, but also the intermittency. Since pilots tend to have a "threshold" and only fluctuations in response which exceed this level lead to control action, Mr. Jones proposes a discrete gust model for aircraft control and handling qualities investigations. The discrete gust model is noted as the most logical approach for V/STOL applications also. Our scope is such that only a sampling of the material presented by Mr. Jones and the others is possible. However, for further information on the subject of turbulence models, consult Mr. Jones' paper which gives a good discussion of the turbulence problem and would be a good point of departure for those interested in more detail. The reader is also referred to AFFDL-TR-69-67 which presents a good Non-Gaussian turbulence model.

FLEXIBLE AIRCRAFT RIDE QUALITIES

The impact of flexible aircraft on handling qualities is a problem. While solutions may exist, these solutions can present additional problems. For example, J.G. Wykes presents a paper (Reference 15) which includes discussion of the flexibility problem as it affects riding qualities, with two approaches to minimize motion at the pilot's location. Before going into these solutions, let's look at the problem of flexibility. Figure 21 (Figure 1 from Reference 15) shows that the trend appears to be toward more flexible aircraft for other than handling qualities reasons.

IMPACT OF FLEXIBILITY ON AIRCRAFT DESIGN

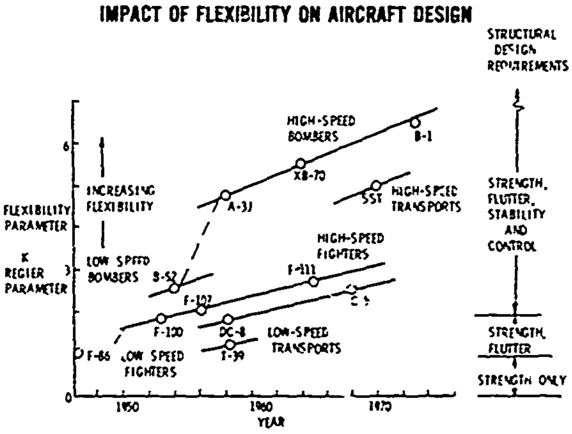


FIGURE 21

The two approaches to solve the flexible aircraft pilot ride quality problem discussed by Wykes are: (1) a seat isloation system, and (2) a structural mode control system. A comparison of their effectiveness is shown on Figure 22. Each approach is shown in conjunction with use of a stability augmentation system (SAS) which markedly reduces the short-period response.

RIDE QUALITY SYSTEMS

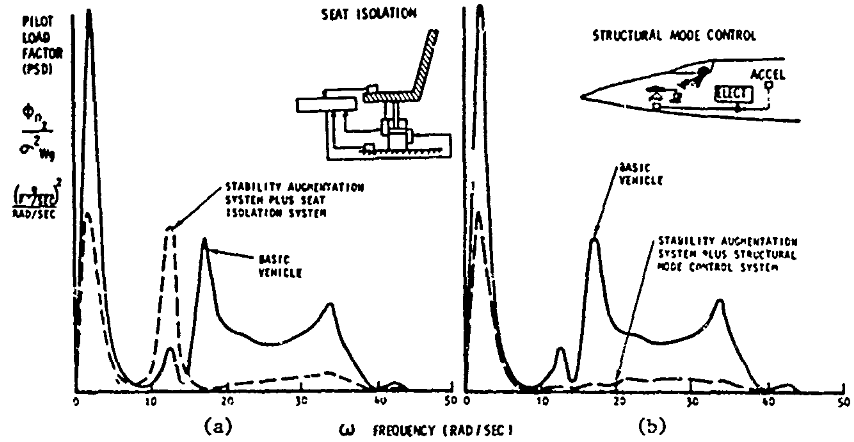


FIGURE 22

The "notch" for the seat isolation system shown by dotted lines in Figure 22a is at 18 rad/sec (approximately $3H_z$) to reduce the large structural mode peak of the basic vehicle at that frequency, as shown by the solid lines. However, pilot motions at frequencies below the "notch" are amplified, as can be seen by the new peak at 12 rad/sec. Figure 23 shows in more detail such displacement (or motion) amplifications at frequencies below the "notch" frequency, in this case at 4-5 H_z.

SEAT ISOLATION SYSTEM

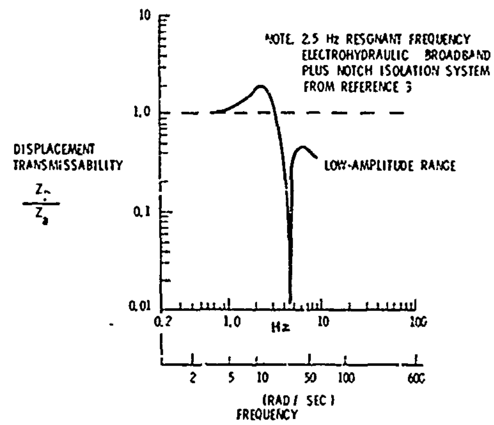


FIGURE 23

This situation results in a relative motion between the pilot and his controls and instruments which can seriously degrade handling qualities and ability to read instruments. For that reason, Wykes recommends that handling quality requirements should limit use of such systems.

The structural mode control system, on the other hand, is seen on Figure 22 to be effective across a broad band of frequencies. While good knowledge of the flexible vehicle characteristics and careful iterative system design is required, the structural mode control system appears to provide a solution to the pilot ride quality problem.

PILOT RATINGS

Still another problem associated with the development of handling qualities criteria is the handling of pilot ratings and comments. This particular problem was discussed by R.P. Harper in Reference 12, who emphasized the desirability, or even the necessity, of obtaining pilot comments along with ratings. The subjective nature of pilot ratings is enough of a problem in itself. However, the main means of handling pilot comments, but this is only a start and a better means of presenting this information is needed. J-C. Wanner voiced his viewpoint that use of pilot rating scales, such as that by Cooper-Harper, provide only an index for measuring pilot workload. Wanner also noted the large impact that changes in displays can have on pilot ratings. This points out the importance of considering the complete man-machine system in determining adequacy of flying characteristics.

PILOT RATINGS AND PILOT COMMENTS

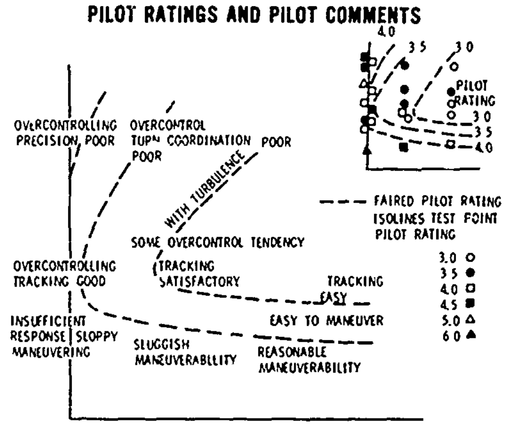


FIGURE 24

CONTROL SYSTEMS

The influence of stability systems on flying qualities of the Concorde was addressed by Deque. In accord with the TSS-5 philosophy the design standard links the level of required flying qualities to the probability of encountering various states of the aircraft. After ranking these on a probabilistic basis, items were selected for evaluation on a simulator and in flight. The objective that unaugmented airplanes be safe may lead to cg limitations that penalize airplane operational economy.

As a result Deque notes "it would probably be possible to improve the operational economy of an aircraft by not observing this rule, if there is sufficient redundancy of systems." All indications lead us to believe that this step will be taken in future generations of transport aircraft. In relation to fixed base simulator tests, it was found that the absence of motion cues tends to generate requirements for unnecessary stability augmentation.

ADDITIONAL CONSIDERATIONS

Numerous examples of specific criteria and test findings were discussed at the meeting. The following examples are representative. In Kehrer's paper (Reference 11), he discusses the fact that the Boeing SST criteria permitted an instability by specifying a minimum time to double amplitude based on Figure 25 (Figure 3 from Reference 11). The cutoff point is the minimum safe condition (P.R. = 6.5), and occurs at approximately 3 seconds. The SST design requirement of 6 seconds thus provided a reasonable time margin and a pilot rating of 5 or so. Adolph's paper (Reference 9) also discussed stick forces per g which were less than 3.0 lb/g limit of MIL-F-8785B; however, the gradients were highly linear. As stated in Reference 9, "When evaluated during tracking tasks, the low gradients were not considered to be objectionable; on the contrary, the flying qualities were considered to be excellent." These are only food for thought and point out that requirements based on experience do exist which might be overly restrictive to the designer.

PILOT RATING OF LONGITUDINAL HANDLING QUALITIES
FOR UNSTABLE SST CONFIGURATION

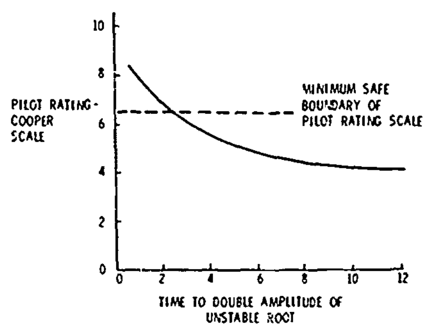


FIGURE 25

RESEARCH

Handling qualities research, noted by Wasicko and Siewert, indicated a wide range of activities and the type of information that AGARD should assist in sharing between the NATO countries. NASA research encompasses many types of aircraft, with emphasis given to instrument flight approaches, steep ILS approaches, simulator studies of stability and control derivatives and handling qualities of new designs, high angle-of-attack flights, and experimental V/STOL aircraft tests.

The U.S. Navy work emphasized the problems of carrier operations. Research to correlate effects of approach speed of 95 to 125 knots on carrier landing performance showed no significant correlation with carrier landing accidents. Inflight simulation efforts, with a small variable stability aircraft to determine effect of the principal handling quality parameters on carrier approaches performance, indicated desirable limits to values of the short period frequency and n_z/α, the basic parameter governing longitudinal response characteristics.

PROBLEMS REMAINING

In addition to all of the problems discussed above which need solutions, there are some additional problems which need to be addressed.

The effects of displays on the development of handling qualities criteria were noted by Wanner, but not addressed specifically at this meeting. This may be due, in part, to the limited knowledge regarding how to include display effects in handling qualities criteria. In any case, this area of handling qualities is still in the embryo stage and it should be coming of age soon.

During this meeting, the level of flying qualities (as they apply to MIL-F-8785B) were addressed. The problem in this area arises from multiple-degraded levels. For example, if there are two or more systems, which are Level 2, what is the overall effectiveness of the aircraft as far as completing its mission? In fact, Barnes stated that "a designer so minded could produce an aircraft meeting Level 1 requirements, but which the pilot would find unacceptable, by diabolical choice of permitted stick forces, frequency, damping, friction and so on." This particular aspect of the level concept needs more study.

CONCLUSIONS

Information from this meeting is summarized in the following three categories:

1. the technical work needed,
2. the non-technical aspects that make the job easier, and
3. the issues and additional problems which need to be addressed.

Figure 26 summarizes some of the major technical areas which need to be addressed by the handling qualities investigators. The non-technical requirements are presented in Figure 27. These items aid in reducing the technical and communication problem down to just a technical problem. Figure 28 is a very brief summary of some of the issues and problems which require attention.

TECHNICAL WORK NEEDED

- TURBULENCE MODEL
  - CONV
  - V/STOL
- DISPLAY EFFECTS ON H.Q. CRITERIA
- PILOT-VEHICLE INTERACTION
- PILOT INFORMATION
  - WORK LOAD
  - LOOP CLOSURES
  - PILOT RATINGS AND COMMENTS
- FLEXIBILITY AND H.Q
- PERFORMANCE, S&C AND H.Q INTERACTIONS
- MULTIPLE DEGRADED LEVELS
- MEANS OF SPECIFYING REQUIREMENT
  - HIGH ORDER SYSTEM
- STALL/SPIN

FIGURE 26

NON-TECHNICAL NEEDS

I STANDARDIZATION OF TERMS AND UNITS

II IMPROVED COMMUNICATION
- RESEARCH PROGRAMS AND PLANS
- WORKING LEVEL INFORMATION

III IMPROVED ACQUISITION OF DOCUMENTATION

FIGURE 27

ISSUES AND ADDITIONAL PROBLEMS

- NEED FOR CRITERIA UP-DATING
- NEED FOR VALIDATION OF CRITERIA
  - PARAMETERS
  - MISSION CONSIDERATIONS
- WHAT IS BEST APPROACH FOR AIRCRAFT DEVELOPMENT?
  - SPECIFICATIONS
  - CRITERIA
- WHAT ARE THE PRIORITIES?
  - HANDLING QUALITIES
  - PERFORMANCE
  - STABILITY AND CONTROL
  - COST

FIGURE 28

As Melchior De Santa Cruz once said, "The wiseman profits more from the fool than the fool from the wiseman; for the wiseman takes warning by the fool, but the wiseman's sense has no value to the fool." And so let us as handling qualities investigators continue to learn from the mistakes of others, for it is by doing this that we can remain wisemen.

REFERENCES

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9.	Adolph, C.E.	Criteria Trends Obtained from Analysis of Current Aircraft
10.	Gallagher, J.T.	Role of Simulation and Analysis in Criteria
11.	Kehrer, W.	Criteria for Supersonic Transport Certification
12.	Harper, R.P., Jr.	The Role of Pilot Opinion Ratings
13.	Hancock, G.J.	Criteria for Stall and Post Stall Gyrations
14.	Jones, J.G.	Turbulence Models for Handling Qualities During Take-off and Landing
15.	Wykes, J.H.	Flying Qualities Interaction with Elastic Airframes
16.	Deque, R.	Flight Control System Interface
17.	Doetsch, K-H.	Parameters Affecting Pilot's Assessment of Handling Qualities
18.	Hosman, R.J.A.W.	Pilot Vehicle Analysis
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20.	Anderson, R.O.	Theoretical Pilot Rating Predictors
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22.	Siewert, R.F.	Recent U.S. Navy Flying Qualities Research
23.	Cooper, G.E. Harper, R.P., Jr.	The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities, NASA TN-D-5153, April 1969

CONSIDERATIONS FOR STABILITY AND CONTROL OF V/STOL AIRCRAFT – A REVIEW OF AGARD REPORT 577

by

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SUMMARY

Revisions have been made to previous V/STOL handling qualities requirements based on criteria rather than specifications. To help provide a clearer understanding of the criteria, a discussion of the pilot's desire for a particular characteristic is given. In addition, data and reference material have been provided to back up the proposed criteria to permit the user to understand the limitations of the data on which the criteria are based. A review of several controversial areas including pitch control sensitivity, static longitudinal stability, roll control power, roll-yaw cross coupling, vertical flight path control, and transition indicates that more information is needed to refine the criteria, particularly for operational IFR use. Further, additional work of a systematic nature must be conducted to clarify the effect of several interacting items that influence the pilot's overall impression of the aircraft's behavior.

1. INTRODUCTION

New handling qualities information has recently been published for V/STOL aircraft. Among the many reasons to revise and update handling qualities are the need to reflect the recent requirements of operational type aircraft, to give consideration for the peculiarities of operating with different types of lift-propulsion concepts, and to describe the effects on closed-loop responses of operation with novel control systems.

The first AGARD publication of V/STOL handling qualities recommendations, AGARD Report 408 (ref. 1), was based largely on NASA TN D-331 (ref. 2). Both reports received criticism, not unexpected, on their scope and specific recommendations. They were directed primarily toward VTOL aircraft and did not adequately cover STOL-powered lift characteristics. Since the results were obtained mostly from test bed type aircraft and helicopters, the reports obviously could not reflect the requirements of operational type V/STOL aircraft. To a lesser degree, the same criticism can be applied to the revised AGARD Report 577 (ref. 3) because only limited results are available from operational type aircraft.

In the recommendations, a chief source of controversy was the effect of vehicle gross weight or size on aircraft response. Further, the consequence of providing only minimum acceptable values of each handling quality item was not fully appreciated by the user; a V/STOL aircraft that meets all recommendations individually could still be too demanding of the pilot's skill because several factors could interact to produce an overall unsatisfactory response.

In revising reference 1 it was agreed that a more meaningful and useful document would include:

- Evaluation of the various handling qualities items in terms of *criteria* rather than requirements or specifications.
- A discussion section following each criterion to explain the purpose of the criterion.
- Data and reference material to back up the proposed criteria.

As used in reference 3, criteria were defined as evaluation standards based on numbers that are meant only to be typical and can vary depending on the particular mission and task. Meaningful criteria can then serve as a guide in establishing specifications to be used by a contractor for the design and testing of a particular aircraft.

In the past, handling qualities requirements have been presented without an explanation of why the pilot desired a particular characteristic; in many cases neither the purpose nor the interrelation of the various factors affecting the requirements was understood. Without an understanding of all possible tradeoffs, there may be a tendency to apply the requirements too rigidly to a particular aircraft design, thereby compromising its utility.

Finally, it is helpful to provide background data and reference material for each criteria. If the user understands the limitations of the data on which the criteria are based, he can evaluate the criteria with respect to their optimum application to his design, and, of course, the contractor can then provide more effective specifications.

Examples of several controversial stability and control areas are given to show how the foregoing philosophy was carried out in preparing AGARD Report 577. The purpose is to point out how well the present criteria compare with the available flight results, review areas that need additional work, and indicate how the gaps in knowledge can be filled. Because of length restrictions only the following areas will be covered in this paper:

- Pitch control sensitivity
- Static longitudinal stability
- Roll control power
- Roll cross coupling
- Vertical flight path control
- Transition acceleration/deceleration

2. RESULTS AND DISCUSSION

2.1. Longitudinal Stability and Control

2.1.1. *General* Good longitudinal stability and control characteristics are essential if V/STOL aircraft are to operate routinely into and out of confined areas. In general, longitudinal stability, damping, and control deteriorate at low speeds and the combined effects can result in poor precision in flight path tracking.

Factors that individually influence the longitudinal behavior of conventional aircraft have been studied for several years and detailed handling qualities requirements are available to cover the speed range down to the stall. Since V/STOL aircraft must also fly down to hover, several new requirements are needed in this lower speed regime. Unfortunately, there is less information upon which to base requirements and many factors must be considered individually and in combination for setting up meaningful criteria. Many factors influence longitudinal behavior including the following: control power and sensitivity, linearity of response, pitch damping, control system time constant, control forces, cross coupling, normal acceleration sensitivity, flight path speed stability (backside operation), static and dynamic stability characteristics, lift-drag variation with engine power, effects of proximity to ground, and direct lift or drag control or both. Only control sensitivity and static stability characteristics will be covered in the following discussion.

2.1.2. *Pitch control sensitivity.* The ratio of the maximum acceleration per unit control input (control sensitivity) is an important parameter that strongly influences the pilot's impression of the response of the aircraft. If the control sensitivity is too low, the aircraft will appear sluggish because a large control movement will be needed to obtain the desired response, while excessively high sensitivity can lead to overcontrolling tendencies.

The pitch control sensitivity criteria of reference 3 are presented in table 1 in which the type of control system and the area of flight operation are considered. Note that only the minimum values are specified since they represent the most difficult design challenge. These criteria were based on results of numerous piloted simulator and flight studies and on consideration of (1) total control power available, (2) control travel limits, (3) control stick gearing (linearity), (4) the mission or task, and (5) the dynamic behavior of the aircraft.

Table 1. Pitch Control Sensitivity

Parameter to be measured	Type of control system	Minimum levels for satisfactory operation	
		Hover	STOL
Attitude change per unit control deflection deg/in.	Attitude command	3 – 5	
Pitch angular acceleration per unit control deflection rad/sec^2/in.	Rate	0.06 – 0.1	0.08 – 0.12
Pitch angular acceleration per unit control deflection rad/sec^2/in.	Acceleration	0.08 – 0.16	

2.1.3. *Validity of pitch control sensitivity criteria.* Numerous studies have been made to determine pitch control sensitivity requirements for V/STOL aircraft. Figure 1 shows typical results from these studies and data from flight tests of several VTOL and STOL aircraft (refs. 4-9). The curves show similar shapes for the 3.5 (satisfactory) pilot rating boundaries as determined by systematic variations of control sensitivity and angular rate damping, but different absolute values for minimum, optimum, and maximum control sensitivities. The reasons for the diversity of these boundaries and the scatter in the flight data are discussed next.

The desirable level of control sensitivity depends primarily on the mission or task and the dynamic behavior of the aircraft in turbulence. First, rapid maneuvering may be required for some missions where quick stops and rapid changes in flight direction are needed for rescue or to avoid enemy opposition. These tasks require higher sensitivities because pilots tend to use quick and frequent control inputs rather than long, steady control movements. Thus, the minimum satisfactory control sensitivities are related to the amount of control input needed to perform the specific task. When large control deflections are required (low sensitivity), pilot fatigue and discomfort are aggravated, and control precision may suffer adversely. Large control displacements make it more difficult for the pilot to return the control to the correct trim or hover position when maneuvering or compensating for unwanted pitch changes resulting from effects of gusts, recirculation, or other disturbances.

The aircraft's pitch dynamic behavior is directly influenced by angular rate damping, M_q, the speed stability derivative $M_u g/I_y$, and the longitudinal force derivative, X_u/m.

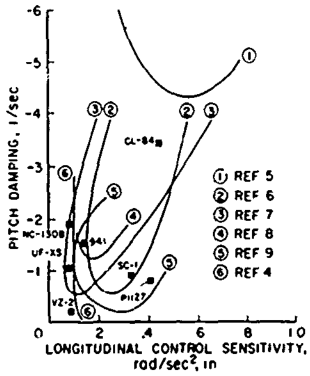

Fig. 1. Comparison of 3.5 pilot rating boundaries from piloted simulator and flight tests.

Minimum M_q is required to prevent pitch attitude overshoots when large, quick control inputs are used and to reduce excursions in pitch attitude. A wide range of satisfactory levels of M_q from a very high optimum level of -4.2 down to 0, is indicated in the data in figure 1. These extremes reflect the particular test conditions and vehicle dynamics. The data for curve 6 were obtained during VFR flight tests of the X-14A VTOL jet-lift aircraft (ref. 4), which in calm air and out-of-ground effect is described by the pilots as exhibiting good hovering steadiness and an insensitivity to gust disturbances. The large optimum value of rate damping described by curve 1 was obtained from tests of a tandem rotor helicopter (ref. 5). Large values of rate damping were inherent to this aircraft; thus, high values of control sensitivity are needed to avoid sluggish response.

Another reason why high values of damping and large control sensitivity may be required is the effect of the speed stability derivative $M_u g/I_y$, which is a measure of the change in pitching moment caused by changes in airspeed. High values of $M_u g/I_y$ require increases in control sensitivity to handle the increased pitching response to gusts. If control sensitivity is too low, large excursions in control position are required to trim for the long period components of gusts. For combinations of high turbulence and large $M_u g/I_y$, high levels of damping are desirable to reduce the effects of the short period gust components.

The stability derivative X_u/m, which is the longitudinal force on the aircraft resulting from changes in airspeed, also influences control sensitivity requirements. For small values of X_u/m (low translational damping) higher values of control sensitivity are required because the aircraft has a tendency to continue in motion until arrested by tilting the thrust vector. High values of X_u/m make the aircraft more susceptible to longitudinal gusts; however, because of the improved translation damping, this may not prove objectionable, as noted in the simulator tests of reference 10.

Finally, the flight results in figure 1 show that a wide range of control sensitivity and damping values exists for the various aircraft for the reasons previously discussed. The main point to be made is that the accumulated data have been used to define only the minimum levels of control sensitivity. Larger values may be required for adequate pitch response when the mission, task, dynamic behavior characteristics, etc. are taken into consideration.

2.1.4. *Stability with respect to speed.* The benefits of stable longitudinal stability characteristics have been recognized for some time for conventional aircraft and the various handling qualities specifications have required both force and position gradients to be stable over a wide speed range. The purpose of static stability is to reduce divergences in airspeed that can cause problems in controlling flight path and in approaching unsafe parts of the flight envelope. For example, a reduction in speed unnoticed by a preoccupied pilot may place the aircraft too close to the stall, and control of flight path may seriously deteriorate.

Depending on the following conditions, V/STOL aircraft may require less conventional static stability (stability with respect to speed):

1. The shape of the power-required curve (airspeed excursions have a smaller effect on flight path control of V/STOL concepts that have a relatively flat power-required curve).
2. The relative importance of pitch attitude stability compared to static stability (since most V/STOL concepts are designed to change airspeed by changing thrust vector angle, pilots are more aware of pitch attitude stability).
3. The airspeed range being considered. (At very low airspeeds flight path changes are made primarily by power; consequently, there is less concern for being "on speed" to provide sufficient "g" margins for maneuvering. Further, at very low airspeeds where the effects of aerodynamic lift are not significant, there is less concern in approaching the stall than at higher airspeeds where a pitch-up could cause the aircraft to enter an unsafe flight condition with insufficient nose-down control for recovery.)
4. The type of control system used. (Attitude command and rate command control systems, in effect, function satisfactorily, regardless of the degree of static stability present.)

The shape of the power-required curve can have a direct effect on the airspeed excursion acceptable to the pilot because it indirectly affects flight path control. Figure 2 shows the variation of the ratio of power available to power required with change in airspeed for two different V/STOL concepts – the tilt wing CL-84 and the fan-in-wing XV-5A aircraft. At low speeds a large change in airspeed will cause a smaller change in rate of climb for the XV-5A than for the CL-84. Consequently, the XV-5A pilot will be less concerned about maintaining precise control of airspeed during the approach and will tolerate a smaller margin of static stability. At airspeeds greater than 50 knots, for example, the XV-5A has negative force and position stability gradients, yet the pilots rated the longitudinal stability characteristics as satisfactory (ref. 11). Further, when approaches are made in the 50-knot speed range, the XV-5A operates on the front side of the power-required curve, resulting in more favorable flight path response characteristics. In contrast, the pilots were more critical of the CL-84 flight path control and static stability (ref. 12).

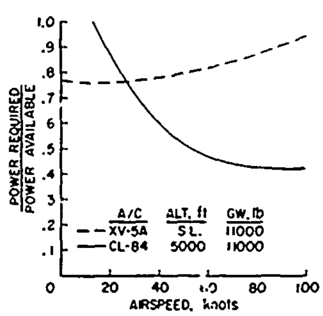

Fig. 2. Comparison of power required curves for the XV-5A and CL-84 aircraft.

The relative importance of pitch attitude stability compared to static stability was brought out during flight tests of the X-22A tilt duct aircraft (ref. 13) and the XC-142 tilt wing aircraft (ref. 14). Although both attitude stability and static stability were negative for the X-22A, the pilot was more concerned with attitude stability, as noted by his comment that "... attitude instability caused difficulty in trimming the aircraft, and increasing aft stick position with increasing nose-down pitch attitude was disconcerting." (He rated it unacceptable for IFR operation, PR 7.) The XC-142 was flown in approaches with the fuselage essentially level and the pilots used pitch attitude instead of airspeed as a primary flight reference. They did not comment adversely on the neutral and negative force and position stability.

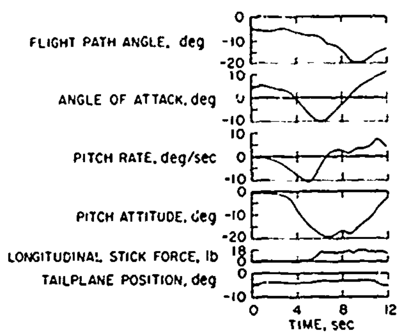

Fig. 3. Attitude divergence characteristics; P.1127 85 knots.

Another example of the relative importance of attitude and static stability is shown in figure 3 for the P.1127 aircraft. In hands-off flight (approximately 5 sec), the aircraft pitched down 15° during which time the airspeed (not shown) increased approximately 15 knots. In this maneuver the pilot was primarily concerned about the steep nose down attitude and the resultant increased flight path angle.

Although longitudinal static stability is undoubtedly desirable, its relative importance decreases as powered lift effects increase. Thus, in the low airspeed range, the pilot uses attitude as the primary reference.

2.2. Roll Control Power

2.2.1. *General background.* One of the more controversial areas that has persisted over the years is a definition of how much roll control moment must be supplied for hover and STOL operation. Pilots have been more critical of the control of V/STOL aircraft about the roll axis than about any other axis partly because the lateral positioning must be quick and precise, and partly because of the large influence of crosswinds during landing. Precise control is essential during approach because even small bank angles result in relatively large heading changes at low speeds. Undoubtedly, some of the difficulty in addressing this problem has arisen because many items interact to determine the overall roll response apparent to the pilot. These include:

- Control needed for maneuvering
- Control needed for trim
- Control needed for upset (due to gusts, recirculation, ground effect, etc.)
- Type of control system used
- Control sensitivity
- Aircraft size (mission considerations)
- Angular rate damping
- Control lag
- Turn entry characteristics (e.g., adverse yaw, yaw due to rolling)
- Mechanical characteristics of control system (e.g., friction, breakout, force gradient)

The total amount of control needed is made up by a combination of these individual requirements; the first four are the major inputs. The pilot desires certain values of roll control for maneuvering, for trimming in sideward flight, and for controlling upsets due to turbulence or self-generated disturbances. Control power requirements depend on many factors: (1) the mission to be performed, (2) the susceptibility of a particular configuration to unsymmetric moments resulting from aerodynamic or thrust-induced cross flow as well as turbulence and ground-induced disturbances, (3) aircraft size (in general, large aircraft are maneuvered less briskly and because of their higher inertias they are disturbed less by turbulence), (4) the type of control system used (more stabilized systems require less control power), and (5) the amount of angular rate damping available.

For trim in hover, various amounts of roll control moment are needed to maintain desired velocities in sideward flight. The amount differs for each VTOL concept because of the difference in magnitude of rolling moment introduced from both aerodynamic and engine-induced flow sources. For aircraft with inherently large rolling moments induced by side velocity, ample control moment is needed to avoid the development of excessively large bank angles, which may occur very abruptly causing a sudden loss in altitude when the aircraft is suddenly turned sideward from a headwind approach. Some types of V/STOL aircraft require that any asymmetric rolling moments associated with power plant failure be trimmed out. Further, the amounts of trim required depend on the crosswind magnitudes specified for a particular mission and VTOL concept.

The amount of control power available to counteract upset due to gusty air or self-induced flow effects in ground proximity (which are also configuration dependent) directly affects the precision of the approach and touchdown. In vertical takeoffs and landings, the pilot needs to adjust attitude rapidly to avoid excessive side drift. Bank angle excursions are undesirable in STOL approaches because of the tendency to induce large heading errors. In these cases, the pilot is interested primarily in returning to the initial bank angle in a given time. In addition, the type of control system used has a pronounced effect on control power requirements for upset. More sophisticated control systems, such as attitude command, automatically reduce or eliminate the need for the pilot to correct for the upset. Because corrections can be sensed and made more quickly by the SAS, large amplitude excursions in bank do not develop and there is a resultant savings in control power requirements.

Because of the foregoing considerations, the criteria for roll control power were broken down in the form shown in table 2. Although only examples of roll control power are presented here, a similar system has been used for the pitch and yaw axes. The chief purpose in breaking the requirements into separate parts is to force the user to examine how each one affects his particular aircraft design or flight evaluation. Different values of roll acceleration are given to take into account the type of control system used and the type of operation (i.e., VTOL or STOL). The reasons for selecting these values are given in the following paragraphs.

2.2.2. *Control needed for maneuvering.* Table 2 lists a range of values for maneuvering control requirements that reflect differences in the mission requirements. In reference 3 the criteria states "that ... aircraft whose missions require extensive maneuvering should be capable of at least the larger values indicated, while those for which maneuvering is only incidental to the mission and those for which direct side force control can also be used should be capable of at least the lower value noted." The validity of the values listed in table 2 is certainly open to question because ultimately the values must come from real operational experience with different classes of V/STOL aircraft. Until such results are available, we can only speculate on the basis of limited data obtained primarily from nonoperational type V/STOL aircraft, some of which have attempted to simulate operational type maneuvers. There is the further

Table 2. Roll Control Power Criteria

PARAMETER TO BE MEASURED	CONTROL POWER REQUIRED FOR:	TYPE OF CONTROL SYSTEM	MINIMUM LEVELS FOR SATISFACTORY OPERATION	
			HOVER	STOL
ROLL ANGULAR ACCELERATION, rad/sec²	MANEUVERING	ATTITUDE COMMAND	0.2 – 0.4	0.1 – 0.6
		RATE	0.2 – 0.4	
		ACCELERATION	0.3 – 0.6	
BANK ANGLE AFTER 1 sec, deg	MANEUVERING	ATTITUDE COMMAND		
		RATE	2 – 4	2 – 4
		ACCELERATION	2 – 4	2 – 4
ROLL CONTROL DEFLECTION AT ZERO ROLLING VELOCITY, in.	TRIM	ALL	SUFFICIENT CONTROL IN EXCESS OF MANEUVERING REQUIREMENTS TO TRIM OVER DESIGNATED SPEED AND CG RANGE AND FOR MOST CRITICAL ENGINE FAILURE	
TIME TO RECOVER TO INITIAL ATTITUDE OR CONTROL DEFLECTION, sec	UPSET (DUE TO GUSTS, RECIRCULATION, GROUND EFFECT, ETC.)	ALL	SUFFICIENT CONTROL IN EXCESS OF MANEUVERING AND TRIM REQUIREMENTS TO BALANCE MOMENT DUE TO A SPECIFIC GUST; FOR EXAMPLE, 30 ft/sec GUST	
			BUILDING UP IN 1 sec	BUILDING OVER A 100 ft DISTANCE
ROLL ANGULAR ACCELERATION, rad/sec²	TYPICAL RANGE OF VALUES USED BY V/STOL AIRCRAFT FOR MANEUVERING, TRIM, AND UPSET	ATTITUDE COMMAND	0.4 – 1.5	0.2 – 2.0
		RATE	0.8 – 2.0	0.3 – 2.5
		ACCELERATION	0.8 – 2.0	–

problem of determining from data obtained during these maneuvers the amount of control used uniquely for maneuvering and the amount used concurrently to correct for trim and upset due to turbulence such as gusts and recirculation. Perhaps the best answers can be derived from records of aircraft for which trim changes, by virtue of their engine and aerodynamic layout, are minimum. Further, if these aircraft use an attitude command type of control system, the effects of external disturbance are minimized. Further confirmation of the lower value of roll angular acceleration for STOL operation has been obtained from "flights" in a piloted motion simulator (ref. 15). A slightly higher value (0.6 rad/sec²) was selected for the upper end of STOL operation to reflect the need for agile maneuvering into confined areas.

2.2.3. *Control needed for upset.* The amount of control needed to compensate for upset depends chiefly on the magnitude and character of the disturbance. It is in this area that the proposed criteria are weak. Although improvements have been made in gust measurement techniques, data analysis, and prediction effects, a well-defined gust model suitable for hover and STOL operation remains to be defined. The criteria for upset used in table 2 attempt to establish a base for firmer values. It was considered necessary to specify a discrete gust effect rather than the usual rms random noise type for simplicity of analysis and to provide meaningful results for control power assessments.

2.2.4. *Validity of roll control power criteria.* The range of values for total control power given in table 2 reflects the speculative nature of the criteria and shows the need for flexibility in choice for design purposes. The values in the bottom row are typical ranges used by various aircraft and are not intended to represent firm numbers that must be met. An examination of flight test data and a discussion of how some of the aforementioned items interact to produce a given overall impression of roll response to the pilot follows. Figure 4 shows results of STOL aircraft tests (taken from ref. 16) obtained during approach and takeoff. The results are presented in terms of maximum angular acceleration obtainable as measured by the conventional roll reversal technique. For convenience the data are presented as a function of gross weight, which was used as a sizing formula $(W + 1000)^{1/3}$ in reference 1. Also shown are the pilots' ratings of the overall roll response for each aircraft. It should be recognized, however, that angular acceleration is only a convenient parameter to use as a yardstick and that it relates only indirectly to the pilot's impression of controllability. Further, when weight is used as a parameter it only approximates the effects of size and, as noted previously, reflects maneuverability requirements and sensitivity to turbulence.

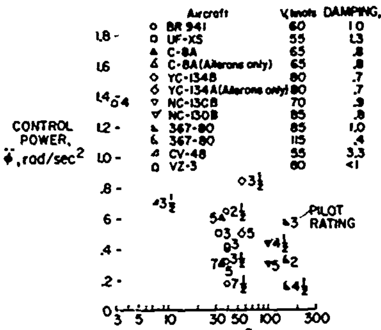

Fig. 4. Roll control power values for STOL aircraft.

Note first that a large acceleration value does not necessarily indicate satisfactory pilot impression of roll response. The VZ-3 aircraft has more than three times the roll acceleration capability of most of the other aircraft tested and still has only a pilot rating of 4. The ability to maintain a desired bank angle while maneuvering in turbulance has been the most critical requirement for roll control of these STOL aircraft at takeoff and landing speeds. In tests of the BR 941, less than 40 percent of the available control was used during extensive maneuvering. Remember that this aircraft needs little lateral trim for crosswind operation and the propellers are interconnected to remove any engine-out asymmetry trim requirements. The 941 is perhaps the most documented of these aircraft. It has been flight tested with several lateral control modifications and has been investigated extensively in piloted motion simulators. Flight tests with this aircraft in IFR operation and moderate turbulence (ref. 17) indicated that roll control was satisfactory with a control power of 0.4 rad/sec² under these more adverse conditions Note that for heavier aircraft, the NC-130B, poorer ratings are evident for this same control power value (based again on IFR operation in gusty air). The poorer overall roll controllability was due in part to low control sensitivity and to the fact that at 70 knots almost full roll control was required to trim for an inoperative engine. Therefore, too small a margin was left for maneuvering. The heaviest (and largest) aircraft tested was the 367-80 (707 jet transport) modified to incorporate a high lift BLC flap system. With the combined aileron spoiler system, the roll acceleration produced by large control deflections was so large for that size aircraft that the pilot was concerned about possible structural damage. In the initial tests with this aircraft the ailerons were equipped with an aerodynamic tab control that was rated unsatisfactory (PR 4-1/2) because of high forces and nonlinear response characteristics. Changing to a hydraulic powered control system with essentially the same rolling moment capabilities improved the pilot rating because of the lower forces. These data show that an improved pilot rating resulted when a higher approach speed was used, even though less acceleration was available. In this case, the cross coupling effects ($C_{N_{\delta_a}}$, C_{ℓ_β}, C_{n_p}) were greatly reduced at the lower C_L associated with the higher approach speed. A further example of interrelated effects is brought out by results obtained on the BLC equipped YC-134A aircraft. Even though very large lateral acceleration was available with the spoiler and aileron combination, precise use of this capability was difficult because of nonlinear response. At approximately 30° wheel position, the region most frequently used in controlling the aircraft, the rapid increase in response and the large increase in force when the spoilers were engaged combined to produce an unsatisfactory characteristic that masked the control power ratings of this aircraft.

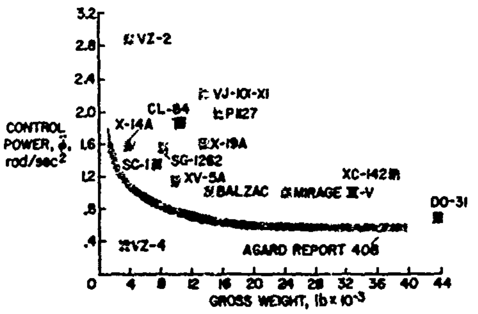

Fig. 5. Comparison of hover roll control values with AGARD 408 requirement.

Figure 5 shows the same parameters for VTOL aircraft in hover. Note the wide range of values for the various aircraft. These values are generally well above the former AGARD 408 sizing formula $(W + 1000)^{1/3}$, which was really meant to be a minimum maneuvering requirement. Because of lack of clarity in this respect, it was conveniently used in many paper designs (and for a few aircraft) as the total control power needed. A sizing rule is difficult to establish from these data for several reasons.

One of the first points to notice is that the X-14A has one-fourth the weight of the P.1127 but can get by with less control power mainly because the P.1127 requires a major portion of its available roll moment to trim for sideward flight. In fact, for the Harrier VTOL aircraft sideslip is restricted in forward flight by a warning device on the rudder pedals (ref. 18). Further, the aircraft would have required even more roll control power if the control sensitivity and the mechanical characteristics of the control system had not been optimized for low speed flight. The XV-5B, SC-1, Balzac, and Mirage III-V also require a large percentage of available control power to offset rolling moments associated either with sideward flight in hover or sideslip in forward flight. In fact, this particular trim requirement had been seriously overlooked in operational testing, and as a result, all of the aforementioned aircraft (except the X-14A) have been damaged, some fatally, in accidents attributed to this trim problem. These aircraft all have inboard jet engines whose induced flows produce the major rolling-moment contributions. On the other hand, two aircraft (the VJ-101 and the DO-31) that have wing-tip jet engines do not have the sideslip trim problem. This is reflected in the control power usage of the VJ-101 (ref. 19) that needs only 0.25 rad/sec^2 for roll control in typical takeoff and landing maneuvers. Similarly, the DO-31 (ref. 20) needs only 0.4 rad/sec^2 for roll control in IFR approaches in gusty air. Both aircraft have much more roll control power available because of engine-out trim requirements.

2.3. Cross Coupling

Because of reduced directional stability and damping at low speeds, moments generated by roll control inputs tend to result in larger sideslip angles than in conventional flight. Sideslip angles that result from the yawing moment due to (1) roll control deflection and (2) roll rate are large at high lift coefficients; consequently their influence is greater for STOL aircraft and they increase the requirements for turn coordination to reduce sideslip. The turn entry coordination problem is discussed in detail in reference 21 and illustrated in figure 6, which shows a time history of a roll maneuver performance with the NC-130B aircraft at 70 knots. These results show that although the desired bank angle was obtained in 2-1/2 sec, 7 sec elapsed before the heading changed to the correct direction. When the pilot attempts to coordinate the turn, he must supply different amounts and phasing of the rudder to account for the effects of adverse yaw, yaw due to roll rate, and yaw rate damping that occurs at different times during the turn.

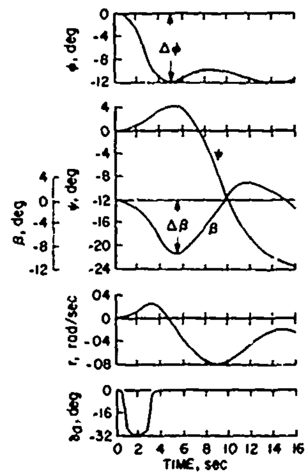

Fig. 6. Time history of the response of the NC-130B to a step bank maneuver; V = 70 knots

The cross coupling that occurs when roll control is used has been expressed as a ratio of maximum sideslip angle to bank angle ($\Delta\beta/\Delta\Phi$), and the maximum allowable values are shown in table 3 of reference 3.

The cross coupling parameter, $\Delta\beta/\Delta\Phi$ is measured during an abrupt bank angle change with rudder fixed. Correlation of $\Delta\beta/\Delta\Phi$ with pilot rating of turn coordination is given in figure 7 for various aircraft and for a range of lateral directional characteristics studied on the simulator (see ref. 16). These data indicate that values of $\Delta\beta/\Delta\Phi$ less than approximately 0.3 were rated satisfactory (pilot rating of 3-1/2). The values shown for the various STOL aircraft point out the need for augmentation during operation at STOL approach speeds. Improvements can be noted for the NC-130B and 367-80 aircraft by the addition of positive N_p and $N_{\dot\beta}$ augmentation.

The lag in changing heading previously pointed out in the discussion of the NC-130B aircraft has been recognized as a major part of the turn coordination problem. It has not been possible, however, to develop a criterion based on heading lag alone as there is a significant interaction between roll and heading control depending upon the roll-mode time constant (ref. 22). These simulator tests indicated that when good roll damping existed, a larger heading lag was tolerable and vice versa.

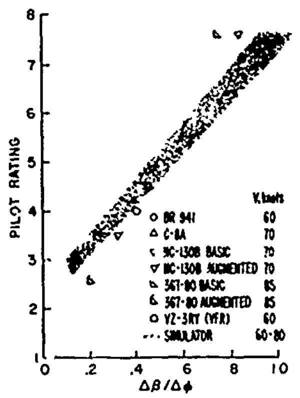

Fig. 7. Relation of turn entry coordination and pilot rating.

2.4 Vertical Flight Path Control

2.4.1 *General background.* Vertical control of flight path angle during approach, flare, touchdown, rotation, and climb-out is an important consideration for STOL operation because of the short field length requirements. Satisfactory routine operation from short fields with obstacles in the approach and climb-out paths depends on precise control of flight path angle. During STOL operation of V/STOL aircraft, vertical flight path cannot be controlled adequately by pitch control alone, and the pilot must use additional methods to develop normal acceleration.

Powered lift is used for flight path control in three general modes: controlling rate of sink at flare and touchdown, acquiring and tracking a particular flight path angle during approach, and making gross changes in flight path for waveoff and turning flight. Satisfactory performance of these tasks depends on the amount of normal acceleration available from powered lift, the aircraft response time, and the degree of cross coupling. The values needed by the pilot depend on how critically the particular flight mode must be controlled. For example, altitude control during flare and touchdown requires a short response time and must be precise It is equally important that cross-coupling effects between powered lift and aircraft rotation be minimized so that the pilot can precisely adjust rate of sink and aircraft attitude independently as required for optimum landing and takeoff performance.

These points are considered in the criteria presented in table 3.

Table 3. Vertical Flight Path Control Criteria. STOL Operation.

ITEM	MODE*	PARAMETER TO BE MEASURED	LEVEL FOR SATISFACTORY OPERATION	MINIMUM LEVEL FOR ACCEPTABLE OPERATION
CONTROL POWER	A	INCREMENTAL NORMAL ACCELERATION	±0.1g	INSUFFICIENT DATA
	B	INCREMENTAL NORMAL ACCELERATION	±0.1g	INSUFFICIENT DATA
	C	STEADY-STATE CLIMB ANGLE	6° OR 600 ft/min	200 ft/min
	ALL	INCREMENTAL DESCENT ANGLE	2° GREATER THAN SELECTED APPROACH ANGLE	INSUFFICIENT DATA
RESPONSE TIME	A	AIRCRAFT RESPONSE	ACHIEVE MODE IA IN LESS THAN 0.5 sec	INSUFFICIENT DATA
	B	AIRCRAFT RESPONSE	ACHIEVE MODE IB IN LESS THAN 1.5 sec	INSUFFICIENT DATA
	C	AIRCRAFT RESPONSE	ACHIEVE MODE IC IN LESS THAN 2.0 sec	ACHIEVE MODE IC IN LESS THAN 4 0 sec
CROSS COUPLING	ALL	PITCHING MOMENT	NOT OBJECTIONABLE	NOT OBJECTIONABLE

*MODE A. FOR FLARE AND TOUCHDOWN CONTROL WHEN LESS THAN 0.15g CAN BE DEVELOPED BY AIRCRAFT ROTATION USING PITCH CONTROL ALONE

MODE B. FOR FLIGHT PATH TRACKING WHEN MORE THAN 0 15g BUT LESS THAN 0.30g CAN BE DEVELOPED BY PITCH CONTROL ALONE.

MODE C FOR GROSS FLIGHT PATH CHANGES REGARDLESS OF THE NORMAL ACCELERATION DEVELOPED BY PITCH CONTROL.

2.4.2. *Criteria.* For satisfactory flight path control during all phases of STOL flight operation below V_{con} (including approach, landing flare, touchdown, and waveoff), the vertical aircraft response characteristics obtained at a constant attitude resulting from any combination of inputs from throttle, collective, and thrust vector controls should meet the values listed.

2.4.3. *Validation of data.* Different modes of operation are specified, in table 3, for STOL operation of V/STOL aircraft depending on the precision required for flight path control. As expected, the pilot desires improved vertical response time and g from power the closer he gets to the ground. To determine whether the criteria for Mode A or B apply, the pilot performs abrupt longitudinal control steps at the appropriate trimmed flight path angle. Compliance with the criteria is demonstrated by steps performed with the flight path control device while the aircraft attitude is maintained constant with the pitch control. Mode C applies equally to all aircraft regardless of the means to produce the response.

In tests of the BR 941 aircraft (ref. 17) engine response to small throttle changes had a 0.5 sec lag plus a first-order time constant of 0.7 sec. There was no appreciable lag between vertical g and power changes (i.e., no aerodynamic slipstream lag). It was possible with throttle alone to obtain more than ±0.1 g, which resulted in satisfactory flight path tracking down to about 15.24 m (50 ft). The pilot felt that longer engine time lags and time constants would have degraded his ability to track the ILS glide slope. This response was not adequate when he used power to arrest the sink rate at touchdown. In general, none of the STOL aircraft tested thus far (ref. 16) could be flared by using engine thrust because (1) engine response was too slow, (2) the aircraft had to be rotated for proper ground attitude, and (3) power changes produced undesirable changes in air speed. As a result, g was obtained, as for conventional aircraft, by rapidly increasing aircraft attitude. The touchdown maneuver for STOL aircraft is, of course, similar to the height control problem for VTOL aircraft. In this respect, values of overall thrust response should not be greater than 0.5 sec and 0.1 g should be available. The response for gross changes in flight path (away from the ground) are less stringent; for example, a 2.0 sec delay was considered satisfactory.

2.4.5. *Additional data requirements.* Admittedly, the vertical flight path criteria, in their present form, are weak, and more firm quantitative values are needed for both control power and thrust response. As is true for control of other axes, cross-coupling effects and interrelated items affect the pilot's assessment of precision of control. Included are the following:

- Static longitudinal stability
- Short period and phugoid frequency and damping
- Direct lift control
- Effe.. of automatic power compensation
- Ground effect on lift, drag, and pitching moment
- Gust sensitivity (lift curve slope)
- Power "backsidedness"
- Trim change with power (magnitude and direction)
- Thrust and control system response (lags)

A systematic evaluation of the foregoing items is a formidable task, and it is difficult to generalize on answers from specific aircraft because significant parameters cannot be varied over wide enough ranges. Steps are underway to examine the effects of these parameters on vertical flight path control using a piloted motion simulator at NASA Ames Research Center, at the RAE, Bedford, and by flight tests of the Bell X-22A aircraft.

2.5. Transition – Acceleration/Deceleration

2.5.1. *General background.* Good transition characteristics are essential for successful use of V/STOL aircraft for a number of reasons. First, it may be desirable to perform transitions quickly to minimize time spent in the terminal area. Second, transitions are usually performed in the critical landing approach phase of flight, where the pilot must be able to maintain precise control of flight path particularly for IFR operation. Finally, transistions occur during the pilot's peak work load, which includes making configuration changes such as selection of landing gear and flaps, starting lift engines, communications, and navigation duties. In the following paragraphs attention is given to those handling-qualities items that govern aircraft behavior in going from powered lift flight to aerodynamic lift regime and vice versa for both VTOL and STOL aircraft.

2.5.2. *Criteria.* VTOL aircraft should be able to accelerate rapidly and safely from hover to V_{con} in climbing flight or at constant altitude. From V_{con} they should be able to decelerate rapidly and safely at constant altitude or in a descent up to the maximum approach angle required by the mission, acquire and maintain both shallow and steep flight path angles, and stop quickly and precisely over a preselected hover spot. Depending on the mission, acceleration and deceleration values up to 0.5 g in level flight are desired. In addition, the ability to accelerate continuously from a rolling takeoff (RTO) to V_{con} and decelerate smoothly to a rolling landing is desirable.

STOL aircraft should be able to accelerate from V_{app} to V_{con} in level flight or climbing flight; decelerate quickly from V_{con} to V_{app}; and precisely acquire and maintain shallow and steep flight path angles.

It should be possible to carry out the above maneuvers with the precision and performance specified for the mission without restriction due to control power, trim, stalling or buffeting, engine thrust, or response characteristics.

The pilot should be required to operate only primary flight controls, power setting, and thrust vector tilt. If other devices required for transitions are operated automatically, it should be possible for the pilot to monitor their performance easily. Inadvertent operation of any transition control should be prevented.

2.5.3. *Discussion.* The purpose of these criteria is to ensure that in going from powered lift flight to aerodynamic lift flight and vice versa, the pilot can perform the necessary maneuvers as expeditiously as needed without undue attention to aircraft attitude, angle of attack, airspeed, and trim-factors that would compromise his ability to fly the aircraft accurately along a chosen flight path in all environmental conditions. Further, good control characteristics are needed for STOL operation when going in and out of ground effect because ground-induced recirculation may cause unsteady flow over the aircraft. In addition, the pilot should have the capability to decelerate as needed at any portion of the speed range to quickly attain a particular approach speed or to avoid overshooting a desired touchdown area.

The time required for making a transition can vary according to the mission; however, it is necessary from safety considerations that the rate desired by the pilot should not be governed by limitations in controllability about *any* axis. If the pilot must handle a large number of separate operations to accomplish the transition, his performance in terms of airspeed, angle of attack, and flight path angle control will suffer during this critical flight phase. Due consideration should be given to multicrew functions in transport configurations where, for example, lift engine startup and shutdown could be handled by a copilot.

2.5.4. *Validation of data.* Operation of various VTOL and STOL aircraft indicate that the V/STOL concept itself has certain built-in limitations with the acceleration/deceleration handling characteristics. Further, these characteristics vary depending on the direction of transition. Typical acceleration and deceleration characteristics are shown in figures 8 and 9 for several V/STOL aircraft.

The P.1127 aircraft, for example, is equipped with a proportional-position thrust vector control that operates only on the engine thrust vector. The magnitude and direction of the aerodynamic (lift and drag) vectors are controlled indirectly through changes in aircraft attitude. The pilot, therefore, can change the magnitude and direction of the engine thrust vector independently on the aerodynamic vectors. As discussed in reference 23, the rate at which the proportional thrust vector control was moved related directly to the magnitude of the vector. When a large engine thrust vector was used (e.g., during takeoff), a rate of approximately 4°/sec was selected. (Note that 90°/sec is available.) This provided an initial acceleration of approximately 0.2 g and an overall average acceleration (0 to 160 knots) of 0.43 g. A higher thrust vector rate would have produced higher accelerations but a loss in altitude since aerodynamic lift could not be gained rapidly enough to offset the change in vertical thrust. During a decelerating transition (160 knots to 0), however, the pilot commanded a thrust vectoring rate of approximately 45°/sec. This was possible, of course, because of the small magnitude of the engine thrust vector. A typical decelerating transition was initiated at 160 knots with +6.5° pitch attitude and a low-power setting. From 160 to 80 knots, a maximum deceleration of 0.46 g was attained. At 80 knots the thrust vector was rotated

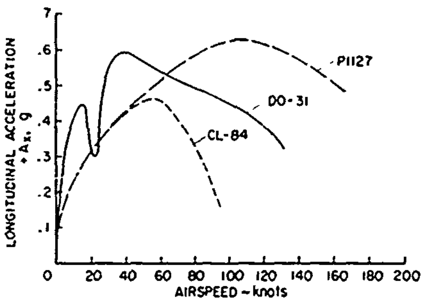

Fig. 8. Accelerating transition characteristics for several V/STOL aircraft.

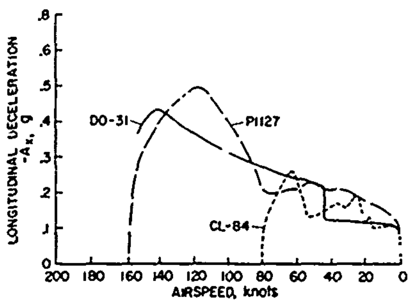

Fig. 9. Decelerating transition characteristics for several V/STOL aircraft.

from the 5° forward position to the vertical position after which the aircraft pitch attitude was increased to +14° to decelerate from 80 knots to zero with an average deceleration of approximately 0.2 g.

In tilt-wing aircraft, such as the CL-84, the aerodynamic vector is rotated with the engine thrust vector. The pilot, therefore, must command a thrust vectoring rate that is compatible with the magnitude of the aerodynamic vector and of the engine thrust. Further, maximum thrust vectoring rate is a function of wing angle and the direction of thrust vectoring rate is a function of wing angle and the direction of thrust rotation. The CL-84 wing could be rotated up at a rate of 6°/sec. The maximum downward rate of 12°/sec was linearly decreased to 2.6°/sec between wing angles of 45° and 5°. The pilot did not have direct control of thrust vectoring rate because his control was only an on/off switch. The approximate thrust vectoring rate desired could be achieved by intermittently turning the switch on and off.

In an accelerating transition the pilot commanded a vector rate of approximately 7°/sec, which produced an initial acceleration of 0.2 g. After a brief 2-sec period the pilot commanded maximum thrust vectoring rate for the remainder of the transition. This produced a maximum thrust vectoring rate of approximately 10°/sec and a maximum acceleration of 0.44 g. Since the initial aerodynamic vector is small in this accelerating transition, a high thrust vectoring rate could be used without experiencing control coordination problems. In this respect the CL-84 is very similar to the P.1127.

Decelerating transitions of the CL-84 tilt-wing aircraft is completely different, however, because the pilot is required to manage the control coordination problem caused by tilting the large aerodynamic vector. This requires selecting a wing tilt rate that is compatible with the aerodynamic vector and the magnitude of the engine thrust vector. This completely unfamiliar technique (as stated in ref. 12) was difficult to perform. It was further complicated by the need to operate the wing-tilt switch intermittently to get a variable rate to match the lift required. Holding deceleration at any fixed rate was thus very difficult. A typical decelerating transition shows that the pilot commanded a thrust vectoring rate of 3°/sec for the major portions of the maneuver (15° to 60°) and then commanded a maximum available rate of 6°/sec for the remainder of the transition (60° to 86°). This produced a nearly constant deceleration of 0.15 g. The aircraft was capable of higher decelerations, but the pilot control coordination problems increased. Different characteristics are shown for the fan-in-wing XV-5A aircraft (ref. 11). At low speed, the wing fan louvers are used to control height, roll, yaw, and speed (thrust vectoring). In addition, the angle of the louvers determines the amount of roll control available to the pilot (roll control is phased out as a function of louver angle as speed and aileron control increase). Specific attention was required to insure that a "rule-of-thumb" relationship of 2 knots of airspeed for each degree of louver angle was maintained to avoid a loss of lateral control power. A high degree of pilot attention was required to maintain the louver angle-airspeed schedule (a pilot rating of 5 was assigned). The maximum thrust vectoring rate built into the XV-5A aircraft was 3°-4°/sec. During an accelerating transition from hover, the pilot commanded an overall average thrust vectoring rate of 1.6°/sec and an acceleration of 0.13 g.

2.5.5. *Additional data requirements.* There is enough data to show that one minimum or maximum rate will not satisfy all VTOL concepts, but there is not enough data to establish a satisfactory rate for each. In addition, the limitations for IFR operation have not been clearly defined. It is to be expected that only relatively low deceleration values will be used to reduce pilot workload in the landing approach task. Early experience with the DO-31 aircraft indicate that deceleration values of 0.07 g were used to provide sufficient tracking time on the ILS to assess the approach such that confidence is gained to proceed to the landing. Further real life operation is needed to assess the passenger comfort aspect for civil use.

3. CONCLUDING REMARKS

Revised V/STOL handling qualities criteria have been prepared to provide updated information that reflects recent requirements of operational-type aircraft, the peculiarities of operating with different types of lift-propulsion concepts, and the effects of operation with novel control systems.

A review of several controversial areas indicates that although improved guidelines have been set down and some form of quantitative criteria are available for most areas, additional information is needed to refine the criteria for operational IFR use.

Some of the areas that need further refinements include (1) control requirements as affected by the mission and task, (2) control power and control usage for various types of control systems, (3) the amount of longitudinal static stability needed in the powered-lift flight regime, (4) cross-coupling effects about all axis, (5) vertical flight path control in landing approach, and (6) transition/acceleration-deceleration characteristics. Further, additional work of a systematic nature must be conducted to clarify the effects of several interacting items that strongly influence the pilot's over-all impression of the aircraft's behavior.

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MISSION EFFECTS ON STABILITY AND MANEUVERABILITY

by

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SUMMARY

This paper has the objective of defining the relationship between the mission requirements of a piloted aircraft and its stability and maneuverability. The framework utilized in current U.S. Air Force handling qualities requirements, i.e., classification of aircraft, flight phases, levels, states, etc., is described. Examples of various aircraft designed for one mission and then utilized for other missions are given. A discussion is presented of the problems encountered when the detailed mission requirements are not clear, such as with V/STOL aircraft, reentry vehicles, etc. Problems encountered with off-design conditions and operation at the limits of the flight envelope are discussed with examples. The various methods open to the designer for achieving the proper compromises in design of an aircraft are outlined.

INTRODUCTION

This paper has the objective of defining the relationship between the mission requirements of a piloted aircraft and its stability and maneuverability. All aircraft, at their inception, have their mission requirements set up in either broad or narrow terms, definitive or vague. From these statements, together with the many other subsystem requirements and design guides, the designer must proceed to make the tradeoffs needed to achieve the best solution to the problem presented to him. Often these mission requirements change either during the design or during operational use. Examples of aircraft designed for one mission and then utilized for other missions are given. A discussion is presented of the problems caused by off design conditions, operation at the limits of the envelope, different operational tactics, failures modes, economic factors, etc. The problems encountered when the detailed mission requirements are not clear such as with V/STOL and reentry vehicles are also discussed.

The framework and philosophy utilized by the U.S. Air Force in current handling qualities requirements to reflect properly the mission requirements is shown. The goal of this framework and philosophy is to allow for adequate definition of the needed stability and control so that the required characteristics can be assured and yet avoid over specification with resulting penalties to the aircraft. Some of the methods open to the designer of achieving the proper compromises in design of an aircraft are also outlined.

DEFINITION OF MISSION

Even before the mission requirements reach the aircraft designer, many tradeoffs have been made by the customer, based on past operational experience, evaluation of the threat, consideration of present or future weapons and tactics, new technology possible for application, and many other factors. These tradeoffs may have been based on detailed knowledge or intuition; be brilliantly visionary or naively hopeful; be detailed or vague. In any event these reach the designer as requirements that he must meet. In many cases underlying requirements may not be expressed. Requirements difficult to quantify are often downgraded or not stated and little help may be given the designer in how essential a particular requirement is to the basic job the aircraft is to do.

The mission requirements may be written very simply and broadly. For example, the requirements prepared by the U.S. Signal Corps in 1907 for the procurement of a heavier-than-air flying machine were very simple and the resulting requirements on stability and control were contained in one sentence. "During the trial flight of one hour it must be steered in all directions without difficulty and at all times be under perfect control and equilibrium".

Research aircraft tend to have very broad and vague mission requirements. The intent of these aircraft is to explore and expand technology and it is to be expected that the mission requirements will be technology oriented. The stability and control requirements, likewise, tend to be subordinated to questions primarily of flight safety. The recent trend towards prototype procurement leads to broad statements of mission requirements. However, if the prototype configuration is later expected to be able to perform other missions than contemplated by the designer, large compromises may result.

Mission requirements may be written for a very narrow and well defined mission. The requirements may be very clear cut and quantitative. Such a case is the Concorde. Other examples that would fit the category of narrowly defined missions include the U-2.

"Off the shelf" aircraft are often procured by the military services for various uses. Generally these aircraft have been in the transport or utility classes. In many cases, the use of the aircraft is essentially identical with commercial use. Examples are the C-9 (DC-9), C-131 (340 and 580), C-140 (Jetstar), and others. In these cases, compromises in their military use are minimal. In other cases, extensive modifications are made to the aircraft and it may be used operationally in ways that were not contemplated by the designer. Examples are the KC-135 (707), AWACS (707), P-3 (Electra), etc. Compromises that may then result must be traded off with such factors as the economic benefits of using available aircraft.

Certain combat aircraft have started their design cycle with detailed mission requirements covering a broad range of uses of the aircraft. Penalties and rewards were specified for those items that could be so quantified, the aim, of course, being to induce the designer to make every effort to achieve those goals. Examples would include the F-111 and the C-5.

CHANGES IN DESIGN MISSION

The history of manned aircraft is replete with examples of aircraft designed for one mission and then later used for another. Possibly this should not be cause for any surprise. As new regimes of flight are explored and new configurations evolve the threat or problem to be solved varies, is it any wonder that the aircraft is used differently than planned? With the five to ten years involved from the concept to operational use of an aircraft, an extraordinary perception would be involved if the mission were to remain identical in detail. A few examples will be given to illustrate the point made above about changes that often occur in the original design mission.

The B-47 was a high-altitude, horizontal bomber, originally. Its very flexible wings were adequate for that mission, but then came low-altitude penetration and lofted bomb delivery. The maximum speed of the airplane on the deck was limited by aileron reversal: during design, no need had been seen to fly so fast at that altitude. B-47 pilots also experienced some difficulties recovering from unusual attitudes in attempted bombing maneuvers. Yet another difficulty was air-to-air refueling with the then-standard tanker, the KC-97. The performance mismatch of the propeller and jet airplanes had the B-47 flying not far from stall at the KC-97's top speed.

Like the B-47, the B-52 heavy bomber had a low-altitude mission added. It also started its service with a near-incompatible KC-97 tanker for a refueling partner. Poor Dutch roll characteristics with the original yaw dampers aggravated the refueling problem, though not to the point of spending money immediately to fix the airplane. The poor damping, it later was found, actually affected the fatigue life of the airplane. To extend the life of the B-52 fleet, the airplanes have been rebuilt several times, improving the Dutch roll damping in the process.

The F-105 was designed primarily for strike missions using nuclear weapons. It has been used operationally as a strike aircraft but the way the mission is performed is not at all as first envisioned. The original sophisticated fire control system found little use in southeast Asia, where F-105 pilots used iron sights to drop iron bombs. Credible performance of this task under very trying conditions says much about both the airplanes and their pilots. One glaring deficiency that developed stems from the original concept of the F-105's use. It was designed to survive as well as possible in an environment of nuclear weapons. Vulnerability to small arms fire, although recognized, was not considered a design objective. In places the hydraulic systems were routed side-by side; thus a single projectile could cause a fire that would burn through all the hydraulic lines, leaving the flight controls powerless. Original design consideration could have brought about a much better and more economic solution than the fixes that were made.

The F-4, our current first line fighter was developed originally for the U.S. Navy. The first mission was as a long range attack aircraft. Shortly after, the mission was changed to that of a missile carrying fighter. Currently, various models of the F-4 serve in all weather, air superiority, ground attack, and reconnaissance missions. Because of the early use of the aircraft deficiencies of the aircraft at high angles of attack were not considered to be critical. With the change in use of the aircraft these characteristics have assumed a great deal more importance and corrective action was necessary. Early consideration would have been far more effective and saved a considerable number of aircraft.

Modifying existing aircraft to have STOL performance capability may be regarded as an extreme change in mission. This has not been successfully accomplished to date. The addition of high lift capability has usually required extensive development of the control system and an augmentation system to cure handling qualities problems. One example is the NC-130B. The unaugmented lateral-directional characteristics degraded to unacceptable at the reduced operating speeds.

The F-111 was designed to perform many missions, some of them exceedingly demanding. The resulting aircraft, naturally, is complex, large, heavy, and expensive. Any aircraft such as the F-111 designed for diverse missions certainly will suffer by comparison with an aircraft designed for more compatible uses.

The C-5A is an example of another sort. The C-5A was designed for a wide variety of transport uses. Included was a requirement for a low altitude-terrain following-cargo drop mission and a requirement for an ability to land and take off on unprepared fields. The requirements are entirely rational and desirable if they could be attained without excessive compromise. These capabilities were designed into the aircraft at considerable penalty. Besides the weight and drag there was considerable complication added to the flight control system to assure good flying qualities. Currently these capabilities are not being utilized. This example illustrates that restraint must be exercised by the customer in asking for too many "good" things or he may not appreciate the resulting product. Examples could be selected from the aircraft of the World War II era, examples that come readily to mind are the Mustang and the B-26. The B-36 lived through its life as a high altitude horizontal bomber but even here extensive modification was made to attempt to preserve its effectiveness.

The question might well be asked, Has any aircraft been utilized as originally conceived and designed, with some trainers and cargo aircraft possibly excepted?

OFF DESIGN CONDITIONS

Operational flight envelopes can be drawn to define the boundaries of speed, altitude, and load factor within which the aircraft must be capable of operating in order to perform its mission. Such an envelope for a typical fighter in the combat flight phase is shown in figure 1. Within the operational envelope the aircraft should have very good flying qualities. There are many conditions outside of this operational envelope which an aircraft can easily attain. What should the mission requirements state about these conditions? Certainly they may affect the flight safety. Another envelope can be drawn which represents flight conditions that can be encountered without exceeding airplane limitations. Beyond this envelope is a boundary which the airplane is capable of safely encountering. Stall, spins, zooms, and some dives may be representative of such conditions. The buffet characteristics, engine limits and many such factors may set these limits. Characteristics in the transonic range may be tolerated which would not be in other regimes. This will be dependent on the importance of this range of flight to the combat mission.

Aircraft have often been designed with internal stores or armament or a few external pods and then used in quite a different manner. Stores have been loaded on in many combinations and permutations, making somewhat of a mockery of the careful aerodynamic design. Very large effects on stability and control and flying qualities are obvious when one views the range of munitions and weapons which are loaded on an A-4, A-7 or F-4. Such effects as the ability of the aircraft to get rid of its stores whenever needed must be given consideration. Gunfire effects may cause disturbances to the aircraft which must be considered in the design tradeoff process, even to such extremes as causing engine stall.

Use of flight control systems of various kinds to modify and improve the basic stability and control characteristics of aircraft is a clear trend and an accelerating one. When this equipment operates as designed the mission performance is probably met and the pilot may be very satisfied. What should the mission requirements say about the tolerable frequency of failure, failure effects, etc.? Should multiple failures be considered? How far should requirements go in considering failures that may have extremely remote probability of failure, even probabilities similar to basic structure? Do errors by the pilot need to be considered or delays in his response to an emergency? Is it satisfactory to turn over to a pilot in an emergency a marginally stable or unstable aircraft that has exceeded a boundary or experienced an equipment failure of some kind? Decisions on such questions can be made by the designer deliberately. If ignored, this is still a decision.

OTHER EFFECTS

There are many other factors that have an influence on the stability and control requirements. Some of these factors may be stated to some degree in the mission requirements; others are implied by the configuration and subsystems that are likely to result.

One of the more obvious of these factors is the concept of use of the aircraft. What weapons are contemplated, missiles, rockets, guns, or a mix? Is the aircraft to be used in "dog fight" or "stand off" tactics? What enemy environment must the aircraft operate against? An example from the past illustrates this point very well. In 1957 the Air Force was under considerable attack to reduce its roll rate requirements for fighter aircraft. Research performed at NASA had indicated that for the missions contemplated the roll rates being required were grossly excessive. High roll rate does cause a penalty in the aircraft and as a result of the studies by Harry Goett of NASA, a re-evaluation of the roll requirements was made. The research was essentially correct in its conclusion, with the concept of fighter engagement consisting of firing long range missiles. In the same time frame, official Air Force policy was indicating that the last manned fighter would be the F-103. During the reevaluation, even the airframe manufacturers took a very cautious attitude on retreating from previous roll rate requirements. A small reduction was made, however, the wisdom of avoiding any drastic changes based on logical analysis using assumptions that did not hold true, is self evident. No clairvoyance is claimed; stubborn conservatism can be equally wrong.

An area that is likely to be treated very lightly in the mission requirements is that of turbulence. The turbulence environment that the aircraft is expected to operate within will have very strong influences on the stability and control, the flight control system, the displays, the airframe structure, and in extreme or long continued environments even the ability of the pilot to perform. At what level of turbulence do we expect missions to be performed? If the turbulence is above this level, what should the pilot do -- abort the mission or accept the degradation in performance? Discrete as well as random turbulence must also be considered. Cross wind landing and takeoff characteristics have been critical on some of our aircraft. The effects of discrete gusts are critical on some of our aircraft. The effects of discrete gusts are especially important for V/STOL aircraft.

Recent aircraft procurement contracts have attempted to set requirements on reliability and maintainability, subjects of vital concern once the aircraft is in operational use. It is a very difficult task, however, to make clear what the Air Force requirements are and what the tradeoffs with other characteristics should be. A related point concerns what is called the "amber light" problem. With sophisticated and redundant flight control systems, deficiencies in aircraft can be corrected. With a failure in one element in the system, the system continues to operate but the "amber light" glows, indicating to the pilot a potential hazardous condition. The pilot normally aborts the mission. With the number of elements in some systems the number of such indications can become discouragingly high. What do the mission requirements state as an acceptable level for such a situation? With no definitive statement, the designer may very well ignore this problem.

That there is extensive interface between the various subsystems of a modern aircraft is an obvious truism. It is equally true that the various specialists and subsystem engineers tend to solve their problems somewhat independently unless constrained by clearly defined mission requirements or a very wise chief designer. For example, the interaction between the propulsion subsystem and the flight control system is very important but sometimes requirements important to one area do not get transmitted to another. It becomes very embarrassing to the pilot, if in a spinning condition, the engine flames out, rapidly runs down, with resultant loss in hydraulic power. Requirements to prevent this situation do not currently exist.

In response to a set of mission requirements a designer may evolve a design with a great deal of airframe flexibility. Another designer may come up with a design with a greater or lesser amount but for many missions there may not be much variance. Similarly, the location of the pilot at some distance from the c.g. may be the natural configuration that different designers would arrive at. Both of these examples may have an effect on the ability of the pilot to perform his job, possibly compromising the mission. Such effects can compound, witness the B-70 pilot reporting turbulence with the chase aircraft reporting none.

One final factor of a different sort that will be discussed concerns the use of operational aircraft for training. Hopefully, operational aircraft will live their life through without actual combat use; some have. What does this mean to mission requirements? One example that can be cited concerns the way in which external fuel tanks are utilized in peace time. In the interest of economy, dropping of such tanks has been avoided. This has obvious implications to the mission, tactics, the airframe, and its subsystems. Other such examples can be cited, the point being that the mission requirements or the designer must consider such factors or the operator will have to accept the consequences.

V/STOL VEHICLES

Consideration of mission requirements for V/STOL vehicles introduces the point that there is almost no operational experience on which to base the mission requirements of V/STOL vehicles except for helicopters. Consequently it becomes difficult to determine the required stability and control characteristics. This point can best be illustrated by an example from our recent experience in the development of Specification MIL-F-83300, "Flying Qualities of Piloted V/STOL Aircraft", Reference 7. It became apparent in our discussions of required characteristics for speeds between hover and conversion to conventional flight that our thinking was conditioned towards a mission that involved the pilot moving from takeoff to conventional flight as rapidly as possible. This may well be the case for some V/STOL vehicles such as cargo or transport types. To limit a V/STOL vehicle away from maneuvering in the low speed regimes, even to sideways or backward flight, may be to deny it from capability that is unique to such aircraft and possibly very valuable tactically. However, without operational experience to validate such ideas, it becomes very difficult to judge the worth of requiring such capability.

In a similar vein, much of our thinking is conditioned by configurations that have to tilt to translate. If there is an operational utility to vehicles that do not have to tilt to translate, the resulting mission requirement will have a considerable effect on stability and control requirements.

Additional examples that can be enumerated where additional knowledge of the mission requirements of V/STOL vehicles are needed, include maneuvering in turbulence, characteristics in engine failure conditions, and IFR flight requirements.

SPACE VEHICLES

With aerospace vehicles we face the same problem as with V/STOL vehicles, i.e., limited operational experience. Many of the space missions fall more into a category of research aircraft missions, where the main objectives are to explore technology. In attempting to prepare a general handling qualities criteria document, Reference 8, this lack of definite mission requirements was a most severe handicap. What is it that you wish to do with an aerospace vehicle, perform reconnaissance? If so, how accurately must the tracking or stabilization be? Do you wish to maneuver, if so, how rapidly? For aerospace missions to date, the weather conditions, the turbulence that must be considered, and other such mission factors are held to the most favorable condition.

A most fundamental question related to such vehicles is the question of how often do we wish to perform this mission, whatever it is. Ultimately this question translates back into the economics of the situation. Can we afford to have the U.S. Navy on standby for the mission? The magnitude of the support operation that can be tolerated and the extent to which every day regular operation is expected translates directly back to stability and control requirements and of course to other subsystem requirements also.

The designer is faced with integrating all the stated and unstated mission requirements that have been discussed above, into a machine. In many cases, the requirements are fuzzy, not defined, and contradictory. How can he provide the flexibility that is needed to adapt to this situation without creating something that is a "jack of all trades and master of none"?

FRAMEWORK OF U.S. HANDLING QUALITIES REQUIREMENTS

The framework of the U.S. handling qualities requirements document (Reference 2) was designed to make use of all the knowledge about mission requirements that is available or should be available. The framework allows the flexibility in use that will be necessary in practical use. Reference 3, "Background Information and User Guide" provides the additional information needed for intelligent utilization.

The requirements have a framework based on the following considerations:

1. the kind of airplane (Class)
2. the job to be done (Flight Phase)
3. how well the job must be done (Level)

Figure 2 indicates the division of aircraft into several classes. Historically flying qualities specifications have recognized the need to specify different values of parameters for vehicles of different size and different operational missions. It is intuitive to expect the handling qualities of sport cars to be different from those of trucks, speed boats to handle differently than ocean liners, and small utility airplanes to fly differently than large transports. In addition, there may be significant differences in the way each vehicle responds to external disturbances such as road roughness, sea state, and atmospheric turbulence or wind. The quantitative requirements of the specification are specified as believed necessary for the various Classes. At the inception of a design the procuring agency decides to which Class the new aircraft belongs and then the proper requirements apply. In most cases this assignment to a Class is obvious.

Figure 3 indicates the division into Flight Phases that is utilized. Experience with airplane operations indicates that certain Flight Phases require more stringent values of flying qualities parameters than do others (e.g., air-to-air combat requires more Dutch roll damping than does cruising flight). In many instances, therefore, the flying qualities specification should state requirements as a function of mission Flight Phase. This degree of breakdown gives the designer additional guidance in optimizing his design. For the most part, the Flight Phase titles are descriptive enough to determine those applicable to a given design. The similarity of tasks in many Flight Phases, plus the limited amount of evaluation data on specific Flight Phases, led to grouping the Phases into three Categories. First, the possible Flight Phases were divided into two groups on the basis of terminal and nonterminal operation. Then Non Terminal flight was further divided into two groups based primarily on the degree of maneuverability and/or precision of control required. The requirements of Reference 2 are generally stated in terms of these three Flight Phase categories, however, a number of requirements are directed at Specific Flight Phases. Not all of the Flight Phases apply to a given airplane, thus the procuring agency may delete Phases and may also add Phases as new mission requirements are generated.

Figure 4 gives descriptive words to define the levels of flying qualities, where possible the specification states the requirements in terms of three values of the parameter being specified. Each value is a minimum to meet one of the three Levels of acceptability. There is a relationship between these Level definitions and the Cooper Harper pilot rating scale. For further discussion of this relationship, see Reference 3.

Figure 5 gives the framework for relating these three considerations. It illustrates that use of this framework would permit stating 36 different values for a given flying qualities parameter, even after combining the Flight Phases into the three Categories A, B and C. Seldom will such a fine breakdown be required, nor will there be sufficient information available to make such fine discriminations. Thus, in most cases, the 36 possible requirements are combined to some extent, but not necessarily in the same pattern for all requirements.

There are many factors involving the configuration of an aircraft, loading, control positions, etc. that must be considered when specifying requirements. The concept of Airplane State has been introduced in the specification to aid in codification. The State of the airplane is defined by the selected configuration, together with the functional status of each of the airplane components or systems, throttle setting, weight, moments of inertia, center of gravity position, and external store complement. The trim setting and positions of the rudder, aileron, and elevator controls are not included in the definition of Airplane State since they are often specified in the requirements.

A crew selected configuration is defined by the positions and adjustments of the various selectors and controls available to the crew except for the rudder, aileron, elevator, throttle and trim controls. Examples are the flap control setting and yaw damper, ON or OFF. Selected configurations to be examined under the specification must consist of those required for performance and mission accomplishment.

The specification required consideration under all loading conditions associated with the airplane's operational missions. The loading is determined by what is in internal loading and attached to (external loading) the airplane. The parameters that define different characteristics of the loading are weight, center of gravity position, and moments and products of inertia. External stores affect all these parameters and also affect aerodynamic coefficients. Since there is an almost infinite number of possible loadings, each requirement is generally only examined at a critical loading. Additional guidance on this area is presented in Reference 3.

Under the specification the contractor is required to describe the Normal States associated with each of the applicable Flight Phases. This tabulation is required to be in the format of Figure 6. Certain items such as weight, moments of inertia center of gravity position, wing sweep, or thrust setting may vary continuously over a range of values during a Flight Phase. The contractor is required to replace this continuous variation by a limited number of values of the parameter in question which will be treated as Specific States, and which will include the most critical values and extremes to be encountered.

The specification requires that envelopes be drawn as shown in Figure 1. Operational Flight Envelopes are regions of speed-altitude-load factor space, where it is necessary for an airplane, in the configuration and loading associated with a given Flight Phase, to have very good flying qualities, as opposed for example to regions where it is only necessary to ensure that the airplane can be controlled without undue concentration. The Operational Flight Envelopes are intended to permit the design task to be more closely defined and to reduce the cost and complexity of the airplane to essentials.

Service envelopes are also to be drawn which surround the Operational Envelopes. Its larger volume denotes the extent of flight conditions that can be encountered without fear of exceeding airplane limitations. Requirements are less severe than in the Operational Flight Envelopes but still stringent enough that the pilot can accomplish the mission Flight Phase associated with the Airplane Normal State. Mission effectiveness or pilot workload, or both, however, may suffer somewhat even with no failures. This envelope is intended to insure that any deterioration of handling qualities will be gradual as flight progresses out from the limits of the Operational Flight Envelope. This serves two purposes. It provides some degree of mission effectiveness for possible unforeseen alternate uses of the airplane and it also allows for possible inadvertent flight outside the Operational Flight Envelope.

Permissible Flight Envelopes are to be drawn to encompass all regions in which operation of the airplane is both allowable and possible. These are the boundaries of flight conditions outside the Service Flight Envelope which the airplane is capable of safely encountering.

In Figure 7 are shown the requirements of the specification with respect to application of the Level concept for Airplane Normal States. From all points in the Permissible Flight Envelope it shall be possible to readily and safely return to the Service Flight Envelope without exceptional pilot skill or technique.

The specification establishes a procedure to consider effects of various malfunctions on the handling qualities. The contractor is required to define and tabulate Airplane Failure States which consist of Airplane Normal States modified by one or more malfunctions in airplane components or systems. There is more to determining Failure States than just considering each component failure in turn. Two other types of effects must be considered. First, failure of one component in a certain mode may itself induce other failures in the system, so failure propagation must be investigated. Second, one event may cause loss of more than one part of the system. When Airplane Failure States exist, a degradation in flying qualities is permitted only if the probability of encountering a lower Level is sufficiently small. This requirement is shown in Figure 8. In no case shall a Failure State (except an Approved Special Failure State) degrade any flying quality outside the Level 3 limit. The concept of Special Failure States was introduced to allow for components, systems or combinations that may have extremely remote probability of failure but may be very difficult to predict with any accuracy. By approval of the procuring agency such conditions may be excepted from considerations. Certain items might be approved as Special Failure States, more or less categorically, such as dual mechanical failures or basic airframe or control surface failure. In other cases a considerable amount of engineering judgment may be required.

From the foregoing it is clear that an elaborate framework has been created. The complexity and the many parameters in this framework are dictated by the complexity of the task of defining the job that the designer is faced with. Any attempt to gloss over or ignore some of the important parameters can only lead to inferior designs. It is obvious that engineering judgment and teamwork between the contractor and the procuring agency must be liberally exercised if excessive and unnecessary analyses are to be avoided. It is our intent that this will be the case.

ACHIEVING PROPER COMPROMISES

The designer has a most difficult task in taking into account the widening spread of characteristics, missions, conditions and regimes of flight, states, automatic and emergency modes, etc. and making the necessary tradeoffs. U.S. procurement practices of the recent past, in which fixed price contracts were awarded and in which the design and development was speeded up have aggravated the designer's task. With this procedure the contractor is under extreme pressure to meet his schedule and to meet the definitive guarantees of the contract. Further, there was a tendency for reduced interaction, and sharing of the problems, with the Air Force engineers. The somewhat more leisurely process of design and redesign during prior years, all with a cost plus fixed fee base, was more tolerant of changing mission requirement and loose criteria.

It is not made entirely clear to the contractor what the Air Force really wants, a safe, effective, maintainable, reliable low cost flying machine. Many of the important decisions on tradeoffs which must be made early in the design stage are left almost entirely up the contractor's judgment of what we want and how we intend to use the vehicle. We either have to make our criteria much better or give the contractor more of the total picture of the use of the vehicle. It is highly important that a rapport be established between the contractors' design team and the government engineers, based on mutual respect and confidence, so that problems and questions are solved as they arise.

As pointed out in Reference 1 there is no very tangible reward for a contractor who achieves a design with excellent stability and control or achieves an optimum tradeoff of flight control system-airframe characteristics. There is no effective penalty for doing a poor job. Basically, stability and control provisions cost the designer weight and drag. If definitive "pay off" functions related to mission effectiveness, safety, reliability and maintainability can be specified, the designer would be able to make intelligent tradeoffs. Such definitive functions must be found and specified.

In the early design stage the mission requirements and the criteria are loose and subject to argument and no amount of work will eliminate all of these cases. In cases where the requirements and criteria are hurting the overall design the contractor will naturally search for all the relief he can get. At the present time we often do not know if he has achieved an acceptable solution or not until the aircraft has flown. At that time it is too late to do anything, unless it is a clear cut and absolute safety of flight item.

Several factors offer some hope of alleviation of some of the problems of achieving proper compromises. Analytical capability has improved immensely in the past few years. Mathematical formulations and the computer capability to go with these formulations are now available. A difficult problem is still present in determining the proper aerodynamic input data, especially in the early design stages.

The other hopeful factor is the rapidly improving capability for simulation, both ground and airborne. If the simulation capability that is now entirely feasible were built and properly utilized in the early design stages of development, many vexing and difficult decisions could be worked out in the laboratory prior to construction of the prototype. The optimum solution is a mix of advanced techniques of analyses and use of sophisticated simulation techniques.

SUMMARY AND CONCLUSIONS

In the foregoing discussion some of the multitudinous and varied factors that must be considered in specifying the mission requirements for an aircraft have been discussed using examples from the manned aircraft of the recent past. Examples of changes in mission requirements from the original concept to actual operation have been given. The manner in which the U.S. military specification has attempted to relate the stability and control and handling qualities of the aircraft to mission requirements has been outlined. The intent is to provide an aircraft with characteristics necessary to perform the mission but without unwarranted penalty to performance or other characteristics. Some possible methods for achieving proper compromises and tradeoffs in the design of new aircraft have been suggested.

It is recognized that more questions have been raised and problems stated than answers provided. This is inevitable in such a broad and complex subject as the title of this paper.

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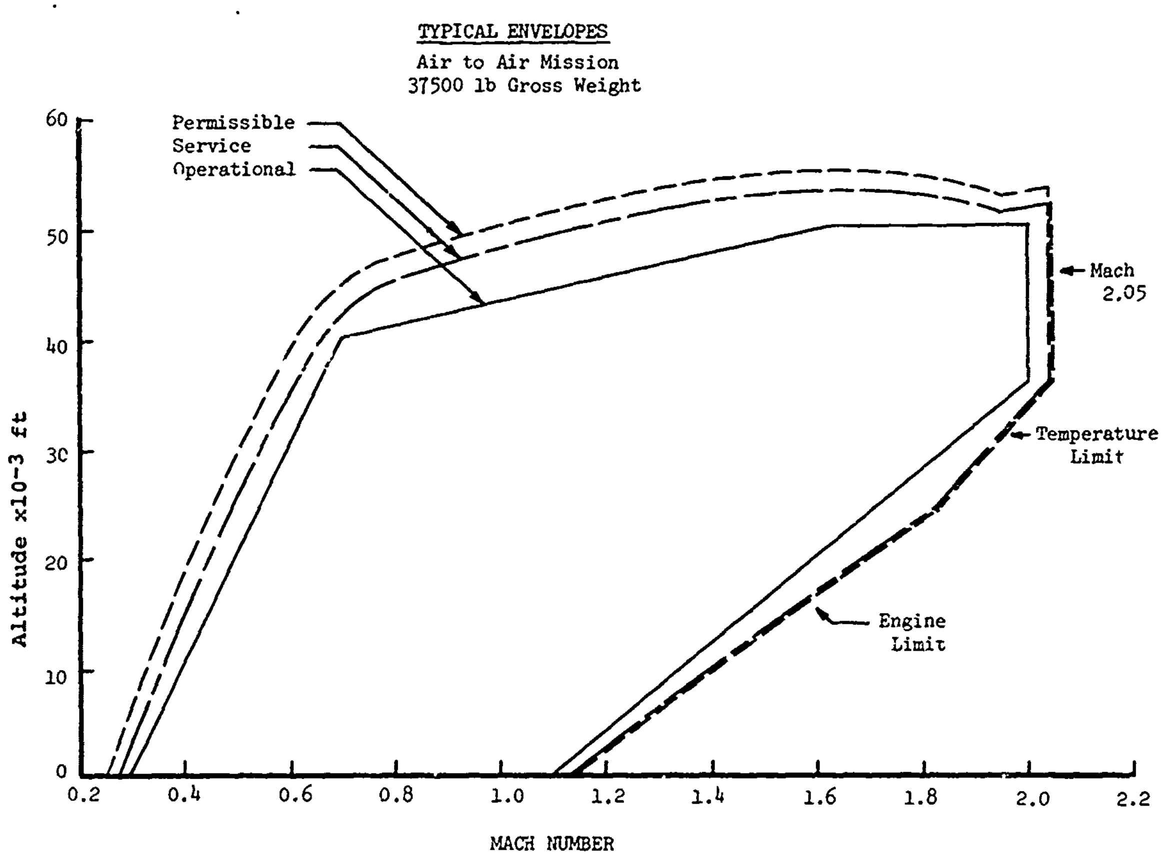

FIGURE 1

CLASSES OF AIRCRAFT

CLASS I — Small, light airplanes such as
- Light utility
- Primary trainer
- Light observation

CLASS II — Medium weight, low-to-medium maneuverability airplanes such as
- Heavy utility/search and rescue
- Light or medium transport/cargo/tanker
- Early warning/electronic countermeasures/airborne command, control, or communications relay
- Antisubmarine
- Assault transport
- Reconnaissance
- Tactical bomber
- Heavy attack
- Trainer for Class II

CLASS III — Large, heavy, low-to-medium maneuverability airplanes such as
- Heavy transport/cargo/tanker
- Heavy bomber
- Patrol/early warning/electronic countermeasures/airborne command, control or communications relay
- Trainer for Class III

CLASS IV — High-maneuverability airplanes such as
- Fighter/interceptor
- Attack
- Tactical reconnaissance
- Observation
- Trainer for Class IV

FIGURE 2

PHASES

Nonterminal Flight Phases:

Category A - Those nonterminal Flight Phases that require rapid maneuvering precision tracking, or precise flight-path control. Included in this Category are:

a. Air-to-air combat (CO)
b. Ground attack (GA)
c. Weapon delivery/launch (WD)
d. Aerial recovery (AR)
e. Reconnaissance (RC)
f. In-flight refueling (receiver) (RR)
g. Terrain following (TF)
h. Antisubmarine search (AS)
i. Close formation flying (FF)

Category B - Those nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this Category are:

a. Climb (CL)
b. Cruise (CR)
c. Loiter (LO)
d. In-flight refueling (Tanker) (RT)
e. Descent (D)
f. Emergency descent (ED)
g. Emergency deceleration (DE)
h. Aerial delivery (AD)

Terminal Flight Phases:

Category C - Terminal Flight Phases are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this Category are:

a. Takeoff (TO)
b. Catapult takeoff (CT)
c. Approach (PA)
d. Wave-off/go-around (WO)
e. Landing (L)

FIGURE 3

LEVELS OF FLYING QUALITIES

Level 1	Flying qualities clearly adequate for the mission Flight Phase
Level 2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists
Level 3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.

FIGURE 4

FRAMEWORK FOR STATING FLYING QUALITIES REQUIREMENTS

Class	Flight Phase Category	Level		
		1	2	3
I	A			
	B			
	C			
II	A			
	B			
	C			
III	A			
	B			
	C			
IV	A			
	B			
	C			

FIGURE 5

AIRPLANE NORMAL STATES

Flight Phase		Weight	C.G.	External Stores	Thrust	Thrust Vector Angle	High Lift Devices	Wing Sweep	Wing Incidence	Landing Gear	Speed Brakes	Bomb bay or Cargo Doors	Stability Augmentation	Other
Takeoff	TO													
Climb	CL													
Cruise	CR													
Loiter	LO													
Descent	D													
Emergency Descent	ED													
Emergency Deceleration	DE													
Approach	PA													
Wave-off/ Go-Around	WO													
Landing	L													
Air-to-air Combat	CO													
Ground Attack	GA													
Weapon Delivery/ Launch	WD													
Aerial Delivery	AD													
Aerial Recovery	AR													
Reconnaissance	RC													
Refuel Receiver	RR													
Refuel Tanker	RT													
Terrain Following	TF													
Antisubmarine Search	AS													
Close Formation Flying	FF													
Catapult Takeoff	CT													

FIGURE 6

REQUIRED LEVELS FOR NORMAL STATES

Within Operational Flight Envelope	Within Service Flight Envelope
Level 1	Level 2

FIGURE 7

AIRPLANE FAILURE STATES

Probability of Encountering	Within Operational Flight Envelope	Within Service Flight Envelope
Level 2 after failure	$< 10^{-2}$ per flight	
Level 3 after failure	$< 10^{-4}$ per flight	$< 10^{-2}$ per flight

FIGURE 8

OPEN DISCUSSION

H.Schmidtlein, Germany: Mr Westbrook mentioned the way in which external fuel tanks are utilized in peace time and made a point on the obvious implications to the mission, tactics, the airframe, and its subsystems. My question is: Are there still other peace time influences on the utilization of military aircraft?

C.B.Westbrook, USA: Yes, there are other examples, some of which may be less obvious than the fuel tank example. For example, it is to be expected that with war time motivation to complete the mission, the pilot may not put safety first. He may push the aircraft to the limits of its envelope, continue the mission after system failures, press on under adverse weather, etc., where he might not under peace time conditions.

Another factor that may be mentioned in this regard relates to the level of experience and the state of pilot training. It is obvious that what might be an acceptable aircraft to an experienced senior pilot may be a disaster to a wartime pilot with much less experience, but with "tiger" tendencies.

A.G.Barnes, UK: In designing an aircraft to meet MIL-F-8785 B, it is necessary to define flight envelope boundaries for operational, service and design cases. The definition of these boundaries is critical, since the requirements are all based on these boundaries. Should the contractor or the procuring agency define these boundaries, and have difficulties arisen in agreeing on such boundaries?

J.W.Carlson, USA: In answer to the question of Mr Barnes in regard to who prepares the flight envelopes of MIL-F-8785 B, the Government or the contractor, it is intended that the contractor prepare the operational envelope after being given the mission requirements from the Government. This should be done during the evaluation of several contractor's designs in order to obtain as large an operational envelope as possible. The service and permissible envelopes, for which rules exist in the specification, must come later as the design evolves and lift, propulsion, and structural limitations become known.

W.T.Hamilton, USA: A comment on the B-52 which lost its vertical tail. The vertical tail was sized for high altitude, high C_L and approach and landing flight conditions. It was broken at high speed and low altitude where less tail area is required. It had to be flown to landing at relatively high speed and low C_L where the airplane was still controllable.

M.Hacklinger, Germany: Our colleagues from Wright-Patterson have explained that handling qualities flight envelopes are being used in early design stages to distinguish between competitive designs. These envelopes are only meaningful, however, together with all the numerical requirements for Dutch roll damping, stick force per g, etc., at the different failure states of systems. It appears to be almost impossible to fix all these parameters at an early stage – therefore I tend to conclude that these envelopes can only be defined with reasonable credibility after a design project has been defined in all its essential components.

C.B.Westbrook, USA: Obviously, the process of determining and validating the envelopes is an iterative one throughout the design process. Clearly, the airplane manufacturer cannot promise compliance in minute detail in the early design stage; in fact, all the idiosyncrasies of the design, the actual performance of equipment, etc., may not be known until well into the service life of the aircraft, and possibly never. However, this is not to say that the envelopes are only a recording of the way the design turns out. In the early design they record the desires of the customer and the manufacturer's promises, even guarantees, to meet these desires. Backed up by proper analysis, simulation, experience, and judgment, the manufacturer can have considerable confidence that he can do what he has promised.

DESIGN CONSIDERATIONS FOR THE SATISFACTORY STABILITY AND CONTROL OF MILITARY COMBAT AEROPLANES

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SUMMARY

Specifications for new military aeroplanes rarely define the stability and control characteristics required, yet these can have a profound effect on the development programme. Official design requirements in general specify only minimum acceptable standards and are deficient in many respects.

In this paper design criteria for satisfactory stability and control are reviewed; official requirements are considered; gaps and inconsistencies are noted; where no accepted standards exist, possible criteria are suggested. Some of the difficulties of designing to meet such criteria are mentioned and some gaps in existing data sheet methods are noted.

The following topics are discussed:-

Longitudinal Stability and Control
- Definition of aft c.g. limit
- Definition of forward c.g. limit
- Sizing the tailplane

Lateral Stability and Control
- Roll Control - design criteria
- Directional Stability - criteria for minimum stability
  - sizing the fin
- Choice of Dihedral/Anhedral
- Choice of Wing/Body incidence
- Rudder design
- Directional Control on the Ground
- Choice of wing planform - stall characteristics

(The latter point is treated only briefly because it is felt to be a different specialist topic.)

LIST OF SYMBOLS AND ABBREVIATIONS

a.c.	Aerodynamic Centre
b	Wing Span
b_T	Tailplane Span
$\bar{c}$	Mean chord
c.g.	Centre of gravity
C_L	Lift Coefficient
C_l	Rolling Moment Coefficient
C_m	Pitching Moment Coefficient
C_{m_0}	Pitching Moment Coefficient at zero lift
C_n	Yawing Moment Coefficient
C_{l_p}	Rolling Moment due to rate of roll derivative
C_{n_p}	Yawing Moment due to rate of roll derivative
C_{l_β}	Rolling Moment due to sideslip derivative
C_{n_β}	Yawing Moment due to sideslip derivative
$C_{l_{\delta A}}$	Rolling Moment due to roll control derivative
$C_{n_{\delta A}}$	Yawing Moment due to roll control derivative
$C_{l_{\delta R}}$	Rolling Moment due to rudder derivative
$C_{n_{\delta R}}$	Yawing Moment due to rudder derivative
d	Fuselage width
E.A.S.	Equivalent Air Speed
H_m	Manoeuvre margin $= -\dfrac{dC_m}{dC_L}$ in manoeuvring flight

I_x	Moment of Inertia in roll
I_y	Moment of Inertia in pitch
I_z	Moment of Inertia in yaw
I_{xz}	Product of Inertia in roll/yaw
k_y	Radius of Gyration in pitch
kg.	Kilograms
K.E.	Kinetic Energy
l_T	Horizontal tail arm
lb.	Pounds
L_p	Dimensional rolling moment due to rate of roll derivative
L_v	Dimensional rolling moment due to sideslip derivative
m.	Metres
mm.	Millimetres
M	Mach number
M.P.	Manoeuvre Point (H_m = 0)
n	Normal Acceleration
N.	Newtons
N	Yawing Moment
N.W.S.	Nosewheel Steering
p	Rate of roll
q	Rate of pitch
Q	Dynamic pressure
r	Rate of yaw
ss (suffix)	Steady State
S.F.	Stick Force
T_{90°	Time to roll to 90° bank from wings level
V	Air Speed
V_{NWL}	Nosewheel Lifting Speed
V_s	Stalling Speed
$V_{u/s}$	Unstick Speed
α	Angle of attack
α_o	Angle of attack at zero lift
β	Angle of sideslip
δ_A	Roll control angle
δ_R	Rudder control angle
ϵ	Downwash angle affecting tailplane
Λ	Wing sweep angle
θ	Angle of pitch
η	Longitudinal Control Angle
ζ_D	Damping Coefficient of dutch roll
ϕ	Angle of Bank
ω_D	Frequency of dutch roll
ω_{SP}	Frequency of longitudinal short period oscillation
Ω	Total rate of rotation in a spin

1. INTRODUCTION

Traditionally the specification for a new military aeroplane defines the required performance in precise terms : consequently it is performance considerations largely which define the configuration. Manoeuvrability is normally specified only in broad terms such as maximum normal acceleration and maximum rate of roll required at certain points in the flight envelope; little or no guidance is given in specifications as to the level of handling qualities required. British and French design requirements manuals specify only the minimum acceptable standards of handling qualities. The latest U.S.A.F. Mil. Spec. is much more comprehensive; nevertheless there are some aspects of stability and control which can affect the deisgn quite fundamentally that are ignored.

In this situation the project designer is understandably reluctant to accept the need for stabilising and control surfaces larger than necessary to meet the minimum requirements, or to accept the need for autostabilisation to improve handling beyond the acceptable standard : there is an identifiable drag and weight penalty on the one hand offset by rather intangible benefits on the other.

However the more advanced the performance of a new aeroplane and the more comprehensive its weapons system, the more exacting is the piloting task to exploit its full potential. It is essential therefore that due attention be given to stability and control in the project design stage to achieve a standard of handling qualities to match the performance and to avoid a lengthy flight development programme.

In this paper some of the considerations for satisfactory stability and control are raised, possible solutions discussed and design criteria suggested.

2. LONGITUDINAL STABILITY AND CONTROL

On a conventional aeroplane the tailplane has two distinct jobs to do:-

(i) Stabilise the aeroplane with centre of gravity aft.
(ii) Trim the aeroplane with centre of gravity forward.

The margins of stability and trim required at these limits and the desired c.g. range dictate the tailplane size required.

2.1 Aft c.g.

The considerations which define the aft c.g. limit are as follows:-

2.1.1 Stick Force per g

Although at first sight there is a reasonable uniformity of opinion between U.K., French and U.S. requirements (Refs. 1 to 3) for stick force per g (Table 1) - all demand a minimum level of about 10-15 Newtons per g and a maximum of 30-40 - the differences are significant when their full design implications are considered.

TABLE 1

Stick Force per g. Requirements applicable to Combat Aeroplanes. (n_1 = 8g assumed)

	U.K. (Av.P.970)	French (Air 2002c)	U.S. (Mil.Spec.8785)
Minimum (aft c.g.)	3.42 lb. 15.8 N	1 kg. 10.2 N	3 lb. 13.9 N
Maximum (forward c.g.)	7.75 lb. 36 N	4 kg. 40.8 N	8 lb. 37 N
Ratio $\frac{\text{max.}}{\text{min.}}$	2.27	4.0	2.67
(SF/g ratio - 1)	1.27	3.0	1.67

With artificial feel it is theoretically possible to provide the minimum required stick force gradient with any value of manoeuvre margin * (H_m) however small, provided that it is positive, so the minimum stick force per g requirement does not in itself locate the aft c.g. limit. However if a linear Q-feel system is used (i.e. stick force per degree of tailplane is constant at a given flight condition), stick force per g is proportional to manoeuvre margin. In order to satisfy the stick force per g requirements at the forward as well as the aft end of the c.g. range therefore the following relationship must apply:-

$$\frac{H_m\,(\text{min}) + \text{c.g. range}}{H_m\,(\text{min})} = \frac{\text{S.F./g max. (fwd. c.g.)}}{\text{S.F./g min. (aft c.g.)}}$$

or $$H_m\,(\text{min}) = \frac{\text{c.g. range}}{\text{S.F./g ratio} - 1}$$

With a linear feel system therefore, given the c.g. range required, the minimum manoeuvre margin is defined and consequently the aft c.g. limit located. There are two drawbacks to this definition however:-

* The term "manoeuvre margin" used here has the British meaning, that is the stability margin ($-\frac{dC_m}{dC_L}$) in manoeuvring flight. (The French term "marge de manoeuvre" has quite a different meaning, being associated with thrust - limited 'g'.)

(i) The disparity between the stick force per g ratios leads to a wide variation in the minimum manoeuvre margin based on different national requirements (Table 1, bottom line).

(ii) Where a large c.g. range is desired, meeting the requirements leads to an unduly generous level of stability at the aft c.g. limit and consequently over the whole c.g. range.

The latter is not detrimental to handling qualities unless it results in excessive short-period frequencies in certain flight conditions (e.g. on a variable sweep aeroplane transonic at low altitude, fully swept). But it does lead to a larger tailplane than would otherwise have been required, with attendant drag and weight penalties. The situation is depicted in figure 1. This figure shows that doubling the c.g. range requirement from Case (a) to Case (b) while respecting stick force per g requirement at the extremes of the c.g. range, results in a manoeuvre margin at the aft limit twice as large as is necessary for satisfactory handling and a tailplane area 10% greater than that required from stability and trim considerations (Case (c)).

One way of avoiding the latter situation is to use a non-linear feel system; if the feel force gradient is made dependent on stick position in the sense of reducing the feel forces per degree of tailplane as the stick moves aft then the variation of stick force per g with c.g. position can be reduced as illustrated in the lower diagram of figure 1 (Case c). This can be done by linking the feel unit to the stick and shaping the curve of stick to tailplane angle to achieve the required stick : tailplane gearing (and consequently force gradient) as a function of displacement from the zero lift trim point. * The principle is illustrated in figure 2.

The extent to which such a system can be applied to a military combat aeroplane is limited by the fact that changes of configuration (e.g. combat flaps, external stores) alter the zero lift trim point and consequently destroy the unique relationship between 1g tailplane angle and stability margin. In practice therefore the "hump" of the gearing curve shown in figure 2c must enclose the zero lift trim point in all relevant configurations, rather than merely attain a unique value at its crest.

An alternative method of defeating the stick force to manoeuvre margin relationship is by the use of a bob-weight to provide a proportion of the stick force per g as illustrated by figure 1 case (d). However the application of bob-weights to high speed aeroplanes has to be approached with extreme caution because of the coupling between aircraft motion and control circuit motion that is inevitably introduced and which can so easily lead to short period instability and pilot-induced oscillations.

2.1.2 Transient Response to Control

The level of stability at the aft c.g. limit does not affect the pilot's judgement of the steady-state manoeuvring characteristics provided that the feel system provides adequate forces. However, as manoeuvre margin is reduced, smaller tailplane angular displacements are required to apply the g; consequently initial angular acceleration response becomes more sluggish. The pilot can compensate for this by applying more control to initiate the manoeuvre then relaxing it as the desired steady-state response is approached, but the increased concentration required to avoid overshooting the required 'g' leads to pilot criticism when the stability margin is too small.

This effect (and the opposite effect - too lively a response with too large a manoeuvre margin) is recognised in the latest USAF Mil. Spec. requirements which define upper and lower limits for the ratio of:

$$\frac{\text{transient angular acceleration in pitch per unit incidence } (\omega_{SP}^2)}{\text{steady-state normal acceleration per unit incidence } (n/\alpha)}$$

for various tasks and failure situations.

Since this ratio is directly proportional to manoeuvre margin and for a given aeroplane + is independent of flight condition

$$\text{since} \quad \frac{\omega_{SP}^2}{n/\alpha} = \frac{\bar{c}\, H_m\, g}{k_y^2}$$

* An associated design feature of such a system is that rearward stick displacement from the trim position would involve a reducing force gradient and forward displacement an increasing gradient unless a compensating non-linear stick force/stick position gearing were introduced to linearise the stick force to normal g relationship at a given flight condition/c.g. position, i.e.

$$\underset{\text{(non-linear)}}{\frac{\text{stick force}}{\text{stick angle}}} \times \underset{\text{(non-linear)}}{\frac{\text{stick angle}}{\text{tailplane angle}}} \times \underset{\text{(Linear)}}{\frac{\text{tailplane angle}}{\text{normal g}}} = \underset{\text{(Linear)}}{\frac{\text{stick force}}{g}}$$

\+ Provided that the radius of gyration in pitch does not change significantly with loading.

it is an eminently sensible and usable criterion for defining the c.g. limits for aeroplanes without control augmentation. It is rare for the required c.g. range to be such that the variation of stability between forward and aft limits spans the allowable range of values of $\frac{\omega^2_{SP}}{n/\alpha}$, although this situation can occur on a variable sweep aeroplane. The criterion is therefore most useful for defining the aft c.g. limit.

The criterion is illustrated in figure 3; values of $\frac{\omega^2_{SP}}{n/\alpha}$ for two BAC military aeroplane at their aft c.g. limits are shown in the lower part of the diagram. The longitudinal handling qualities of both these aeroplanes are rated as good at their aft c.g. limits, which suggests that the Mil. Spec. Level 1 requirement is rather severe and that the Levels 2 and 3 requirement is adequate.

The unique relationship between initial angular acceleration in pitch and steady-state normal acceleration can be broken if a control augmentation term is inserted between the stick and the tailplane power control, giving a "manoeuvre boost" effect, as illustrated in figure 4. This enables a smaller manoeuvre margin to be tolerated without degrading handling qualities.

2.1.3 Response to Configuration Changes

Selection of airbrakes or combat manoeuvring devices, release or jettison of external stores in general causes a change of trim and stability which, if unopposed, results in disturbance of the flight path. The incremental normal acceleration due to a change of pitching moment is inversely proportional to manoeuvre margin so considerations of the response to configuration changes can dictate the minimum acceptable manoeuvre margin in high speed flight.

National requirements differ slightly in the allowable response, but, broadly speaking, if the resulting disturbance is within $\pm \frac{1}{4}$g this is acceptable. What is not usually made clear is to what extent simultaneous release of stores from different positions is required, to what extent hang-ups in the release sequence must be catered for and over what speed range release is required.

It is not enough to define these things in due course during the development phase; they should be specified at the outset of the project, to avoid over-design with attendant weight and drag penalties or under-design and subsequent development problems.

2.1.4 Power Control Discrimination

A pilot requires to control the flight path of an aeroplane to an accuracy of about ± 0.1g. The discrimination of the power control actuator must therefore match this accuracy to avoid the necessity for continuous control movement to direct the flight path; this imposes a requirement for actuator linear discrimination of:-

$$\frac{\text{Actuator Stroke} \times \text{Minimum Tailplane Angle per g} \times 0.1}{\text{Tailplane total angular Travel}}$$

Putting typical values to this, for example 0.3° per g with a total angular travel of 30°, the actuator discrimination would have to be 1/1000 of its total stroke. Such a requirement is by no means impossible to achieve but it can be expensive. It is certainly an important consideration in defining minimum tailplane angle per g and therefore minimum manoeuvre margin.

2.1.5 Inertia Coupling Effects

In defining c.g. limits it is not enough merely to consider the uncoupled longitudinal response to pilot control and to disturbances. The effect of longitudinal stability on rapid rolling behaviour must also be studied. In a rapid roll simultaneous rates of roll and yaw generate a gyroscopic pitching moment, resulting in excursions of incidence and normal acceleration which increase as the stability margin is reduced. Superimposed on this are the effects of inadvertent pilot's control movements : in a full aileron roll the pilot has to apply a considerable force and displacement laterally and cannot control the fore and aft stick position with the same precision as in normal flight. This is perhaps the best reason for retaining a reasonable stick displacement (as well as stick force) per g and a minimum of 5 mm per g is suggested as a design aim.

From rolls entered in 1g flight these control movements are generally random although a bias in the direction of moving the stick aft in left rolls and forward in right rolls at 1g has been noted on one aeroplane (figure 5B); this is believed to be due to the pilot's moving the stick in an arc centered at his elbow. From rolls entered under positive or negative g, fortunately the tendency seems to be for any stick movement to be in the sense of returning towards the 1g trimmed position; this is presumably due to pilots relaxing the fore and aft force when applying a lateral force. Some examples of these inadvertent control movements are shown in figure 5.

The excursions in incidence and normal acceleration which occur in rapid rolls must be considered from the following standpoints:-

Structural overloading
Danger of stalling

Disturbance of the flight path
Crew comfort (accelerations at the cockpit rather than at the c.g.)

There are no official design requirements for acceptable rapid rolling characteristics in terms of longitudinal disturbances but the following rules are known to provide a reasonable safeguard.

1. The normal acceleration or incidence limits for initiation of rapid rolls should allow at least 1g margin from the values which would result in structural overloading or stalling with longitudinal control fixed.

2. From rolls entered in trimmed 1g flight the application of forward or aft stick movement which would result in ± 1g in normal flight, should not cause structural overloading or loss of control.

These requirements can impose unduly restrictive limits if longitudinal stability is inadequate and are therefore an additional consideration in defining the aft c.g. limit.

2.2 Forward c.g. Limit

Stick force and transient response characteristics in relation to the forward c.g. limit have already been mentioned in earlier paragraphs; it is unusual for these to be allowed to dictate the forward c.g. limit. The more usual situation is that nosewheel lift on take-off, flare and touchdown capability on landing impose the limit and provide the forward boundary for sizing the tailplane.

Control power for supersonic manoeuvring may be an additional consideration, but if airfield performance is important and effective high-lift devices are used, it is likely that take-off and landing considerations will be overriding.

The following criteria have been shown to be satisfactory for defining the forward limits for take-off and landing:

(i) It must be possible to raise the nosewheel on take-off at a speed such that rotation to the nominal take-off attitude can be accomplished at a mean rate of 5 degrees/second without delaying unstick speed beyond the nominal value (generally 1.1 V_s)

$$\text{i.e.} \quad (V_{u/s} - V_{NWL}) = \frac{\alpha_{u/s}}{5} \times \frac{(dV)}{(dt)} \text{ mean}$$

(ii) It must be possible to apply 1.1g at 1.15 V_s in the landing configuration, away from ground, using not more than 90% * of full negative tailplane travel.

(iii) It must be possible to touchdown in 1g flight at 1.1 V_s in ground effect (mainwheels touching) using not more than 90% * tailplane travel.

(* the remaining 10% being "thrash margin" to allow the pilot continuously to assess response)

The above are somewhat idealised definitions because both the nosewheel lift unstick and landing flare are dynamic manoeuvres involving the transient response characteristics. However the landing limits have been shown to correlate well with forward c.g. limits determined from flight trials. Nosewheel lift calculations have generally been shown to be pessimistic, by quite a large amount in one case, as illustrated in figure 6.

2.3 Sizing the Tailplane

Sizing the tailplane to meet the c.g. range requirements without incurring weight and drag penalties due to over-generosity demands accurate knowledge of:

C_{m_0} in the take-off and landing configuration. Tail off a.c. in the take-off and landing configuration. Tailplane C_L max.	) to establish the forward c.g. boundary
Tail off aerodynamic centre of the rigid aeroplane. Tailplane lift-slope. Downwash gradient. The effects of Mach number on the above. Aeroelastic effects on lift-slope and a.c. of wing and tailplane. Wing fuselage inertial bending effects. Intake and jet flow effects on stability.	) to establish aft c.g. boundary

The former boundary is usually the easier one to establish. The latter is difficult due to the large number of parameters involved and the necessity of basing project decisions on preliminary data. A typical breakdown of these effects on the longitudinal stability of a strike aeroplane with external stores at high subsonic speed, low altitude, is shown in figure 7. The following points are worth noting:

(1) Since it is usual, for structural reasons, for the tailplane to have a smaller aspect ratio than that of the wing, the tailplane lift slope "grows" less with Mach number than the wing lift slope (and consequently the downwash gradient affecting the tailplane). The tailplane contribution to stability therefore reduces with Mach number subsonically, producing a "trough" in stability at high subsonic speed. The bigger the tailplane, the deeper the trough.

(2) Aeroelastic losses on the wing and tailplane can act in opposite senses. A stiff tailplane and a relatively flexible wing can result in improved stability compared with the rigid aeroplane but the bigger the tailplane the more likely there is to be a nett loss.

(3) The effect of wing distortion on downwash gradient can be large and is difficult to estimate, although it is easy to define an upper limit to the effect

i.e. $\frac{d\epsilon}{d\alpha}$ is proportional to flexible wing lift slope

or $\frac{d\epsilon}{dC_L}$ = constant

(4) Underwing stores in front of a low tailplane are strongly destabilising (Ref. 4) but the rigid effects are alleviated by increased wing inertial bending and reduced losses due to tailplane aeroelastics.

(5) Fuselage inertial bending has a significant favourable effect on stability.

(6) In the case illustrated (as in several others known to the author) the algebraic sum of aeroelastic, inertial and mass flow effects on stability is close to zero. (However it is not implied that this is a universal rule!)

2.4 Summary of Longitudinal Stability and Control Considerations

For satisfactory longitudinal stability and control definition of the centre of gravity limits must take proper account of:-

Stick force per g
Transient response to pilot control
Response to configuration changes
Power control discrimination
Inertia Coupling in rapid rolls
Nosewheel lift capability
Landing flare capability.

Official design requirements are deficient in many respects and both amplification of these and uniformity within NATO countries would ease the designer's task.

The most useful criterion for defining minimum stability levels for satisfactory handling is $\frac{\omega^2_{SP}}{n/\alpha}$ but the Level 1 minimum required by USAF Mil. Spec. 8785 is felt to be excessive and the Level 2/3 value is normally adequate. Smaller levels can be tolerated with control augmentation provided that due account is taken of response to release of weapons, inertia coupling effects in rapid rolls etc.

3. LATERAL STABILITY AND CONTROL

The designer's task of providing a satisfactory level of lateral stability and control is more difficult than in the case of longitudinal stability and control because there is no single measurable end product equivalent to c.g. range, against which to evaluate the lateral characteristics.

Since the primary control mode of a conventional aeroplane involves control of bank angle it is logical to deal first with roll control.

3.1 Roll Control

3.1.1 Combat Rate of Roll

Inadequate rolling power restricts the operational capability of an aeroplane; excessive rolling power entrains inertia coupling problems. The task of the designer is to provide the right level of rolling performance between these extremes.

Theoretical studies (References 5 and 6) have yielded the following values of roll performance required for different combat tasks.

Task	Performance	Ref.
Ground Attack:	2.5 radians/second	5
G.W. Interceptor:	1 radian/second	5
Air to Air Combat:	90° bank in 1 second	6

It is not difficult to provide this level of rolling performance at high speed; the difficulty lies in deciding down to what speed these requirements apply.

Figure 8 compares French and U.S. requirements for rolling performance, together with pilot comments from a flight evaluation. The wide difference between the two national requirements at high speed is evident. Comparison with pilot comments suggests that an alternative, more logical, form of requirement would be as follows:

a) Time to 90° bank = 1.3 seconds at V/V_s = 2.0 (the 4g stall boundary).

b) At lower speeds the wing tip helix angle should not be less than that given by (a) above.

c) At higher speeds the rate of roll should not be less than that given by (a) above.

Apart from the air to air combat situation of acquiring an evading target, the main requirement for rolling performance is in the breakaway manoeuvre for debris and collision avoidance. Data from instrumented combat aeroplanes (Ref. 7, 8), reproduced in figure 9, provides some guidance on the acceptable variation of roll rate with normal acceleration. In particular the very steep cut-off of roll rate used at less than 1g suggests that a relaxation of the structural design requirements for rolling push-over manoeuvres in this part of the flight envelope should be possible. This is because the requirement to demonstrate structural integrity in full aileron rolls at negative g is usually incompatible with handling considerations; such manoeuvres invariably involve very high rates of roll and autorotational tendencies and the flight envelope for asymmetric manoeuvres must, of necessity, be restricted on this account.

3.1.2 Asymmetric Stores

Asymmetric release of stores, intentionally or due to a hang-up in the release sequence, in general produces a roll and yaw asymmetry; dominant effects are the roll asymmetry due to asymmetric underwing stores and the yaw asymmetry due to asymmetric fuselage-mounted missiles. It is not enough to ensure that the aeroplane can be trimmed laterally in such a situation; adequate controllability and manoeuvrability must also be ensured to avoid restricting the evasive manoeuvres following weapons attacks.

There is little or no guidance in official design requirements on these matters. The following are suggested as target requirements for satisfactory control:-

In the event of an asymmetry resulting from a deliberate selection, a single failure in, or interruption of the weapon delivery sequence:-

1. It must be possible to sustain a pull-out at 80% n_1 or 80% C_{Lmax} using roll control alone to prevent the aircraft rolling.

2. It must be possible to use full aileron in either direction to roll through:-

   90° for ground attack manoeuvres
   180° for air to air combat manoeuvres

   at normal accelerations up to the limit for asymmetric manoeuvres in symmetric configurations, without encountering a dangerous flight condition.

3. It must be possible to trim out the asymmetry at normal cruising speeds and in the landing configuration at instrument approach speed.

3.1.3 Landing Approach

The requirements here are fairly well known. In recent years there has been a tendency to apply requirements closer to the old carrier-based values, to land-based aeroplanes. This is not illogical when operation from semi-prepared airfields is required, as is the current trend. The Table below summarises requirements from various sources. Requirements are often defined in aircraft specifications.

TABLE 2

Rolling Performance Requirements for the Landing Approach

Origin	Requirement
U.K.	$\frac{Pb}{2V}$ = .07 land-based $\frac{Pb}{2V}$ = .09 carrier-based
France	$\frac{Pb}{2V}$ = .06 land-based $\frac{Pb}{2V}$ = .09 carrier-based
U.S.A.	30° bank in/second

3.1.4 Crosswind Landing

Usually on swept-wing aeroplanes with high lift flaps the crosswind landing case imposes a more severe requirement for rolling power than rate of roll on the approach. This is due to the large rolling moment due to sideslip at high lift combined with the large sideslip angle in a touchdown without drift at low forward speed.

Of the two extremes of crosswind landing technique, the wing-down approach and the crabbed approach culminating in the kick-off drift manoeuvre, the latter imposes the more severe requirement for roll control. The USAS Mil. Spec. 8785 requires that it be possible to hold wings level in the maximum specified crosswind using not more than 80% of the available roll control. The rudder position is not specified; some slight relief of the roll control is obtained if the rudder is assumed to be deflected to apply the sideslip, and certainly this interpretation seems to give an adequate margin of control.

It is usual to design to meet this requirement at the nominal touchdown incidence (usually 1.15 V_s) but in addition it is considered necessary to ensure that control does not deteriorate too rapidly with increasing incidence. A useful rule is that with a 10% reduction of speed below the nominal touchdown speed, 100% of the available roll control will be sufficient to maintain wings level.

3.1.5 Spin Recovery

The best insurance against unsatisfactory spin characteristics in the project design stage is to provide adequate control power for recovery. On an inertially slender aeroplane ($I_y >> I_x$) the control having the most powerful effect for spin recovery is the roll control; the combination of in-spin roll rate with nose-up pitch rate generates an out-spin gyroscopic yawing moment, leading to recovery.

Reference 9 gives a correlation of aileron lateral moment of area x deflection angle versus Spin Angular Momentum for satisfactory spin recovery. This correlation is reproduced here in figure 10 with additional points added from aeroplanes which use differential tailplane for roll control; the correlation seems to fit differential tail equally well.

3.1.6 Choice of Roll Control

Typical curves for the variation of rolling moment and yawing moment with wing sweep, angle of attack and Mach number for ailerons, spoilers and differential tailplane are shown in figures 12 and 13. The advantages and disadvantages of these three types of roll control are summarised in the following Table.

TABLE 3

Roll Controls - Design Features

Type of Control	Advantages	Disadvantages	Means of Countering disadvantages
Ailerons	Plenty of background experience Linear characteristics Small yaw effect at low incidence	Ineffective at extreme sweep/high angle of attack, (where rolling moment due to sideslip is large)	
		Adverse yaw at high angle of attack	Differential deflection but this produces nose up pitch on swept wing
		Loss of usable span for flaps	Drooped ailerons - but this aggravates adverse yaw
Spoilers	Full span flaps can be used	Loss of effectiveness near stall	Leading edge droop or slat
	Effectiveness increases with flaps down - good for crosswind landing	Ineffective at high sweep	
		Ineffectiveness over small angles	Vent under spoiler
	Can be used also as lift dumpers	Over-sensitivity over small angles flaps down	Shroud under spoiler

Type of Control	Advantages	Disadvantages	Means of Countering disadvantages
Spoilers Contd.		Nose up pitching moment on swept wing	Suitable spanwise position to influence tailplane
Differential Tailplane	Full-span flaps can be used Retains effectiveness at extreme angle of attack Linear characteristics Pro-yaw assists roll at high angle of attack	Inadequate effectiveness for low sweep/high AR wing: Pro-yaw large; bad at low angle of attack	 Rudder interconnect

The following additional points are worth noting:-

1. The nose-up pitching moment due to spoiler on a swept wing can be reduced or eliminated by suitable spanwise positioning, to influence the downwash over the tailplane. Pitching moments cannot easily be estimated reliably and wind tunnel testing is necessary to establish the range of satisfactory spanwise positions, as illustrated by figure 11.

2. The yawing moment due to spoiler can be strongly influenced by fin interference; on a supersonic aeroplane the ratio C_n/C_l can change significantly as the Mach lines from the spoiler move away from the fin (figure 14). Data Sheets give no information on this matter and early wind tunnel tests are essential.

3. The yawing moments due to differential tail are strongly dependent on afterbody width (i.e. the distance of the tailplane root from the base of the fin). This is because a large proportion of the yawing moment is due to induced sidewash on the fin as illustrated in figure 15. The resulting sidewash on the fin reinforces the yawing moment and reduces the rolling moment. No data sheet methods are yet available for calculating these effects.

The choice of roll control type for a military combat aeroplane will be influenced largely by its mission requirements; the trend in recent years has been towards short airfield performance and good low altitude gust ride comfort, placing design emphasis on high lift capability with minimum wing area. Consequently ailerons have fallen somewhat out of fashion, yielding place to spoilers and differential tailplane for roll control. However this trend could well be reversed if jet lift gains favour.

3.2 Directional Stability and Control

Whereas the pilot has to excite the longitudinal modes of motion of an aeroplane continually to control its flight path, directional motion is induced only inadvertently in normal flight. Ideally, bank angle is controlled by rolling the aeroplane about its flight velocity vector, which should always be contained in the plane of symmetry. The question facing the designer is how far the flight characteristics can be allowed to depart from this ideal without incurring criticism or degrading operational capability.

3.2.1 Criteria for Sizing the Fin

The fin has to offset the destabilising effect of the fuselage forebody, which is approximately proportional to the product of the square of its maximum depth and the distance of the maximum depth from the c.g.; in addition it has to provide adequate stability to restore sideslip to zero within a reasonable time following a disturbance and to react the asymmetric moments due to lateral control deflection, inertia coupling and weapon release, without excessive sideslip. Design criteria are listed below:-

3.2.1.1 Dutch Roll Frequency

Mil. Spec. 8785 gives minimum values of dutch roll frequency related to dutch roll damping (figure 16). These criteria are useful for checking the adequacy of directional stability at low speeds but are unlikely to be sufficient to cover combat manoeuvre requirements. For example in manoeuvring flight the dutch roll frequency can be adequate and damping positive with zero or even negative values of C_{n_β}; in these cases the dynamic $C_{n_\beta} = C_{n_\beta} + \frac{I_{xz}}{I_x} C_{l_\beta}$

provides the stability; but handling in response to control would be totally unacceptable.

3.2.1.2 Directional Stability with Bank Angle Constrained

It has been found that in many control tasks the pilot exercises a tight control over bank angle but is not immediately conscious of sideslip. In the situation where dihedral effect is positive the yawing moment due to roll control is destabilising if adverse, stabilising if proverse.

The effective directional stability under constraint is:-

$$C_{n_\beta} - C_{l_\beta} \frac{C_{n_{\delta A}}}{C_{l_{\delta A}}}$$

On many of the current generation of combat aeroplanes directional instability due to adverse aileron yaw occurs well before the stall and constitutes a limit to usable incidence, as it has been found to also on the BAC 221 research aeroplane (Ref. 10).

It is therefore an important consideration to be taken into account in sizing the fin, in conjunction of course with choice of dihedral/anhedral and type of roll control.

3.2.1.3 Rapid Rolling Behaviour

Within the boundaries where rolling behaviour, as indicated by Phillips' criterion (Ref. 11) is non-divergent, large excursions in sideslip can still occur if directional stability is inadequate. These can result in:-

- Oscillatory rolling behaviour - this is recognised in Mil. Spec. 8785 which defines limits for satisfactory and acceptable behaviour. This criterion has been found to agree well with pilot opinion and is illustrated here in figure 17. The criterion applies equally to rapid rolls and turn entry manoeuvres at moderate rates.

- Excessive lateral acceleration felt by the pilot - a tentative pilot opinion scale is shown in figure 18. Note that cockpit lateral acceleration may differ significantly from c.g. acceleration in a rapid roll, and that important lateral accelerations can arise directly from the side forces due to roll control with spoilers or differential tail, in the absence of sideslip.

- High fin loads - in general increasing fin area reduces fin side load in a given manoeuvre because the loading due to sideslip increases in proportion to fin area but sideslip diminishes in proportion to C_{n_β}, which increases in greater proportion than fin area. (For example if 60% of the fin area is required to balance the destabilising forebody, stability is proportional to the remaining 40%. Consequently if fin area is increased by 10% stability increases by 25%; fin loading per unit sideslip increases by 10% but sideslip in a given manoeuvre reduces by 25%; therefore fin loads reduce in the ratio $1 : \frac{1.1}{1.25}$ i.e. by 12%.) BAC policy is to clear rolling manoeuvres only if the calculated fin load, based on flight-matched derivatives, does not exceed 70% unfactored design load in the normal manoeuvre. The remaining 30% is to cater for inadvertent pilot longitudinal control inputs, tolerances on derivatives, etc.

- Autorotation, that is a continuing roll with roll control centralised. Reference 12 shows that the rate of roll and incidence at which autorotation occurs are strongly related to C_{n_β}; once the rolling performance versus normal acceleration is defined therefore the onset of autorotation defines a lower limit to C_{n_β}, for autorotation must be regarded as unacceptable behaviour except possibly in extreme manoeuvres at high altitude/low speed and only then if there is a clear, instinctive recovery procedure.

  In general a 1g margin from the autorotation boundary is considered necessary, but Official requirements give no guidance on this matter.

A complete investigation over the full Flight Envelope is necessary to define which of these criteria impose limits on rolling performance and consequently affect fin area requirements.

3.2.1.4 Lateral Phugoid Behaviour

On certain configurations, notably with slender or highly swept wings, at low levels of directional stability (C_{n_β}) the roll subsidence and spiral modes of motion combine to form a second oscillatory mode of long period, the lateral phugoid; with further reduction of C_{n_β} the lateral phugoid goes unstable, as illustrated by a root locus plot in figure 19.

The Civil Airworthiness requirements for supersonic transports recognise this phenomenon and set a lower limit of 10 to the ratio of spiral : roll subsidence time constants for normal operation. Reference 13 suggests that the onset of lateral phugoid sets an absolute lower limit to C_{n_β} for acceptable handling, but this was based on limited evidence and cannot therefore be applied as a universal rule. On a military combat aeroplane if a full manoeuvre clearance is required up to the maximum design Mach numbe the deteriorating rapid rolling characteristics are likely to set the lower limit to C_{n_β} befo lateral phugoid onset; the latter could be of significance however if a "gentle manoeuvres only" clearance is required at the top end of the Mach number range.

3.2.2 Estimation of Directional Stability

Accurate prediction of C_{n_β}, including its variation with Mach number, angle of attack and dynamic pressure (E.A.S.) is difficult for the following reasons:

i) The destabilising contribution due to the forebody is subject to a tolerance of at least 10% due to effects of shape which are not accounted for in data sheet methods.

ii) The effects of fuselage-mounted stores can be large and cannot be estimated accurately.

iii) The influence of body vortices on fin effectiveness at high angle of attack, although calculable for certain families of shapes such as ogive-cylinders, cannot be predicted with confidence for practical aeroplane shapes. In particular estimates vary widely with different assumptions about the point of separation of the body vortices (e.g. canopy crest, wing root etc.). Figure 20 presents a correlation of fin effectiveness at high angle of attack with fin height and body depth; a clear trend may be seen but the spread of the points is fairly wide. Twin fins are not always better. (Fig. 21)

iv) Aeroelastic effects at the design limits are considerable - typically 20 to 25% fin effectiveness may be lost due to spanwise twist resulting from bending and due to rudder distortion. These effects are not easy to estimate in the project stage.

3.2.3 Fin Shape and Size

Having defined the required levels of directional stability at different flight conditions and the destabilising effects of the body and external stores the required stabilising contribution from the fin is defined.

It is not intended to discuss the subject of fin design in depth but the following points are noteworthy:-

a) High incidence considerations favour a tall fin, but aeroelastic effects impose an upper limit to the aspect ratio that can be tolerated.

b) In some cases ventral fins are more efficient than additional upper fin area (as in the case illustrated in figure 22); this is due to the favourable interference effect of the fuselage, to their comparatively high stiffness, and the reduction in dihedral effect which they cause.

c) Ventral fins generally retain their full effectiveness at high incidence but their effectiveness at low incidence may be severely reduced by interference from fuselage-mounted stores (see figure 23).

Fin design is inevitably a compromise between conflicting requirements and the number of aeroplanes that have been subject to fin modifications during their development is indicative of the difficulty in defining requirements and estimating effectiveness precisely.

3.3 Choice of Wing Dihedral/Anhedral

It is generally recognised that a positive dihedral effect is required to ensure that an aeroplane rolls the "right" way when rudder is applied. This implies that the rolling moment due to sideslip must override the rolling moment due to rudder, giving a minimum value of dihedral effect $(-C_{l_\beta})$

$$-C_{l_\beta} > \frac{C_{l_{\delta R}}}{C_{n_{\delta R}}} \cdot C_{n_\beta}$$

Requirements imply this by demanding that crossed controls should be required to trim a straight sideslip.

If this requirement is applied at all flight conditions (e.g. low altitude, high speed without external stores) and used to define minimum value of $-C_{l_\beta}$ and therefore the maximum acceptable wing anhedral, it can result in excessive dihedral effect in other flight conditions and configurations, particularly at high incidence and with external stores carried; the latter invariably increase dihedral effect.

Excessive dihedral effect is an embarrassment because it causes:-

oscillatory rolling behaviour
autorotational tendencies
excessive bank angle response to sidegusts
increased roll control demands for crosswind landing

The requirement for "crossed controls" in sideslip at all flight conditions can therefore lead to a degradation of handling qualities in these other respects.

The underlying reason for this requirement is evidently to ensure that an aeroplane should be capable of being controlled with rudder alone in the event of loss of aileron control. It is reasonable to apply the requirement to aeroplanes without control duplication but with split or duplicated power controls where a single failure results only in reduced roll control authority it seems unreasonable to apply it blindly at all flight conditions. It is suggested that the requirement should be relaxed to apply only to economical cruise, approach and landing conditions, that is for safe return to base.

Rolling behaviour has already been discussed earlier but it is worth noting here that there is a trade-off of fin size against dihedral effect for equivalent rolling characteristics, so that if excessive dihedral effect can be avoided this will ease the fin size requirements. Similarly it will ease the roll control requirements for crosswind landing and reduce or eliminate the necessity for roll autostabilisation to suppress bank angle response to gusts. With regard to the latter, the traditional ϕ/Ve versus dutch roll damping criterion is now considered to be inadequate and there is accumulating evidence that it is the relationship between roll damping (L_p) and dihedral effect (L_β) which governs pilot opinion of bank angle response to turbulence. The form of a possible criterion is illustrated in figure 24 but insufficient work has been done to assign quantitative boundaries.

3.4 Choice of Wing-Body Incidence

The inclination of the principal axis of inertia to the flight path has a significant influence on dutch roll and rolling characteristics, as is well known.

The designer has some freedom to bias the rolling behaviour in the direction of improving the high incidence rolling characteristics by increasing the wing-body incidence setting. However this can only result in deterioration of the low incidence characteristics (increased tendency to autorotation etc.) due to the increased nose-down inclination of the fuselage at low incidence.

To enable the optimum choice of wing-body setting to be made, it would ease the designer's task of providing good handling characteristics at the more important positive g conditions if the current stringent low g rolling requirements were relaxed.

3.5 Rudder Design

The role of the rudder in normal flight is to suppress sideslip. However it is the tasks which require deliberate application of sideslip which size the rudder, so these will be discussed first.

3.5.1 Crosswind Landing

A wing down approach requires more rudder than the kick-off drift manoeuvre following a "crabbed" approach because in the latter case the dynamic overshoot in sideslip in response to rudder can be used to advantage. In view of this, and because

(a) the wing down technique is impractical on many aeroplanes because of the excessive bank angle required to balance sideslip

(b) the control forces required for a wing-down approach are inevitably high

(c) the crabbed approach is the standard technique taught in NATO Air Forces.

It is considered unnecessarily severe to design the rudder to trim the full sideslip equivalent to zero drift at touchdown speed. The criterion for rudder control power which has been found to give a satisfactory margin of control is:-

$$C_n(\delta R) = 0.7 \times \frac{V_x\ \text{wind}}{V_{\text{Touchdown}}} \left(C_{n_\beta} + C_{l_\beta} \frac{C_{n_{\delta A}}}{C_{l_{\delta A}}}\right)$$

The factor of 0.7 implies a 40% dynamic overswing of sideslip in response to rudder.

3.5.2 Spin Recovery

If the aeroplane in question has a high value of I_y/I_x, it may well be that a conscious decision is taken to design for spin recovery by roll control and to forget the rudder in this application.

However if designing for good recovery by means of rudder, rudder power at spin incidence taking account of wing and tailplane shielding must be scaled to the total rotational energy of the spinning aeroplane.

A value based on Lightning and Jet Provost, both of which have outstandingly good spin recovery on rudder alone is:-

$$\frac{N(\delta R)}{\text{Spin K.E.}} = 0.075 \text{ to } 0.10$$

3.5.3 Asymmetric Stores

The rudder must be capable of balancing the aerodynamic asymmetry due to asymmetric carriage of stores and the associated yawing moment due to the roll control required to maintain wings level in all flight conditions. This is unlikely to size the rudder but is one of the considerations to be taken into account in defining the authority required at high speed.

3.5.4 Turn Co-ordination

Ideally for the velocity vector of the aeroplane to be contained within the plane of symmetry while rolling into or out of a turn, a rate of yaw must be generated equal to

$$r\ (\text{ideal}) = p + \frac{g}{V} \sin\phi \cos\theta \ \text{(in body axes)}$$

It is instructive to examine the ratio $\frac{r\ (\text{actual})}{r\ (\text{ideal})}$ on turn entry and exit with different forms of turn co-ordination (e.g. geared rudder : aileron; roll rate to rudder gearing etc.).

The minimum value of $\frac{r\ (\text{actual})}{r\ (\text{ideal})}$ gives an indication of the tendency for the nose of the aeroplane to "hang back" on turn entry and is a useful criterion for defining quality of turn co-ordination. Tentative pilot opinion ratings are shown also in figure 25. However pilot opinion must be related also to the width of the "trough" in figure 25 so this criterion must not be taken as hard and fast. It is mentioned here because turn co-ordination can impose quite large transient rudder demands which need to be taken into account in defining rudder authority.

3.5.5 Transonic Characteristics

Rudder effectiveness diminishes rapidly between subsonic and supersonic speed due to both:-

i) the reduced aerodynamic effectiveness of a flap-type control at supersonic speed

ii) the increased aeroelastic losses at supersonic speed.

In the presence of an asymmetry therefore there is a rapidly changing rudder trim requirement in the transonic region. The effect is most pronounced on transonic deceleration at low altitude (due to the high deceleration rate) and can lead to a significant yaw/roll disturbance as the rudder effectiveness increases sharply below M = 1.0.

The variations of rudder effectiveness transonically for

a) a small chord unswept rudder
b) a large chord swept rudder
c) an all-moving fin

are illustrated in figure 26.

The ratio $\frac{\text{rudder effectiveness at 1.2M}}{\text{rudder effectiveness at 0.8M}}$ may be taken as a figure of merit and it can be seen that the ratio varies between 0.2 and 0.8 for the three designs.

Obviously this is not a major design consideration but a value of 0.6 is a good aim. The following features favour smooth transonic characteristics:-

i) large rudder to fin chord ratio
ii) trailing edge sweep
iii) high torsional stiffness.

3.5.6 Limitations on Rudder Travel

Having decided the rudder travel required to satisfy various requirements over the speed range it is necessary to investigate how much rudder travel can be tolerated before fin loads or aircraft response impose a limit. Generally BAC policy is to define the design fin load from rapid rolling considerations and safeguard the structure in rudder-induced manoeuvres by limiting rudder travel either by feel forces, by mechanical stops, or by jack stall.

Structural design requirements define the rudder inputs required but the type of input differs widely between different national requirements, viz:-

British (Av.P.970) : a) Rudder deflected and held
b) Fishtail manoeuvre

French (Air 2004D) : Triangular input

U.S. (Mil. Spec.) : Trapezoidal input.

Consequently the allowable rudder travels differ widely in designing to different requirements. However none of the requirements recognise the importance of the response in roll and pitch due to rudder, which can impose limits. A rudder always induces roll; this is normally in the same direction as yaw, so roll and yaw combine to produce pitch-up through inertia coupling.

The following criterion is suggested:

"In response to a rudder applied and held using a pedal force of 800 Newtons (173 lb.), with other controls fixed, the following initial peak response values should not be exceeded:-

Rate of roll greater than that corresponding to $\frac{3}{4}$ aileron.
Incremental Normal Acceleration of 2g.
Lateral acceleration of 0.6g."

Even with these safeguards against structural overloading and loss of control due to rudder and with similar limits on aileron usage, combined aileron and rudder usage can easily lead to structural or handling limits being exceeded.

In a tight air to air combat situation the pilot needs to use rudder as well as aileron, frequently in an unco-ordinated manner, to engage or evade the enemy. At present there is no guidance in any official requirements on combined aileron and rudder usage. The need to use both controls should be recognised and official requirements framed, to avoid under-design leading to structural and handling problems.

3.6 Directional Control on the Ground

Control of aircraft track on the ground, particularly at high speeds during the early part of the landing run in a crosswind, requires a separate set of design considerations. It is not sufficient merely to ensure that there is adequate directional control to meet the turning radius requirements at low speed and to resist weathercocking at high speed. The following additional factors must be considered:-

i) The gearing of the directional control (nosewheel steering in particular) should be designed to avoid oversensitivity at high speed. A tentative correlation of pilot opinion with directional control sensitivity for nosewheel steering controlled from the rudder pedals is shown in figure 27.

ii) The response and discrimination of the steering power control should be sufficient to prevent pilot-induced steering oscillations. An example of one such incident is shown in figure 28.

iii) The equilibrium of side forces and yawing moments during the crosswind landing ground roll must be investigated over a range of forward speeds and crosswinds taking account of:-

runway state : dry/wet/flooded
braking parachute or thrust reverser : on/off
wheel brakes : full and reduced braking
nosewheel steering : operative/inoperative
roll control position : into wind/neutral/downwind
longitudinal control : stick forward neutral/back

and the associated equilibrium control positions and tyre slip angles determined. Limits are encountered when either:-

a) full control is required to prevent yaw into wind
b) full control is required to prevent roll downwind
c) yaw and sideforce control cannot be maintained simultaneously and the aircraft slides downwind.

Of the three limiting conditions the latter, (c) is the most difficult to predict theoretically and difficult for the pilot to control, for he is not immediately conscious of loss of control because the directional control applied may be quite small. A typical situation is depicted in figure 29. When equilibrium of yawing moments and sideforces cannot be satisfied with the same values of yaw control angle and tyre slip angle, this is indicated by the fact that the two lines do not cross. In this situation the pilot tends to maintain tight control in yaw but is not immediately conscious of sliding until a significant downwind drift has occurred. At this point the correct (but not instinctive) action is probably to relax wheel braking to restore tyre adhesion for resisting sliding; however it is dangerous to generalise too far because of the large number of variables involved and each case must be studied individually.

3.7 Summary of Lateral Stability and Control Considerations

1. Roll control design should take account of:-

Combat roll performance
Landing approach
Crosswind landing
Manoeuvring with asymmetric stores
Spin recovery.

2. Fin design should take account of:-

Stability with bank angle constrained
Rapid rolling behaviour

Possible coupling of spiral and roll subsidence modes.

3. Choice of wing dihedral/anhedral and wing-body incidence influence fin design.

4. Crosswind landing requirements generally size the rudder but the following additional considerations can influence the design:-

   Spin recovery
   Asymmetric stores
   Transonic change of effectiveness
   Turn co-ordination requirements.

5. Current official design requirements are considered to be illogical in relation to:-

   Combat rate of roll
   Crossed controls in sideslip

   and deficient in relation to:-

   Asymmetric stores
   Release and breakaway manoeuvres
   Rudder-induced manoeuvres - pitch up
   \- roll response
   Combined aileron and rudder usage
   Lateral g induced by asymmetric manoeuvres

6. There is a need for further generalised wind tunnel and theoretical work in the following areas, to develop Data Sheet methods:-

   Fin interference effects due to roll control
   Differential tail effects
   Directional stability at high angle of attack
   Spoiler pitching moments.

7. There is a need for further handling qualities work in relation to:-

   Bank angle response to turbulence to develop suitable design criteria
   Directional control on the ground.

4. STALL CHARACTERISTICS - CHOICE OF WING PLANFORM

This subject has been placed last, not because it is least important; in fact it is probably the most important configuration design decision. However it is not considered at length in this paper because wing design, including the prediction of buffet boundaries and usable lift limits is felt to be a specialist topic.

Suffice it to say here that in an evasive combat situation buffet can never constitute an absolute limit and to avoid being shot at a pilot will pull up to the point where a definite limitation such as stall or roll/yaw divergence imposes a compulsive limit.

The effects of planform on longitudinal stability, lateral stability and control power must therefore be taken into account in the project design stage. The planform must not be decided solely on the grounds of performance, or it could impose unduly severe penalties in the size and weight of stabilising and control surfaces to maintain adequate stability and control up to and beyond the stall. To cite two examples:-

i) Increasing wing sweep allows the designer to use a thicker wing, simplifying flap design and increasing fuel stowage volume. However some of these benefits are eroded when the increased roll control demands for crosswind landing and combat manoeuvres and the increased difficulty in providing satisfactory handling qualities at high incidence are taken into account.

ii) Increasing wing aspect ratio improves cruise and thrust-limited g performance but entrains the possibility of pitch-up and wing drop, which are difficult to control without artificial stall prevention; the latter restricts usable lift capability.

5. THE ROLE OF STABILITY AUGMENTATION

Throughout this paper the emphasis has been on providing fundamentally satisfactory characteristics without stability augmentation, the latter being added as a refinement. It is frequently argued that if stability augmentation is necessary at all, then it should be used to augment the basic static stability of the aeroplane, minimising the size of the stabilising and control surfaces, thus saving weight and drag : if a large authority autopilot is required, with multiplexed sensors and actuators, for terrain-following or automatic weapon delivery, then the hardware is already there and the argument for artificial stability is strengthened.

It is not too difficult to design a "fly-by-wire" manoeuvre demand system which gives satisfactory stability and control in normal flight conditions, but severe difficulties can arise in off-design conditions. The following examples illustrate this point:-

i) In a spin a roll rate demand system will sense in-spin rate of roll and apply opposite roll control. On an inertially slender aeroplane this will speed up and flatten the spin, possibly preventing recovery.

ii) In a fast roll with the principal axis of inertia at negative incidence roll and yaw rates in the same direction combine to produce nose-up pitch, through inertia coupling. A manoeuvre demand system may sense this and apply nose-down pitch control to oppose it, but in a strongly coupled roll this will aggravate the pitch-up, leading to a divergent situation.

There are solutions to these problems, in the form of manoeuvre-limiting devices or reversion to direct position control beyond certain manoeuvre limits. However when account is taken of these safeguards and the handling problems associated with sudden changes of control mode (which cannot be simulated in strongly coupled manoeuvres), some of the apparent advantages are eroded.

The question "Fly-by-wire or big tails?" is one of the main problems confronting the designers of the next generation of combat aeroplanes.

6. CONCLUDING REMARKS

6.1 The provision of satisfactory stability and control characteristics is not merely a flight development task : it must be given adequate consideration in the project stage and throughout the design phase.

6.2 The provision of satisfactory stability and control characteristics invariably compromises performance to some extent. So that stability and control can be given adequate design priority its importance should be recognised in aircraft specifications and design requirements manuals.

6.3 In many respects Official design requirements are outdated; there are notable gaps in many areas where the designer requires guidance; there are inconsistencies between handling and loading requirements; there is a lack of uniformity between requirements originating in different NATO countries.

6.4 Where existing requirements are considered unsatisfactory or do not exist, design criteria have been suggested in this paper. Some of the difficulties of designing to meet these criteria are also highlighted.

6.5 Some gaps in design data sheets for the estimation of stability and control parameters have been noted. There are areas where theoretical work and generalised wind tunnel tests would be of value.

6.6 The question of how far to go in the direction of providing satisfactory handling characteristics by artificial stability requires continued effort, particularly in the solution of off-design problems.

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3	Anon	Flying Qualities of Piloted Airplanes.	Mil. Spec. 8785B (ASG)	
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11	W. H. Phillips	The Effects of Steady Rolling on Longitudinal and Directional Stability.	NACA TN 1627	1948
12	W. J. G. Pinsker	Critical Flight Conditions and Loads Resulting from Inertia Cross-Coupling and Aerodynamic Stability Deficiencies.	RAE TN Aero 2502	1957
13	F. D. Newell	Ground Simulator Evaluations of Coupled Roll - Spiral Mode Effects on Aircraft Handling Qualities.	AFFDL TR 65-39	1965

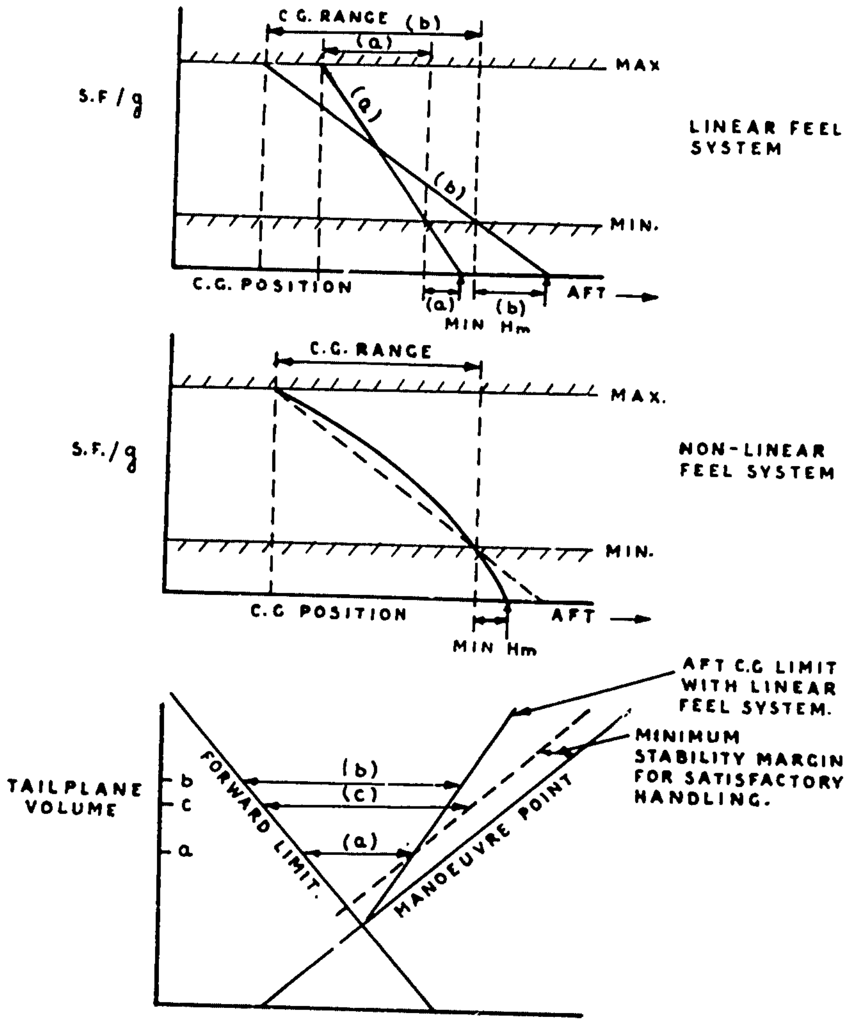


FIG 1 EFFECT OF STICK FORCE REQUIREMENTS ON MINIMUM STABILITY MARGIN AND TAILPLANE SIZE

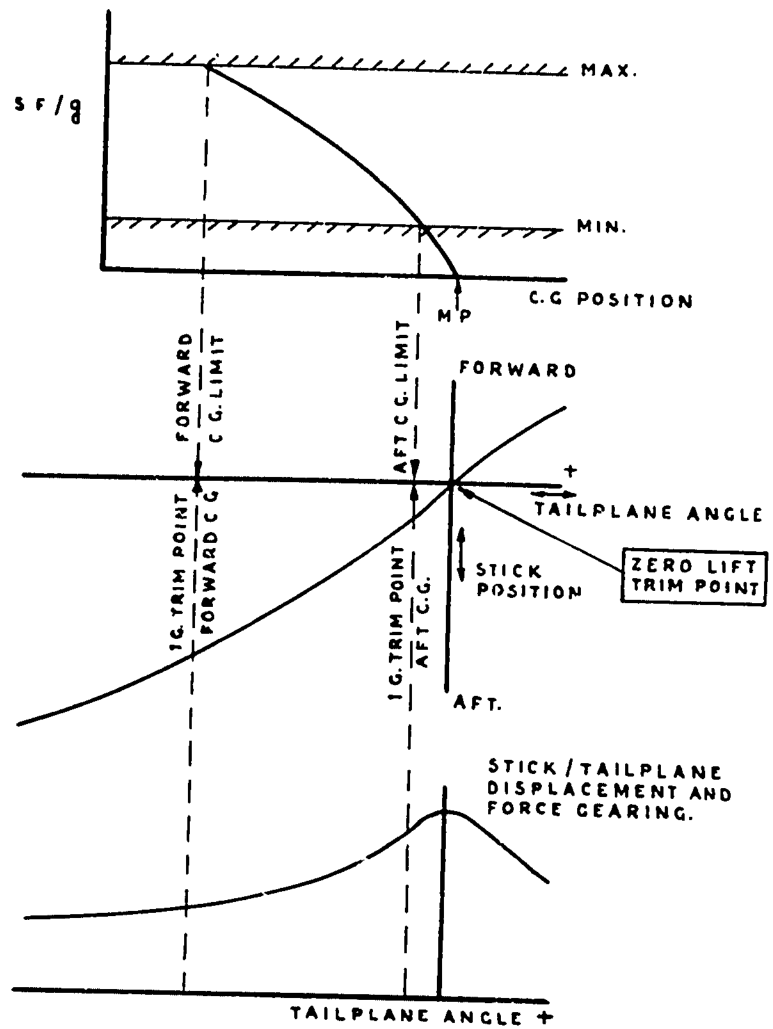


FIG. 2 DESIGN FEATURES OF A NON-LINEAR STICK/TAILPLANE GEARING

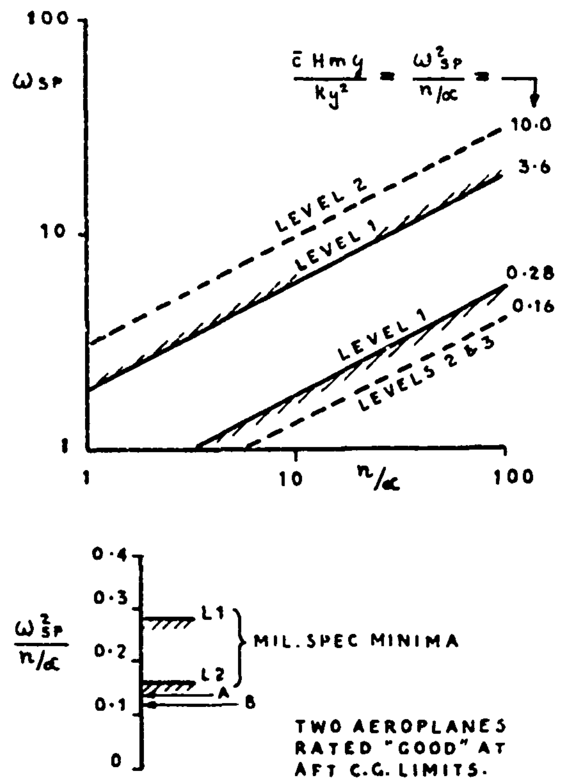


FIG. 3 LONGITUDINAL STABILITY REQUIREMENTS (U.S.A.F. MIL.SPEC.)

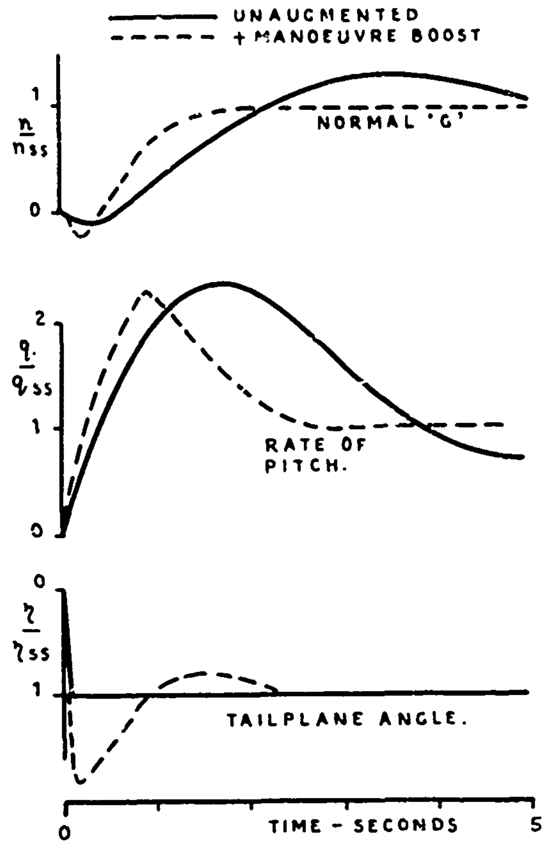


FIG. 4. EFFECT OF MANOEUVRE BOOST ON TRANSIENT RESPONSE TO TAILPLANE.

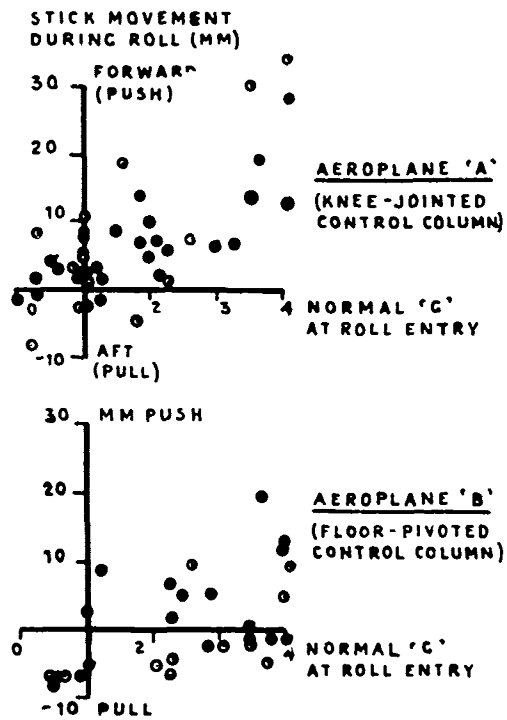


FIG. 5. EXAMPLES OF INADVERTENT FORE AND AFT STICK MOVEMENTS IN RAPID ROLLS ON TWO AEROPLANE TYPES.

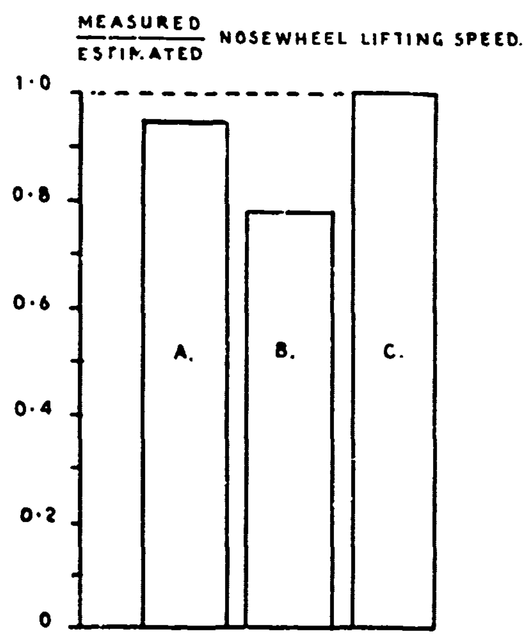


FIG. 6 RELATIONSHIP BETWEEN MEASURED AND ESTIMATED NOSEWHEEL LIFTING SPEEDS ON THREE AEROPLANE TYRES.

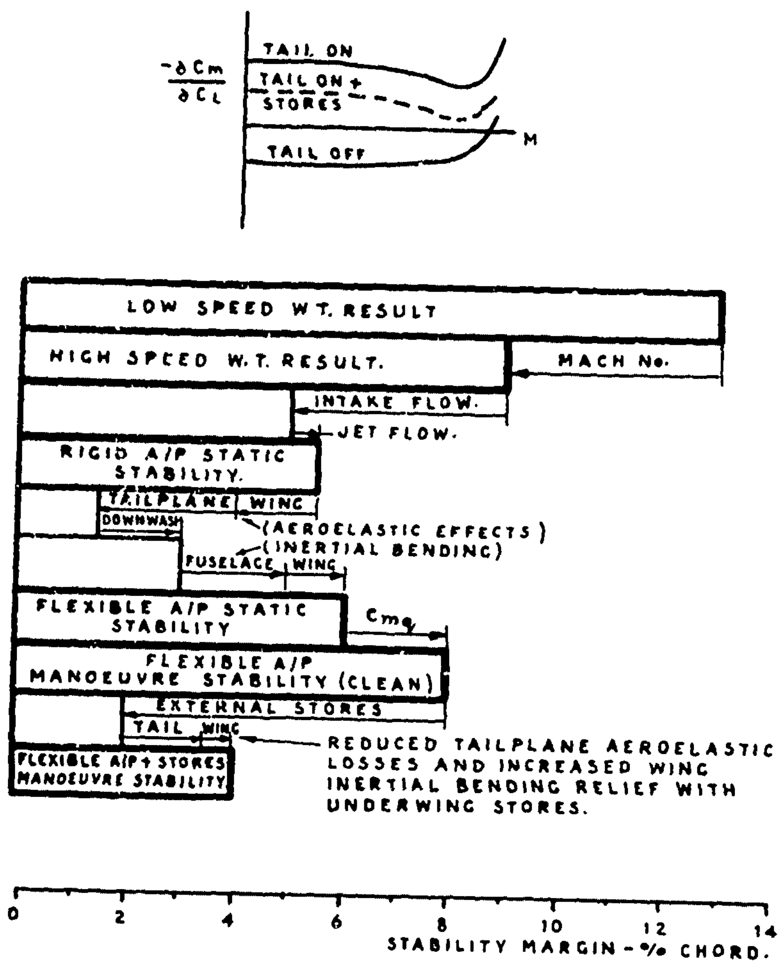


FIG.7. EFFECTS TO BE TAKEN INTO ACCOUNT IN ESTIMATING LONGITUDINAL STABILITY FROM WIND TUNNEL RESULTS.

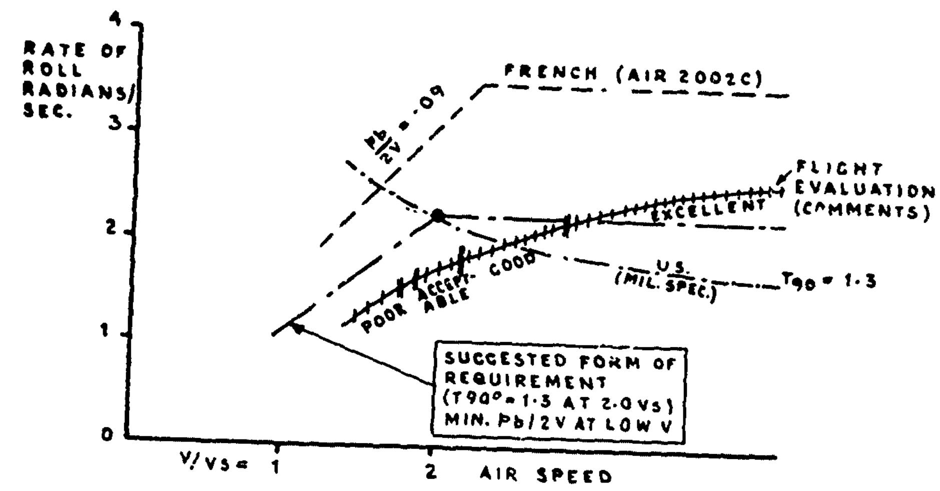


FIG 8 RATE OF ROLL REQUIREMENTS COMPARED WITH PILOT COMMENT FROM FLIGHT EVALUATION

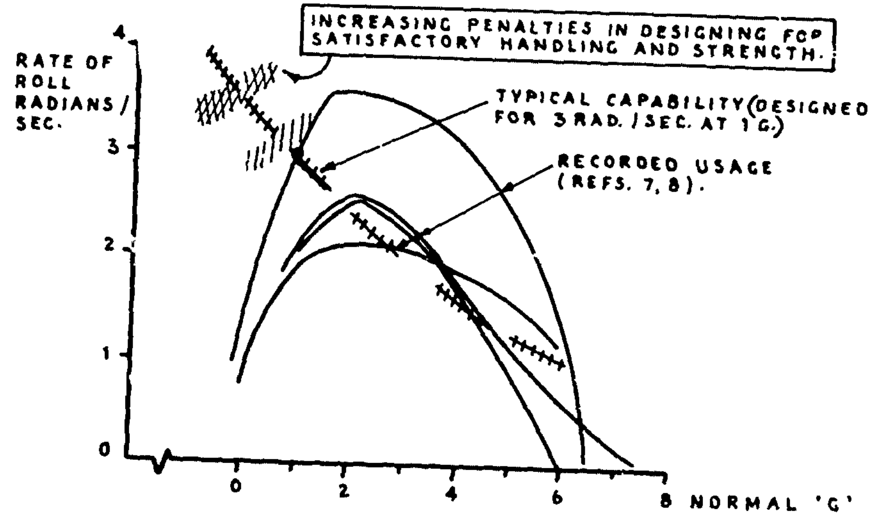


FIG 9 RATE OF ROLL VARIATION WITH NORMAL ACCELERATION.

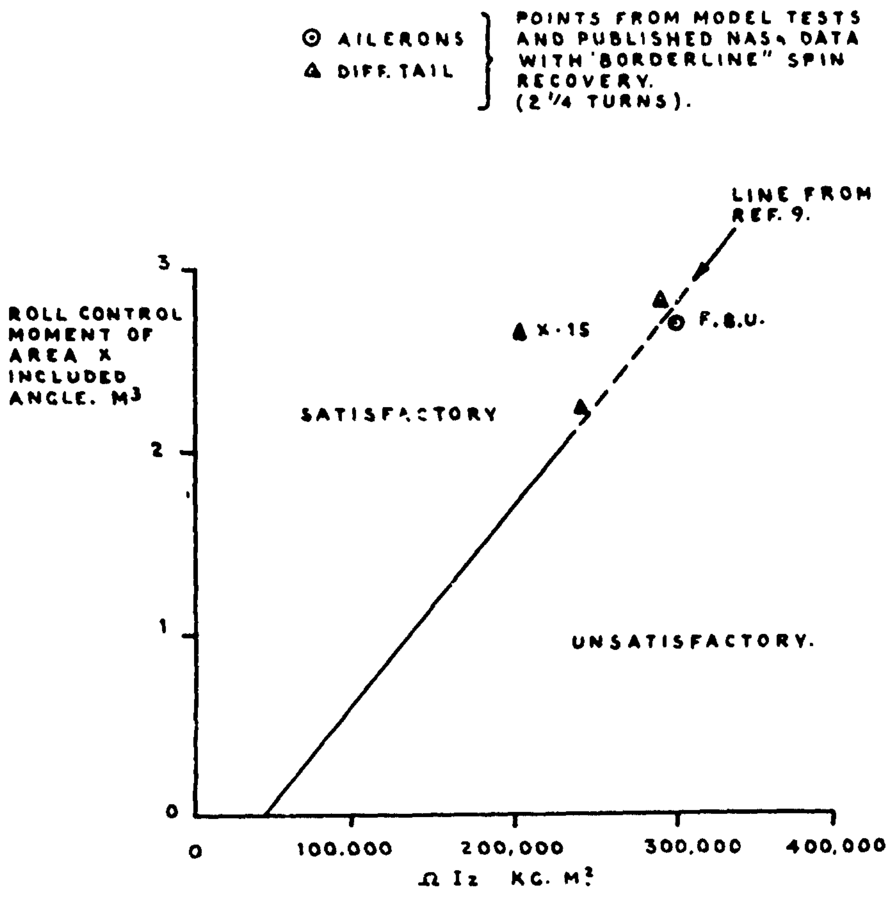


FIG. 10 ROLL CONTROL REQUIRED FOR SPIN RECOVERY

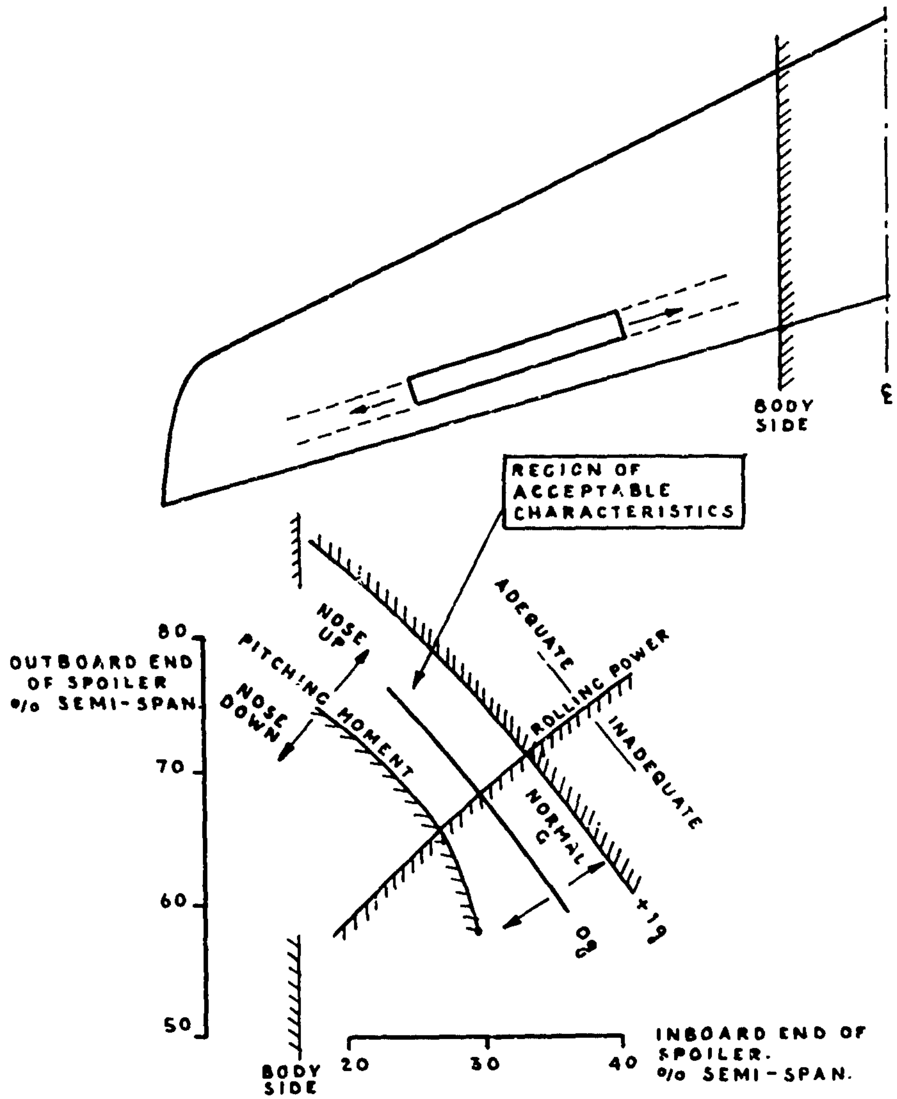


FIG. 11 SPOILER SPANWISE POSITIONING TO MINIMISE PITCHING MOMENT EFFECT

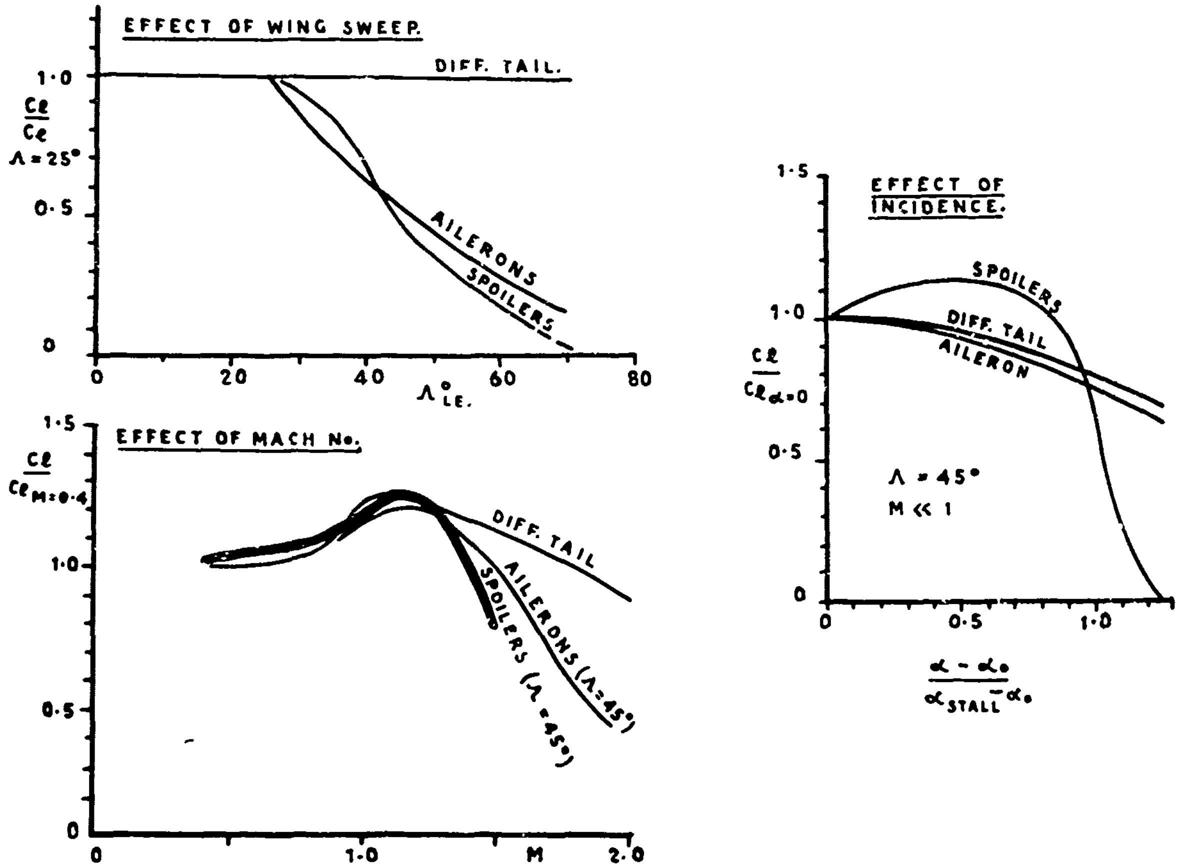


FIG.12 VARIATION OF CONTROL EFFECTIVENESS WITH SWEEP, MACH N° AND INCIDENCE FOR ALTERNATIVE ROLL CONTROLS

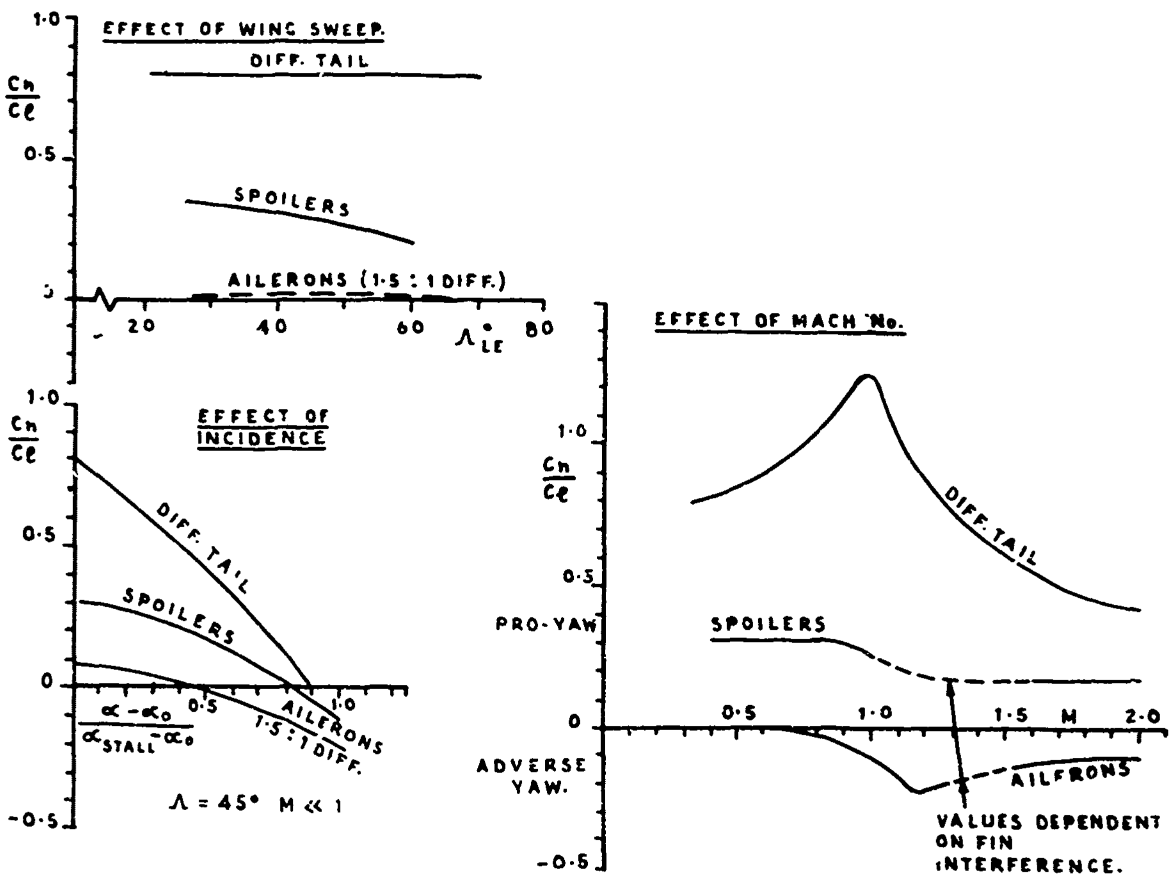


FIG 13 VARIATION OF YAWING MOMENT DUE TO ROLL CONTROL WITH SWEEP, MACH N° AND INCIDENCE FOR ALTERNATIVE ROLL CONTROLS

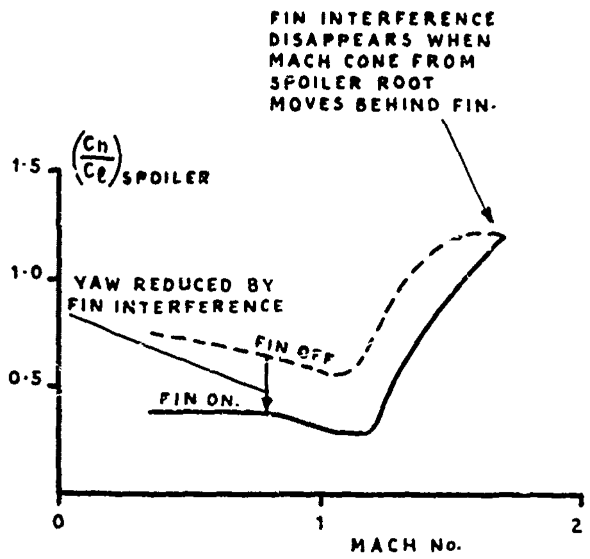


FIG.14. VARIATION OF SPOILER YAW/ROLL EFFECTIVENESS RATIO WITH MACH NUMBER.

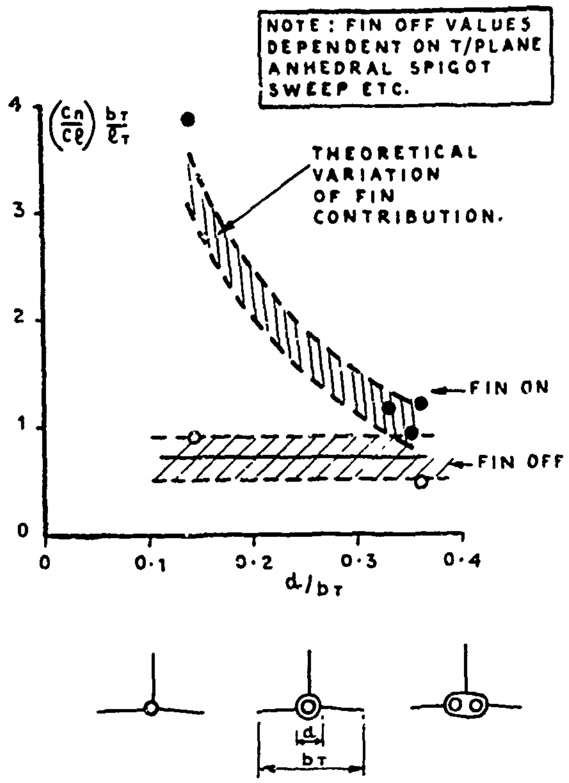
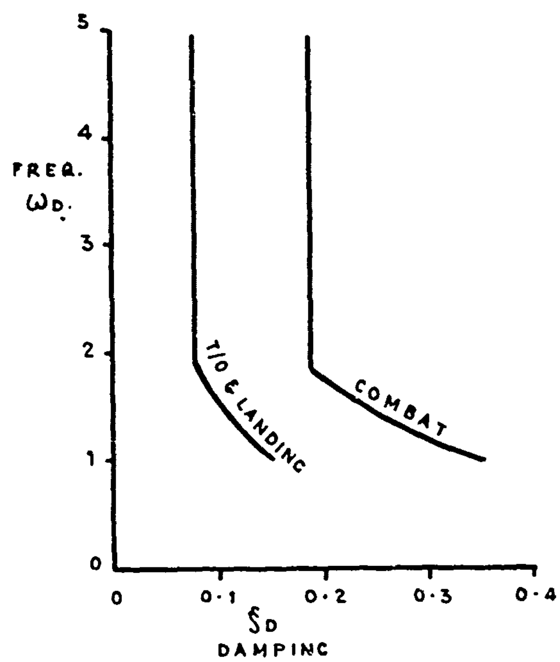


FIG.15. EFFECT OF AFTERBODY WIDTH ON YAWING MOMENT DUE TO DIFFERENTIAL TAIL.
$\alpha = 0$; $\delta_H = 0$; $M < 1$.



REF. MIL. SPEC. F. 8785 B. (ASG)
(APPLICABLE TO CLASS IV. LEVEL 1.)

FIG16. MINIMUM DUTCH ROLL FREQUENCY AND DAMPING.

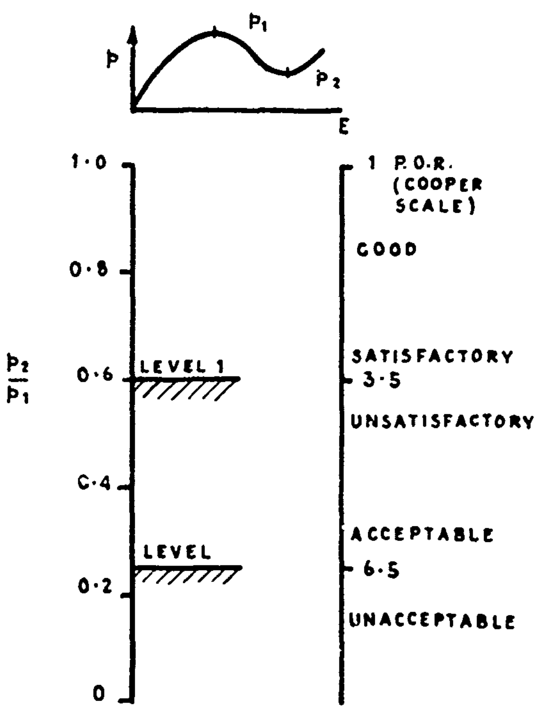


FIG. 17. PILOT OPINION OF RATE OF ROLL OSCILLATION DUE TO ADVERSE SIDESLIP.

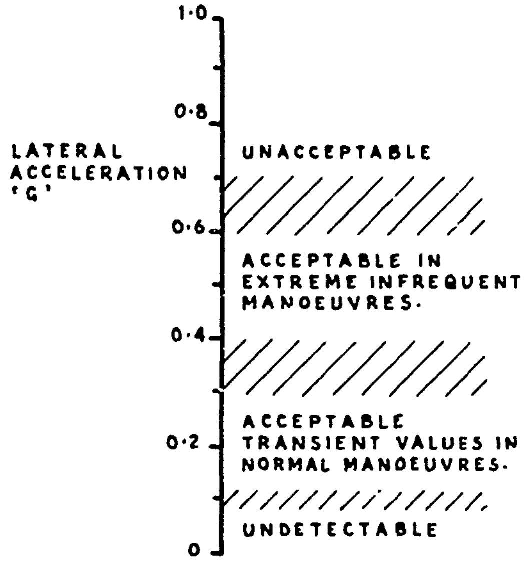


FIG.18. TENTATIVE PILOT OPINION OF LATERAL 'G' IN RAPID ROLLS.

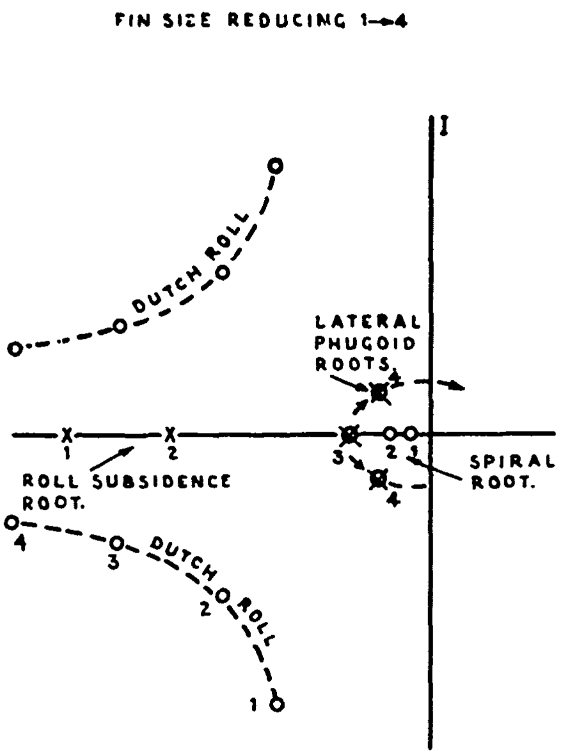


FIG.19. LATERAL PHUGOLD ONSET WITH REDUCING FIN SIZE ON SLENDER AEROPLANE WITH HIGH C_{l_β} & HIGH $\frac{C_{n_p}}{C_{l_p}}$ (ROOT LOCUS PLOT.)

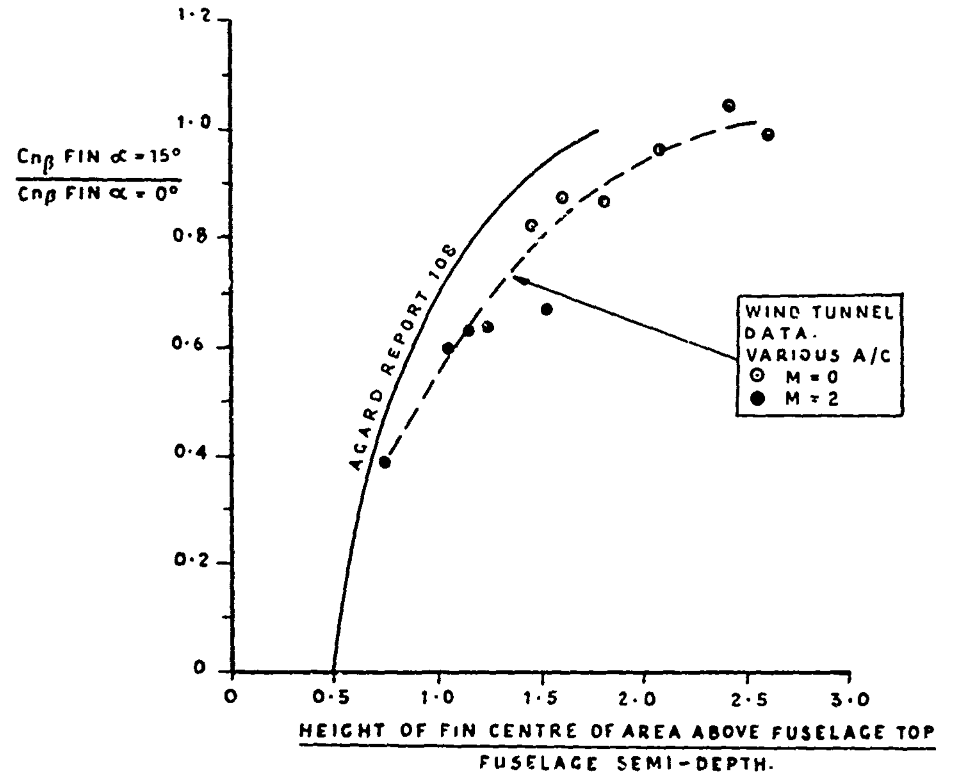


FIG. 20 EFFECTS OF ANGLE OF ATTACK ON FIN CONTRIBUTION TO DIRECTIONAL STABILITY

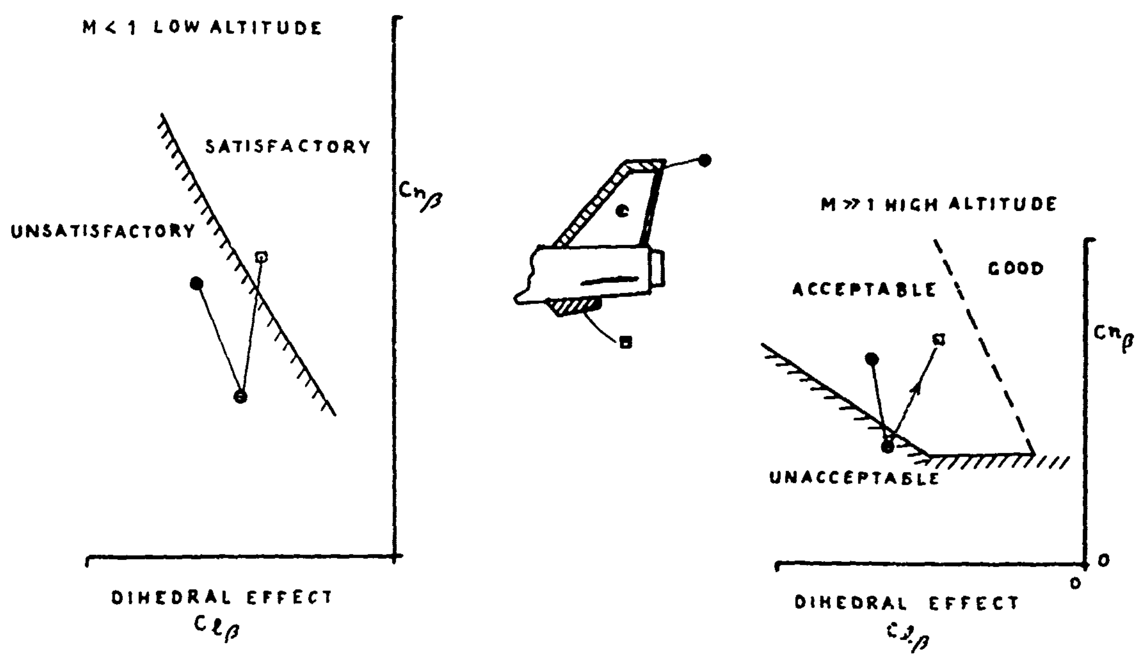


FIG. 21 EFFECTS OF ADDITIONAL BASIC FIN AREA COMPARED WITH VENTRAL FIN OF SAME AREA ON STABILITY AND HANDLING QUALITIES.

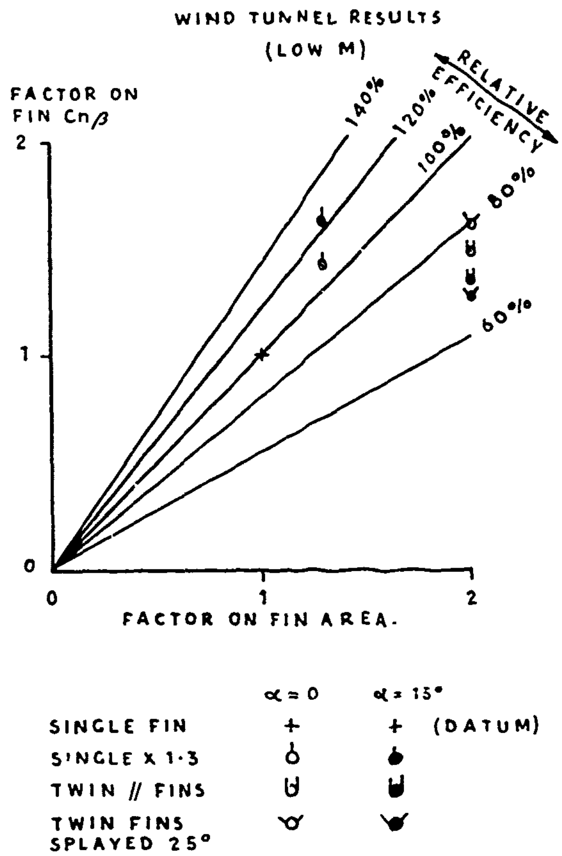


FIG. 22. COMPARISON OF SINGLE AND TWIN FIN CONTRIBUTIONS TO DIRECTIONAL STABILITY.

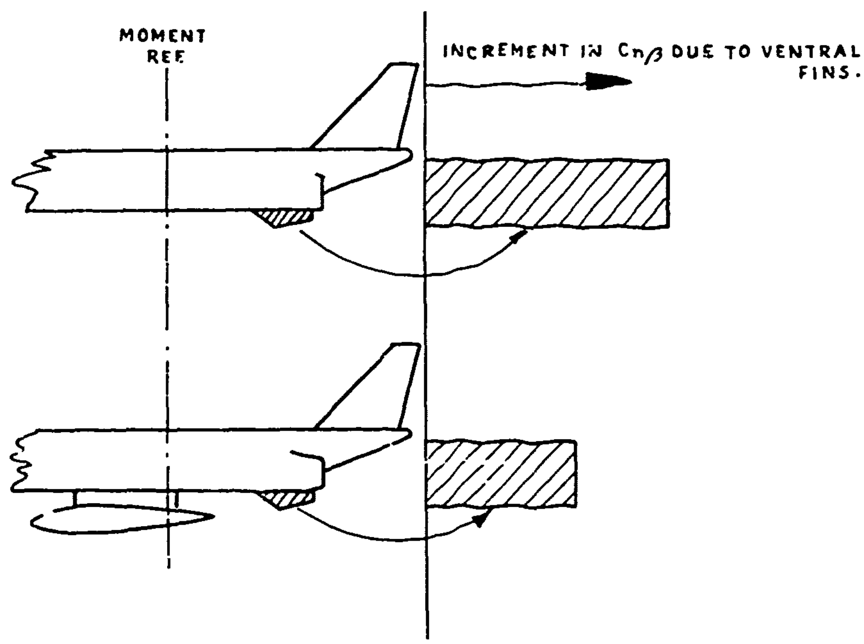


FIG 23 REDUCED EFFECTIVENESS OF VENTRAL FINS IN THE PRESENCE OR A VENTRAL STORE

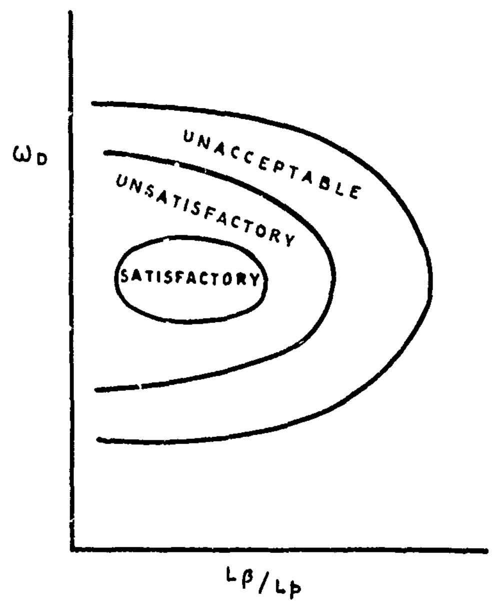


FIG. 24. A POSSIBLE CRITERON FOR LATERAL HANDLING QUALITIES IN TURBULENCE.

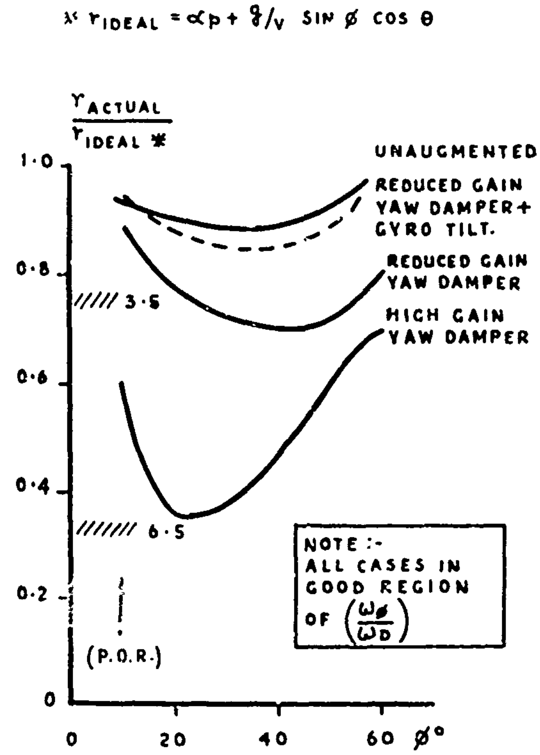


FIG. 25. TURN ENTRY MANOEUVRE PILOT COMMENTS RELATED TO RATE OF YAW DEVELOPMENT.

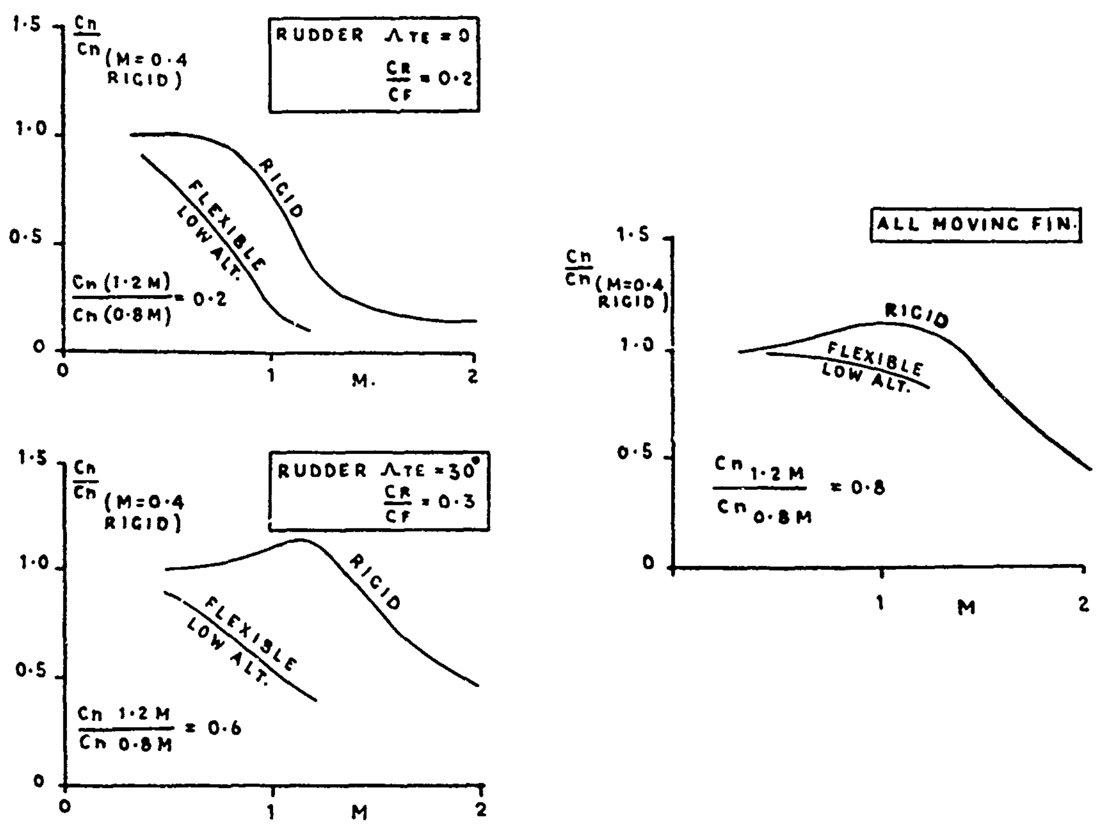


FIG. 26. TRANSONIC CHANGE IN RUDDER EFFECTIVENESS: EFFECT OF FIN & RUDDER GEOMETRY

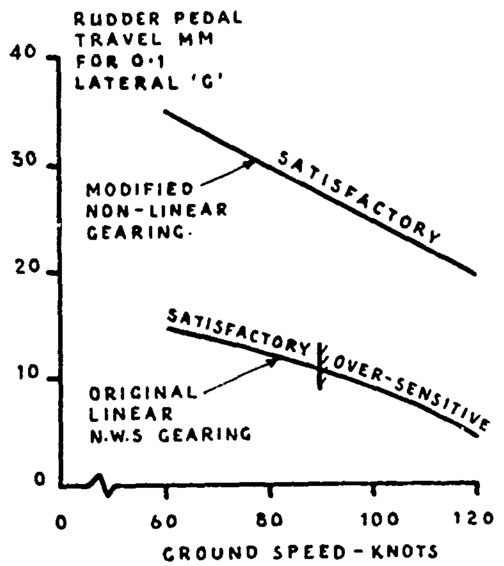


FIG. 27. NOSEWHEEL STEERING PILOT COMMENTS RELATED TO STEADY-STATE RESPONSE.

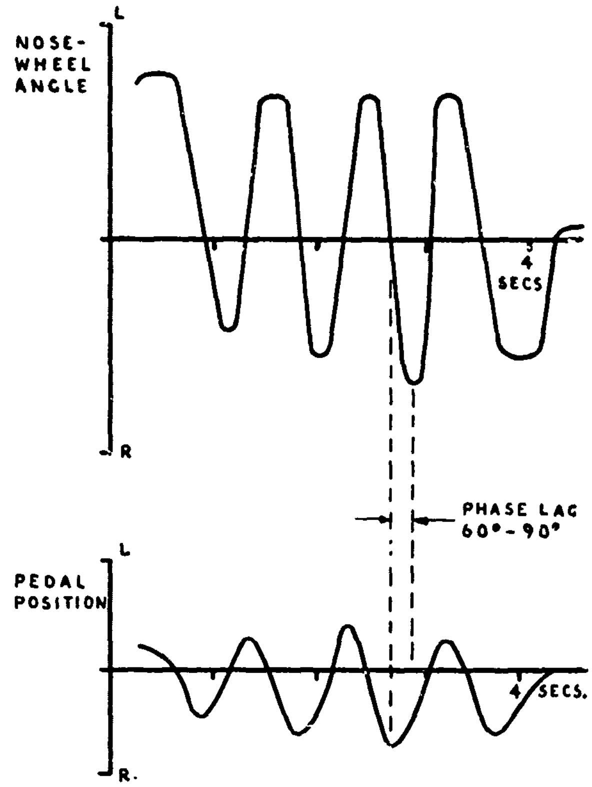


FIG. 28. EXAMPLE OF PILOT-INDUCED STEERING OSCILLATION DUE TO EXCESSIVE PHASE LAG IN RESPONSE.

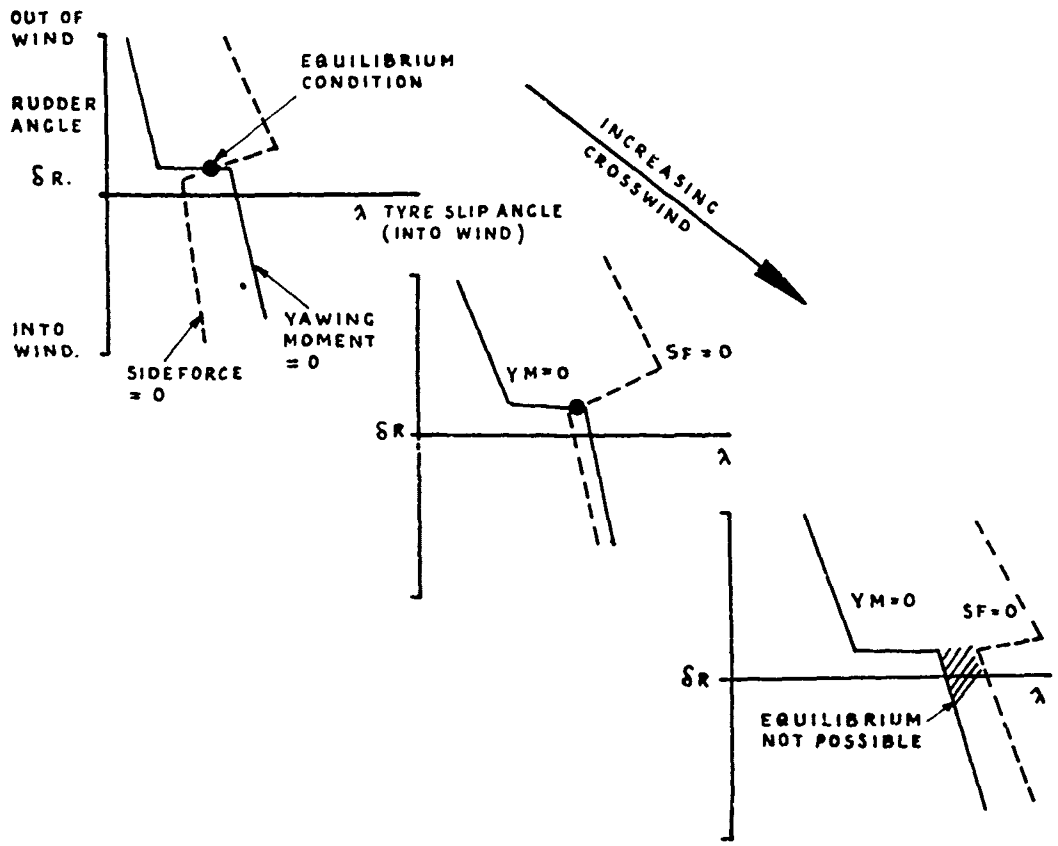


FIG. 29. CROSSWIND LANDING GROUND ROLL EQUILIBRIUM CONDITIONS

OPEN DISCUSSION

H.H.B.M.Thomas, UK: I would first like to congratulate Mr Burns on a most useful paper and Mr Tucker on his presentation. My question concerns your tentative requirements regarding lateral acceleration during rolling manoeuvres. Do the lateral acceleration levels refer to the pilot's station or to the cg.?

J.H.Tucker, UK: Answered, that the lateral acceleration levels refer to the pilot's station.

M.Hacklinger, Germany: Mr Tucker has referred to the MIL-Spec. roll performance requirement as a roll rate requirement. This is not quite true: MIL-F-8785 B defines time to bank, which includes the transient initial period of a roll manoeuvre. But my main question is on the speed parameter. As far as I recall, not only AVP 970 but also the old MIL-Spec. for handling qualities had formulated the roll requirement as a helix angle which included aircraft speed as a parameter. Could Mr Woodcock inform us why the new MIL-Specification has excluded speed from the roll performance requirement?

R.J.Woodcock, USA: The form of the MIL-F-8785 B roll performance requriements is based on both theoretical and experimental considerations. The inadequacy of pb/2V was stated, for example, by Patterson and Spangenburg in AGARD Report 419. Bank angle in a given time was suggested by such investigators as Mazza, Becker et al. (NADC-ED-6282), Ashkenas (AFFDL-TR-65-138), and Creer, Stewart et al. (NASA Memo 1-29-59 A). These same reports describe the need for an upper limit on the roll-mode time constant, in order to assure precise control. The arguments are summarized in the MIL-F-8785 B background material, AFFDL-TR-69-72.

The parameter $C_{n\beta_{eff}} = C_{n\beta} - C_{l\beta}\,C_{n\delta}/C_{l\delta}$ illustrates the danger of considering only one parameter. While the effective Dutch roll frequency (analogous to $C_{n\beta_{eff}}$ for tightly-controlled wings-level flight) indeed increases with proverse yaw, such pilot control may at the same time drive the Dutch roll damping unstable for the closed-loop system.

T.B.Saunders, UK: Commented on Mr Woodcock's objection to the $C_{n\beta_{eff}}$ criterion. The implications of this parameter for roll control with rudder fixed were possibly more easily understood than directional control with bank angle constrained. This relation to the ω_ϕ/ω_d criterion was also recalled.

Some criterion of this type is necessary although $C_{n\beta_{eff}}$ is not necessarily the best parameter to use.

P.Wüst, Germany: A question concerning Figure 18: Could you give a definition of extreme infrequent manoeuvres? For instance, are these rapid rolls at high normal g's?

J.H.Tucker, UK: Answered, that rapid rolls at high normal g's are not extreme infrequent manoeuvres. For combat aircraft such manoeuvres may lie in the design envelope.

THE EFFECTS OF THRUST CHARACTERISTICS ON LONGITUDINAL STABILITY IN SUPERSONIC FLIGHT

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SUMMARY

The influence of the variation of thrust with speed and height on the dynamic stability of the longitudinal motion in supersonic flight is shown. The first part of the paper describes the effects directly related to thrust changes. The second part considers the effects due to pitching moments which, associated with thrust characteristics, depend on speed and height. In addition, the thrust influence on two methods of artificial stabilization of the long-term modes is investigated.

1. NOMENCLATURE

a	speed of sound
a_h	$= (1/a)\ da/dh$
C_D	drag coefficient+)
C_{Do}	drag coefficient at zero lift
C_L	lift coefficient+)
C_m	moment coefficient+)
C_{mo}	aerodynamic trim moment
C^*_{mV}	denoting effective pitching moment change due to speed change ($\Delta C_m = C^*_{mV}\Delta V/V$)
$C^*_{m\varrho}$	denoting effective pitching moment change due to height change ($\Delta C_m = C^*_{m\varrho}\ \varrho_h \Delta h$)
$\bar{c}$	mean aerodynamic chord
$\partial C_D/\partial C_L$	$= C_{D\alpha}/C_{L\alpha}$
$\partial C_m/\partial C_L$	$= C_{m\alpha}/C_{L\alpha}$
D	drag
g	acceleration due to gravity
h	height
i_y	radius of gyration about the y-axis
K_V	control loop gain (speed feedback to moment)
K_h, $K_{\dot{h}}$	control loop gain (feedback of height and height rate to moment)
k	$= \partial C_D/\partial(C_L^2)$ drag due to lift
k_ϱ	$= -(g/\varrho_h)/V^2$
M	Mach number
m	mass

+) Additional subscript denotes partial derivative.

n_K	denoting thrust change due to change in the variable given by subscribt (K = M, Te, V, ϱ)
n_V^*	denoting effective thrust change due to speed change ($\Delta T/T = n_V^* \Delta V/V$)
n_ϱ^*	denoting effective thrust change due to height change ($\Delta T/T = n_\varrho^* \varrho_h \Delta h$)
q	angular velocity in pitch
$\bar{q}$	dynamic pressure
S	wing area
s	Laplace operator
s	transfer function zero (with subscripts denoting transfer function)
s_5	height mode root
s_τ	$= (\mu\bar{c}/V)s$
T	thrust
Te	temperature
T_T	thrust time constant
V	speed
z_T	thrust line offset
α	angle of attack
α_T	angle between flight path tangent and thrust line
β_h	$= (a_h/\varrho_h)\beta_M$
β_M	$= M^2/(M^2-1)$
γ	flight path angle
Δ	denoting change of a variable, e. g., ΔV
δ_e	elevator angle
δ_T	throttle deflection
ε_P	$= \sin^{-1} \zeta_P$
ζ_P, ζ_{SP}	damping ratio of the phugoid, short period mode
μ	$= 2m/(\varrho S\bar{c})$ relative density of the airplane
ϱ	air density
ϱ_h	$= (1/\varrho)\, d\varrho/dh$ density gradient
σ	real part of complex variable
σ_P	phugoid damping
ω	imaginary part of complex variable
ω_{nP}, ω_P	undamped natural frequency, frequency of the phugoid
ω_{nSP}	undamped natural frequency of the short period mode

2. INTRODUCTION

In high speed flight, the longitudinal motion and dynamic stability are considerably influenced by the change of atmosphere with height (Ref. 1, 2, 3, 4). These effects increase with the flight Mach number and are especially important in the supersonic region (Ref. 5, 6, 7). This is closely connected with the behavior of the shapes of the long-term modes: The height changes, as compared with the speed changes, are much more significant in supersonic flight than in the subsonic case. From this point of view, the effects of speed and height on thrust gain increased importance upon stability in the supersonic region. This is possible in two ways: Directly, by affecting the thrust-drag relationship and, indirectly, by inducing pitching moments due to speed and height changes.

The purpose of this paper is to show these effects in explicit form and to illustrate their numerical size using a supersonic transport at a Mach number of 3 and an altitude of 21000 m as an example.

3. EQUATIONS OF MOTION AND THRUST MODEL

The equations of motion for longitudinal flight including the effects of the change of the atmosphere with height can be written in linearized form as

$$\begin{pmatrix} s_\tau+(2-\beta_M)C_D & C_{D\alpha}+C_D\tan\alpha_T & s_\tau(C_L+C_D\tan\alpha_T)+\mu\bar{c}\varrho_h C_D(1+\beta_h-n_\varrho^*) \\ (2-\beta_M)C_L+n_V^*C_D\tan\alpha_T & C_{L\alpha}+C_D & -s_\tau^2+\mu\bar{c}\varrho_h(C_L(1+\beta_h)+n_\varrho^*C_D\tan\alpha_T) \\ \mu C_{mV}^* & -s_\tau^2(i_y/\bar{c})^2+s_\tau(C_{mq}+C_{m\dot{\alpha}})/2+\mu C_{m\alpha} & -s_\tau^3(i_y/\bar{c})^2+s_\tau^2C_{mq}/2+\mu^2\bar{c}\varrho_h C_{m\varrho}^* \end{pmatrix} \begin{pmatrix} \frac{\Delta V}{V} \\ \Delta\alpha \\ \frac{\Delta h}{\mu\bar{c}} \end{pmatrix} =$$

$$= - \begin{pmatrix} C_{D\delta_e} & -C_D \\ C_{L\delta_e} & C_D\tan\alpha_T \\ \mu C_{m\delta_e} & \mu(z_T/\bar{c})C_D/\cos\alpha_T \end{pmatrix} \begin{pmatrix} \Delta\delta_e \\ \Delta\delta_T \end{pmatrix} . \quad (1)$$

In the thrust model, the variation of thrust with flight condition is taken into consideration according to

$$\Delta T/T = n_V\Delta V/V+n_M\Delta M/M+n_\varrho\Delta\varrho/\varrho+n_{Te}\Delta Te/Te. \quad (2)$$

With regard to the variables used in the equations of motion, i.e., speed and height, the coefficients $n_{V,M,\varrho,Te}$ can be combined to yield the effective values n_V^* and n_ϱ^*, which are given by

$$n_V^* = n_V+n_M ,$$
$$n_\varrho^* = n_\varrho+(a_h/\varrho_h)(2n_{Te}-n_M) . \quad (3)$$

A parameter which indicates the significance of all height-dependent effects and which will be repeatedly referred to is

$$k_\varrho = -(g/\varrho_h)/V^2 = -C_L/(\mu\ \bar{c}\ \varrho_h) . \quad (4)$$

The US-Standard Atmosphere (Ref. 8), which is used in this paper to describe the variation of atmospheric conditions with height, shows that the density gradient is approximately constant in the altitude range of interest. From this it follows that

$$k_\varrho|_{M>2} \ll 1$$

for Mach numbers $M>2$ (Fig. 1). Due to this fact, the influence of height-dependent factors on stability is significant. The paper presented here concentrates upon this Mach number range. Moreover, Fig. 1 also shows, that

$$|a_h| \ll |\varrho_h| .$$

From this it follows that the compressibility effects caused by height changes are small as compared with the density effects ($M>2$), i.e.,

$$|\beta_h| \ll 1 .$$

As a consequence of the density and temperature gradient, the characteristic equation of the linearized equations of motion is of the fifth order and, thus, yields 5 roots. Corresponding to these, there are usually three modes of motion. These are the short-period and phugoid modes, and, in addition to the low speed case, the so-called height mode. The latter is an aperiodic motion in which mainly height and speed changes are involved.

4. DIRECT THRUST EFFECTS

In the cases considered first, it is assumed that there are no speed- or height-dependent effects on the pitching moment ($C^*_{mQ} = 0$, $C^*_{mV} = 0$). The influence of thrust characteristics is then dire ly related to thrust changes, which are described by Eq. (3).

As a result of the approximate factorization of the characteristic equation, the height mode root can be expressed as

$$s_5 \cong (1-k_Q)[n^*_V - (2-\beta_M)n^*_Q] \frac{g}{V} \frac{C_D}{C_L} . \qquad (5)$$

From this it follows that the height mode root is determined by thrust characteristics, i.e., decrease of thrust with either speed or height are stabilizing and vice versa with the stability boundary given by

$$n^*_V \cong (2-\beta_M)n^*_Q . \qquad (6)$$

This is shown in Fig. 2 for a supersonic transport at a Mach number of 3 and an altitude of 21000 m (referred to as "basic airplane") the characteristics of which are given in table 1. The range chosen for the thrust parameters n^*_Q and n^*_V is such as to include moderate gains of a control loop with height and/or speed feedback to thrust. With regard to aerodynamics, the only quantity of importance is the lift-drag ratio, to which the value of s_5 proportional (Fig. 2). However, this ratio - within certain limits - can be considered a fixed number in respect to a given design Mach number or cruising speed.

As to the phugoid mode, the approximate factorization for the undamped natural frequency yields

$$\omega^2_{nP} \cong -g\varrho_h \frac{1+k_Q}{1+C_{mq}/(2\mu\partial C_m/\partial C_L)} . \qquad (7)$$

In case of $|2\mu\partial C_m/\partial C_L| \gg |C_{mq}|$, ω_{nP} is mainly a result of the density gradient and, thus, approximately constant within the altitude range of interest. Flight condition and airplane configuration, including thrust characteristics, have no effect (Fig. 3).

In contrast, phugoid damping is strongly influenced by configuration-dependent factors, and especially by thrust characteristics. It is given by

$$\sigma_P \cong (n^*_Q + k_Q n^*_V - 1)(1 - \frac{\beta_M}{2}) \frac{g}{V} \frac{C_D}{C_L} + \Delta\sigma_{P1}(C_{m\alpha}) \qquad (8)$$

where ($|2\mu\partial C_m/\partial C_L| \gg |C_{mq}|$)

$$\Delta\sigma_{P1}(C_{m\alpha}) \cong \bar{c}\varrho_h \frac{g}{V} \frac{C_{L\alpha}}{8C_{m\alpha}} [4(\frac{i_y}{c})^2 + \frac{C_{mq}(C_{mq}+C_{m\dot{\alpha}})}{\mu C_{m\alpha}}] .$$

The first part in Eq. (8) represents the influence of the forces on phugoid damping, whereas the term $\Delta\sigma_{P1}(C_{m\alpha})$ denotes the contribution due to the interaction of forces and pitching moments. The first part, being the decisive component, shows the thrust characteristics to be the dominant effect on phugoid damping. As shown in Fig. 3, it is destabilizing if thrust decreases with height or increases with speed. Because of

$k_Q \ll 1$, speed influence is of minor importance. It is interesting to note that the density gradient effects, when limited to aerodynamics only, depend solely on the pitching moment characteristics. If

$$4\mu C_{m\alpha} \approx -(\bar{c}/i_y)^2 C_{mq}(C_{mq}+C_{m\dot{\alpha}}) \qquad (9)$$

there is no change in the aerodynamic contribution to phugoid damping. In case of larger $|C_{m\alpha}|$-values, the density gradient introduces a destabilizing component.

Combin ng the results for the roots of the phugoid and height mode, there are two main points: The first one is, that the effect of height on thrust is substantial for the stability in supersonic flight. As shown in Fig. 4, it is always destabilizing, no matter how thrust changes with height. This is in contrast to the low speed region, where height-dependent thrust changes have no effect on stability at all. Particularly with respect to the phugoid, the case $n_Q^* = 1$ which can be considered a typical value of air-breathing propulsion systems in the stratosphere (Ref. 9) shows that thrust characteristics are the main reason for the reduction of the damping in supersonic flight, so that the oscillation is practically undamped at Mach numbers $M > 2$.

The second point is the effect of speed on thrust. As also shown in Fig. 4, the height mode is destabilized by increase of thrust with speed and vice versa. In terms of n_V^* and n_Q^*, speed and height influence are of equal significance. With regard to the phugoid, the effect of speed on thrust is of minor importance, being steadily reduced with an increase in flight Mach number ($k_Q \propto 1/V^2$). This is further illustrated when investigating thrust control by speed feedback, as shown in Fig. 5. Due to the density gradient, those zeros of the speed-thrust transfer function which are of interest in this connection are located close to the phugoid roots, i.e.,

$$\begin{aligned} s_{VT3}+s_{VT4} &\approx 2\sigma_P|_{n_Q^*=1,n_V^*=0} + (g/V)\tan\alpha_T , \\ s_{VT3}\cdot s_{VT4} &\approx (1-k_Q)\,\omega_{nP}^2 . \end{aligned} \qquad (10)$$

Thus, the closed loop roots of the phugoid remain near the open loop roots, with the result that there is almost no stabilization. If α_T is positive, as it usually is, the stabilization still existing is further reduced or even turned to the opposite. The only quantity to be stabilized is the height mode root. These speed effects, which are a consequence of the density gradient, represent again a difference to the subsonic case, and especially to low speed flight. Here, the phugoid and thus the whole motion are strongly stabilized by speed feedback to thrust (Ref. 10, 11). This is shown in Fig. 6 for a Mach-2-SST in low speed flight ($M = 0.24$).

In order to give physical interpretation of the thrust effects in supersonic flight, a suitable procedure is to investigate the shapes of the phugoid and height mode and to apply the time vector method (Ref. 12). The dominant components in both modes are speed and height. With regard to the height mode, the ratio of speed and height changes is approximately a linear function of the flight Mach number, i.e.,

$$\frac{\Delta V}{\Delta h} \approx -\frac{1}{2-\beta_M}\,\varrho_h\,V . \qquad (11)$$

This means that the ratio of the forces induced by speed and height is constant, which indicates that the influence of n_V^* and n_Q^* is of equal significance (Fig. 7). In contrast, the ratio of speed and height changes of the phugoid decreases with flight Mach number according to

$$\left.\frac{\Delta V}{\Delta h}\right|_1 \approx \frac{g}{V} . \qquad (12)$$

The ratio of the forces, then, decreases even with the square of flight Mach number. The result is that the speed effect on the forces becomes very small as compared with the height effect for Mach numbers $M > 2$ (Fig. 7). The thrust characteristics have an influence in so far as they introduce a second component in case of $n_Q^* \neq 1$. It is given by

$$\frac{\Delta V}{\Delta h}\Big|_2 \approx (n_Q^*-1)\sqrt{-g\varrho_h}\,\frac{C_D}{C_L} \qquad (13)$$

with a phase angle of 90° relative to $\Delta V/\Delta h|_1$. Combining the results of the investigation of the roots and the mode shapes, the time vector method makes evident the dominance of all height-induced effects on the phugoid. This is illustrated in Fig. 8. Comparison with the case $M = 0.24$ of a Mach-2-SST points out the difference between the supersonic and the low speed region.

5. THE EFFECTS OF PITCHING MOMENTS DUE TO THRUST CHARACTERISTICS

The propulsion system can cause speed- and height-dependent pitching moments in various ways. Without going into details, it can be stated that there is a moment if the thrust line does not pass through the center of gravity. Another moment results from thrust effects on the flow field, especially at the tail location. A third contribution is associated with the change of the direction of the mass flow passing through the engines. The effects of such moments cannot be separated from the influence of other speed- and height-dependent moments, which, for example, may be the result of aeroelasticity or compressibility. Therefore, the moment effects due to thrust characteristics are shown by using the effective derivatives

$$\begin{aligned} C_{mV}^* &= n_V^*(z_T/\bar{c})C_D + MC_{mM} + C_{mV} + 2C_{mo} + 2C_{m\bar{q}} \;, \\ C_{mQ}^* &= n_Q^*(z_T/\bar{c})C_D - (a_h/\varrho_h)MC_{mM} + C_{mo} + C_{m\bar{q}} \end{aligned} \qquad (14)$$

which are considered to combine all such moment components. For example, the C_D-terms denote the contributions due to thrust offset.

With use of C_{mV}^* and C_{mQ}^*, the height mode root can be expressed as

$$s_5 = \frac{g}{V}\,\frac{(n_V^*-(2-\beta_M)n_Q^*)C_D\partial C_m/\partial C_L - a_1 C_{mQ}^* + a_2 C_{mV}^*}{C_L\partial C_m/\partial C_L - C_{mQ}^* - k_Q C_{mV}^*} \;. \qquad (15)$$

The coefficients $a_{1,2}$, which are determined by thrust and drag characteristics, are given by

$$a_1 = n_V^*\frac{C_D}{C_L} + (2-\beta_M)\left(\frac{\partial C_D}{\partial C_L} - \frac{C_D}{C_L}\right) ,$$

$$a_2 = n_Q^*\frac{C_D}{C_L} + (1+\beta_h)\left(\frac{\partial C_D}{\partial C_L} - \frac{C_D}{C_L}\right) .$$

There are two main properties of the effects due to C_{mQ}^* and C_{mV}^*. First, positive values of C_{mQ}^* or C_{mV}^* do not change the height mode root appreciably. This is a direct result of the approximation for s_5. It is confirmed by investigating the root locus of a moment control with height or speed feedback, in respect to which C_{mQ}^* and C_{mV}^* can be interpreted as gains. This investigation shows that the real open loop pole, i.e., s_5 in case of $C_{mQ}^* = 0$ and $C_{mV}^* = 0$ (Eq. (5)), and the corresponding zero of the height-moment or speed-moment transfer function, i.e.,

$$s_{h\delta 1} \approx s_5|_{n_Q^*=0} - (2-\beta_M)\,\frac{g}{V}\left(\frac{C_D}{C_L} - \frac{\partial C_D}{\partial C_L}\right) \qquad (16)$$

and $(1/(\mu k\bar{c}) \gg |\varrho_h| C_D/C_L)$

$$s_{V\delta 1} \approx \varrho_h\, V\left[(n_Q^*-1)\frac{C_D}{C_L} + \frac{\partial C_D}{\partial C_L}\right] , \qquad (17)$$

are located closely together. Thus, the height mode root stays within the limits given by these quantities, if the gain is such as to correspond to positive C^*_{mQ} and C^*_{mV}, respectively.

In contrast, the second point is that negative values of C^*_{mQ} and C^*_{mV} can change the magnitude of the height mode root substantially. This is the case if

$$C^*_{mQ} + k_Q C^*_{mV} \longrightarrow C_L \partial C_m/\partial C_L \ . \tag{18}$$

Due to $k_Q \ll 1$, the C^*_{mQ}-effect is dominating, while the C^*_{mV}-influence is reduced with square of flight Mach number. The strong stability or instability of the height mode which then exists depends on the nominator of Eq. (15), which is mainly affected by the thrust parameters n^*_V and n^*_Q .

These effects are illustrated in Figs. 9 and 10, where the important parameters shown in the expression for s_5 (Eq.(15)) are varied. If not otherwise stated, it is assumed, that there is a combined pitching moment according to $C^*_{mV} = 2\cdot C^*_{mQ}$. In Fig. 9, the case of decreased short-period frequency, i.e., $\omega_{nSP} = 0.8\ \mathrm{sec}^{-1}$ as compared with $\omega_{nSP} = 1.5\ \mathrm{sec}^{-1}$ of the basic airplane (with regard to flying qualities, refer to Ref.13, 14), shows the effect of reduced $C_{m\alpha}$, which, in general, causes an equivalent increase of the influence of C^*_{mQ} and C^*_{mV}. The reduction of both C_L and ω_{nSP}, which is illustrated by the example M = 3.5, points out the increased susceptibility of s_5 to negative C^*_{mQ}-values. The case $C^*_{mV} = 0$, in comparison with all other examples, shows that the C^*_{mV}-influence is mainly restricted to the nominator of s_5. This means that C^*_{mV} affects the sign of s_5, i.e., stability instead of instability when $|s_5|$ is large, but it has only a minor influence on the strong increase of $|s_5|$ which is determined by C^*_{mQ}.

Fig. 10 makes evident that n^*_V and n^*_Q have significant effects on the sign of s_5, while the strong increase of $|s_5|$ is independent of them. This is particularly obvious in the case $n^*_V = n^*_Q = 0$ where s_5 is almost equal to zero when $C^*_{mQ} > 0$.

The effects of C^*_{mV} and C^*_{mQ} on phugoid frequency can be expressed as

$$\omega^2_{nP} \cong \omega^2_{nPo}\left[1 - \frac{C^*_{mQ} + k_Q C^*_{mV}}{C_L\ \partial C_m/\partial C_L}\right] \tag{19}$$

with ω_{nP_o} denoting the case $C^*_{mV} = C^*_{mQ} = 0$ shown in Eq. (7). The damping is given by

$$\sigma_P \cong \sigma_{Po} + \Delta\sigma_{P2} - s_5/2 \tag{20}$$

where

$$\Delta\sigma_{P2} = \frac{2g/V}{2\partial C_m/\partial C_L + C_{mq}/\mu}\left[kC^*_{mV} - \frac{\bar{c}\varrho_h}{C_L}\left(\left(\frac{i_y}{\bar{c}}\right)^2 - \frac{C_{mq}+C_{m\dot{\alpha}}}{2C_{L\alpha}}\right)\frac{C^*_{mQ}+k_Q C^*_{mV}}{2\partial C_m/\partial C_L + C_{mq}/\mu}\right] \ . \tag{21}$$

σ_{Po} which represents phugoid damping in case of $s_5 = 0$ with $C^*_{mV} = C^*_{mQ} = 0$ can be disregarded in this context.

Eq. (19) shows that phugoid frequency increases with C^*_{mQ} and/or C^*_{mV}. It becomes zero in the range of $C^*_{mQ} + k_Q C^*_{mV} \cong C_L\ \partial C_M/\partial C_L$, and thus, the phugoid motion aperiodic. Again, the effect of C^*_{mV}, as compared with that of C^*_{mQ}, decreases due to k_Q with the square of flight Mach number. The effects of both moments are directly related to C_L and $\partial C_m/\partial C_L$ in that they proportionally increase with a reduction of C_L or $\partial C_M/\partial C_L$.

The main property of phugoid damping is the strong interaction with the height mode root in case of large $|s_5|$-values. As shown above, this occurs when C^*_{mQ} and C^*_{mV} are negative. The result is that stability of the height mode is combined with instability of the phugoid and vice versa. From this it follows in general, that negative C^*_{mQ}-values being sufficiently large always introduce dynamic instability of the longitudinal motion. In case of positive C^*_{mQ} and C^*_{mV}, the interaction between height mode and phugoid roots is small, because the change in the height mode root itself is small. Thus, the second contribution to phugoid damping, i.e., $\Delta\sigma_{P2}$ (Eq. (21)), becomes more significant. It adds to phugoid damping in case of positive C^*_{mV} with $|C_{m\alpha}|$ assumed to be sufficiently large, and it reduces damping due to $C^*_{mQ} > 0$.

These effects are illustrated in Figs. 11 and 12. In particular, comparison with Figs. 9 and 10 shows the strong interaction between height mode and phugoid roots which always results in strong instability of the longitudinal motion.

6. THE EFFECTS OF THRUST CHARACTERISTICS ON TWO METHODS OF STABILIZING THE PHUGOID AND HEIGHT MODE

From the investigation of the airplane's inherent stability in supersonic flight, it follows that the phugoid and/or height mode are instable or, at best, slightly damped. For example, the phugoid damping of the basic airplane does not meet the phugoid damping requirement of $\zeta_P = 0.04$ as given in MIL-F-8785 B (Ref. 15). This is true, even if the case $n^*_Q = 0$ favorable for phugoid damping is considered and the increased instability of the height mode is ignored. Though the instabilities occuring for $C^*_{mQ} = C^*_{mV} = 0$ are small when considering the divergence times and an alert pilot can successfully control the airplane, it is nevertheless necessary to improve stability in order to relieve the pilot from such tasks. This is especially true for supersonic cruising flight in an air traffic control system.

In this chapter, the effects of thrust characteristics on two simple methods of artifical stabilization are investigated, the influence of which is mainly restricted to the phugoid and height mode.

The first method consists of feedback of speed to moment. The root locus plotted in Fig. 13 shows that the phugoid and height mode are effectively stabilized in the case of the basic airplane. However, the stabilization of the height mode depends on thrust characteristics, since the zeros of the V/δ_e-transfer function which are of interest in this context are strongly affected by the variation of thrust with height:

$$\begin{aligned} s_{V\delta 1} + s_{V\delta 2} &\cong -\frac{V}{2\mu k\bar{c}} , \\ s_{V\delta 1} \cdot s_{V\delta 2} &\cong \varrho_h V (s_{V\delta 1} + s_{V\delta 2}) [(n^*_Q - 1)\frac{C_D}{C_L} + \frac{\partial C_D}{\partial C_L}] . \end{aligned} \tag{22}$$

From this it follows that the stabilization of the height mode decreases with a reduction of n^*_Q. A favorable aspect is given by the fact that the size of the zeros increases with flight Mach number. Thus the effectiveness of the stabilization also improves with an increase of Mach number.

The second method utilizes feedback of height and rate of height to moment. Here again, the height mode is affected by thrust characteristics, while the phugoid can be individually stabilized by appropriate choice of the ratio of height and height rate feedback (Fig. 14). Due to the fact that the zero of the h/δ_e-transfer function which determines the stabilization of the height mode is closely located to the height mode root of the open loop (Eq.(16)), the stabilization of the height mode requires decrease of thrust with speed. If necessary, this can be accomplished by an additional control loop using speed feedback to thrust.

7. CONCLUSIONS

The thrust characteristics have significant effects on the longitudinal stability in supersonic flight, particularly for Mach numbers $M > 2$. The effects differ considerably from the subsonic case, and especially, from the low speed region. The main points can be summarized as follows:

1. The variation of thrust with height destabilizes the supersonic motion, no matter how thrust changes with height.
2. The variation of thrust with speed has little influence upon the phugoid, but it strongly affects the height mode. Increase of thrust with speed is destabilizing.
3. The phugoid cannot be effectively stabilized by a control loop with feedback of speed to thrust.
4. The height mode root is determined by thrust characteristics.
5. In case of pitching moments due to thrust characteristics, the size of the height mode root can substantially increase. As a consequence of the interaction with the phugoid, stability of the height mode is then combined with instability of the phugoid and vice versa, thus always resulting in instability of the longitudinal motion. The decisive factor in this context is the pitching moment due to height changes. The large changes of the height mode root occur if the derivative of this moment is positive ($C^*_{m_Q} < 0$).
6. With regard to artificial stabilization of the long-term modes, feedback of speed to pitching moment can stabilize both modes. In particular, the stabilization of the height mode depends on the variation of thrust with height. In case of feedback of height and rate of height to moment, stabilization of the height mode requires decrease of thrust with speed.

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TABLE 1: CHARACTERISTICS OF THE BASIC AIRPLANE +)

$m = 135\,000$ kg	$C_{Do} = 0.0082$
$S = 375$ m^2	$k = 0.585$
$\bar{c} = 19$ m	$C_{D\delta_e} = 0$
$i_y = 10$ m	$C_{L_\alpha} = 1.55$
	$C_{L\delta_e} = 0.375$
$n_V^* = 2$	$C_{m\dot{\alpha}} = -0.30$
$n_Q^* = 1$	$C_{m\delta_e} = -0.25$

The values of $C_{m\alpha}$ and C_{mq} were chosen in such a way as to yield $\omega_{nSP} = 1.5$ sec^{-1} and $\zeta_{SP} = 0.5$. $|C_{mq}|$ is considered to be artificially increased as compared with the basic aerodynamic configuration.

+) The characteristics presented here correspond to the airplane of Ref. 14.

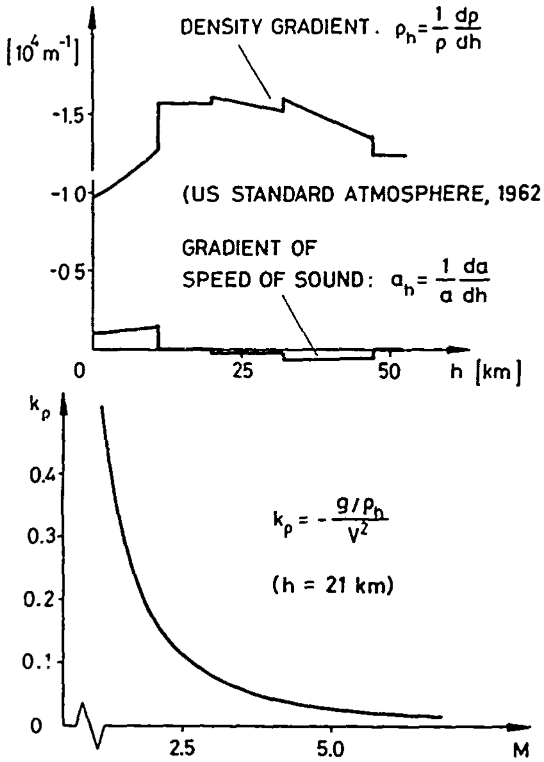


FIG.1: DENSITY GRADIENT, GRADIENT OF SPEED OF SOUND AND k_ρ

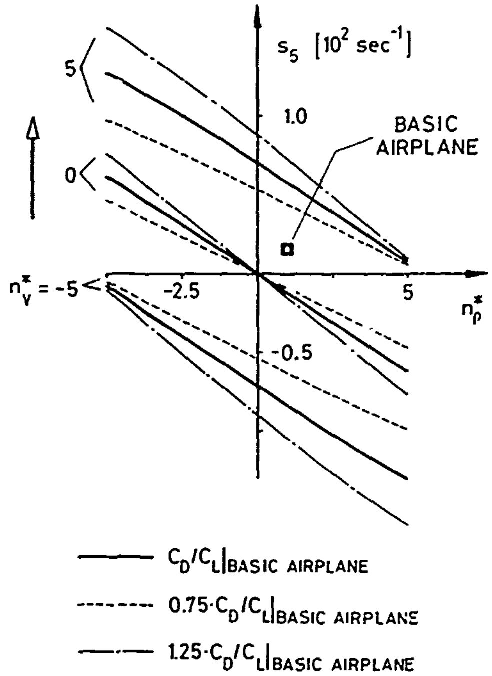


FIG.2: EFFECT OF n_ρ^* AND n_V^* ON THE HEIGHT MODE

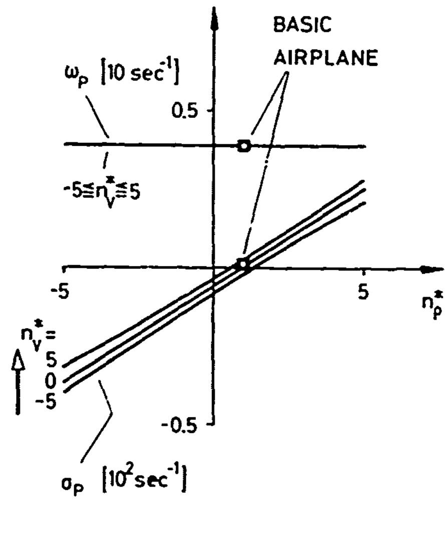


FIG.3: EFFECT OF n_ρ^* AND n_V^* ON THE PHUGOID

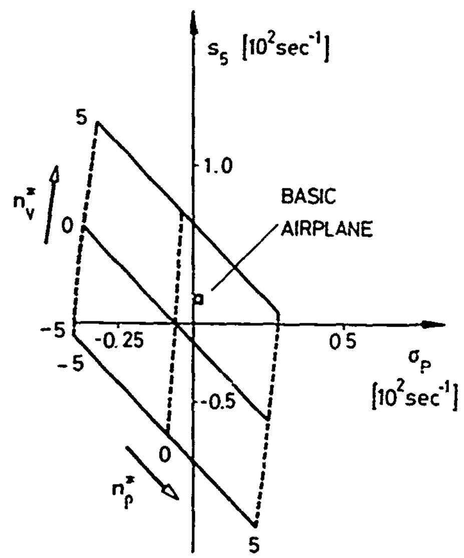


FIG.4: EFFECT OF n_ρ^* AND n_V^* ON THE HEIGHT MODE ROOT AND PHUGOID DAMPING

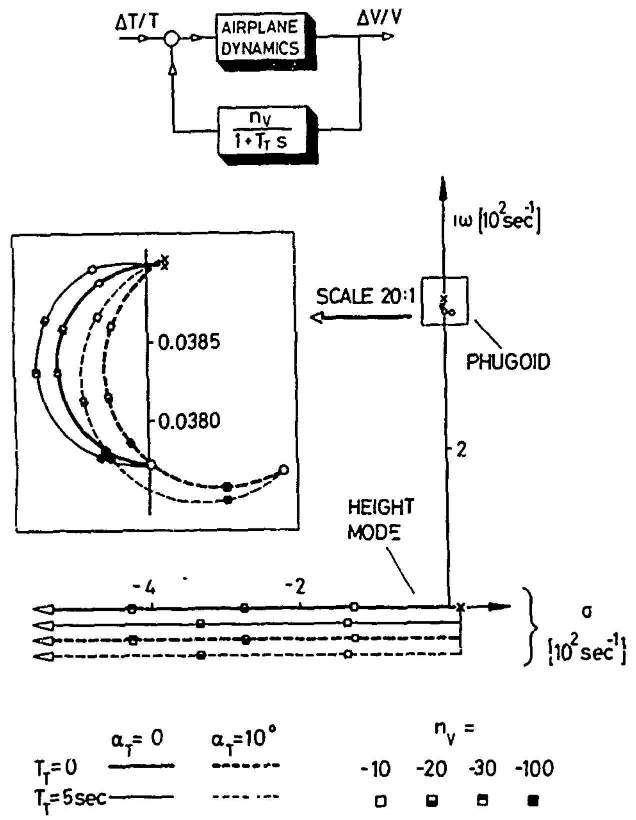


FIG.5: EFFECT OF SPEED FEEDBACK TO THRUST ON THE PHUGOID AND HEIGHT MODE IN SUPERSONIC FLIGHT (M=3)

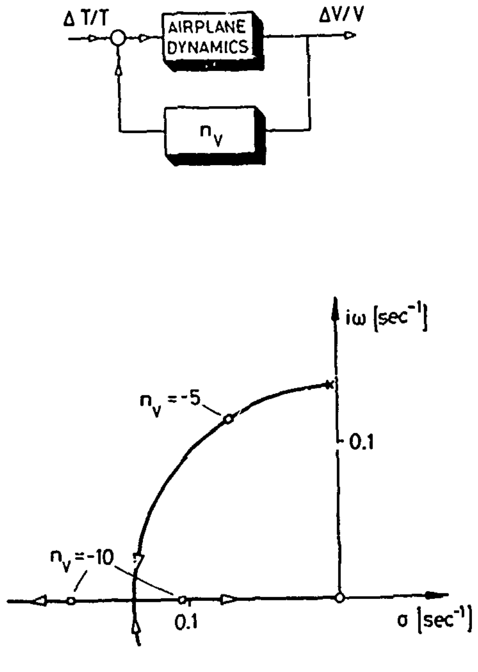
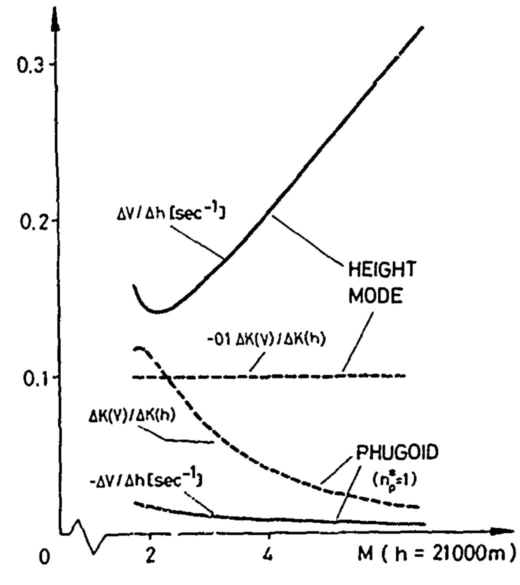


FIG.6: EFFECT OF SPEED FEEDBACK TO THRUST ON THE PHUGOID IN LOW SPEED FLIGHT (M=0.24)



ΔV/Δh . RATIO OF SPEED AND HEIGHT CHANGES (——)

ΔK(V)/ΔK(h). RATIO OF AERODYNAMIC FORCES DUE TO SPEED AND HEIGHT CHANGES (------)

FIG 7: SPEED-HEIGHT-RATIO AND AERODYNAMIC-FORCES-RATIO OF THE PHUGOID AND HEIGHT MODE

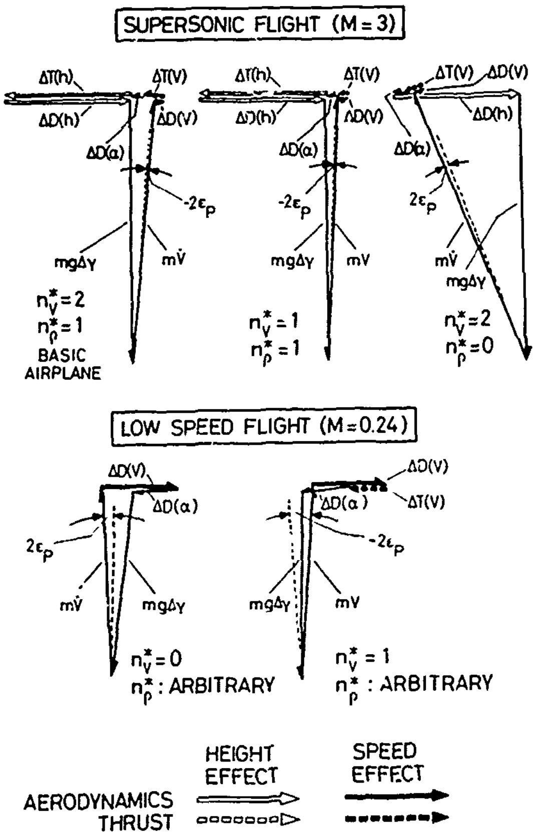


FIG.8: TIME VECTOR DIAGRAM OF THE PHUGOID FOR SUPERSONIC AND LOW SPEED FLIGHT (POLYGON OF FLIGHT PATH TANGENTIAL FORCES)

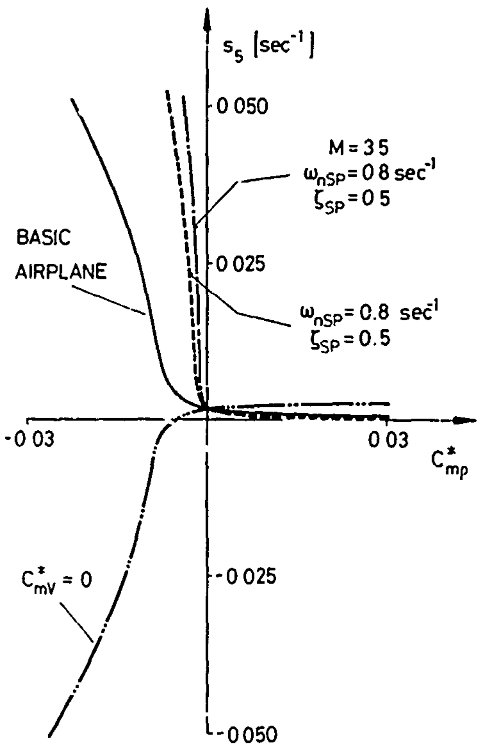


FIG.9: EFFECT OF C^*_{mp} ON THE HEIGHT MODE (PARAMETER. ω_{nSP}, M, C^*_{mV})

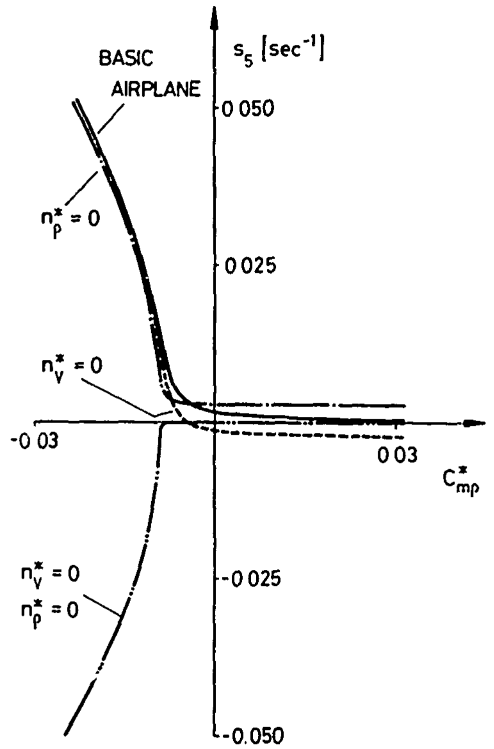


FIG.10: EFFECT OF C^*_{mp} ON THE HEIGHT MODE (PARAMETER: n^*_ρ , n^*_V)

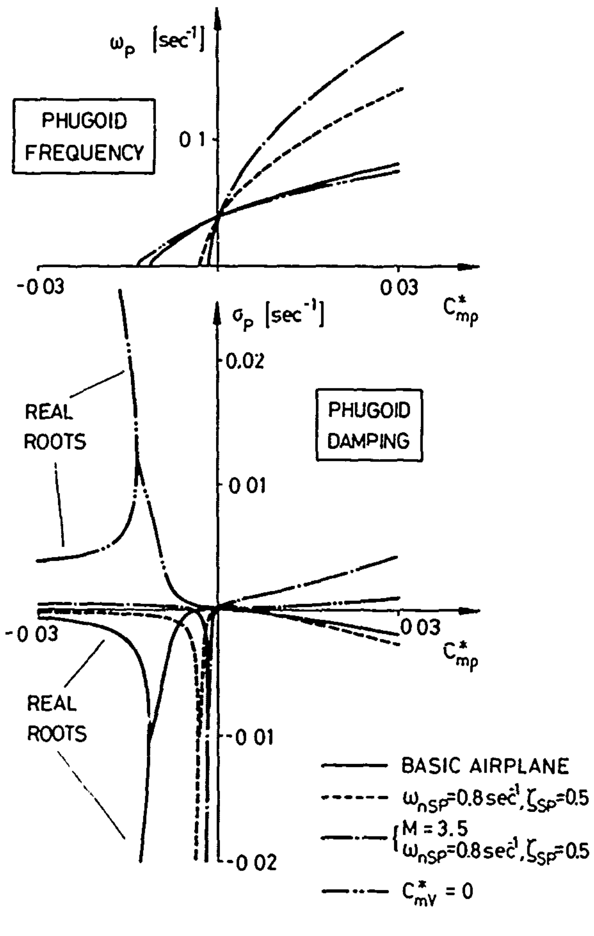


FIG 11: EFFECT OF C^*_{mp} ON THE PHUGOID MODE (PARAMETER ω_{nSP}, M, C^*_{mV})

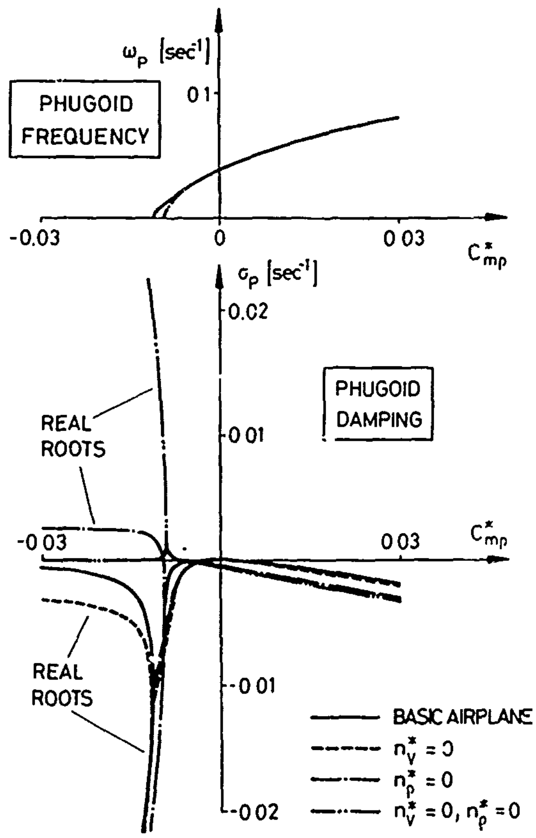


FIG.12: EFFECT OF C^*_{mp} ON THE PHUGOID (PARAMETER: n^*_ρ , n^*_V)

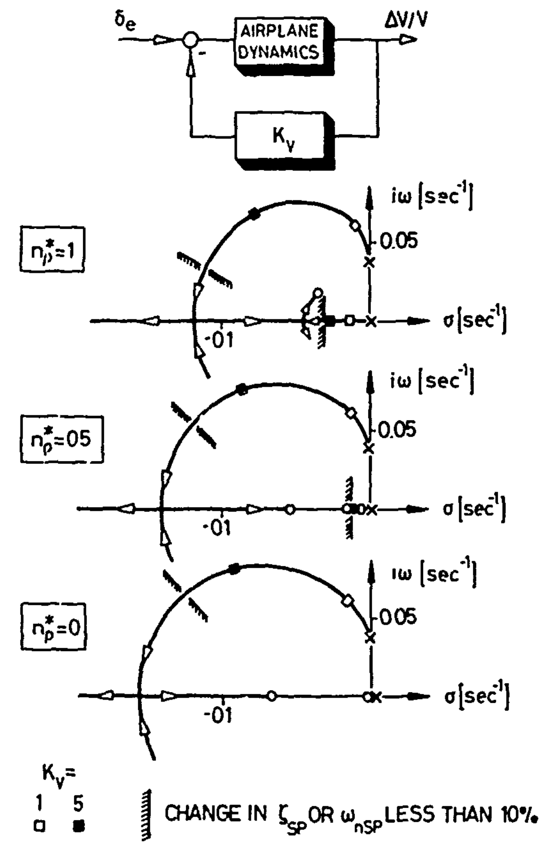


FIG.13: THRUST EFFECTS ON THE STABILIZATION OF THE PHUGOID AND HEIGHT MODE BY SPEED FEEDBACK TO MOMENT

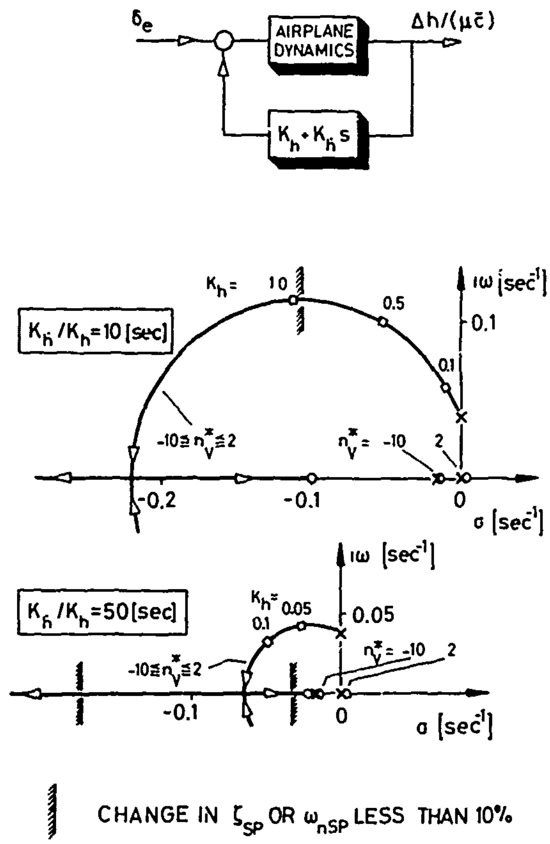


FIG.14: THRUST EFFECTS ON THE STABILIZATION OF THE PHUGOID AND HEIGHT MODE BY FEED-BACK OF HEIGHT AND RATE OF HEIGHT TO MOMENT

OPEN DISCUSSION

J.F.Renaudie, France: The major assumption of the study of Mr Sachs is that the density of the air is constant at the cruise height. From the most recent knowledge in that field, this assumption is true for average values but during the period of the phugoid or other oscillations which are considered, there are great local variations of the temperature in small distances producing significant changes of thrust. What should be the influence of these variations on the conclusions of Mr Sachs?

G.Sachs, Germany: The influence of local variations of the temperature depends on the relation between the time scale in which they are encountered by the airplane and the models of motion. In so far as thrust changes are considered, the long term motions are mainly affected. For a given local distribution of the temperature variations, the time scale is reduced with an increase of flight Mach number. This is especially important with regard to the phugoid, since its period – being almost independent of speed and altitude – is approximately constant for Mach numbers $M > 2$ (standard atmosphere).

On the whole, the disturbances due to local variations of the temperature aggravate the situation of an airplane which, as has been shown for the case of the standardized atmospheric conditions, is instable or, at best, slightly damped. This is also indicated by the comment of Mr Leyman.

A.A.Clark, UK: I am not familiar with what you mean by the height mode in longitudinal dynamic stability. Could you please clarify the point in physical terms.

G.Sachs, Germany: The height mode can be interpreted as an aperiodic motion in which mainly height and speed changes are involved. The changes, which are slowly varying, occur in such a way as to maintain constant lift, i.e., a decrease in height is combined with a reduction of speed and vice versa. The height mode being a result of the density gradient can be ignored in low speed flight. It becomes significant at high subsonic speeds and especially in the supersonic region.

C.S.Leyman, UK: Comment: On M.Renaudie's remarks, BAC experience is that atmospheric temperature disturbances vary in time scale from 2 to 20 seconds and therefore can affect phugoid roots. With regard to the manifestation of these disturbances to the pilot, it seems to show up as a rather long term P.I.O. in cruising flight.

G.C.Howell, UK: What assumptions on the values of $n_\rho^\star$ and $n_\rho^\star$ did you make relative to the engine intake and engine control systems? For a $M = 3$ aircraft, the intake control system is of necessity fairly complex and so could influence the equivalent values of the longitudinal forces and pitching moments due to speed and height variations. Were the values used in your paper realistic?

G.Sachs, Germany: With regard to the engine intake and engine control system, it is assumed that the time constants describing the behavior of the systems are small compared with the periods and/or time constants of the long term modes of motion. As can be seen in Figure 5, where the case $T_T = 5$ sec can also be interpreted as the time dependent behavior of the engine, the influence of the engine time constant is small, in spite of such a large value. As to the coefficients $n_\rho^\star$ and $n_V^\star$ describing the influence of speed and height on thrust, the values chosen for these parameters cover a wide range of thrust variations. For example, $n_V^\star = 5$ denotes a thrust change of 50% relative to a speed change of 10%. n is defined in an analogous manner. This range of $n_e^\star$ and $n_V^\star$ also includes the values generally used in investigations of supersonic flight. In a similar way, the speed and height dependent pitching moments are treated.

INFLUENCE DE LA MASSE ET DE LA REPARTITION DE LA MASSE SUR LES QUALITES DE VOL

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RESUME

Nous examinerons tout d'abord l'influence de la masse, c'est-à-dire de la taille d'un avion de transport sur sa stabilité dynamique latérale et longitudinale et sur sa manoeuvrabilité. Nous essaierons, autant que possible, de n'introduire aucune variation de conception aérodynamique de manière à bien séparer le problème des poids. Nous nous référerons aux deux avions Frégate et Transall de conception voisine qui nous sont bien connus.

La répartition des masses peut varier de nombreuses façons et avoir des répercussions diverses. L'influence la plus connue et la plus marquée est celle qui est apparue sur les avions rapides de faible allongement. Nous ne nous étendrons pas sur ce sujet fort connu. Pour des avions plus classiques, la répartition des masses peut également poser des problèmes. En dehors de l'influence de la variation d'inertie proprement dite, l'inclinaison des axes principaux d'inertie par rapport aux axes avion joue un rôle dans la manoeuvrabilité. Nous prendrons comme exemple le Corvette. Il est apparu sur cet avion un autre problème lié à la répartition des masses. C'est le cas du vol avec des bidons supplémentaires de carburant situés en extrémité de voilure. Ce problème a été traité en utilisant un simulateur.

1. INTRODUCTION

Les différents problèmes de qualités de vol d'un avion que peuvent poser les masses et leur répartition sont parfaitement connus et nous ne chercherons pas à apporter ici des révélations. Nous allons plutôt tenter de montrer sur des exemples les plus concrets possibles comment l'on doit tenir compte de l'influence des masses sur les qualités de vol. Nous essaierons donc de bien séparer les influences des caractéristiques aérodynamiques de celles des masses proprement dites ; et ce n'est pas toujours facile car les deux sont souvent intimement liées. Nous nous placerons, par ailleurs, surtout au stade du projet, ou même de l'avant-projet et nous essaierons de montrer les diverses méthodes d'études nécessaires, certaines approximations pouvant entraîner des erreurs graves. Il est certain qu'avec les performances demandées actuellement aux avions, quelle que soit leur catégorie, l'étude des qualités de vol doit débuter dès l'avant projet et être suivie tout au long du développement de l'avion. Bien des possibilités existent en ce domaine et il faut en tenir compte.

2. INFLUENCE DE LA MASSE

Nous allons tout d'abord regarder l'influence de la masse d'un avion sur sa stabilité dynamique et ses qualités de vol. Nous nous appuierons sur deux avions connus le Frégate et le Transall pour pouvoir extrapoler les conclusions aux avions qui sont prévus pour l'avenir.

Rappelons les caractéristiques des deux avions que nous connaissons (Cf Planche Nº 1) ; le Frégate et le Transall. Ce sont deux avions de transport, l'un militaire, l'autre civil. Ils ont des configurations très semblables : leurs ailes sont en position haute, ce sont des biturbopropulseurs. Nous voyons que leurs masses sont dans un rapport 5 environ, ce qui représente une différence d'échelle importante. Disons que ces deux avions ont des qualités de vol qui ont donné satisfaction à tout le monde. Il faut noter que le Frégate possède des commandes entièrement manuelles tandis que le Transall est lui, servo-commandé sur tous les axes. Nous voyons quelques différences de caractéristiques aérodynamiques. L'allongement de 10 pour le Transall n'est que de 8,7 pour le Frégate. De même la charge alaire est plus faible sur le Frégate. Les chiffres sont respectivement égaux à 200 kg/m^2 et 300 kg/m^2. Ceci représente approximativement 40 et 60 lb/sq.ft. Le volume d'empennage horizontal est également légèrement différent. Ajoutons que la motorisation plus forte du Transall pose des problèmes que nous n'évoquerons pas car ils sont spécifiques d'un tel avion.

Nous allons quand même comparer les stabilités dynamiques (planche Nº 2) de ces deux avions dans des configurations de croisière et d'approche, ceci à des centrages équivalents, à 20 % de la corde de référence. Nous ne nous intéressons qu'à l'oscillation d'incidence dans le comportement longitudinal et à l'oscillation de roulis hollandais dans le mode latéral. Les variations observées dans les autres modes dépendent beaucoup plus de facteurs secondaires. La phygoïde, par exemple, est fonction de la vitesse et de la variation de la traction avec la vitesse. La masse de l'avion n'intervient pratiquement pas. Nous avons également sélectionné deux cas de vol : tout d'abord l'approche qui représente toujours une phase de vol délicate et qui est effectuée sur les deux avions à des vitesses très voisines, l'hypersustentation plus poussée du Transall compensant presque sa charge alaire plus forte. Nous avons pris également une phase de croisière à 4.000 m d'altitude et une vitesse indiquée de 200 kt. Nous constatons que les pulsations propres du Frégate sont plus grandes notamment en latéral. Ceci ne nous surprend pas. L'avion qui a le moins d'inertie répond le plus vite. L'influence de la taille sur l'amortissement est beaucoup moins évidente puisque les valeurs sont assez voisines en latéral et notablement différentes en longitudinal, l'avion le plus léger étant le moins amorti. Ces différences constatées, il apparait difficile de tirer une conclusion générale, trop de paramètres variant simultanément, charge alaire, allongement, etc... C'est pourquoi, nous avons cherché à construire, à partir de ces deux avions connus, une famille logique d'appareils où la seule variable essentielle serait la masse. Nous admettons donc des avions ayant tous la même charge alaire et la même forme de voilure. Il s'agit d'abord de définir les lois de variation des moments d'inertie. Nous avons fait les raisonnements simples suivants qui nous ont conduits aux lois présentées sur la figure 3 et qui ont été très bien recoupées par les comparaisons Transall - Frégate. Si la charge alaire reste constante, l'envergure varie alors comme la racine carrée de la masse, et l'inertie en roulis comme m^2 si l'on admet que le rayon de giration est proportionnel à l'envergure. Ceci est certainement vrai dans le cas où beaucoup de carburant est dans la voilure, moins si le fuselage est très chargé et la voilure vide. Si on admet que la charge marchande est proportionnelle à la masse au décollage et que la densité de cette charge est constante, on trouve que la dimension linéaire, donc la longueur du fuselage varie comme la racine cubique de la masse. On obtient alors, en supposant un rayon de giration en tangage proportionnel à la longueur du fuselage, un moment d'inertie variant comme $m^{5/3}$. Nous supposons enfin que l'inertie en lacet est très voisine de la somme des inerties de tangage et de roulis. Nous avons comparé avec les formules statistiques données par la référence 1 et nous voyons que les différences ne sont pas suffisantes pour fausser la comparaison que nous allons entamer.

Nous voulons examiner l'influence de la masse d'un avion sur ses qualités de vol et ceci jusqu'à des masses jamais réalisées c'est-à-dire jusqu'à 1.000 t. Entre le Frégate et le Transall, il existe un rapport de masse de 4,7 environ. Ce coefficient appliqué une fois au Transall conduit à un avion de 220 tonnes et une deuxième fois à un avion de 1.000 t. Nous avons envisagé deux possibilités quant aux coefficients aérodynamiques. L'hypothèse la plus simple est de les garder constants et égaux à ceux du Transall.

On peut également adopter d'autres hypothèses, par exemple, en longitudinal, supposer que le volume d'empennage est conservé. Nous avons comparé ces deux hypothèses dans le cas de l'approche (planche 4). Dans la représentation classique du lieu des racines, nous avons tracé dans le demi-plan supérieur les valeurs obtenues en conservant des coefficients aérodynamiques constants et dans le demi-plan inférieur celles obtenues en gardant le volume d'empennage. Les différences sont faibles. Le tableau donnant les pulsations propres et les amortissements réduits montre également des différences faibles. Il est très intéressant de noter également que les valeurs du Frégate diffèrent relativement peu de l'avion ayant une masse identique mais une aérodynamique assez différente. La masse est donc un paramètre prépondérant pour des appareils de formule voisine. On voit que l'oscillation d'incidence se décompose assez rapidement en deux mouvements apériodiques. Ceci n'est pas gênant en soi puisque la norme américaine militaire MIL 8785B (référence 2) tolère des amortissements réduits de 1,3 pour les phases de vol A et C et même de 2 pour les phases B. On est encore loin d'atteindre ces valeurs. Ceci traduit néanmoins une tendance nette des gros avions à une réponse molle. Il faut également se méfier de ce qu'une des racines apériodiques ne devienne divergente. Bien que la variation envisagée pour les coefficients aérodynamiques ne semble pas faire ressortir de telles craintes, il faut noter qu'un calcul rapide par les expressions simplifiées habituelles tenant compte toutefois du mouvement vertical fait apparaître une divergence. Il est donc nécessaire de faire un calcul complet.

La figure suivante (planche 5) montre les mêmes caractéristiques dynamiques longitudinales dans la configuration de croisière. Les conclusions sont sensiblement identiques mais relativement moins sévères ce que l'on pouvait attendre. Ici le Frégate est vraiment très proche de son cousin bâti sur le modèle du Transall et la séparation de l'oscillation d'incidence en deux modes apériodiques ne se produit que pour des avions de masse beaucoup plus élevée. Nous n'avons pas tracé le cas des avions ayant un volume d'empennage constant, les différences avec le cas de coefficients aérodynamiques égaux à ceux du Transall devenant insignifiantes.

Nous avons calculé des caractéristiques des modes propres de divers avions et nous avons constaté que les chiffres obtenus restaient dans l'enveloppe permise par une norme militaire. Mais si l'on nous demande de réaliser un avion répondant à un certain programme, comment savoir si les caractéristiques prévues sont suffisantes. Nous venons de noter qu'un calcul simplifié de caractéristiques dynamiques n'était pas toujours suffisant. Ici nous voyons qu'un calcul même exact de caractéristiques dynamiques n'est pas suffisant. Il faut pouvoir calculer la réponse de l'avion à une commande de gouverne pour pouvoir juger de l'aptitude d'un avion à répondre à un programme. C'est ce que nous avons fait (planche 6). On peut juger de la détérioration extrêmement rapide des qualités de l'avion avec l'augmentation de masse, que ce soit en incidence ou en altitude ce qui est finalement la variable essentielle lorsque l'on est en approche. Il n'existe actuellement aucun critère pouvant caractériser l'agrément de pilotage des gros avions. Bisgood (référence 3) a examiné cette question très en détail et n'a pu conclure. Il semble ici que la manoeuvrabilité se détériore très vite dès que le mouvement devient apériodique.

Si l'on retient ce critère et que l'on raisonne sur le seul degré de liberté de tangage, on obtient (planche 7) une relation simple entre le coefficient de rappel C_{m_α} et le coefficient d'amortissement C_{m_q}. Si l'on suppose un volume d'empennage constant, on voit que C_{m_q} varie comme $m^{-1/6}$ puisque la corde de référence varie comme $m^{1/2}$ et le bras de levier comme $m^{1/3}$. Dans ces conditions, il faudrait que C_{m_α} donc la marge statique, augmente comme $m^{1/2}$. Nous avons supposé dans ce raisonnement que la charge alaire restait constante. Un calcul simple montre que, pour que la marge statique reste constante avec $\mathfrak{Z} = 1$, il suffit que la charge alaire augmente comme $m^{1/3}$. On peut dire que ce sont les progrès de l'aérodynamique qui ont permis des charges alaires élevées qui ont, à leur tour, facilité la manoeuvrabilité des avions géants.

Nous avons également examiné l'influence de la masse sur la stabilité latérale en approche. Nous avons considéré des coefficients aérodynamiques égaux à ceux du Transall et le cas d'une dérive dimensionnée par la VMC c'est-à-dire dont la surface varie comme $m^{7/6}$. Les différences sont imperceptibles. La planche 8 montre la variation des racines du roulis hollandais en fonction de la masse. On constate un amortissement croissant avec la masse comme en longitudinal mais on n'arrive pas à la décomposition du mouvement oscillatoire en deux mouvements apériodiques. Les différences aérodynamiques entre le Transall et le Frégate font apparaître une forte différence d'amortissement bien que les pulsations propres pour deux avions de 10 tonnes soient très voisines.

Ici encore les caractéristiques de manoeuvrabilité sont très affectées par l'augmentation de masse. A aérodynamique donnée, $\frac{pb}{V}$ est constant. Une augmentation de charge alaire est, ici encore, bénéfique et même doublement bénéfique. Elle fait diminuer l'envergure et augmente donc la vitesse de roulis. De plus elle nécessite l'installation de dispositifs hypersustentateurs plus efficaces et des spoïlers peuvent alors donner une excellente maniabilité. C'est ce qui est réalisé sur le Transall.

3. INFLUENCE DE LA REPARTITION DES MASSES

Nous allons aborder maintenant l'influence de la répartition des masses sur les qualités de vol d'un avion. C'est un sujet qui a déjà fait l'objet d'un très grand nombre de communication à des Congrès variés lors de l'étude du couplage roulis tangage des avions élancés. L'étude des qualités de vol spécifiques des avions élancés a fait l'objet également de travaux dont ceux de Pinsker (référence 4). Nous ne parlerons pas de ces problèmes sauf pour faire remarquer que c'est un magnifique exemple de couplage entre les problèmes massiques, inertie en roulis très différente de celles sur les autres axes, et les problèmes aérodynamiques liés aux ailes en forte flèche, effet dièdre important à grande incidence et amortissement en roulis très faible notamment.

Nous allons évoquer ici deux petits problèmes qui nous ont été posés par l'avion Corvette. Voici une photographie du prototype (planche Nº 9). C'est un biréacteur léger de transport conçu pour recevoir une dizaine de passagers. La masse au décollage est de 6,1 tonnes. La voilure de 22 m^2 a une flèche modérée de 20º au quart des cordes. Les deux réacteurs Pratt and Whitney Canada JT 15 D-4. sont situés latéralement à l'arrière du fuselage selon une formule classique.

Une telle formule d'avion appliquée à un appareil de faible tonnage implique une position de l'axe principal d'inertie à piquer par rapport à la référence fuselage. On sait (Cf réf. 5) qu'une telle configuration peut amener des conclusions variées selon les coefficients aérodynamiques de l'avion et notamment les valeurs respectives de l'effet dièdre $C_{l\beta}$ et de la stabilité de route $C_{n\beta}$. C'est pourquoi, dès le projet, nous avons vérifié la stabilité du mode roulis hollandais en fonction de l'inclinaison de l'axe principal d'inertie ε. La planche 10 montre cette influence pour l'approche et la croisière. Nous voyons que la valeur de -4º qui était celle calculée pour le Corvette donne une stabilité fort convenable et qu'il serait nuisible d'essayer de rendre l'axe principal d'inertie moins piqueur.

Ce problème d'inclinaison de l'axe principal d'inertie s'est posé au moment du projet. Au stade de définition de l'avion de série, il nous a fallu envisager l'influence du montage de ballonnets contenant du carburant supplémentaire en bout de voilure. Cette solution était mise en balance avec un réservoir de fuselage qui était moins bien centré et pesait plus lourd du fait des sécurités nécessaires. Il restait à voir si les qualités de vol n'étaient pas trop affectées par ces bidons ou si un remède simple pouvait être trouvé. Disons que les bidons qui ont chacun une capacité de 350 litres font doubler l'inertie en roulis quand ils sont pleins. L'inertie en lacet est également augmentée très notablement. Il n'y a évidemment pas que des changements d'inerties et de masses. L'effet dièdre notamment est augmenté.

Nous avons examiné la stabilité latérale du Corvette (Planche Nº 11) avec et sans bidons et nous avons envisagé un léger accroissement de la hauteur de la dérive. Nous voyons ici la stabilité en configuration approche avec les bidons pleins. C'est un cas relativement rare puisque le combustible des ballonnets est généralement le premier épuisé. Toutefois c'est un cas que l'on doit envisager. On s'aperçoit que l'amortissement est très détérioré. Par contre l'augmentation de la dimension de dérive procure une amélioration notable. Devant les chiffres que nous voyons, que doit-on conclure ? Pour notre part, nous avons décidé que cette simple étude était insuffisante et qu'il fallait dans ce cas utiliser un simulateur pour pouvoir mieux juger des qualités de vol comparées des diverses versions possibles.

Cette étude a été faite sur le simulateur du centre d'Essais en Vol à Istres. Cet appareil possède trois degrés de liberté : roulis, tangage et mouvement vertical. L'expérimentation sur simulateur n'était valable qu'à la condition de retrouver correctement le comportement de l'avion prototype après affichage de ses coefficients. C'est ce qui a été tout d'abord acquis. Nous avons pu ensuite afficher d'autres configurations avec un niveau de confiance suffisant. Et nous avons eu une réponse très nette de la part des pilotes et des enregistrements également quant à la question posée.

L'avion avec bidons nécessitait un agrandissement de la dérive. Quand on examine les enregistrements (planche Nº 12) de la version dérive normale sur une prise de cap, on voit certes que le travail principal a été accompli. Mais le dérapage, la vitesse de lacet sont très perturbées et le travail effectué aux ailerons par le pilote est très important. Par contre, avec la dérive agrandie (planche N° 13), bien que l'amélioration de l'amortissement soit faible, on s'aperçoit que le pilotage est beaucoup plus calme et il n'est pas étonnant que les pilotes aient trouvé une différence considérable entre les deux avions.

4. CONCLUSION

Nous venons de présenter ici un ensemble de plusieurs petites études qui semblent disparates. Elles le sont et c'est normal, car nous avons voulu montrer, tout en traitant l'influence des masses et de leur répartition, que chaque problème appelait un traitement spécifique. Tel problème peut être rapidement résolu par un simple calcul approché utilisant un seul degré de liberté. Tel autre, au contraire, nécessitera l'emploi d'un simulateur relativement complexe.

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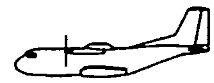
	FREGATE	TRANSALL	Pl. 1



	FREGATE	TRANSALL
Masse (kg)	10 800	49 000
Surface alaire / Wing area (m^2)	55	160
Allongement / Aspect ratio	8,7	10
Charge alaire / Wing loading (kg/m^2)	196	305
Volume d'empennage horizontal / Horizontal tail volume	0,9	0,97

Stabilité dynamique — Pl. 2

Dynamic stability

				Transall $m = 47\,000$ kg	Frégate $m = 10\,000$ kg
Croisière / Cruise	Altitude 4000 m, $V = 200$ kt	Longitudinal	ω_{nsp}	3,1	3,65
			ξ	0,7	0,52
		Latéral	ω_{nd}	1,15	2,05
			ξ_d	0,19	0,18
Approche / Approach	Altitude 0, $V \sim 115$ kt	Longitudinal	ω_{nsp}	1,75	1,8
			ξ	0,8	0,5
		Latéral	ω_{nd}	0,93	1,26
			ξ_d	0,22	0,29

Lois de similitude pour les inerties — Pl. 3

Similarity laws for inertia

m/S = Constante

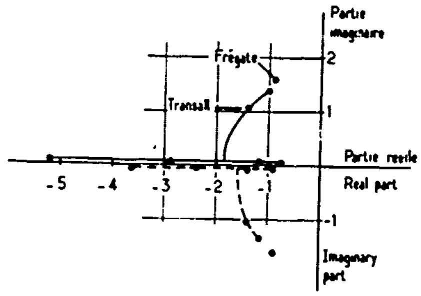
		Lois adoptées / Adopted laws	AGARD Report 472
Roulis / Roll	ρ_x / b = constante $\longrightarrow$	$I_x = O(m^2)$	$I_x = O(m^{5/3})$
Tangage / Pitch	$l_{fuselage} = O(m^{1/3}) = O(\rho_y)$ $\longrightarrow$	$I_y = O(m^{5/3})$	$I_y = O(m^{[illegible]})$
Lacet / Yaw	$\longrightarrow$	$I_z = 0{,}95(I_x + I_y)$	$I_z = A m^{5/3} + B m^{[illegible]}$

Stabilité longitudinale
Approche

Pl. 4

— Cas n°1 . Coefficients aérodynamiques Transall

-- Cas n°2 . Volume d'empennage / Horizontal tail volume $\bar{V} = 0{,}97$

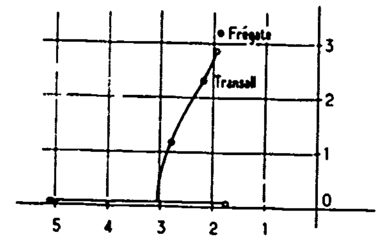


Masse avion / Airplane mass	Cas n°1		Cas n°2	
	ω_{nsp}	ζ	ω_{nsp}	ζ
10 t	1,67	0,6	1,73	0,66
47 t	1,75	0,8	1,75	0,8
220 t	1,86	1,1	1,82	1,02
1 000 t	2,4	1,25	2	1,16

Stabilité longitudinale
Croisière

Pl. 5

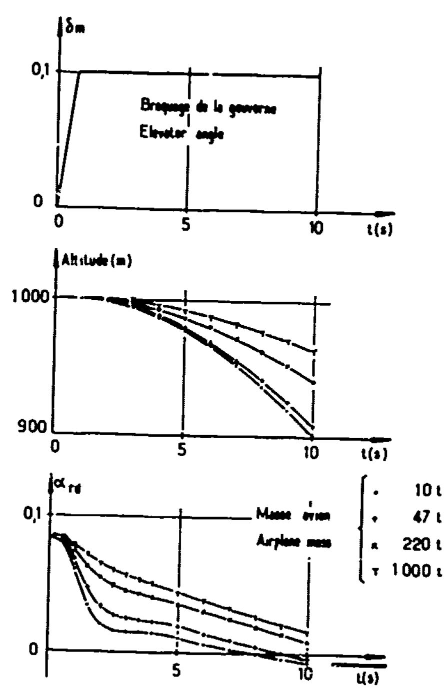
Coefficients aérodynamiques Transall



Masse avion / Airplane mass	ω_{nsp}	ζ
10 t	3,48	0,56
47 t	3,1	0,7
220 t	3,05	0,92
1 000 t	3,02	1,12

Manœuvre en approche
Approach manœuver

Pl. 6



Oscillation d'incidence — Short period

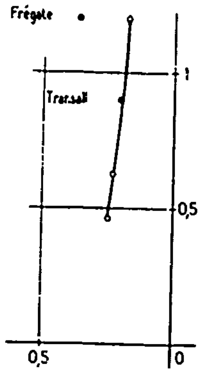
Pl. 7

$$\zeta < 1 \longrightarrow \left| C_{m_\alpha} \right| > \frac{\rho S l^3}{8 I_y} C_{m_q}^2$$

$$\bar{V} \text{ constant} \longrightarrow C_{m_q} = O(m^{-1/2}) \text{ et } \frac{\rho S l^3}{8 I_y} C_{m_q}^2 = O(m^{1/2})$$

Stabilité latérale
Approche

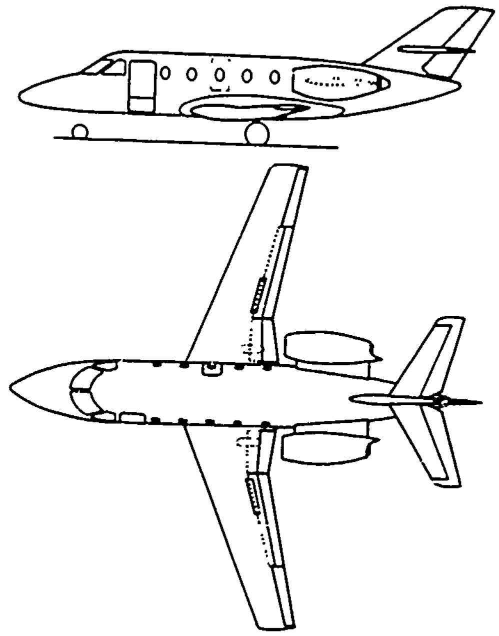
Pl. 8



Masse avion Airplane mass	ω_{n_d}	ξ_d
10 t	1,20	0,15
47 t	0,93	0,22
220 t	0,62	0,37
1000 t	0,46	0,54

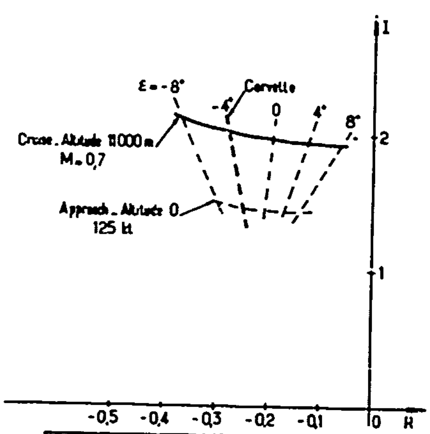
Corvette 601 A

Pl. 9



Influence du calage de l'axe principal d'inertie
Influence of the principal axis tilt angle

Pl. 10



ε°	Cruise		Approach	
	ω_{n_d}	ξ_d	ω_{n_d}	ξ_d
-8	2,18	0,17	1,52	0,20
-4	2,05	0,135	1,48	0,17
0	2,00	0,098	1,47	0,143
4	1,95	0,064	1,46	0,117
8	1,94	0,032	1,45	0,097

Stabilité latérale — Pl. 11

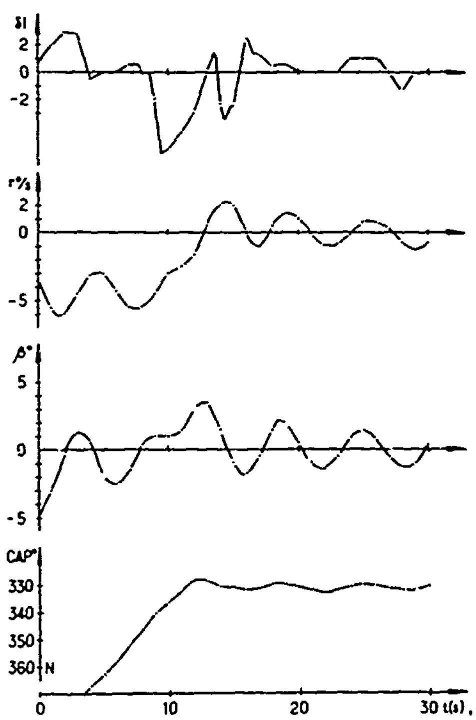
Influence des bidons _ Tip tanks influence

Approche

	Avion sans bidons / Without tip tanks		Avion avec bidons / With tip tanks	
	ω_{nd}	ξ_d	ω_{nd}	ξ_d
Dérive initiale / Normal fin	1,48	0,17	1,05	0,06
Dérive agrandie / Greater fin	1,61	0,21	1,15	0,09

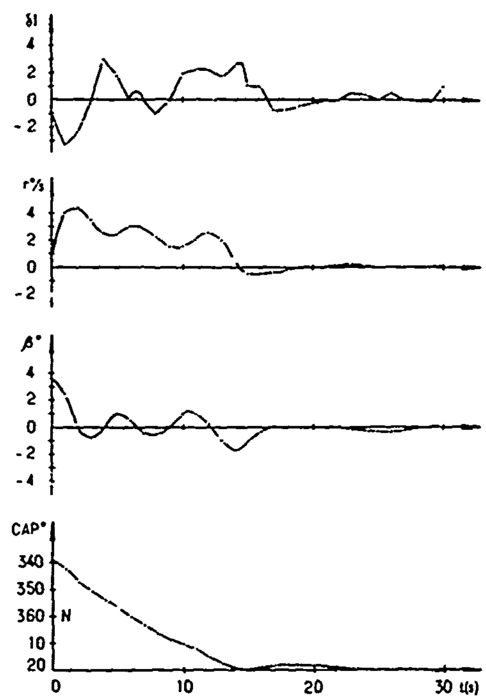
Avion avec bidons _ Dérive initiale — Pl. 12

Aircraft with tip tanks _ Normal fin



Avion avec bidons _ Dérive agrandie — Pl. 13

Aircraft with tip tanks _ Greater fin



OPEN DISCUSSION

J.F.Renaudie, France: This is not a question, only a comment. It is shown in the paper of Mr Mesniere that the wing loading is greater for heavier aircraft; this inevitably leads to great differences in behaviour. This variation of the wing loading with the mass was discovered by Helmholtz a long time ago when there was no aircraft flying; only birds were flying at that time. Helmholtz established by simple dimensional analysis that the wing loading is proportional to the cube root of the mass. He checked this law by applying it successfully to about 60 birds. I myself tried to apply this law to present day aircraft. It is always true; only the constant coefficient is different reflecting a state of technology different for aircraft as compared to birds. One cannot prevent the wing loading increase with increase in the take-off weight.

You can see the Helmholtz law as applied to the birds in one of the last books of Th. v. Kármán (Selected Themes of Applied Aerodynamics).

X.Hafer, Germany: You showed the influence of an increase of the mass only for the short period motion. As we found in similar investigations there could be a severe influence for the phugoid mode which could lead to instability effects. Did you investigate this case also?

M.Mesniere, France: We have not investigated the influence of the mass on the phugoid motion. This mode is primarily affected by the speed of the aircraft and we did not think that the influence of the mass could be significant.

A.G.Barnes, UK: Although a weight increase at constant wing loading will change the short period frequency ω_{nsp} and damping ζ_{sp} there will be no corresponding change in the lift parameter $L_\alpha = g/V\ n_{z\alpha}$. A critical handling qualities parameter is the ratio of ω_{nsp} to L_α; this ratio will change with weight changes. The new Mil. Spec. includes requirements based on ω_{sp} and $n_{z\alpha}$. Therefore, a study of weight variation should include the L_α effect on handling.

M.Mesniere, France: We have not considered changes in the lift parameter. The Mil. Spec. effectively requires a variation of natural frequency with the lift parameter. All our values of natural frequency are well in the good range required.

THE ROLE OF THEORY AND CALCULATIONS IN THE REFINEMENT OF FLYING QUALITIES

by

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SUMMARY

The present state of the art in handling qualities research and design is broadly surveyed with particular emphasis on the role of theory and paper studies in this field. First the significance and scope of handling criteria is critically discussed as setting targets for flying qualities design. The capabilities and limitations of theory are then considered in such areas as derivative prediction, rigid-body stability and response calculations, predictions of stability under partial constraint and under active pilot control. Finally some general consideration is given to novel flying qualities problems associated with the introduction of stability and control augmentation systems.

1 INTRODUCTION

When we discuss flying qualities we are considering normally those properties of the aircraft that are relevant to the pilot in the context of control. When the aircraft is under fully automatic control it clearly still has flying qualities but they confront the pilot as a passive observer and have fundamentally different significance. We may therefore exclude automatic control from our discussion and address ourselves specifically to those aircraft characteristics which the pilot meets during the exercise of manual control. The term handling qualities is perhaps more appropriate here.

The need for good flying qualities is so self-evident that it needs no justification. However, like all good things in life, good handling has to be paid for and there is certainly good reason to consider the point at which the quest for perfection in this area becomes economically questionable. I feel that this argument is too frequently ignored by the proponents of such ultimate refinements as manoeuvre demand or adaptive control. One begins to wonder whether these techniques are not aiming at a degree of perfection that shows diminishing returns. Do we for instance, really require flying characteristics which are invariant with flight condition and thereby underutilize the pilot? And what is more important, do they really contribute to safety? These are legitimate questions which must be considered seriously in a balanced approach to the refinement of flying qualities and we shall find occasion to return to them in this paper.

In order to be able to control and refine handling qualities the aircraft designer requires three things:

i A proper design aim, ie a quantitative definition of the flying characteristics to be achieved. This is the purpose of handling qualities research and the framing of certification requirements.

ii He must have the means of accurately predicting the flying characteristics of a given design so as to be able to identify deficiencies at the earliest possible moment.

iii He must have at his disposal an efficient armoury of palliatives and the means of predicting their effects and side-effects. Today we think in this context immediately of automatic feedback control.

In all these areas, theoretical exploration and routine calculations play an important and sometimes dominant part. Not only is this true in the design stage but equally when flight trials show handling deficiencies. It requires sound theory to associate complaints that are often only expressed qualitatively with quantifiable aircraft properties.

This paper will attempt, albeit in broad outline, to indicate the capabilities and limitations of current theory in these areas. The aim is not to present a detailed survey or indeed a complete catalogue. Instead specific points of interest will be highlighted and controversial concepts critically discussed.

The subject of handling is unique amongst the engineering disciplines in that it brings the engineer face-to-face with the human sciences, in particular with physiology and psychology. By training and inclination he is clearly more at home in the more sober world of physical science and will be normally somewhat sceptical of the pseudo-scientific pretentions so frequently found in psychological argument. It is surprising therefore to note the often uncritical enthusiasm with which he will indulge in such activities as pilot opinion gathering which is very plainly a technique borrowed from the psychologist. We should be very much on our guard when entering this field, in particular as at best we can only claim amateur status in this area. I shall return to this topic again when discussing handling criteria, a most important aspect of all work on flying qualities.

Design for flying qualities exploits, as any other engineering activity, both theoretical and experimental techniques and one cannot fruitfully discuss one in isolation from the other. The ideal function of theory is of course, the accurate and reliable prediction of the performance of an engineering design. If and when such theory is available experimental checks and verification are superfluous. We are certainly not discussing a field here in which such claims can be made for theory. Not only is accuracy alone often disappointing but also and perhaps more seriously we must still expect flight trials to reveal handling problems of a kind not previously met or at least not anticipated. It is of course, the continuing aim of research in the field of flying qualities to improve the power of prediction methods in accuracy as well as in scope.

A good example of what is possible today in this area is Concorde. In spite of its unorthodox shape and the many novel handling problems associated with its configuration, this aircraft has not presented in flight any important handling features that were not anticipated and predicted during the design stage. In fact practically all the surprises, that emerged in flight, were of a positive and reassuring nature.

Design by theory is only as reliable as the assumptions fed into it. This is as true for work on flying qualities as in any other field of engineering. The assumptions made in stability and control analysis eg are generally of two kinds. In the first place there are assumptions in the mathematical formulations say of the equations of motion. Linearization is one such assumption, that occasionally has been found impermissible. Secondly, the aerodynamic and inertial properties of a design are often only available as estimates, especially early in the design process, and they are therefore no more than assumptions in the stability analysis. It is always prudent to check a solution for sensitivity to such assumptions by allowing variations of the more powerful parameters in the calcualtions.

It is also important clearly to distinguish between rigorous theory on the one hand and simplified theoretical calcualtion methods on the other hand. Because the latter may have proved adequate in practical work over a period of time, they frequently assume the status of 'classical theory' but fail if applied to new situations or configurations where the implied assumptions are not justified. Many disappointments have resulted from the failure on the part of the analyst to be alive to the existence and the nature of such assumptions. We shall consider therefore, conditions where phenomena not normally allowed for in classical theory may be important.

2 THE IDENTIFICATION OF HANDLING CRITERIA AND OPTIMA

One need not go far back in the history of aircraft design to recall the time when handling characteristics were not treated as a primary design aim. They merely happened to fall out from a design procedure orientated predominantly towards performance. Only if in this process handling deficiencies of a really prohibitive nature emerged could the airframe designer be persuaded to make major concessions or indeed to abandon a particular configuration altogether. Normally the stability and control specialist could expect accommodation only in such relatively inexpensive features as adjustments in wing dihedral and the size of the tail surfaces. In this climate it made little sense for him to search for sophisticated handling optima. There was little practical prospect of achieving these.

The introduction of powered flying controls and the consequent possibility of employing automatic feedback control for stability augmentation have altered all this. Now we are in a position to design specific flying qualities into the aircraft without necessarily constraining the airframe designer in his preoccupation with performance. Having thus liberated the possibilities of design for good handling, it became essential to establish proper design criteria.

Research into handling criteria follows two main streams. One is predominantly empirical and the other is more analytical. The empirical approach exploits the capabilities of the ground based simulator and the variable stability aircraft. In these facilities chosen aircraft characteristics are varied systematcally and note is taken of the conditions at which the pilot judges the aircraft handling characteristics to pass certain landmarks of acceptability, say from satisfactory to unsatisfactory. In this manner design criteria for this parameter are established. There are, however, practical difficulties in this procedure. The linearized equations of the aircraft are defined by something of the order of 17 aerodyanmic derivatives and 4 inertia parameters. In addition we have 8 control derivatives to consider and to the list may be added factors defining the mechanical properties of the control systems. The aircraft is only uniquely defined by a complete set of these factors and the prospect of exploring the entire field defined by all possible values and combinations of these parameters is clearly discouraging. There are of course, many configurations in which some of the aerodynamic derivatives are negligible in their effect and can be ignored, but this may not be true in every case and it has often been found rash to make such assumptions too readily. But even then the size of the remaining describing matrix will still be prohibitive.

In order that systematic handling research be directed along some profitable lines it was necessary to postulate some particular aircraft parameter as having prominant handling implications and to establish by experiment pilot ratings against this parameter or possibly against combinations of such parameters. Initially the choice of such parameters was more or less intuitive, being restricted to those factors which are normally emphasized in classical stability analysis. These are in particular the frequency and damping of the short period rigid-body modes and the time constants for the aperiodic modes, such as the roll subsidence and the spiral mode. Although the early work along such lines was successful in establishing a general foundation to the rationalization of handling, sooner or later inconsistencies appeared and it became evident that factors other than those specifically controlled in these experiments were also involved and in need of identification. This again left the doors wide open and one was faced with the prospect of systematically working ones way through the whole range of all possible combinations and permutations. There was clearly the need for a more rational framework and only theory would provide that.

Before discussing the function of theory in this field I would, however, just briefly like to dwell on another aspect of this work, the technique namely of pilot opinion gathering. The idea of condensing pilots comments on flying qualities into a single scalar quantity was pioneered by G. Cooper and R. E Harpur independently until both combined to produce the now generally accepted standard pilot opinion rating scale of Ref. 1. Such scales are today used almost univerally in all systematic handling research and assessment and it is important therefore fully to appreciate the nature and the possible limitations in this method. As the term properly signifies pilot opinion is essentially and inevitably a subjective judgement.

It must be expected to be conditioned by experience. The pilot may be expected for instance, to be more happy with a characteristic he has met before and is likely to rate poorly the novel, whatever its intrinsic value.

Pilots' standards will also be conditioned by expectation. What was adequate yesterday is unacceptable today. What is satisfactory today may well be criticized tomorrow. Expectations rise with the progress of technology and this aspect must be reflected in a degree of impermanence of handling criteria.

The most difficult phenomenon to cope with rationally, however, is scatter of pilot ratings. Some degree of inconsistency between ratings given by one pilot on repeat-trials or by different pilots testing the same configuration is of course unavoidable but in some instances, especially when relatively poor configurations are tested, ratings varying from three to eight or even nine are frequently attained. What is one to make of such a result? Clearly the worst rating obtained deserves serious consideration. Is there any possible justification for taking arithmetic means of pilot opinion ratings? If in one out of four trials a pilot lost control, is it any consolation that on the remaining three occasions no difficulties were experienced? Pilot opinion rating numbers are no more than convenient labels summarizing lengthy verbal comment, they can in no sense be taken as numbers capable of arithmetic manipulation. I would suggest that mathematical theorizing here is somewhat suspect.

The systemisation of handling research and assessment has been greatly assisted by the idea and the standardisation of pilot rating scales. Let us be careful therefore, not to discredit a most valuable tool by too careless use.

Let us return now to the role of theory in the field of handling criteria research. The fundamental idea in this work is the proposition that it is possible to visualize the human pilot as a mechanistic controller with response characteristics that can be modelled mathematically. There are in fact, two concepts which have been useful in this field.

The simplest idea is the assumption that the pilot is capable of suppressing completely a particular freedom of the aircraft response. Stability analysis of the remaining aircraft motion will then frequently reveal the existence of a response mode not shown by conventional theory. If and where applicable such an approach leads to attractively simple stability criteria. The identification of the speed stability mode by Neumark[2] is perhaps the best known example, the practical relevance of which is generally beyond dispute. We shall consider other applications of this concept in the detailed discussion of partially constrained flight later in this paper.

The concept of pilots' control leading to the virtual suppression of a freedom of aircraft response is of course only viable if the control activity implied is comfortably within the pilots' capabilities. The most important condition to be satisfied is that the mode suppressed is sufficiently slow so that dynamic limitations in pilot response are not strained.

The method is therefore not suitable for studying pilots' control of high frequency aircraft modes. such as the longitudinal and lateral short-period oscillations. For the meaningful analysis of such situations the pilot must be represented as a dynamic agent, by a model that represents faithfully all the physiological and psychological factors defining his capabilities as a controller. Such an approach has been pioneered and developed by Ashkenas and McRuer[3] and their followers and has given rise to a literature too numerous to quote here. By representing the human pilot by an equivalent transfer function it is possible then to treat the complete system of aircraft plus pilot as a closed loop servo system and investigate its stability and response characteristics by the mathematical techniques of automatic control theory. This approach has provided a sound foundation for the understanding of many important piloting problems that would otherwise not be capable of rationalisation. We shall return to this subject in a more detailed discussion of flight under active pilots' control.

However, there are wide areas of pilot control where the implied concept of the pilot as a deterministic continuous controller does not apparently apply, where perhaps higher functions of his intellect are brought into play, where sophisticated judgements rather than automatic reactions predominate. The assessment of handling criteria for such situations still awaits a theoretical formulation. In particular one thinks here of what might be called discrete manoeuvres, such as the execution of the landing flare or take-off rotation, or kicking off drift in crosswind landings, reactions to sudden failures, such as power failures and control runaways. Very often in these cases the pilot applies well memorized and judged patterns of control application. In other words, he appears to operate open loop control with discrete checks and corrections at certain intervals. This is very evident in records of landings where the final phase to touchdown is often seen to consist of a sequence of controlled steps.

The lack of a coherent analytical approach to such problems is perhaps best appreciated in a particularly pressing current interest, namely that into steep approaches, especially but not only in the context of STOL. The question concerning everybody involved in this field is simply, can or can we not expect pilots to perform landings from steep approaches with the necessary precision, repeatability and safety? We can of course, calculate the amount of controllability that we have to provide in the aircraft to make the manoeuvre physically possible but this does not really answer the question. We have no way of solving this problem on paper, we do not understand in any quantitative sense the factors that influence the pilot here; how well for instance can he judge height and vertical velocity; will a flight director help; what is the influence of the aircraft response characteristics; will the more immediate response provided by direct lift control make a major difference?

Although we have come a long way in flying qualities research and prediction there are still large and important areas where theory has so far been unable to make an impact.

3 PREDICTION OF AERODYNAMIC AND INERTIAL AIRCRAFT CHARACTERISTICS

To predict the dynamic behaviour of an aircraft relevant to its flying characteristics, we must in the first place have accurate knowledge of its aerodynamic and inertial properties. The estimation of aircraft inertia requires essentially no more than accurate book-keeping and arithmetic in the weights department and is therefore mainly a matter of organisation and less of science. Since the importance of aircraft inertias especially to lateral stability and in fast rolling manoeuvres has been properly appreciated, the quality of the estimates provided today appears to be satisfactory, judging by the isolated occasions where it was possible to check such estimates by tests on the actual airframe[4,5]. It appears that inertias supplied by aircraft manufacturers now lie within a few per cent of the true value, sufficiently accurate for most practical purposes.

Unfortunately no such success can be claimed in the field of derivative estimation . This subject generated great interest and activity in the fifties, judging by the amount of literature devoted to this topic then. This activity has now virtually ceased and the reason is certainly not that there is little more to be done. I would suggest that there are in fact two quite different reasons. In the first place there are now available a range of wind tunnel facilities allowing the measurement of unsteady aerodynamic data, rolling balances and oscillatory rigs as for instance that developed at the RAE by J S Thompson[6]. On the other hand the accuracy and reliability of theoretical procedures for estimating aerodynamic derivatives has proved disappointing, especially for configurations where interference between the various components of the complete airframe is significant.

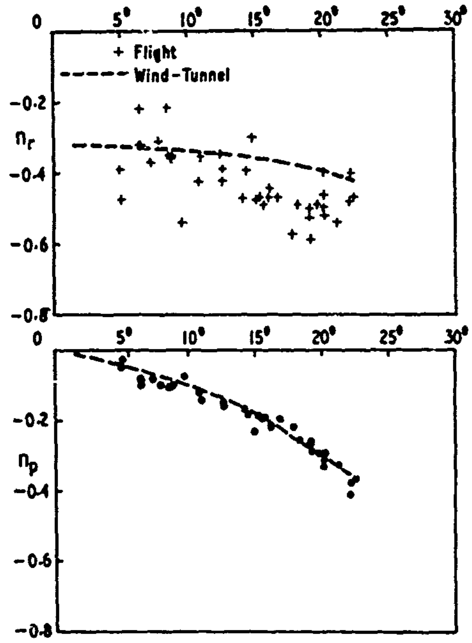

FIG. I YAWING MOMENT DERIVATIVES OF HP115 FROM ANALYSIS OF DUTCH ROLLS

An example of the precision with which carefully conducted wind tunnel tests today allow derivatives to be measured is illustrated in Fig. 1. These results are reproduced from Ref. 7 and compare two unusually elusive derivatives namely n_r and n_p as obtained with the apparatus described in Ref. 6 and presented in Ref. 8, with actual flight measurements. I am only showing this particular result as an example, comparisons of the other derivatives have been made and are equally impressive. It may be noted that the case quoted here is perhaps somewhat exceptional as the inertias of this aircraft were actually measured on a ground rig, which as previously mentioned agreed well with estimates, and this allowed the last few per cent of accuracy to be extracted from flight. Also in the oscillatory wind tunnel tests a range of values of reduced frequency was realised so that it was possible to extrapolate to full scale flight values, a procedure which proved important in the case of n_p, the yawing moment due to rate of roll.

It is difficult to generalize about the potential accuracy of theoretical derivative estimates, as this will obviously vary from configuration to configuration and from method to method. However, a useful survey of this field was recently made by Fletcher[9] of NASA with results that can be taken as typical, judging by our own experience. Some examples from his Report are shown in Figs. 2 and 3. In each case, estimates for a given derivative calculated by five different current methods are compared with wind tunnel results and where available with flight data for two aircraft configurations. It is not the intention here to discuss in any detail the merits of the various theoretical procedures used but merely to use this interesting comparison to illustrate the current state of art in this field.

It is possible to summarise by saying that the power of theory for predicting aerodynamic derivatives is only just adequate to permit very broad assessments of handling features early in the design process and that these must be replaced at the earliest possible moment by wind tunnel results, if predictions are to be at all realistic.

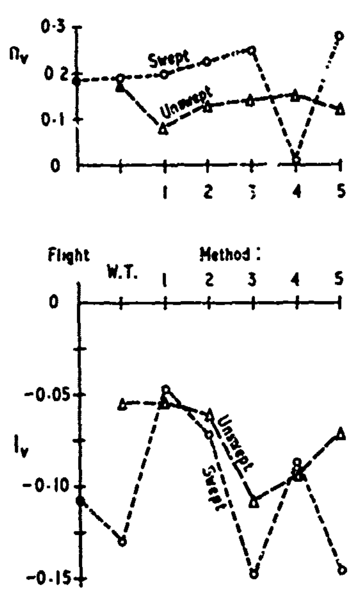

FIG. 2 COMPARISON WITH EXPERIMENT OF LATERAL DERIVATIVES ESTIMATED BY VARIOUS THEORETICAL METHODS

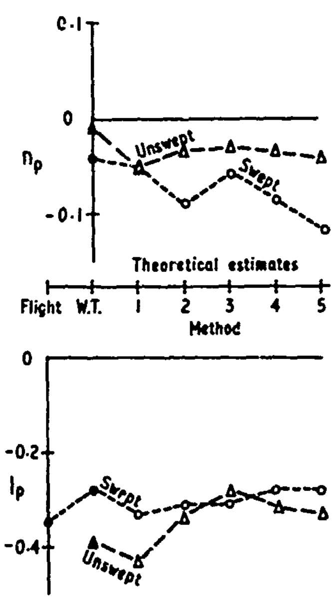

FIG. 3 COMPARISON WITH EXPERIMENT OF ROLLING MOMENT DERIVATIVES ESTIMATED BY VARIOUS METHODS

4 PREDICTIONS OF RIGID-BODY STABILITY AND RESPONSE

Although the assessment of the flying characteristics of an aircraft frequently requires much more sophisticated considerations, the basis of all work in this field is still the determination of the fundamental stability characteristics of the uncontrolled configuration. A good deal of paper study is therefore devoted to the prediction and if necessary to the improvement of these basic aircraft properties.

Much of this work follows a classical pattern of linearized analysis treating the longitudinal and lateral motion separately. As we have stressed at some length in the introduction, the validity of this approach is based on a string of assumptions, which it is alway prudent to keep in mind, especially if the design under consideration shows unusual geometric or inertial features. Factors which predispose an aircraft to departures from linear behaviour and the possibility of lateral-longitudinal couplings are for instance extremes in inertia distribution[10] and engines generating large rotary momentum[11].

Theory plays perhaps its most important role in the field of handling qualities during the early design phase in predicting the aircraft dynamic behaviour and in identifying major difficulties. In extreme cases such analysis may give such an unfavourable forecast that a project may have to be abandoned altogether or drastically reconfigured. Such a case is illustrated in Fig. 4 where the estimated dutch roll damping characteristics are shown as they were calculated in the project stage of a fighter design proposal considered in the early post-war era. The features of the design of particular interest in the present context are the high tail location and especially the installation of the engine high up on the rear fuselage. This resulted in a steeply inclined principal inertia axis which in turn was responsible for unacceptably low dutch roll damping. As you will note, at higher altitude this mode was expected to become severely unstable. The design was rejected. It must be noted that this story relates to a period when stability augmentation was in its infancy and could not be trusted to turn this project into an efficient service aircraft. Today one might consider such a solution more seriously if the offending design features offer otherwise attainable performance advantages. In either case the early recognition of such major stability shortcomings is as important today as it was then, if only to make appropriate provisions for the required performance and integrity of stability augmentation aids.

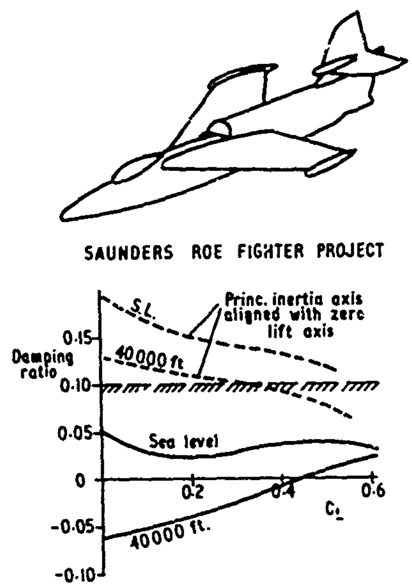

FIG. 4 PREDICTION OF DUTCH ROLL CHARACTERISTICS OF AN AIRCRAFT PROJECT REVEALING TOTALLY UNACCEPTABLE BEHAVIOUR

An interesting version of a similar problem presented itself during the design of the HP 115 slender wing research aircraft. Simplicity and cheapness were dominant requirements in this design, which had the primary purpose of demonstrating the viability of the slender wing concept in low speed flight. In fact this design evolved from an originally considered unpowered glider version and the engine was installed with ease of construction foremost in our mind. Hence its unusual location as evident from Fig. 5, again leading to a very adverse principal inertia axis inclination. Although six component model tests were made, little was known about the rotary derivatives for this unconventional shape which had to be either guessed or extrapolated from available tests on simple wing-only models. The resulting estimates for the dutch roll characteristics as they were available just prior to first flight are shown by the fine dotted lines of Fig. 5. Since there was some uncertainty about the precise position of the principal inertia axis, the calculations were made for a range of values of this parameter. You will note that even with the most favourable assumption, the result was disconcerting and as a consequence first flight was held up. Fortunately at about this time the first results became available of free-flight tests of a dynamically similar model of the aircraft which were commissioned because of the general doubts existing at the time about the flying characteristics of this type of aircraft. These flight results gave a reassuring picture and suggested that in our estimates we had made some significant errors in the assumed aerodynamic derivatives. We choose to modify n_p below $C_L = 0.4$ in the manner indicated in Fig. 6. This matched the free-flight model results and produced an estimate for the aircraft, which, in conjunction with now refined estimates for its inertia distribution, gave sufficient confidence to allow us to authorise flying. We note from Fig. 5, that this estimate closely follows the actual aircraft behaviour since measured in careful flight tests. However, it is interesting to record, that this apparently excellent agreement was to a certain extent accidental. In Fig. 6 the original and the modified estimate for n_p are compared with the correct values established much later both in

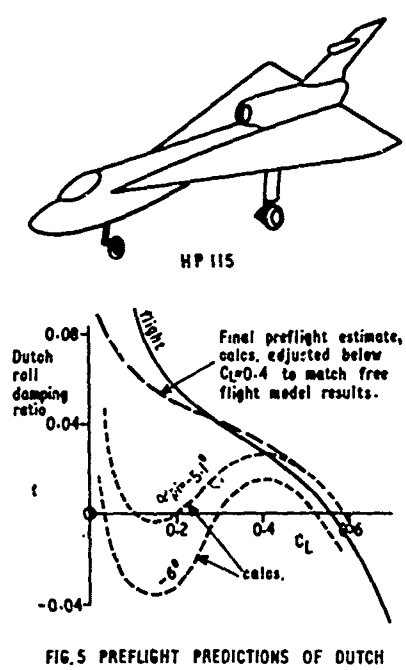

FIG. 5 PREFLIGHT PREDICTIONS OF DUTCH ROLL STABILITY OF THE H.P. 115

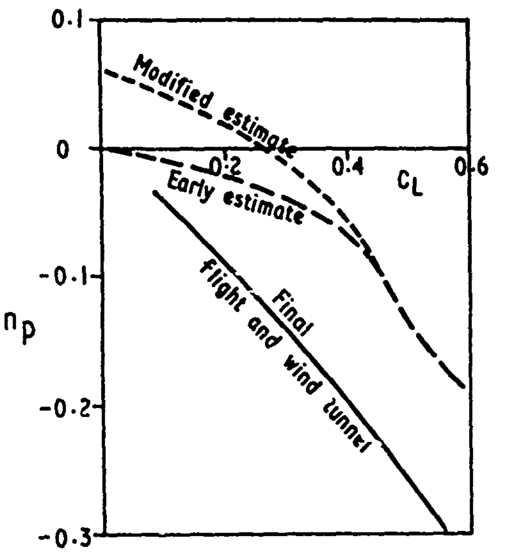

FIG. 6 COMPARISON OF EARLY ESTIMATES AND LATER FLIGHT AND WIND TUNNEL DATA FOR THE DERIVATIVE n_p.

flight and in model tests on the oscillatory rig described in Ref. 8. These, if anything, are closer to the early estimate than to the modified estimate which gave such apparently good agreement of the actual dutch roll behaviour. The explanation to this complex story is that early estimates of derivatives other than n_p and equally the principal inertia axis were also seriously in error and later when all derivatives became accurately defined in appropriate tunnel tests. in combination they produced virtually perfect agreement.

We may conclude from this discussion that theoretical predictions are clearly of the greatest importance in ensuring at least safety for initial flying, but that such calculations can be seriously in error if not based on sound aerodynamic and inertial assumptions.

5 COUPLED LONGITUDINAL-LATERAL MOTION AND NON-LINEAR BEHAVIOUR

The most commonly encountered departure from linear behaviour occurs with the modern aircraft in fast rolling manoeuvres, especially in inertially slender designs. Phillips[12] had identified as early as 1948 the phenomenon later to be known as inertia-crosscoupling, but some time passed before the practical significance of this fundamental work was fully appreciated. Today inertia-crosscoupling is considered as a matter of routine in every military aircraft design and flight acceptance. Although general criteria[7] and generalized design data[13] are available to allow the designer to get a broad appreciation of the general sensitivity of a configuration to this phenomenon, detailed design predictions and preparations for flight test usually require elaborate theoretical calculations, using analogue or digital computers. Inertia-crosscoupling is perhaps the most challenging subject for the analyst because he is dealing here essentially with a resonance phenomenon, which is much more sensitive to small changes in the aerodynamic data than is the case in classical linearized stability analysis. This becomes vitally important in the flight test stage, when calculations are used in each step of the progressive exploration of the manoeuvre-flight envelope of the aircraft. It has been found essential in this process, frequently to update the assumptions by matching against the latest flight records available. In many cases nonlinearities in aerodynamic forces and moments must be carefully represented, because manoeuvres involving inertial coupling result in large excursions, for instance in sideslip, ie, past the small perturbation regime.

When all these precautions are taken we find theory today to be capable of producing very accurate predictions in this field with matches that are often virtually faithful overlays of flight records.

However, there is some doubt whether inertia-crosscoupling is strictly a handling problem or simply a structural stressing case. It arises from the flying characteristics of the aircraft, but it is generally accepted that it is a phenomenon virtually outside the control of the pilot and therefore possibly not a proper subject for this paper.

There are other factors capable of inducing nonlinear aircraft behaviour or lateral-longitudinal coupling with important consequences to flying qualities. We have already briefly mentioned the role of the rotary mass of engines. In certain circumstances as elaborated in Ref. 11 this can couple especially the dutch roll with the longitudinal short period oscillation and lead to destabilization of one or the other of these modes. Engine coupling is potentially most significant at low speeds, where aerodynamic forces are relatively weak by comparison with inertia reactions, ie, when the parameter

$$K = \frac{(h/\omega)^2}{BC} \geqslant 0.01$$

is large. h is the angular momentum of the rotating engine machinery. B and C are the aircraft inertias in pitch and yaw respectively and ω is the frequency of an uncoupled aircraft oscillatory mode.

Fan engines are likely to be more significant in this context than conventional jets, because they generate about four times as much angular momentum for a given amount of thrust. Although in current designs this effect was found in Ref. 11 to be not serious, even though noticeable, it may well become important with STOL aircraft, as these are likely to combine high installed power with extreme low speeds. In VTOL designs the gyroscopic engine effect dominates in the hover and this forces the designer to minimise engine rotor momentum by either installing engines in handed pairs or constructing engines with counter-rotating parts as with the Pegasus engine in the Harrier.

Aerodynamic nonlinearities can also be significant by themselves. Again we may refer to the much-laboured HP 115 for an interesting example. In the low speed regime linearized analysis and flight observation predict instability of the dutch roll (Fig. 5). However. if the pilot allows the motion to develop it is soon seen to settle into a limit cycle oscillation with a stable amplitude which increases when incidence is increased. Since this behaviour is entirely innocuous and can be stopped instantly by either conventional use of the ailerons or by pushing the stick forward, this particular manoeuvre has been performed at many flying displays and may therefore be familiar to some in this audience.

The phenomenon was investigated in the so called flight dynamics simulator of the RAE[14] in which wind-tunnel-measured aerodynamic coefficients are fed on-line into an analogue computer representing the aircraft kinematics and dynamics (including also the missing rotary derivatives) and the computed aircraft response is used to drive the wind-tunnel model. This procedure reproduced the limit cycle oscillation phenomenon and gave the values for the steady oscillation amplitude in sideslip indicated by X in Fig. 7. No quantitative flight data are available to check these results in detail, but qualitatively they appear to be of the right order although the onset of instability in flight occurs at much higher incidence. The nonlinear behaviour giving rise to the limit cycle was suspected to be largely the result of nonlinearity in n_v, ie, in the trend of yawing moment with sideslip as illustrated by an example in the insert in Fig. 7.

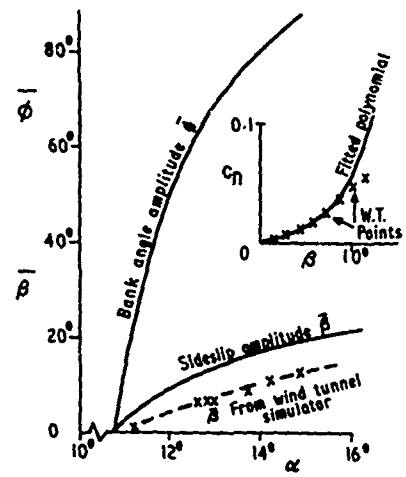

FIG. 7 RESULTS OF CALCULATION OF LIMIT CYCLE AMPLITUDE $\bar{\beta}$ AND $\bar{\phi}$ OF THE HP115 DUTCH ROLL AT HIGH INCIDENCE (FROM REF. 16)

Beecham had developed in Ref. 15 an approximate method for solving dynamic problems involving such nonlinearities and this method was then applied in Ref. 16 to this case to see if it is possible to obtain a purely analytical answer of adequate accuracy. The results of these theoretical calculations are shown in Fig. 7 and compared in the case of sideslip amplitude $\bar{\beta}$ with the semi empirical results of the procedure described before. The yawing moment chacteristic was approximated by a fitted third order polynomical giving the kind of fit shown in the insert. Even though the increased 'stiffness" in n_v at larger values of sideslip was exaggerated by this approximation, it is seen that this simplified theoretical approach predicted larger limit cycle amplitudes than the more sophisticated 'calculations' performed in the wind tunnel-simulator experiment. From the calculated bank angle amplitudes it can be said with assurance that they are substantially larger than anything observed in flight. These large bank angles give in fact. a clue to the partial failure of the approximate theory, where all kinematic relationships were linearized. Clearly for such a large perturbation motion a more realistic treatment of the kinematics is required. Nevertheless, the relatively simple theory of Ref. 16 has reproduced the essential features of the observed phenomenon and can certainly be recommended, provided the validity of all the assumptions is carefully observed.

The most spectacular, and practically important nonlinear, large perturbation flying characteristic is the spin. It has attracted theoreticians for a long time without, however, any really convincing results. The real difficulty in this area is the provision of aerodynamic wind tunnel data without which such work is doomed to failure. Such tunnel tests are perhaps more difficult to perform and to analyse than spinning tunnel tests which produce directly the desired overall answer.

However, an interesting exercise in this field has recently been reported in Ref. 17 where digital computer calculations were made to investigate the effects of mass variations (ie, the effects of carrying external stores) on the basic spin characteristics of the Mirage fighter aircraft. Since the spin and recovery characteristics of the assumed datum configuration appeared to match the actual aircraft behaviour in all essential features, it was argued that such calculations are likely to give a sound forecast of the likely effects of incremental changes to the basic configuration. It would appear that a further condition for this basically attractive argument to be acceptable is that the stores do not have significant aerodynamic effects. These were not represented in these calculations.

The evidence considered here suggests again that existing theory is well capable of dealing with nonlinear and high coupled aircraft motion but that in such work considerable care must be taken to ensure the validity of the assumptions made and that sufficiently precise aerodynamic data are fed into the calculations.

6 DYNAMIC AIRCRAFT BEHAVIOUR INVOLVING AN ACTIVE PILOT

The ultimate concern of the aircraft designer is not just the behaviour of the aircraft when left to itself but the situation that may develop when the pilot begins to exercise active control. Many instances are known where a basically docile aircraft begins to develop vices in this situation, and conversely, theoretically existing instability may disappear in flight when the pilot gets hold of the stick. The most interesting phenomena are of course those in which apparently rational control by a skilled pilot generates an instability not inherent in the basic airframe. Such conditions must be identified in advance and only theory can provide the tools for such predictions.

There are in fact two classes of such phenomena which we shall discuss separately. Analytically the simplest is the important class when the pilot supresses more or less perfectly a particular freedom of the aircraft motion only to find the remaining aircraft motion becoming unstable.

The other group involves conditions where the dynamic response of a pilot attempting to control a particular rigid-body mode, usually a short period osciallation, destabilizes this mode like a badly designed and overgeared feedback servo.

6.1 FLIGHT UNDER PARTIAL CONSTRAINT

Normally we consider under this heading situations where the pilot controls the aircraft so that one motion freedom is for all practical purposes constrained, but we shall widen the discussion and consider also a case where an automatic control system (namely the autothrottle) generates such constraint.

The theoretical work is then simply concerned with the stability of the aircraft in the remaining freedoms taking careful account of the aerodynamic effects produced by the control used to force the constraint.

The concept of flight under constraint was first introduced by Neumark[2] and the first practical problem to which this approach was applied was flight under glidepath constraint by pilot's elevator control. This work identified the now well known speed stability mode and minimum drag speed (or more precisely, the minimum power-required speed) as the critical point below which this mode goes unstable.

The assumption of effective constraint by the pilot, however, begs one important question, namely, is this constraint physically realizable without inducing in its trail destabilization of the mode assumed to be contained. In general it has been found that the assumption of simple constraint is generally viable if the mode suppressed is slow in relation to the pilot's reaction time. This can be broadly taken to be no more than 0.5 seconds. Since the mode suppressed in glidepath constraint, namely the phugoid, normally satisfies this condition it is not surprising to find that the theory of speed stability works well and leads to results of great practical significance.

Very close to the ground, when the pilot becomes preoccupied with the flare, he is often seen to increase his control gain to such an extent that the longitudinal short period mode becomes involved and as shown in Ref. 18, this can lead to instability of a kind occasionally seen in flight records. Strictly speaking this kind of problem, ie, the pilot-induced oscillation, belongs to the field to be discussed in the next section; however, it appeared opportune to mention it here to indicate the limits beyond which the idea of simple constraint can only be taken at some peril.

Another condition to which the concept of partial constraint gave a convincing explanation was observed on the BAC 221, the high speed companion to the HP 115 slender-wing research aircraft operated at RAE Bedford. During exploration of the limits of low speed flyability, the aircraft experienced directional divergence or a mild form of 'nose slicing'. This happened in a flight condition where classical stability analysis predicted no difficulties. The observed instability could, however, be readily reconstructed by a theory[19] assuming the pilot to maintain wings level by aileron control. The analysis leads again to an extremely simple stability criterion:

$$n_v - \ell_v \frac{n_\xi}{\ell_\xi} > 0$$

An interesting observation is that this criterion applies irrespective of the system of axes in which the derivatives are expressed, provided of course, a consistent set is used.

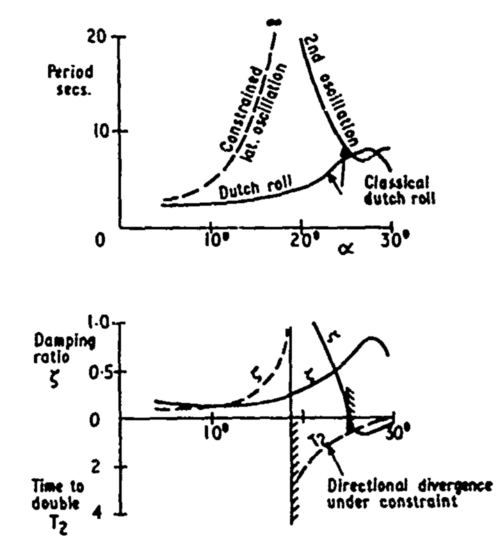

FIG. 8 CLASSICAL DUTCH ROLL AND BANK CONSTRAINED MODE FOR BAC 221 AIRCRAFT

The results of this analysis are compared with the classical dutch roll solution in Fig. 8. The significant point is that for this aircraft bank angle constraint leads to instability of about 19° incidence, whereas without pilot's control the lateral motion would be stable up to 24° incidence, where in fact the so-called second oscillation becomes undamped. Flight difficulties appeared in fact just above 19° incidence. The interesting fact revealed by the analysis is that this apparent loss in directional stability can be simply cured by reducing adverse aileron yaw, say by a suitable interconnect with the rudder. Since at the stability boundary pilot's constraint leads to an infinitely slow divergence mode, the concept of constraint is of course perfectly plausible and there is no difficulty in accepting the basic assumption in this theoretical approach.

To introduce the next topic I would like to draw your attention back to the problem of speed stability. The answer to the problem of flight on the back side of the drag curve is the autothrottle. In order to get the full benefit from such a device, the modern autothrottle is usually designed not only to correct the speed-instability of the aircraft but also to attain effective speed-lock capability. Seemingly an ideal piloting aid, allowing him to forget speed, almost obviating the need for control altogether. This is in fact when trouble was met. In flight with such a perfect autothrottle aircraft have been observed to suffer substantial and even dangerous deviations from the proper glide slope.

We have here of course a perfect case of rigorous constraint, that of air speed. When the remaining longitudinal motion is analysed[20] the aircraft is shown to lose its normal glidepath stability and may in fact become divergent in this mode. Again theoretical analysis gives a simple criterion, defining the stability root of the glidepath mode as

$$\lambda = \frac{g}{V}\cos\gamma\left(\tan\gamma + \alpha_E - \frac{z_E}{x_a}\right)$$

The terms used in this expression are defined in Fig. 9. The most powerful term in this expression is the vertical distance of the engine thrust axis from the centre of gravity. Low slung engines are destabilizing, so that designs with engines carried below low wings are particularly affected by this condition.

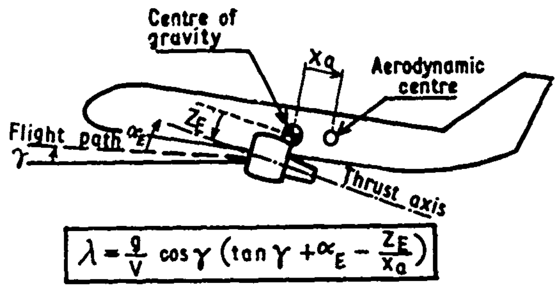

FIG. 9 FACTORS INFLUENCING GLIDE PATH STABILITY IN FLIGHT UNDER SPEED CONSTRAINT

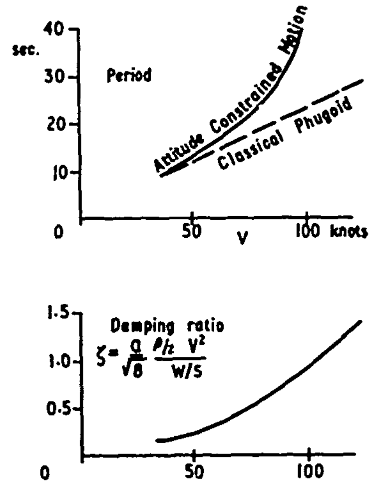

FIG. 10 LONGITUDINAL MOTION IN THE STOL REGIME WITH PITCH ATTITUDE CONSTRAINT BY ELEVATOR CONTROL

In a general theoretical study of STOL flying problems[21] the case was considered when the pilot uses elevator to constrain pitch attitude. This leaves a form of phugoid as the remaining mode. As shown in Fig. 10 this mode is highly damped and has a very long period in the normal speed range. This is of course the reason why control of pitch is normally such a successful strategy. In the STOL range, however, pitch constraint leaves the aircraft with a much less well damped residual motion of relatively short period, which may cause some pilot dissatisfaction. This work so far is pure speculation as no flight observations are available.

It is hoped that this discussion has demonstrated that the theoretical concept of partial constraint of flight proved itself as a most useful assumption and led to the analysis and prediction of a wide range of important handling problems.

6.2 PILOT INDUCED OSCILLATIONS AND ALLIED PHENOMENA

There is of course a wide range of handling problems where the concept of pilot's control leading simply to the effective suppression of a motion freedom is inapplicable. This is particularly true if control of one of the faster rigid modes is considered. Here the dynamic response characteristics of the pilot must be properly accounted for as it is always possible that his control activity may lead to dynamic excitation rather than suppression of the control parameter.

Whereas theoretical analysis of partially constrained flight generally leads to drastic mathematical simplifications, the introduction of pilot dynamics adds complexity. Fortunately the treatment of automatic control systems has led to the development of wide variety of efficient mathematical methods which can be applied to the study of human control if we succeed in modelling the pilot's control response in the form of an appropriate transfer function.

Ashkenas and McRuer have pioneered this concept which has found wide application and given rise to a literature too vast to review here in any depth. The basis of this approach is the assumption that in many practical flying tasks the pilot acts effectively as a continuous feedback controller and that his behaviour in such situations is largely deterministic and can be represented by a pilot describing function[22] of the type:

$$Y_P(s) = \frac{Ke^{-\tau s}}{(T_N s + 1)}\,\frac{(T_L s + 1)}{(T_I s + 1)} .$$

In this transfer function two terms define the 'mechanical' properties of the human controller, namely the response time delay τ and a neuromuscular lag T_N. The remaining terms define his higher functions as an adaptive agent, namely the gain K and lead and lag equalisation T_L and T_I. Within certain limits the human controller has been found capable of generating and selecting these functions so as to optimise overall performance, ie, tightest possible control with maximum stability.

Some investigators have considered even more complex pilot models by for instance, introducing an indifference threshold and higher order representations of the neuromuscular lag. These refinements do not materially alter the results and in most cases a simplified version of the above describing function has been found adequate. The pilot is assumed to have the innate ability to adjust his response to a given control situation until the resulting performance is as close as possible to the desired optimum. If this requires him to adopt strong equalisation, in particular lead, he will register this as an added workload and give the configuration a poor rating. If even with best equalization, he is unable to prevent instability when controlling with adequate gain, the aircraft will be rated as unacceptable. We have then the phenomenon known as a pilot induced oscillation (PIO).

Generally the term PIO is, however, applies to a specific group of handling situations in which apparently acceleration feedback plays a crucial role. This must be so because the closed loop instability often does not materialise in these cases when simulated on a ground based simulator not having cockpit movement represented. This then leads to difficulties in the application of human control theory because the pilot is now clearly responding to at least two stimuli, visually perceived aircraft attitude and the physical sensation of body acceleration. To conduct mathematical analysis we must define how and in what proportion the pilot perceives and mixes these two signals. There is no intuitive concept available to allow the addition of two so fundamentally different stimuli to be reduced to some self-evident principle. This is why human operator theory had difficulty making quantitatively accurate predictions in this field.

When the basic concepts of human feedback control apply without such reservations, theory has given excellent results. The assumptions made in its formulation must be properly understood to avoid disappointment. The main assumptions are:

i The pilot's representation as a mechanistic feedback controller is true only in a statistical sense, he is not 'wired' into the loop and should not be expected to behave with deterministic consistency. At most the human transfer function is an 'average' transfer function, describing that portion of his control output that over a period of time can be seen to correlate with the input stimulus. It does not allow us to predict what he will do precisely at any particular instant. The remainder of his overall control activity, usually defined as the remnant, is effectively random noise that he injects into the aircraft. This remnant is particularly large when a demanding task is performed and may then amount to as much as 90% of his total output. Also he is of course perfectly capable of ignoring the input altogether for a while, using control intermittently or of changing control strategy in a fundamental manner. For instance, pilots are often seen to allow a divergent oscillation to develop up to an amplitude they consider safe and then kill the energy accumulated by the mode with a single well-aimed control pulse.

ii The feedback concept requires that there is a single parameter identifiable as the control stimulus and also that there is a unique control response to this 'input'. A good deal of general flying does not fall into this category, which essentially refers to tracking only. General longitudinal control is a typical example, where the pilot acts with a long term result in view, and considers the total flight situation. He exercises energy management rather than feedback control.

These restrictions imply that there is still a wide range of flying qualities problems to which the present human controller theory does not apply and for which an appropriate theoretical approach is still wanting.

When and where the assumptions of pilot's feedback control apply, however, we have an excellent theory which can deal effectively with handling problems in tracking tasks, tight control of flight in turbulence and similar situations.

7 STABILITY AND CONTROL AUGMENTATION

One cannot today, meaningfully discuss aircraft flying characteristics without reference to automatic augmentation systems. There is no need, however, to consider this field in detail here as M. Deque will later in this symposium give us an excellent exposition. I will confine myself therefore to some novel flying qualities problems which may result from the adoption of stability and control augmentation.

The body of existing handling criteria has been developed round the properties of the natural aircraft. These are governed by fundamental physical principles defining the nature and magnitude of the aerodynamic and gravitational forces acting on the aircraft. Stability augmentation systems can, however, generate forces and moments of a fundamentally different kind and this may result in an aircraft having unusual response characteristics, no longer defined by the conventional rigid-body modes of the natural aircraft. These may not be covered by existing handling criteria and may necessitate the search for new and appropriate design requirements. The example quoted earlier of the appearance of an unstable glidepath mode for aircraft under automatic throttle control is an example which may serve as a warning.

The most disturbing innovation is the self-adaptive stabilization system, which may present us with an aircraft having no longer in any meaningful sense deterministic characteristics. Its present response behaviour is always conditioned by the immediate past history of the flight. How can one rationalize the assessment of the flying qualities of such an elusive device?

However, even more mundane autostabilizers can present problems not found in the natural aircraft. They always have limited authority and at the point of saturation will cease to enhance aircraft stability. This problem is perhaps particularly important when non-transientized feeback signals are used, as for instance that of pitch rate into elevator. Such systems can be saturated for quite prolonged periods, eg. in steep turns, and expose the pilot suddenly to the perhaps rather poor characteristics of the basic airframe. In the transition from the stabilized normal regime to the unstabilized flight condition, handling difficulties may arise which may be treated theoretically by the methods outlined in section 6.2. These will be the greater, the bigger the difference between the stabilized and the natural aircraft behaviour, in other words the larger the gains used by the augmentation system. If such gains are reduced the flying characteristics of the aircraft in normal operation will be less attractive, but the point at which the stabilizer saturates will be moved out, making it less probable for this condition to be met. Also the pilot will have less difficulty adjusting to the smaller change in aircraft stability if and when he exceeds the authority limit.

Such a situation poses a difficult design dilemma, one that forces us to reconsider the real purpose of stability and control augmentation. One has to make a choice between excellent normal handling but a safety risk on the one hand or a less attractive aircraft in normal operations but less risk of control difficulty in extreme manoeuvres on the other hand. It creates a problem in need of careful theoretical and practical consideration.

8 CONCLUSIONS

The many aspects which the modern aircraft presents in the field of flying qualities have been broadly surveyed with particular attention to the role which theory can play in their solution and prediction. Theory has been shown to be well able to predict with great accuracy the stability and response characteristics of aircraft, but this requires reliable knowledge of the aerodynamic characteristics of the airframe. The power of theory for estimating aerodynamic derivatives is still very limited and wind-tunnel experiments are as vital today as they ever were.

There are many many situations in which classical stability analysis with the applied assumption of linearization and independence of the lateral and longitudinal motion is inappropriate. Criteria are available to indicate when more sophisticated treatment is required and sound theoretical methods are available to deal with a wide range of such conditions.

Great strides have been made in the last decade or so in the analysis of flying qualities problems in which pilot's control is an essential agent. Theory has been successful here in two particular areas. One considers situations in which pilot's (or system) interaction virtually suppresses one particular motion freedom of the aircraft leading to the emergence of otherwise unsuspected instabilities. Theoretical work has succeeded in identifying several handling problems of this nature, all of real practical significance.

Pilot's control of relatively fast aircraft modes, creates another potential type of handling difficulty, directly involving the dynamic response characteristics of the human. By modelling the pilot's behaviour in the form of a transfer function, standard methods of servo control analysis can be utilized to study the stability of the assembly of aircraft plus pilot as a closed loop system. Theory has been able to analyse many previously obscure handling problems and has been instrumental in rationalizing flying qualities requirements in this field. There are, however, large areas of flight control when the pilot adopts control strategies not compatible with the concept of simple feedback control. Their solution by theory is still awaiting the formulation of appropriate concepts.

Finally some handling implications of advanced stability augmentation systems are considered. It is suggested that they present some unusual characteristics having no parallel in the natural aircraft and are not covered by existing flying qualities criteria. Caution is advised in their use lest their more obscure characteristics cause safety hazards in extreme conditons.

ACKNOWLEDGEMENT

The author wishes to express his gratitude to Mr Bisgood who provided much of the data presented in this paper and has generally helped by stimulating suggestions.

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OPEN DISCUSSION

H.H.B.M.Thomas, UK: Mr Pinsker has drawn attention to the important question of derivative estimation. My feeling is that wind tunnel tests come late in the design process and estimates are essential for early project work. I wonder if our friends from industry would care to comment.

As to reasons for the present position I would say that as far as the UK is concerned there are two:

(1) Definite fall-off in activity.

(2) Lack of systematic wind tunnel tests, which are essential to the development of methods of estimation.

Is this an area to which the FMP should direct attention?

J.Czinczenheim, France: The n_p derivative estimation has given, in the past, the sort of trouble that you have mentioned. However, the effect of the reduced frequency on conventional stability is less known. Can you give some indication about the values involved?

R.Fail, UK: Answer: Data on the effect of the frequency parameter on n_p is given in Reference 8 of Pinsker's paper.

X.Hafer, Germany: I am not so pessimistic that improvements may not be possible in the future. To my feeling, one of the main reasons for relatively low accuracy of estimated derivatives depends on the interference effects of the different parts of the aircraft which can be calculated with better accuracy by new methods of theoretical aerodynamics, i.e., the finite elements method.

W.J.C.Pinsker, UK: I agree with Professor Hafer that improvements are certainly possible if more attention is given to this almost totally neglected field. Let us not, however, underestimate the magnitude of the task. In many areas it may be necessary to have a mathematical model of the whole aircraft for a meaningful theoretical solution. Worse still, we are not always dealing with attached flow. Vortices shed from wings, fuselage, and intakes are becoming increasingly evident in modern aircraft, where they frequently affect the flow at the tail surfaces. It will be a long time before we can expect theory alone to predict all significant contributions, but a start must be made and this must be supported by systematic wind tunnel tests to obtain empirical data on those features less amenable to theoretical analysis.

ADJUSTMENT OF FLYING QUALITIES BY WIND TUNNEL TESTING

by

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SUMMARY

In the development of the Fokker F28 Fellowship the wind tunnel has been used extensively to predict the flying qualities of the aircraft.
To obtain information which could be valuable in the exploratory phase of flight testing, component detail variations were included in the testing of lifting and control surfaces. Two examples of this wind tunnel approach to flying qualities are described in this paper, one dealing with the development of the elevator surface, the other with the development of stall characteristics.

NOTATION

AC	-	aerodynamic centre, fraction of $\bar{c}$
c	-	wing chord
$\bar{c}$	-	wing mean aerodynamic chord
c_e	-	elevator chord, aft of hinge
$\bar{c}_e$	-	elevator mean aerodynamic chord, aft of hinge
CG	-	centre of gravity, fraction of $\bar{c}$
C_h	-	elevator hinge moment coefficient
C_{h_α}	-	$\partial C_h/\partial \alpha_s$
C_{h_δ}	-	$\partial C_h/\partial \delta_e$
C_L	-	lift coefficient
$C_{L_{max}}$	-	maximum C_L
C_{L_α}	-	tailplane lift curve slope
$C_{L_{\delta s}}$	-	elevator lift curve slope
C_m	-	pitching moment coefficient
C_{m_o}	-	C_m of aircraft less tail at $C_L=0$
F_e	-	elevator control force
M	-	Mach number
M_{MO}	-	maximum operating M
M_D	-	dive Mach number
p	-	atmospheric pressure
Re	-	Reynolds number
S_e	-	elevator surface, aft of hinge
$\bar{V}$	-	tail volume coefficient
V_s	-	stalling speed
α	-	angle of attack of aircraft
α_s	-	angle of attack at tailplane
δ_e	-	elevator deflection
Δ	-	increment
γ	-	ratio of specific heats

Subscripts

M	-	at relative Mach number
M_o	-	M at trimmed condition

INTRODUCTION

The F28 Fellowship aircraft has been developed by Fokker for use over short to medium distances. The standard configuration can accomodate up to 65 passengers, while a stretched version will provide an additional 15 seats. The lines of the standard prototype F28 are shown in figure 1.

Short haul operation is characterized by frequent flight cycles comprising take-off, climb, cruise, descent and landing. In the design of the F28 for this type of operation with a two man crew, the emphasis was laid on easy handling and consequently good stability and control characteristics throughout the operational flight envelope.

Fig. 1 F28 first prototype aircraft

Two main objectives evolved from these aspects of short haul jet operation for the design of the wing, i.e. (1) achievement of relatively high values of $C_{L_{max}}$ and (2) good inherent transonic characteristics, the maximum operating Mach number being M_{MO} = 0.75 with a corresponding dive Mach number of M_D = 0.83.
The F28 wing has a 16° sweep angle at the quarter chord line; its wing sections are modified NACA four digit series sections with rather large nose radii primarily to improve section maximum lift. The maximum lift capabilities are further increased by a Fowler type flap, which is single slotted at settings up to 18° and double slotted at the larger settings, when the flap vane becomes effective after the flap has further expanded to form the second slot.

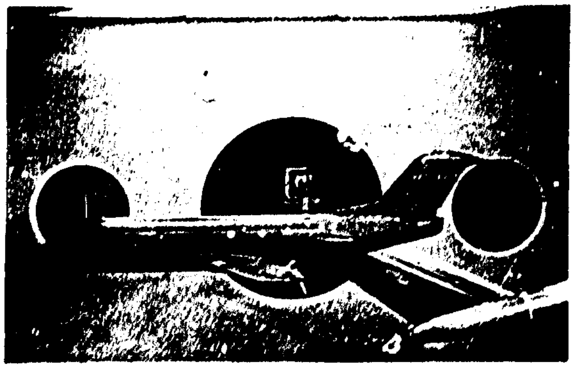


Fig. 2 Final F28 wind tunnel model

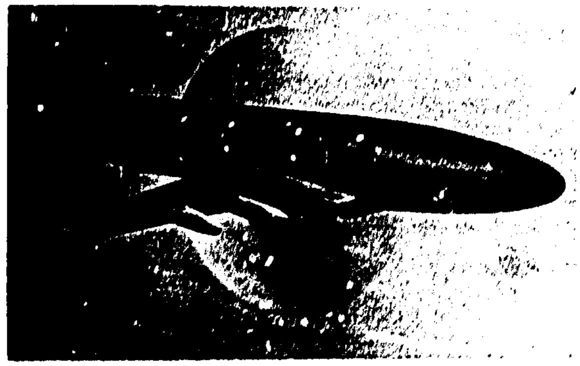


Fig. 3 Halfwing wind tunnel model

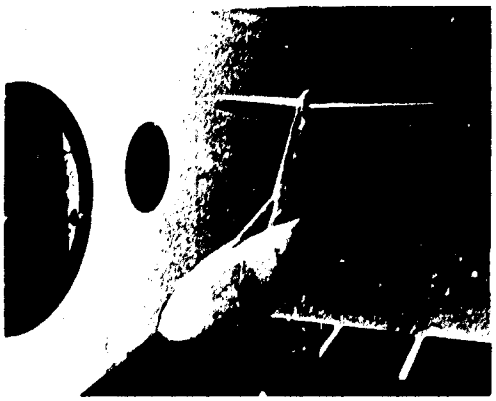


Fig. 4 T-tail wind tunnel model

In the course of the design process elaborate use of the NLR wind tunnels in Amsterdam has been made. The final model of the aircraft is shown in figure 2. Figure 3 shows a larger scale half wing model used to study, among other things, the stall at 5 x 10^6 Reynolds number.
A T-tail model was used to determine rudder and elevator characteristics. This model is shown in figure 4; the data obtained from this model was of particular importance in preparing for the full scale flight testing of the manually controlled elevator.

Two examples of wind tunnel test programmes used to refine the basic design are presented in this paper. The first deals with the development of the elevator surface to achieve satisfactory flying qualities in the pitching plane, the second with the development of stall characteristics.

DEVELOPMENT OF ELEVATOR SURFACE

F28 longitudinal control is obtained by means of an adjustable tailplane for trim, combined with an elevator for manoeuvring. The tailplane, with 27.5 degrees sweep at the quarter chord line and 11 percent thick airfoil sections, was designed to ensure that transonic compressibility effects would be postponed to beyond the design dive Mach number of M_D = 0.83.
The elevator hinges around the 78 percent chord line of the tailplane and is aerodynamically balanced by an overhang balance nose.
Considerations regarding the design philosophy for the control systems to be applied to this aircraft, led to the following approach to the systems in the longitudinal control channels.
Irreversible power operation was selected for the tailplane by the use of a duplicated hydraulic control unit with an electrical back up in the third mode.
For the elevator an essentially manual control was selected, the control forces thus being proportional to the elevator hinge moments, however reduced to the proper level by a duplicated reversible hydraulic booster with a low boost ratio to ensure conditions allowing landing of the aircraft with acceptable control forces in case of a double hydraulic failure. By this arrangement a more complicated system such as triplicated irreversible hydraulic control with the associated artificial feel system, would be avoided. The aim was also to avoid the complication of a Mach trim compensation system by designing for inherent transonic static longitudinal stability to beyond M_D. This will be examined further.

The flying qualities between M_{MO} and M_D were analyzed on the basis of conservatively interpreted results of wind tunnel tests. Some fundamental equations are given on the next page.

In straight flight a change in Mach number from a trimmed condition results in a change of pitching moment ΔC_m, consisting of a wing-body and a tail contribution, which has to be corrected in flight by an elevator deflection and then is recognized by the pilot as stability.
The analysis indicated that a slight stick position instability could appear on the aircraft at speeds halfway between M_{MO} and M_D.
This was mainly caused, see equation (1), by an increase of $\Delta\alpha_s$ due to a decrease of the wing lift curve slope, whereas other aerodynamic coefficients hardly changed.
The corresponding stick force is defined by equation (3).
Stick force stability implies a push force, i.e. a negative F_e, with increasing speed.
The product $C_{h_\alpha}.\Delta\alpha_s$ contributes thus in stick free stability when it is negative. For subsonic conditions a speed increase results in a decrease of α_s which leads to the well known stabilizing effect of a positive C_{h_α}.

F O R M U L A E

A speed change from M_o (trimmed condition) to M at n=1 and in level flight, with fixed elevator and constant tailplane setting, results in a change of pitching moment coefficient ΔC_m.

$$\Delta C_m = \underbrace{\boxed{\Delta C_{m_o} + C_{L_M}(CG-AC_M) - C_{L_{M_o}}(CG-AC_{M_o})}}_{\text{wing-body}} - \underbrace{\boxed{C_{L_{\alpha_s}}.\bar{V}.\Delta\alpha_s}}_{\text{tail}} \quad (1)$$

The corresponding elevator deflection for zero pitching moment is

$$\Delta\delta_e = \frac{\Delta C_m}{C_{L_\delta}.\bar{V}} \quad (2)$$

$$F_e = \frac{\text{gearing}}{\text{boost ratio}}.S_e.\bar{c}_e.\frac{\gamma}{2}.p.M^2\,(C_{h_\alpha}.\Delta\alpha_s + C_{h_\delta}.\Delta\delta_e) \quad (3)$$

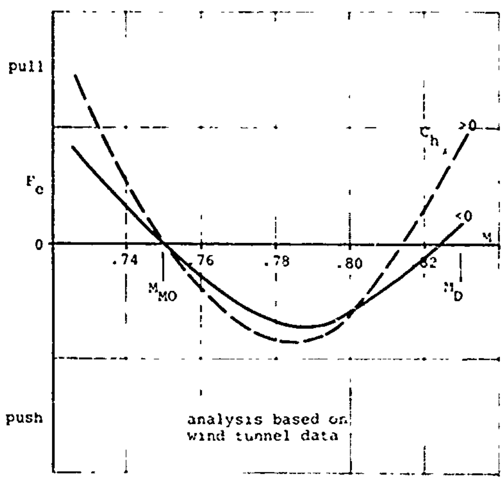


Fig. 5 Effect of C_{h_α}-sign on transonic phenomena

However, when transonic phenomena on the wing cause an increase in α_s upon an increment in Mach number, then a positive C_{h_α} results in a reduction in stick free stability.
The above aspects of a positive C_{h_α} are qualitatively illustrated in figure 5, also indicating that a negative value of C_{h_α} would have a favourable influence on flying qualities between M_{MO} and M_D.

Wind tunnel tests performed to investigate the effect of various balance shapes on the elevator hinge moments revealed, however, that in all cases C_{h_α} was positive at higher Mach numbers, even for an unbalanced elevator.
In figure 6 the plan view of a tail plane half is shown, together with a cross section of the final elevator configuration. This picture also shows the solution to make C_{h_α} negative. For that purpose semi-cylindrical strips, generally called "beads", were attached to the trailing edge of the elevator by which the pressure distribution over the aft part of the surface is boosted up proportional with angle of attack or elevator deflection. This leads to more negative values of C_{h_α} and C_{h_δ}, depending on bead span. This is depicted in figure 7, the radius of the bead being 3 mm (0.12 inch) full scale.
A favourable by-product of the application of a bead is that excellent linearity of the hinge moment with angle of attack or elevator deflection exists up to rather large values of these variables.
As shown in figure 7 the aerodynamic stiffness of the elevator, C_{h_δ}, is also considerably increased for the chosen bead span of 40% elevator span. However, power boost of the elevator was considered necessary on the F28 to ensure full elevator control capability in assumed extreme angle of incidence conditions, so the rather high value of C_{h_δ} was

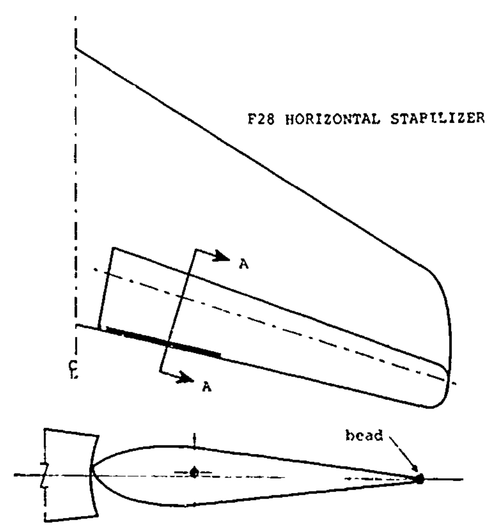


Fig. 6 F28 tailplane and final elevator configuration

fully acceptable. A modest boost ratio of 4 showed to be the best compromise of this application.
Finally in figure 8 it is illustrated that the bead length adopted on the basis of wind tunnel tests produced identical effects in flight. Flight tests also confirmed the predictions regarding the flying qualities during excursions beyond M_{MO}.

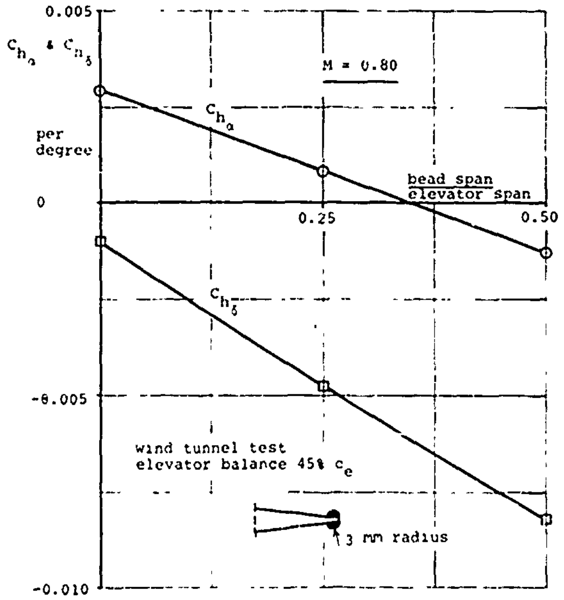


Fig. 7 Influence of bead on elevator hinge moment derivatives

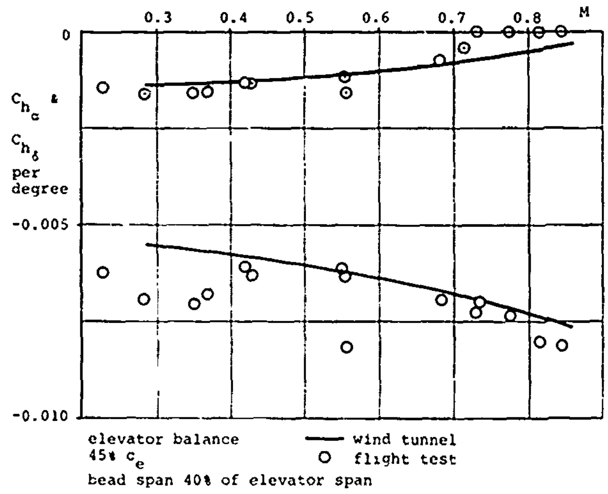


Fig. 8 Hinge moment characteristics of F28 elevator

DEVELOPMENT OF STALL CHARACTERISTICS

Initial flight testing with a prototype aircraft is always afflicted with some uncertainties regarding flying qualities, more specifically regarding stall characteristics. Although it is well known that the correlation between flight and wind tunnel with respect to stall behaviour is rather speculative, extensive use was made of the wind tunnel in preparation for the F28 prototype stall tests in an attempt to establish trends rather than quantitative information. This concerned particularly the investigation of devices which might be required in the course of full scale stall tests to adjust the characteristics to become satisfactory.

A qualification of these characteristics is provided by the civil airworthiness requirements, which ask for easy recognition by the pilot of the developing stalled wing condition and for gentle behaviour of the aircraft in the stall to avoid large attitude changes and consequently losses in height.
In a flight simulator programme which was used to convert wind tunnel characteristics into pilot assessed full scale behaviour, it was recognized that an unmistakable nose down pitching motion at or near the stall would provide satisfactory results. This was particularly the case because of the reduced longitudinal stability which had to be expected for angles of incidence beyond the stall for T-tailed aircraft.

Figure 9 shows the relationship between pitching moment and angle of attack as obtained in the final stage of wind tunnel testing. It can be noticed that immediately beyond the angle of attack for maximum lift a sharp increase in nose down pitching moment appears. The clean wing stall was characterized by a rapid span wise spread of separation. The result on figure 9 was obtained by controlling the location of initial flow separation on the wing by use of a small boundary layer fence near the wing leading edge. Further details of this effect will be shown later.

Figure 9 also shows the characteristic variation of the pitching moment at extreme angles of attack for an aircraft equipped with a T-tail, which is caused by the immersion of the horizontal

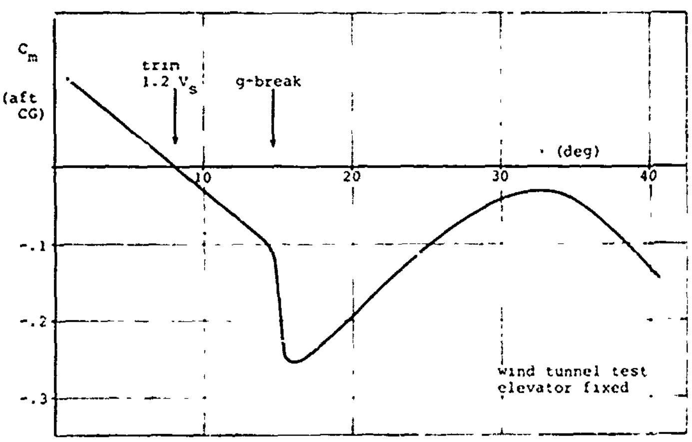


Fig. 9 F28 pitching moment characteristics

tailplane into the wing wake. The associated flight mechanical aspects of this phenomenon were already discussed at an earlier meeting of this panel.

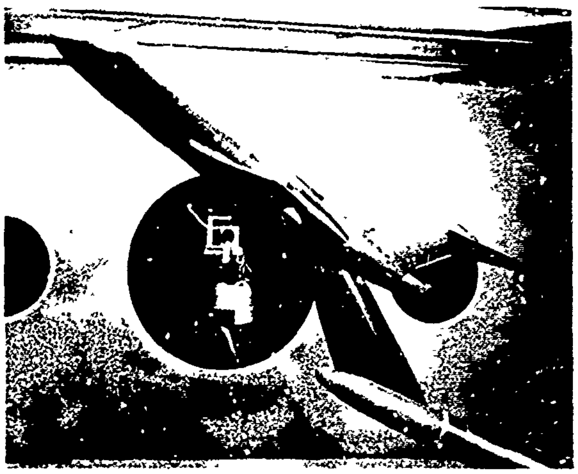
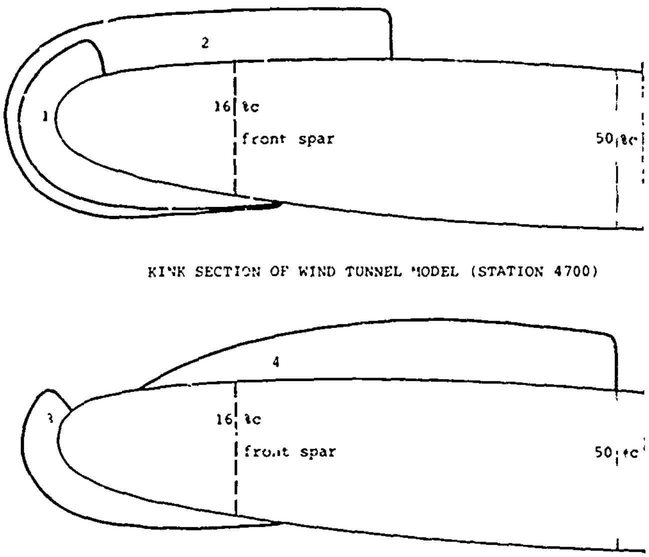


Fig. 10 F28 wind tunnel model at extreme angle of incidence

When the model size is sufficiently small relative to the dimensions of the test section of the tunnel, scale effects are negligible in conditions of separated flow over the full wing span.
Pitching moment data at these large angles of attack, as obtained in the wind tunnel, are therefore valid for the full scale aircraft. A typical picture of the investigation at extreme angles of incidence is shown in figure 10.

It may be noted here that testing at extreme angles of attack in the wind tunnel revealed that a nose down pitching moment was obtained throughout the angle of attack range investigated, i.e. up to 38 degrees, with the elevator deflected fully downward; this applies for all flap settings and up to the aircrafts most aft centre of gravity position. This was as well illustrated by full scale tests showing prompt recovery from angles of attack as high as 30 degrees.

The desired increase of the nose down pitching moment near maximum lift can be obtained by initial flow separation on the inner wing, which causes a favourable change of the downwash field at the horizontal tailplane. Apart from this effect flow separation on the inner wing also results in retention of full lateral control up to angles of attack at which the flow on the outer wing separates.
There is however one restriction relative to early flow separation for the case of the F28 as distortion of the engine intake flow should be avoided up to stall onset. This implies that the wing sector immediately in front of the engines should preferably stall at an incidence angle beyond maximum lift.



In the wind tunnel phase many aerodynamic gadgets were tried out to probe possibly satisfactory configurations in full scale testing. The small boundary layer fence showed to be most promising in relation to high maximum lift in combination with the desired characteristics.
Figure 11 shows a number of boundary layer fence sizes tried at one wing section (station 4700). The intention of this survey was to obtain a minimum fence size for the desired characteristics.
The short fence in front of the suction area and fence 4 located aft of this area on the wing nose failed to produce any effect in stall behaviour. Fence 1 and 2 were almost equally effective.

Fig. 11 Boundary layer fence sizes tested in wind tunnel

The way in which the progression of flow separation is affected by the introduction of a boundary layer fence is depicted in figure 12. It can be observed that the small fence at the leading edge of station 4700 changes the stall progression of the F28 wing completely. Local separation is introduced at the inboard side of the fence at 10 degrees angle of attack, the maximum lift is attained at approximately 13^{o}, the aileron region stalls at 19 degrees, while the wing without fence abruptly loses lift at 15.7 degrees due to full span stall. A very slight loss in lift accompanies the changed separation pattern.

Figure 13 presents the influence of the spanwise location of a fence on the progression of flow separation. This progression is depicted by showing the angle of attack for onset of flow separation, for maximum lift and for separation in the aileron region.
The identical characteristics at root and tip represent in fact the absence of the fence. The figure shows the result of tests on the wing with fully deflected flaps, being the most critical with respect to stall behaviour. It can be concluded from the figure, that a small leading edge boundary layer fence in almost any position largely affects the progression of flow separation. The separation in the aileron region is thereby postponed to much larger angles of attack than without fence.
This improvement is accompanied by a slight loss in maximum lift as can be recognized from the smaller angle of attack for maximum lift. Pitching characteristics in the stall were only satisfactory for the inboard positions of the fence. The initial flight testing was

therefore started with a fence at wing station 4700, the section at the kink in the leading edge of the wing.

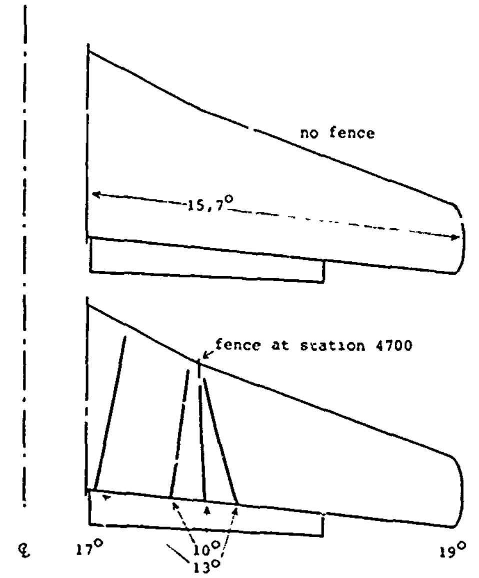


Fig. 12 Effect of fence on progression of flow separation

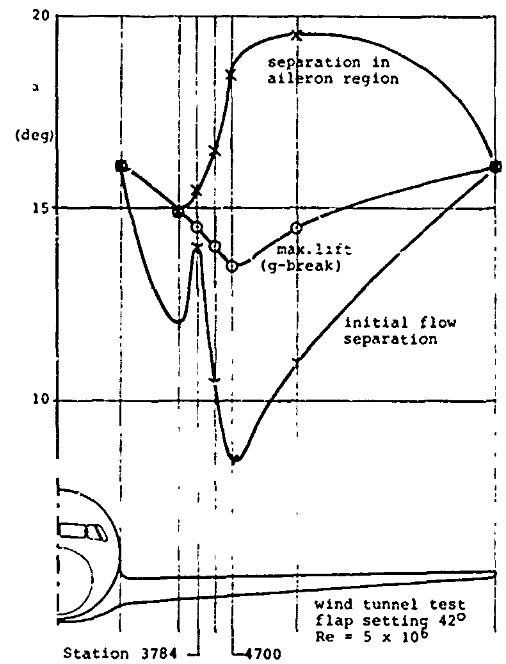


Fig. 13 Effect of span wise fence location on progression of flow separation

The characteristics observed in the wind tunnel were basically confirmed during prototype stall tests. However, the initial buffeting which preceded the stall of the wing was quite strong, and affected adversely the obtainable maximum lift.
Because of this observation it was decided to include in the flight test programme a number of alternative fence positions previously investigated in the tunnel.
Extensive tests revealed that the optimum fence location was 3 ft more inboard than station 4700, both from a point of view of obtainable maximum lift and overall stall characteristics.

OPEN DISCUSSION

H.Max, Germany: Mr Schuringa, from your paper I understand that you have done an optimization of the size, form and position of the boundary layer fence for getting good stall characteristics in the low-speed flight region. Due to a boundary layer fence very often you have to pay penalties in the transonic region. Have you found for example a remarkable influence of the fence on the Mach number for buffet onset?

Tj.Schuringa, Netherlands: We never performed flight tests at transonic speeds without a fence on the wing, so we do not know explicitly any detrimental effect of the fence on transonic characteristics, and particularly the buffet onset boundary. This boundary was determined at two occasions with the fence at different positions, i.e., at the kink and 3 feet more inboard, without any noticeable difference. Furthermore, this buffet onset boundary proved to be slightly higher, in terms of lift coefficient, than predicted from wind tunnel tests, thus there was not much reason to suspect the fence.

FLIGHT SIMULATION -

A SIGNIFICANT AID IN AIRCRAFT DESIGN

by

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SUMMARY

Landing an aircraft on the bobbing deck of a carrier is the most critical piloting operation performed routinely by the U.S. Navy. Because of recognized limitations of specifications in effectively addressing the unique characteristics of a particular design in the carrier approach environment, the most recent aircraft development programs have made extensive use of piloted flight simulation to evaluate the carrier approach characteristics early in the development cycle. This paper will describe the role of simulation in the development of the S-3 and F-14 aircraft, including the facilities used, the problems addressed, and the conclusions reached. In addition, an appraisal is made of simulation technology as applied to aircraft design with a projection of future applications.

1. INTRODUCTION

During the past two decades, the general-purpose, flight simulator has been developed into one of the more powerful tools available to the airplane designer. Making proper decisions at appropriate times is unquestionably the key (albeit trite) to successful aircraft design, and it is within this context of timely decision making that simulation is the "tour de force." The ability of the flight simulator to provide a basis for pilot evaluation years before first flight, in turn, allows the luxury of the "fly before buy" phylosophy to be practiced by the designer.

The U.S. Navy and its associated contractors have been heavily involved in the application of ground based and inflight simulation technology since its inception. As early as 1955, simulation played a significant role in the design of the RA5C Vigilante (ref 1). More recent examples, which for the most part addressed a broad spectrum of flying qualities investigations directed toward establishing criteria, were recently covered (ref 2) by Mr. Siewert at your Ottawa meeting and will not be reiterated here. Suffice to say, it is on the basis of this considerable experience with flight simulators that the Navy has encouraged full use of simulation technology and considers the flight simulator as an integral part of the design and development effort.

Both of the most recent Navy aircraft development programs, the Lockheed S-3 and the Grumman F-14, have effectively utilized general purpose flight simulators during design and development. An examination of these two programs will provide as current a view of the use of simulators as there is available. While considerable contrast exists between the missions and operational envelope of the two aircraft, there was a common focal point for their respective simulation programs, and that was the carrier landing. The intricacies of performing a carrier approach simulation and the associated design problems which can be addressed on such a simulation can hardly be considered typical. But these very specific applications do serve to highlight the potential of flight simulators as an aid to making critical design decisions.

2. DISCUSSION

Before launching into the use of general purpose flight simulation as an aid to aircraft design based upon today's technology let's review some of our more notable design problems of the past. For example, the pilot induced oscillations and the roll-yaw-pitch divergence (classic inertia coupling) would never have achieved the level of notoriety accredited them had flight simulators been in vogue at that time. Both of these problems were symptomatic of our inability to effectively evaluate conglomerate systems prior to flight test. That is not to say that each time a simulator is used to support a design effort some form of catastrophic mismatch is being thwarted, but the probability of such a mismatch occuring is certainly reduced.

Divorced from specific design support, flight simulator investigations are currently concentrated on three areas of concern. These are the landing approach flight characteristics (STOL, SST, and carrier landing); the high angle of attack, stall-spin area; and the combat arena. Fortunately, the S-3 and F-14 simulation programs provide the opportunity to address one of these "areas of concern," namely the carrier approach within the contest of design support.

The reasons for both the S-3 and F-14 simulation programs focusing on the carrier approach are delineated in Figure 1.

Typical questions which were addressed in the approach simulation and which significantly impacted the design are presented in Figure 2.

The facilities used to answer these and many other questions regarding the approach characteristics are described in the following section.

3. SIMULATOR DESCRIPTION

The Vought Aeronautics simulator used during the initial S-3 investigations is composed of a six-degree-of-freedom representation of the aircraft motion, a three-degree-of-freedom (pitch, heave and roll) carrier, a digitally generated display of the night carrier landing situation, and a small-amplitude, moving-base cockpit. An example of the digitally generated scene just prior to touchdown is shown in Figure 3. The datum lights and "meatball" of the Fresnel lens optical landing system (FLOLS) can be seen to the left of the flight deck. The moving base cockpit and landing signal officer instructor stations are shown in Figure 4.

The Lockheed simulator, which has been progressively improved during the S-3 development, started out with a three-degree-of-freedom aircraft mechanization in a fixed base cockpit using a 5 inch cathode ray tube display of the FLOLS and horizon. The CRT display is shown in Figure 5. Early improvements included expansion to six degree of freedom equations of motion for the aircraft and the use of a closed circuit television display to furnish visual cues to the pilot. A black-and-white television camera is moved over the terrain model in 6 degrees of freedom to simulate the motion of the aircraft. The scene is displayed on a 25 inch television monitor mounted behind a collimating lens in the windshield area of the cockpit. The S-3 simulation has several models available, including a terrain map of the flight test facilities and airport at Palmdale, an aircraft carrier model and seascape, and a cloud pattern used for high altitude flights.

The carrier model shown in Figure 6 is a 400:1 scale CVS class aircraft carrier model with flight deck markings, landing area deck lights and a simulated optical landing system. The optical landing system simulation initially consisted of a light source and mirror arrangement which provided glide slope information similar to the old mirror landing system. This optical system has been replaced with a servo-driven light system. The servo is controlled by a computer generated glide slope error signal which moves the "meatball" relative to fixed reference lights. This system provides an improved indication of "meatball" motion, earlier sighting of the "meatball" during an approach, and permits simulation of the effects of ship motion due to rough sea conditions.

The control column and rudder pedals in the simulator were designed to the S-3 geometry. The control column and pedals are both driven by hydraulic servo actuators which reproduce the feel force characteristics of the aircraft including detent, friction, bob weights nonlinear springs, and control system dampers. If the dual hydraulic system fails in the aircraft an emergency flight control system is provided by reversion to direct mechanical control of the surfaces and the artificial feel-force system is disengaged. This emergency condition, including the transfer transients, can also be simulated with the simulator force-feel system.

The most recent improvement to Lockheed's simulator is the addition of a four-degree-of-freedom motion system having the following capabilities:

	Pitch	Roll	Vertical	Lateral
Acceleration	± 25 deg/sec^2	± 50 deg/sec^2	+0.8, -1g	± 0.2 g's
Rate	± 15 deg/sec	± 17 deg/sec	± 12 in./sec	± 15 in./sec
Displacement	± 15 degrees	± 15 degrees	± 12 inches	± 12 inches

This new capability provides improved cueing, particularly for evaluation of failures and external disturbances. The motion system is shown in Figure 7.

The evolution of the F-14 carrier approach simulation followed much the same pattern as S-3's. Initial studies were conducted on the Grumman small amplitude motion system depicted in Figure 8. The aircraft was represented only in the three longitudinal degrees-of-freedom. The visual scene of the carrier and seascape were projected by a point light source onto a transluscent screen located in front of the cockpit. The FLOLS model was located on the face of the projection screen and the sensitivity was varied inversely with range.

The most recent F-14 approach investigations were conducted at the NASA Ames Research Center using their moving transport-cab simulator shown in Figure 9. The characteristics of this system are as follows:

	Roll	Pitch	Heave
Acceleration	± 270 deg/sec^2	± 270 deg/sec^2	± 1 g
Velocity	$\pm$ 13 deg/sec	$\pm$ 13 deg/sec	-
Displacement	$\pm$ 9 deg	+ 14, -6 deg	± 2 ft.

The motion system was coupled with the Redifon closed circuit, color television visual system. This is the same visual system used in the Concord simulation program. The carrier model and seascape are shown in Figure 10. The aircraft was modeled with the six degree-of-freedom equations of motion. The carrier model was driven in two degrees-of-freedom (pitch and heave) for a nominal sea-state simulation. The FLOLS model used a servo-driven fiber optics element for generation of the "meatball."

As with the Vought and Lockheed simulations, the NASA mechanization included the modeling of the complex air wake behind the carrier. The wake model derived from Ref 3, includes the down draft and deterioration of wind-over-deck aft of the carrier, causing the aircraft to settle as it approaches the ramp; the large scale cyclic vorticies shed from the pitching-heaving carrier deck and dissipated down stream; and the small scale, random appearing turbulence generated by the carrier superstructure.

4. CARRIER APPROACH TASKS

A major portion of the following discussion will be devoted to the S-3, primarily because of the more complete documentation. Fortunately, the S-3 program is an excellent example of a well coordinated and aggressively implemented simulation in support of design. The abridged design and development schedule of Figure 11 indicates the simulation program to be a continuing effort from contract initiation through flight testing.

Thus far, the plan has been and is being followed with only short interruptions for simulator modifications. The flight conditions evaluated during the development program are listed in Figure 12. While most of the evaluations were focused on the critical carrier approach task, other segments of the flight envelope have received sufficient attention to assure satisfactory characteristics of the basic airplane and acceptable failure transients and post failure characteristics.

The carrier approach evaluations were conducted using primarily the terminal approach profile, which is essentially a straight-in approach started just prior to glide slope intercept. The straight-in carrier approach was used to evaluate glide slope intercept and tracking capability and general controllability in final approach. The test consisted of a short level-flight segment at 600 feet altitude starting trimmed at approach speed in power approach configuration, followed by glide slope intercept and tracking the FLOLS "meatball" to touchdown. The carrier landing visual display is available through the approach.

A second type of approach profile, called the circling carrier approach, started 1-1/2 miles behind and slightly to the right of the carrier with the aircraft in the cruise configuration. As depicted in Figure 13, this profile envolves approaching the carrier at constant altitude and executing a 360° turn while decelerating to approach speed and transitioning to the approach configuration. The final segment of this test is identical to the straight-in approach. The circling approach allows evaluation of trim changes, aircraft dynamics and overall flying qualities in maneuvers typical of carrier recovery operations. Any portion of the circling approach may be used to evaluate problems associated with specific tasks. Because the carrier aspect relative to aircraft exceeds the visual system capabilities for the first portion of this maneuver, the pilot utilizes the ground track display to monitor the aircraft position.

High speed flight characteristics were evaluated in maneuvers typical of operational requirements or similar to those used to evaluate specific aircraft characteristics during flight test. A visual cloud presentation is available to provide attitude references for these maneuvers. The ASW maneuvers were evaluated using the ground-track display.

5. THE DIRECT LIFT CONTROL INVESTIGATION

One of the earliest design decisions to be made on the S-3 was whether or not a DLC (direct lift control) system should be included in the design. DLC provides a highly responsive vernier control of approximately $\pm$ 0.1 to 0.2 normal load factor at the approach flight conditions. This DLC modulated load factor is generated by rapid reconfiguration of the wing through high response trailing edge flaps or alternately, symmetric operation of spoilers from a biased (DLC neutral) deflection. The appreciable pitching moment generated by the wing reconfiguration is alleviated by a DLC interconnect with the primary longitudinal control surface.

For DLC implementation through the stick mounted thumb wheel, the pilot can independently control flight path with DLC and attitude with the stick. This approach is particularly attractive during the terminal approach when the pilot is simultaneously concerned with maintaining the desired touchdown attitude while making final flight path corrections. The DLC system provides sufficient flight path control for the pilot to change glide slope 3/4° in one second, using an average incremental load factor from DLC of .07 g at an 100 knot approach speed.

The benefits of DLC are logically going to be most apparent on aircraft with marginal approach characteristics e.g., highly wing loaded (i.e., low N_{Z_α}) aircraft requiring large attitude changes to make glide slope correction and short-coupled aircraft with the more pronounced reversal in the initial load factor transient (i.e., high-T_{h_2}).

However, with the S-3 having a moderate wing loading and a respectable tail arm, the benefits of incorporating DLC were not apparent. To quantify the possible levels of enhancement, in terms of pilot rating and/or touchdown performance, Lockheed utilized their fixed-base carrier approach simulator and the moving base facility at Vought. 3,400 approaches were flown with the basic S-3 and several candidate DLC systems. The subjective pilot rating data resulting from these evaluations are summarized in Figure 14 for the 1,200 moving base evaluations. The data indicates the basic S-3 to be satisfactory with a pilot rating of 3.0. The two DLC systems provided a slight (0.2 and 0.4) rating improvement.

Touchdown performance data for the same series of runs is presented in Figure 15 in terms of sink rate and attitude dispersion. The dispersion envelopes for the basic S-3 and the more promising DLC systems are both well within the design boundaries. The sink rate dispersion is shown to be slightly higher for the DLC system and the pitch attitude dispersion is appreciably reduced. However, this decreased attitude dispersion with the DLC does not completely compensate for the characteristic increase in mean pitch attitude associated with the DLC neutral configuration. Thus for the S-3, the DLC system as evaluated on the simulator did not show a significant improvement in pilot rating over the already satisfactory basic aircraft, and indicated a slight deterioration in the margin between the dispersion envelope and the design boundary. The conclusion was obvious that the S-3 did not need DLC, and the associated cost, complexity and weight was saved.

6. TURN COORDINATOR INVESTIGATION

Comparison of the S-3 lateral-directional and roll control characteristics against the imposed flying qualities specification (MIL-F-8785 ASG with additional Navy requirements) indicated compliance of the basic aircraft characteristics. However, simulator evaluation indicated these characteristics to be unsatisfactory (PR-5.5) because of the high level of Dutch roll excitation associated with roll control. A range of the aerodynamic coefficients normally associated with this type of coupling and the primary Dutch roll damping coefficient were then evaluated on the simulator to establish the sensitivity of the various parameters. The results of the evaluation are presented in Figure 16 and indicate that the reduction of the adverse yawing moment due to roll provides the most appreciable benefits. The increase in damping (larger negative values of C_{n_r}) is shown to be an improvement for the base line value of C_{n_p} but reducing in effectiveness as C_{n_p} is reduced toward zero, and actually having an adverse effect at zero C_{n_p}. The turn coordinator mechanizations evolving from the parametric evaluation are presented in Figure 17. The mechanization using both the yaw rate and roll rate feedback and aileron to rudder crossfeed provided the most versatility for optimization and the necessary closures to match the results of the parametric study. The alternate mechanization using only the yaw rate feedback and a filtered aileron to rudder crossfeed was felt to be the simplest mechanization which could provide the desired improvement. The approach, then, was to first optimize the more sophisticated mechanization and then to see how the simplified turn coordinator would compare. Representative results of the simulator evaluations presented in Figure 18 show the simplified coordinator to provide as much improvement as the best of the sophisticated systems.

The example clearly indicates the potential of the simulator in augmenting the flying qualities requirements, in sorting out the aerodynamic coefficients of import, and in synthesising a simple scheme to compensate for the aerodynamic deficiencies to the satisfaction of the pilots.

7. SHORT PERIOD STABILITY INVESTIGATION

The short period stability requirement for the F-14 in the approach configuration have been superimposed on the boundaries of the most recent flying qualities, MIL-F-8785B, in Figure 19.

Early estimates projected the F-14 basic airframe short period characteristics to be quite marginal in terms of specification compliance. Fluctuations in stability associated with configuration refinements tended to migrate even further to the deficiency side of the boundary.

While the deterioration in the basic stability level was disconcerting, the predicted performance of the stability augmentation system was by contrast quite reassuring. The characteristics for a representative condition, presented in Figure 20, show the augmented airframe to have a short period frequency of 1.45 rad/sec versus the 0.77 rad/sec frequency for the basic airframe. Early simulator evaluations indicated the basic aircraft characteristics to be generally satisfactory, but no appreciable improvement could be detected between the augmented and unaugmented characteristics. This was surprising in view of the significant increase in short period frequency predicted with augmentation. Subsequent analysis indicated that while the augmentation system did move the short period to higher frequencies as advertised, higher order terms (a washout and a lag-lead) associated with the augmentation had an adverse effect. A comparison of the higher-order, augmented aircraft dynamics with an equivalent unaugmented aircraft is presented in Figure 21. The response characteristics for the two systems are essentially identical. However, the short period frequency for the augmented aircraft is 1.45 rad/sec versus 0.9 rad/sec for the equivalent aircraft. Thus, the augmented aircraft evaluated by the pilots in the simulation had the appearance of the equivalent aircraft, which explained the difficulty in discriminating between the augmented and unaugmented aircraft. The ability to rely on the pilots evaluation of the approach characteristics helped keep the short period requirement in proper perspective. Knowing that the augmentation did not provide the analytically predicted margin of stability above the required level from a pilot's view served as a cautious reminder for the need to perserve the existing stability levels.

8. FLIGHT TEST

Prior to the first flight of the S-3 a comprehensive training program was conducted on the flight simulator. Thirty-three hours of simulator time were devoted to flying the basic flight profile and to the investigation of the various critical failures listed in Figure 22 for up-and-away conditions and in Figure 23 for landing.

Based upon the limited flight testing completed to date the Lockheed test pilots feel that the simulator is a valid representation of the aircraft.

Likewise, the Navy pilots who flew the F-14 approach simulation just prior to the first Navy preliminary evaluation felt the airplane-simulator match to be in good agreement.

9. CONCLUSION

The simulation of today is a uniquely useful tool for the aircraft designer:

As a means of actively and continually including the indespensable experience of the pilot.

As a monitor on requirements.

As an evaluator of the interface compatibility between systems.

As a demonstrator of new concepts.

LIST OF REFERENCES

1. J. D. Richardson, et al — The Application of Flight Simulators to the Development of the A-5A Vigilante. Journal of Aircraft, Vol. I, No. 4, July-August 1964 pp. 171-177.

2. Raymond F. Siewert — Recent U.S. Navy Flying Qualities Research. AGARD Specialists Meeting, Ottawa, Canada, Sep. 1971.

3. Tulvio S. Durand, et al — An Analysis of Terminal Flight Path Control in Carrier Landing, Rept. No. 137-1 (R) Systems Technology Inc., August 1964.

4. Hogan R. W. — Direct Lift Control Investigation Using A-6A, Grumman Rpt. XAR-102, July 1970.

- First and foremost- satisfactory carrier approach characteristics is very high on Navy list of priorities
- Recognized limitations of requirements
- Evaluation of qualitative requirements
- Minimize impact of achievement of satisfactory carrier approach characteristics (i.e. evaluation of design options and don't overdesign)
- Many interface considerations
  Airframe-engine
  Stability augmentation
  Approach power compensation
  Direct lift control

Figure 1 - Why Carrier Approach?

- Is direct lift control required?
- Are speedbrakes required ?
- Does the approach power compensation engine system compensate for carrier wake effects?
- Is a lateral-directional interconnect required?
- Is the direct lift control to elevator interconnect satisfactory over center of gravity range ?
- Does stability augmentation help approach characteristics?

Figure 2 - Typical Questions Addressed on Simulator

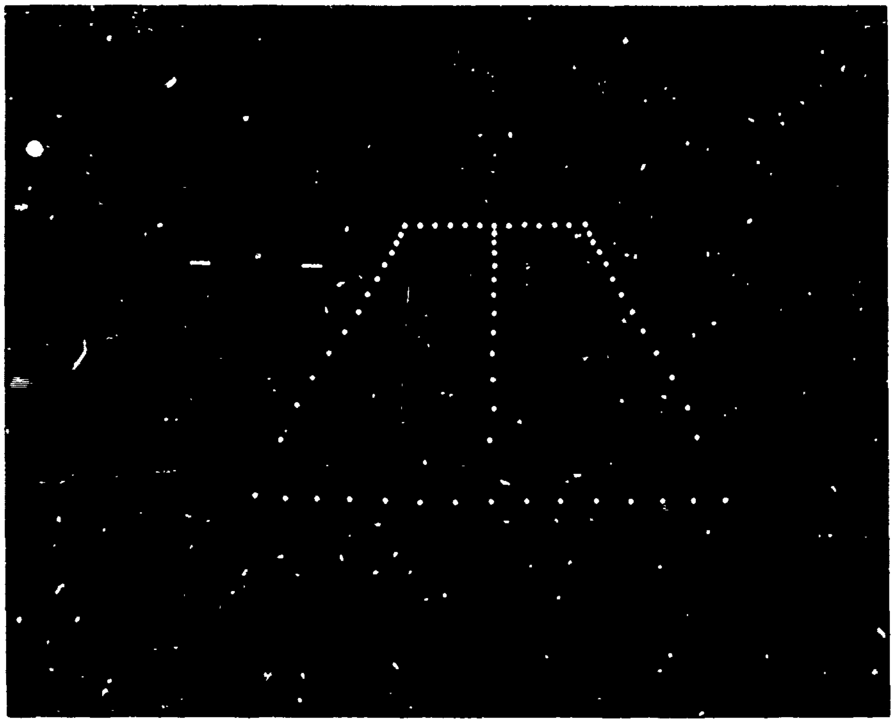


Figure 3 - Visual Presentation at Vought Aeronautics Carrier Approach

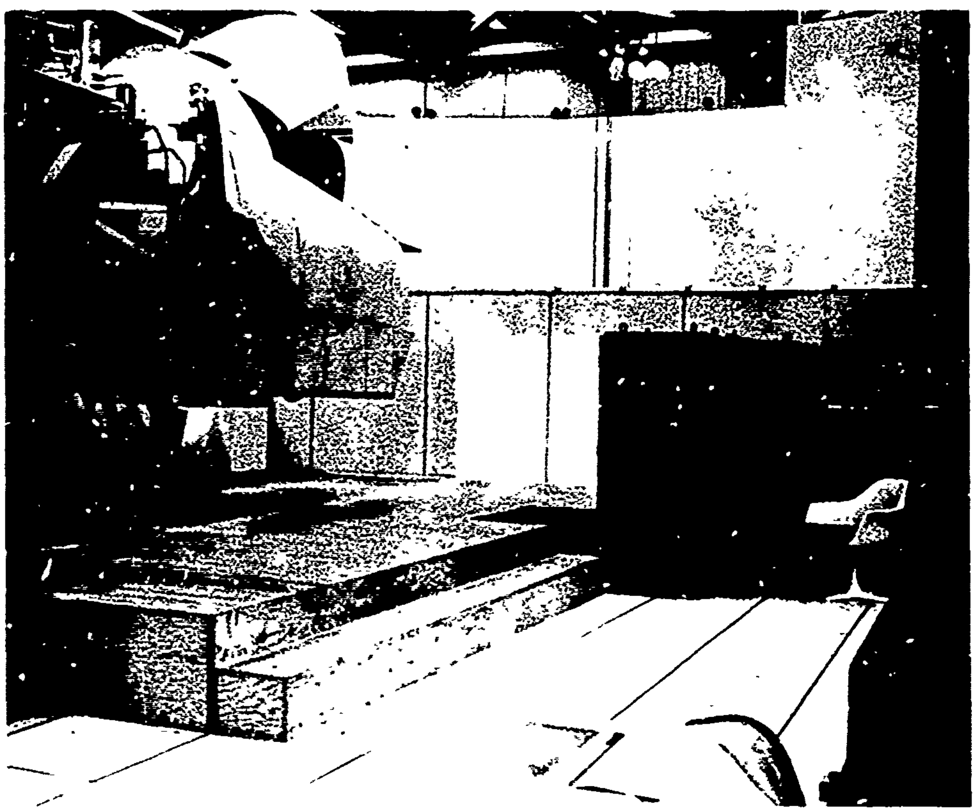


Figure 4 - Vought Motion System

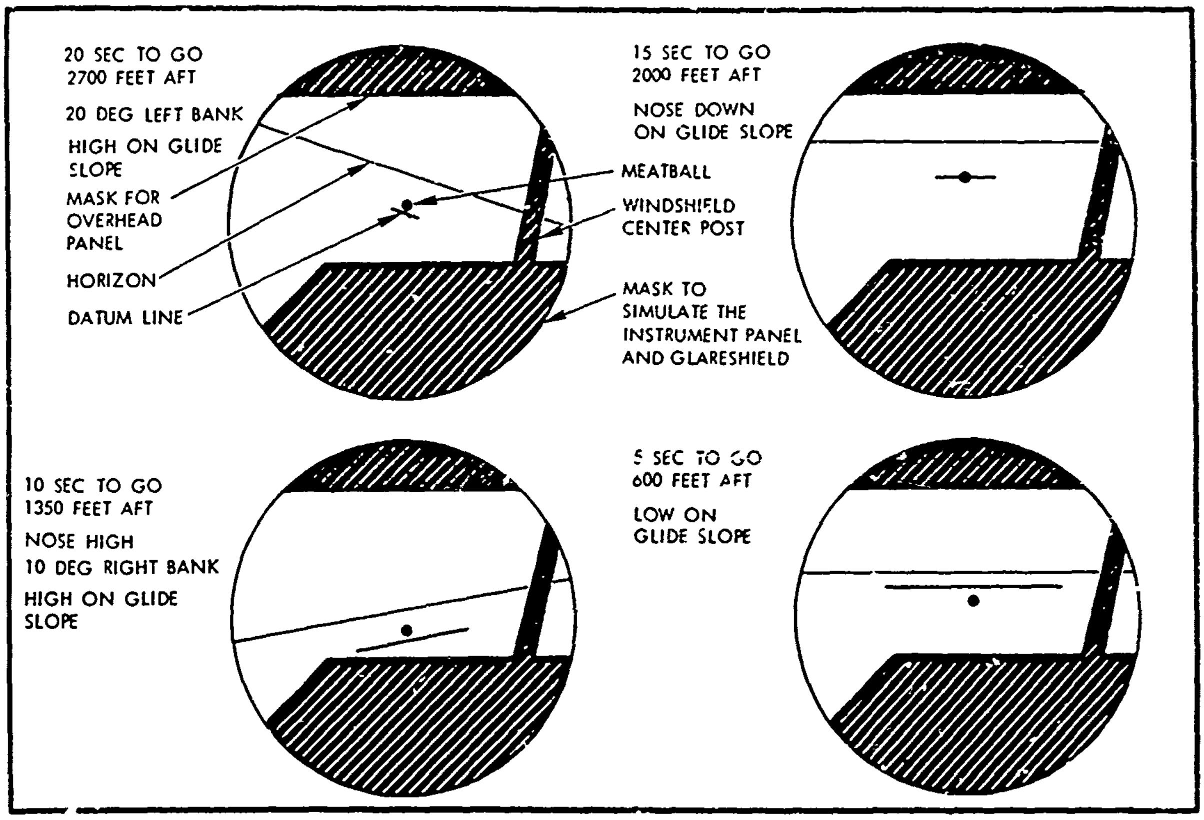


Figure 5 - Lockheed Initial Simulator Display

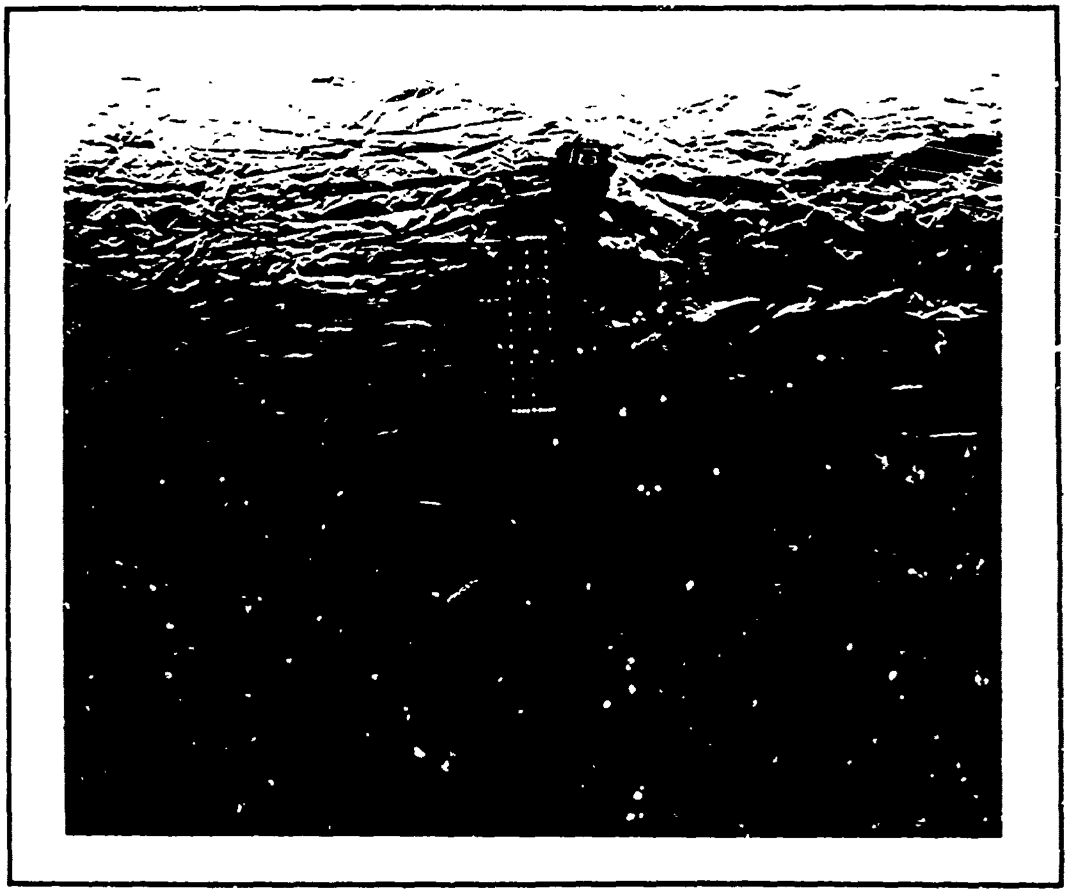


Figure 6 - Lockheed's Carrier and Seascape

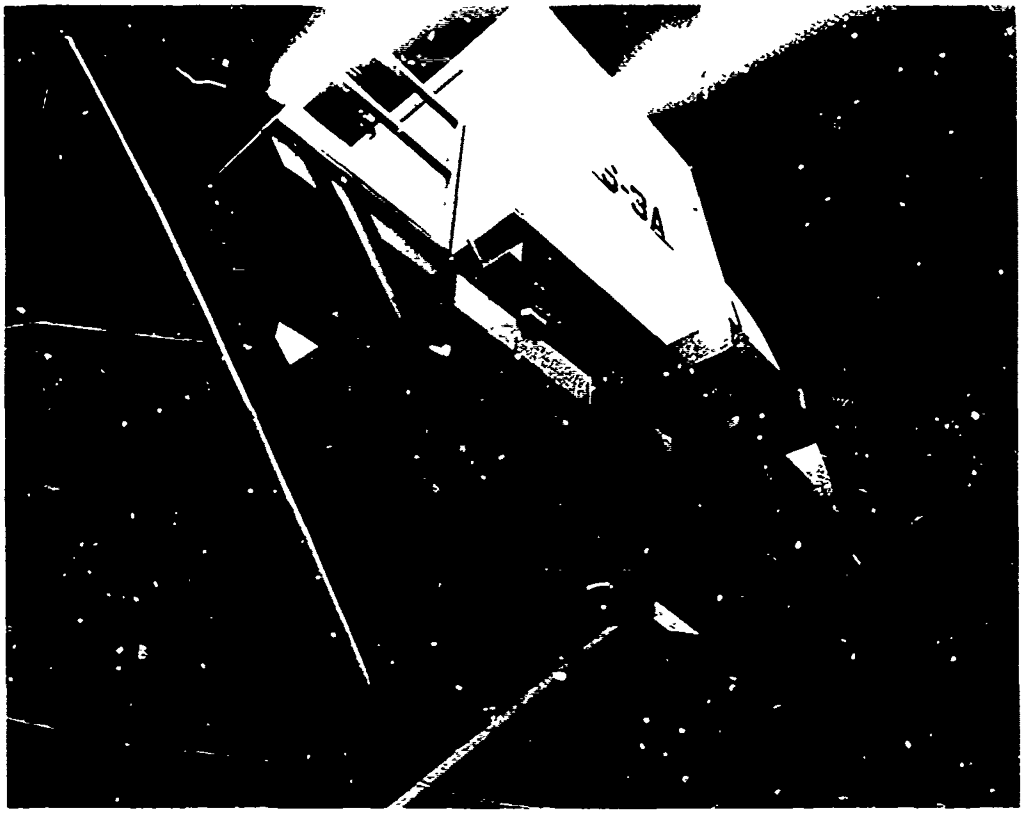


Figure 7 - Lockheed Motion System

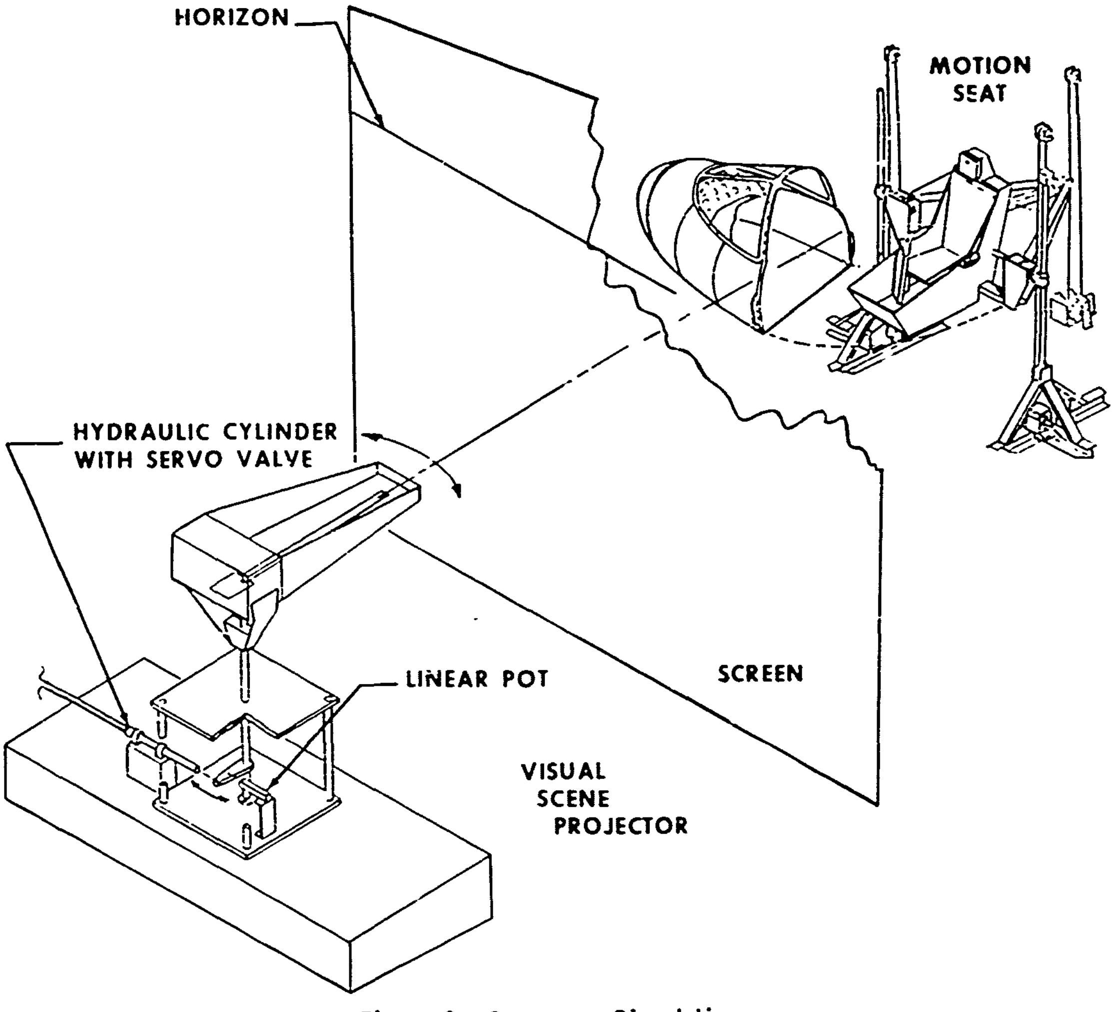


Figure 8 - Grumman Simulation

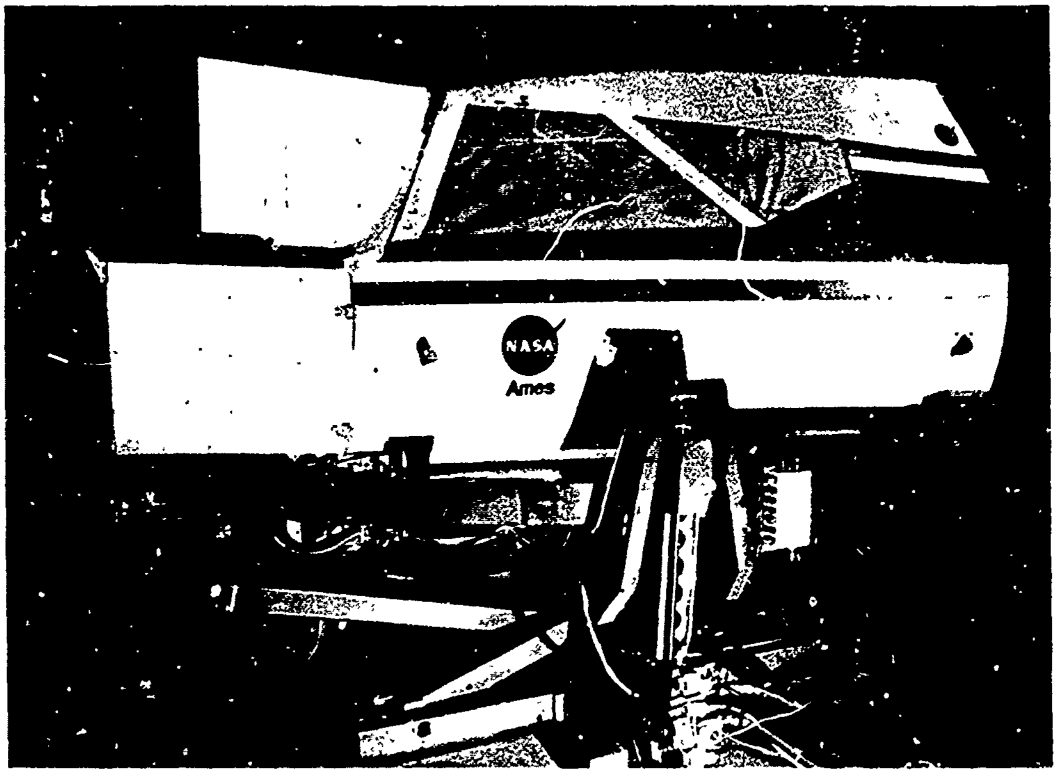


Figure 9 - NASA-AMES Transport CAB Motion System

Figure 10 - NASA-AMES Carrier and Seascape

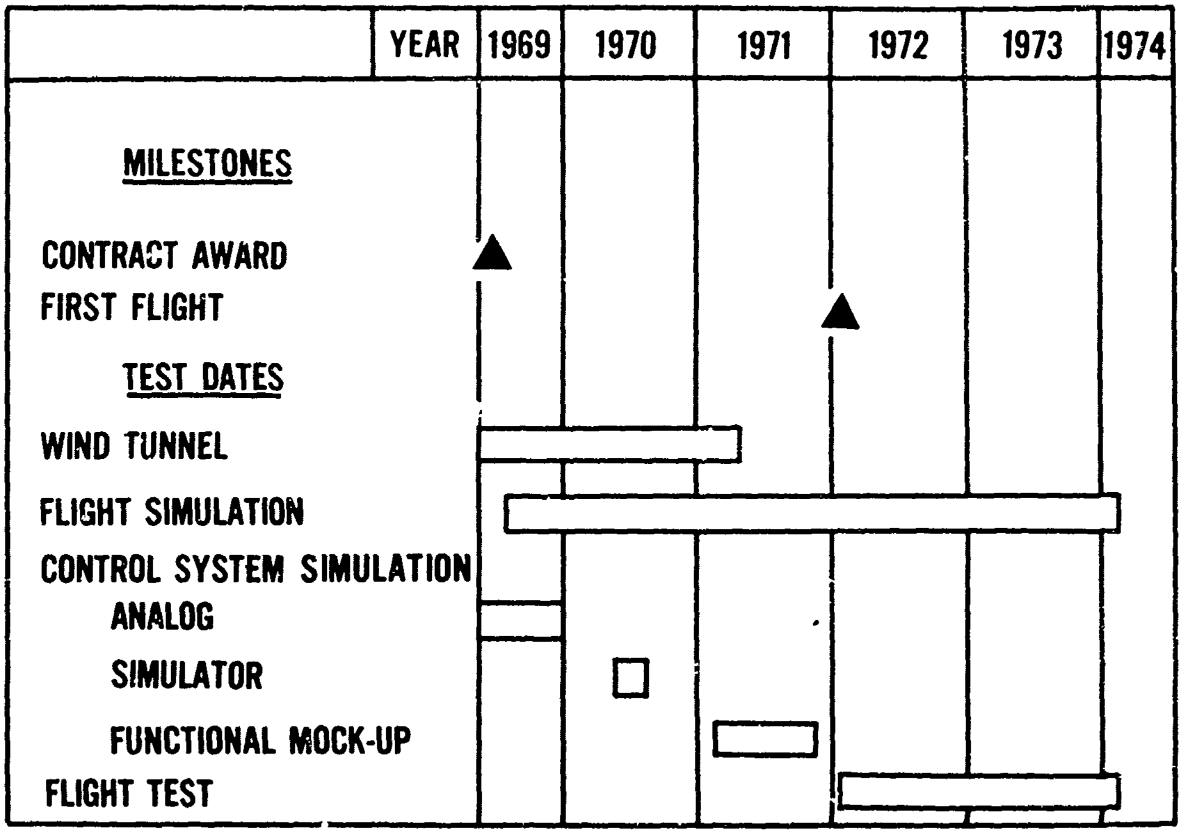


Figure 11 - S-3A Design and Development Schedule

- CATAPULT
- CARRIER APPROACH
- WAVE OFF
- UP AND AWAY
- ASW MANEUVERING

Figure 12 - Flight Conditions Evaluated on S-3 Simulator

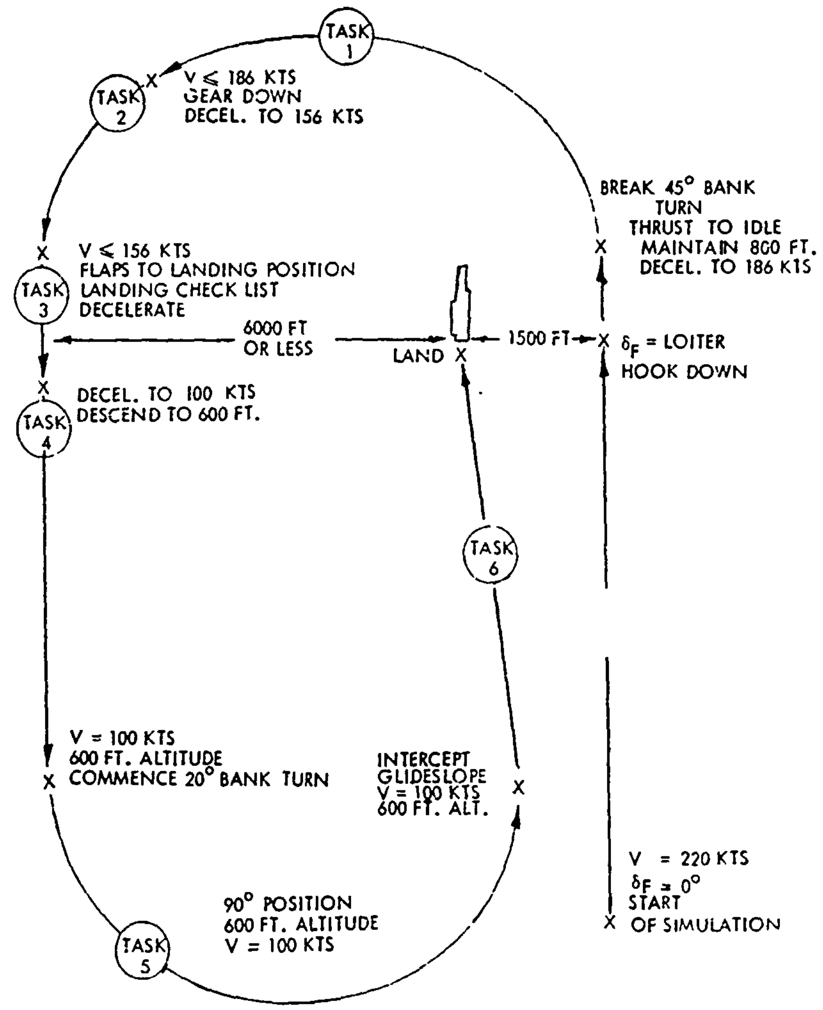


Figure 13 - Circling Approach Profile

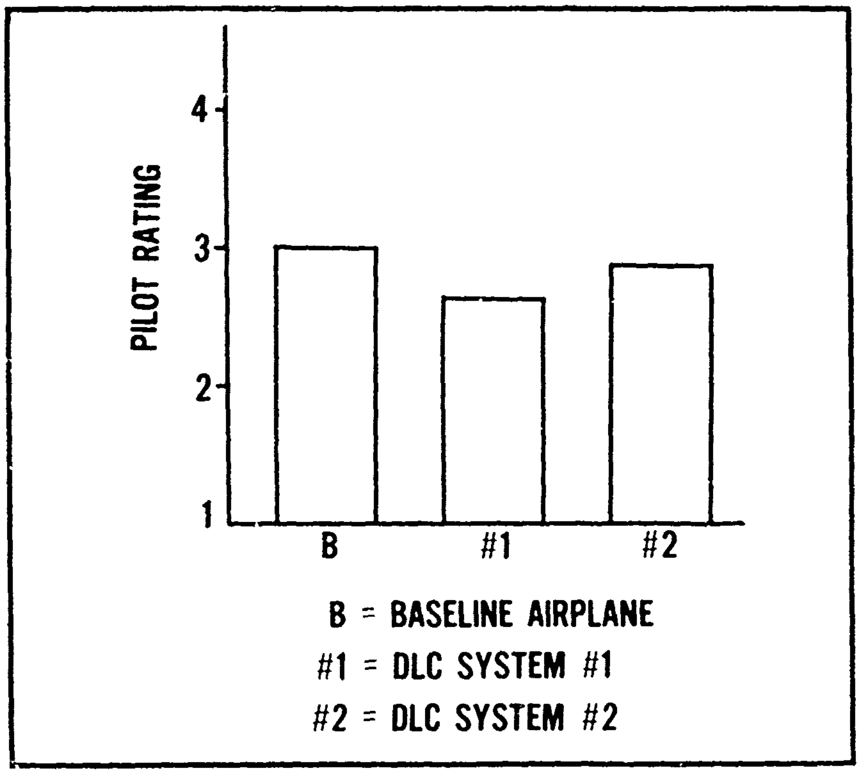


Figure 14 - S-3 Direct Lift Control Evaluation-Moving Base Simulator

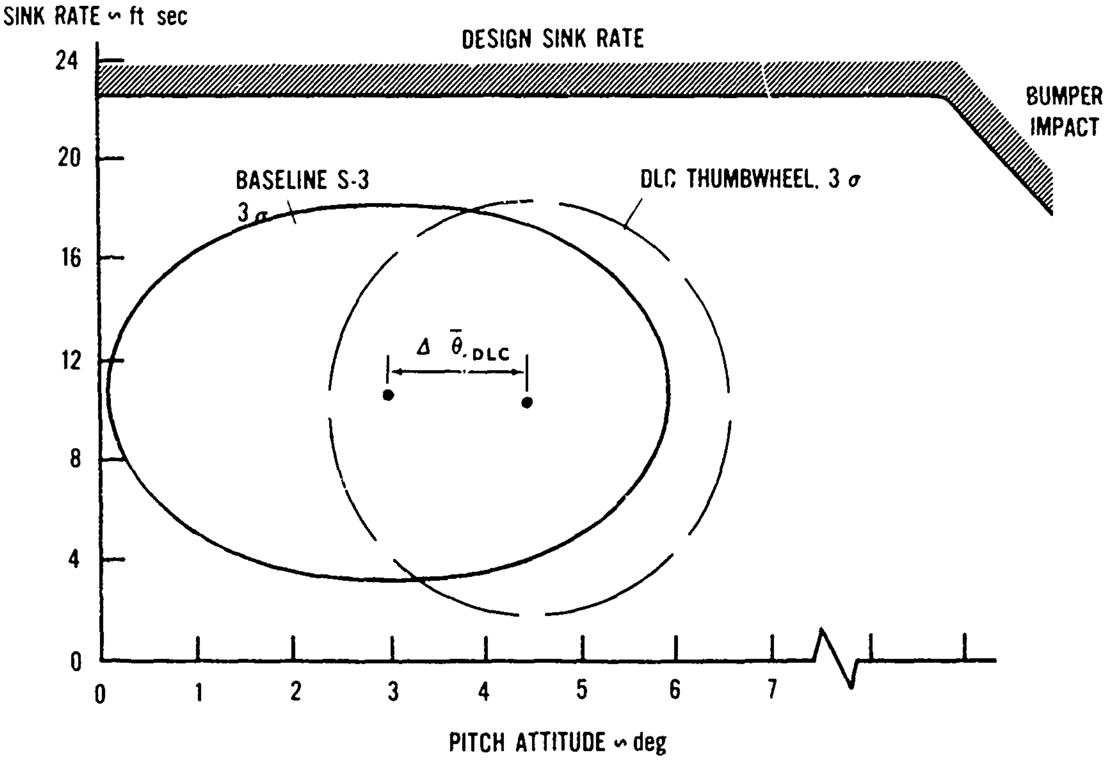


Figure 15 - S-3 Conditions at Touchdown - Moving Base Simulation

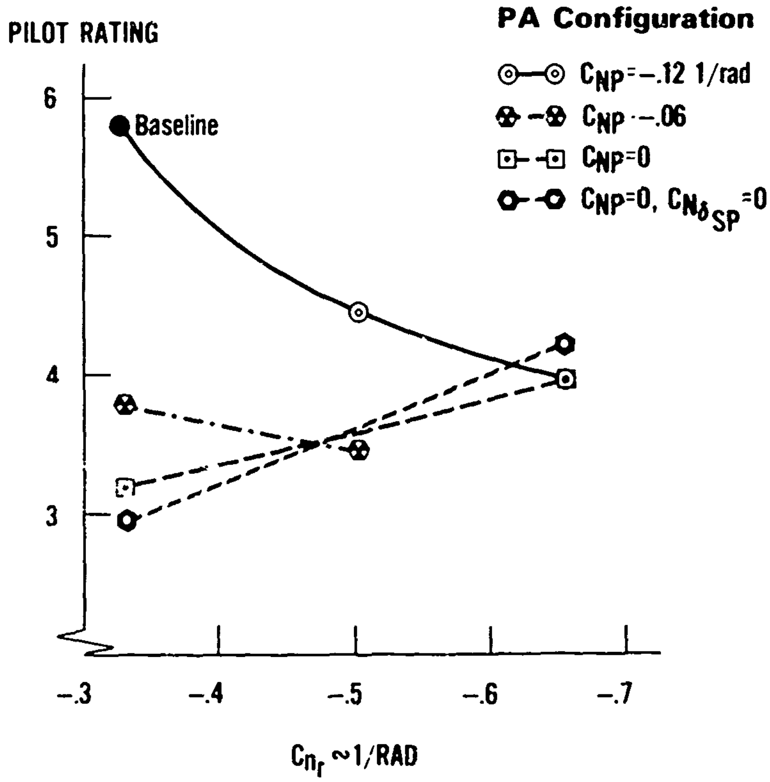


Figure 16 - S-3 Lateral Directional Parametric Study

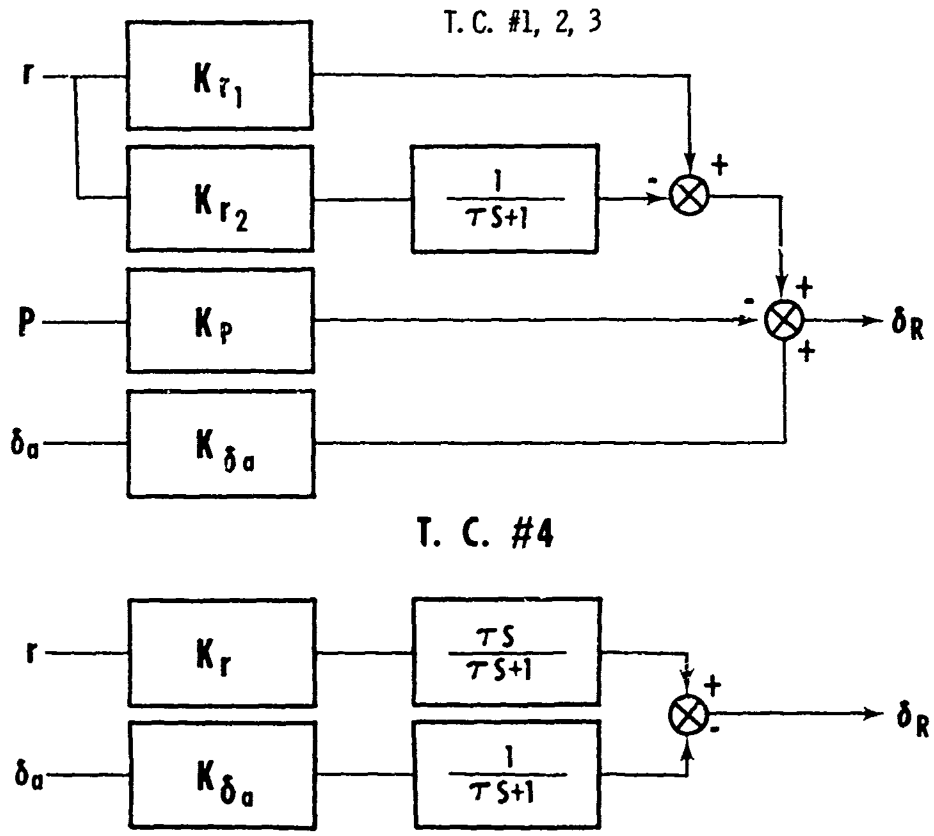
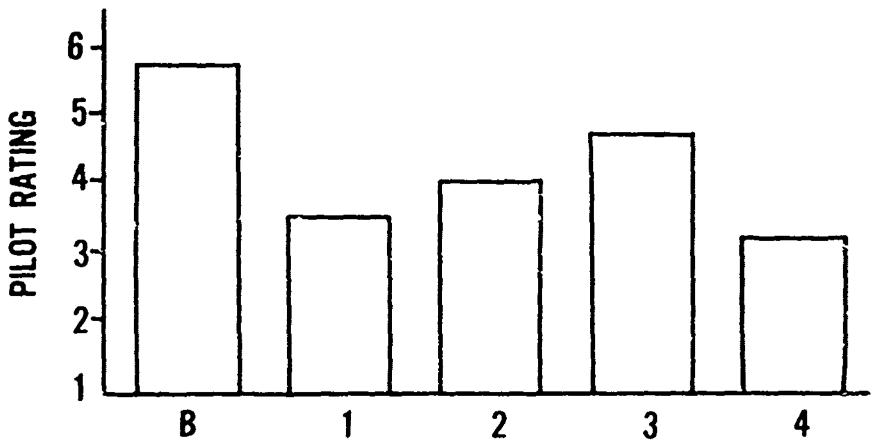


Figure 17 - Turn-Coordinator Mechanizations



B = BASELINE AIRPLANE
1 = TURN COORDINATOR #1
2 = TURN COORDINATOR #2
3 = TURN COORDINATOR #3
4 = TURN COORDINATOR #4

Figure 18 - Pilot Ratings of S-3 Turn Coordinators

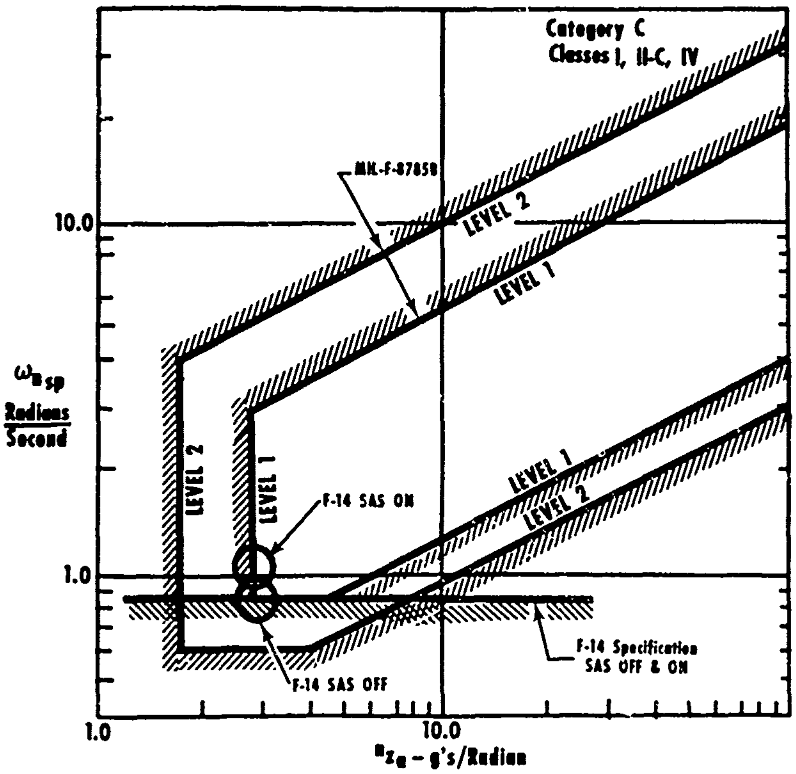


Figure 19 - Short Period Frequency Requirement

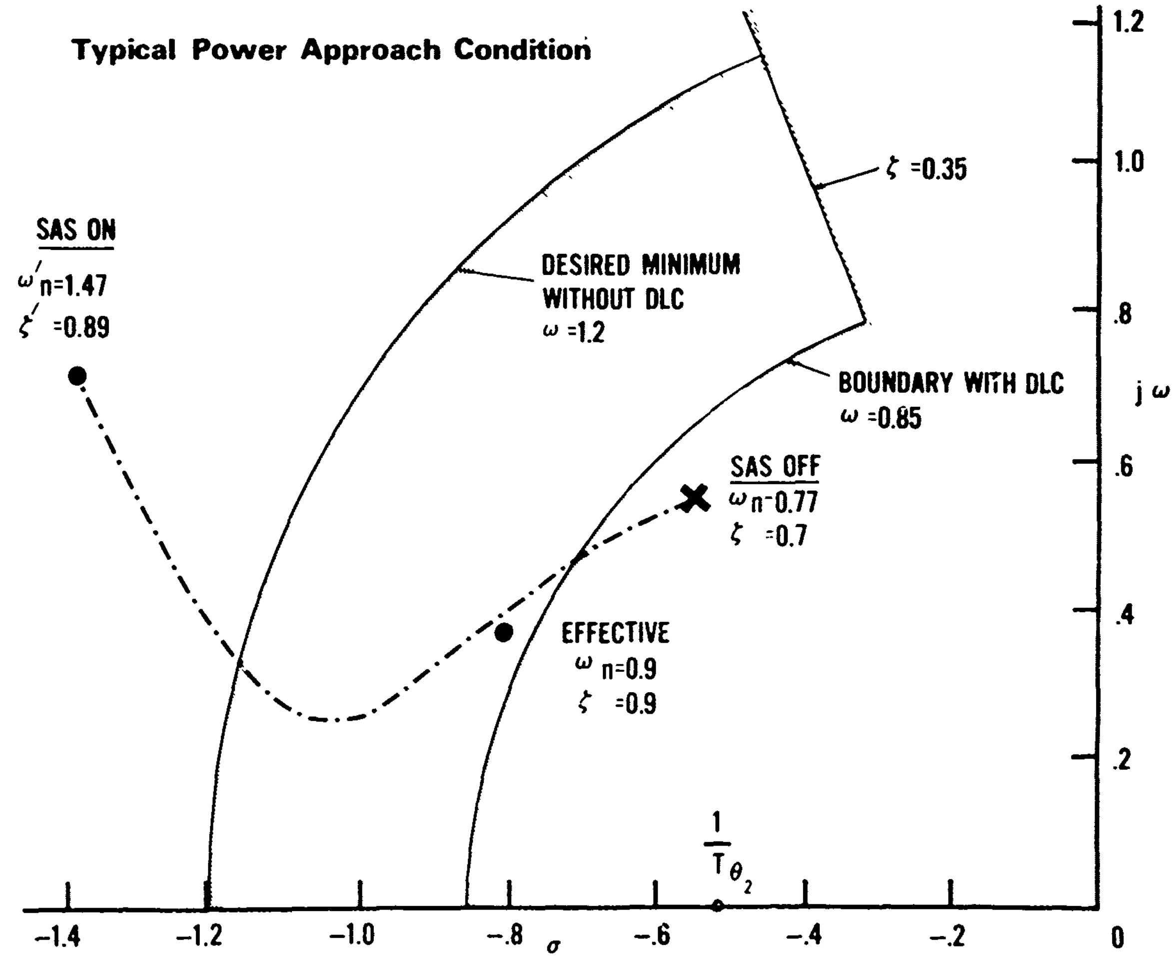


Figure 20 - F-14 Short - Period Dynamics

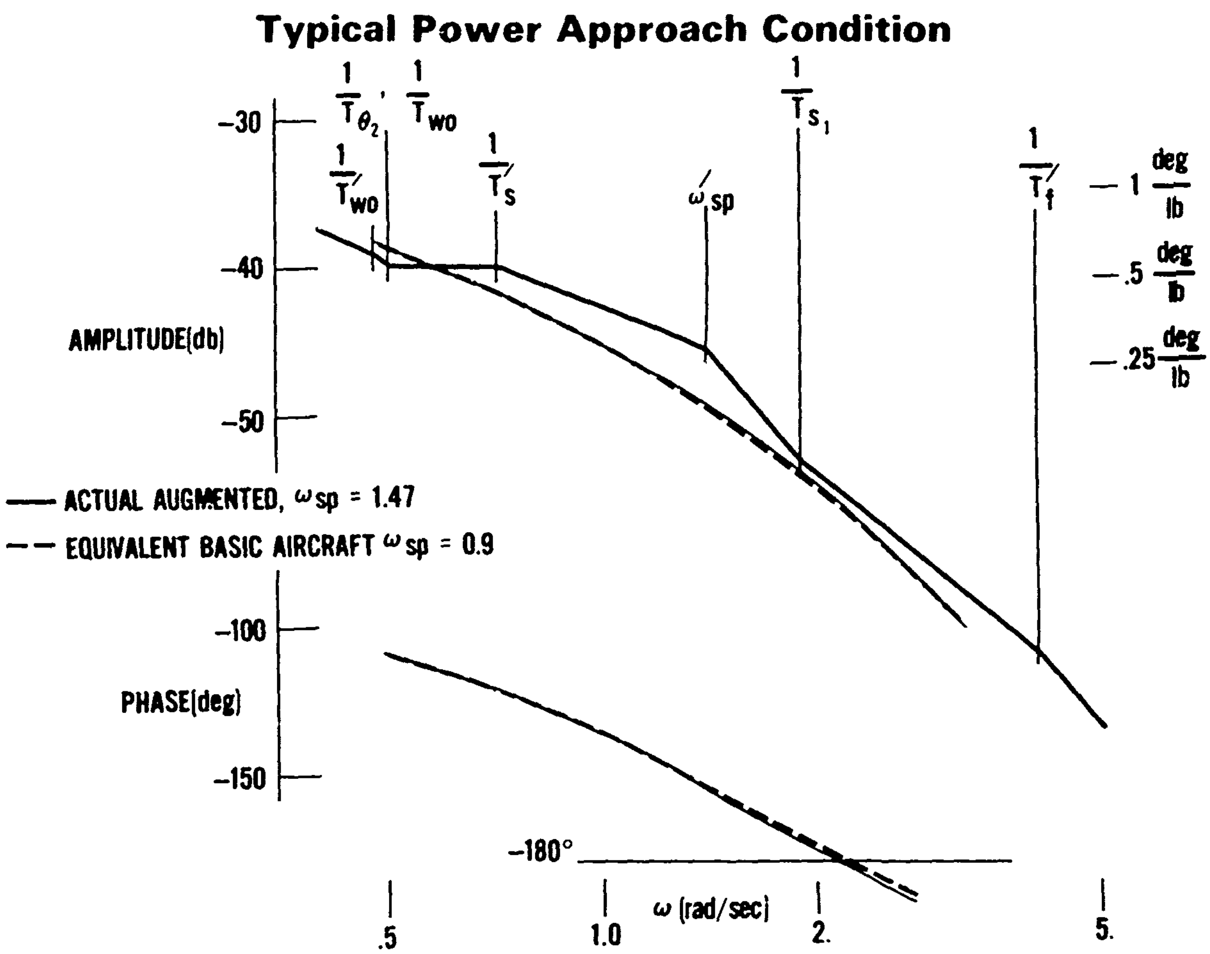


Figure 21 - F-14 Pitch Attitude to Stick Force Transfer Function

-UP AND AWAY FAILURES

- ELEVATOR TO STICK LINKAGE
- STICK DAMPER
- TRIM
- SINGLE ENGINE
- SINGLE ENGINE & HYDRAULICS
- FUEL SYSTEM COMPONENTS
- THROTTLE LINKAGE
- TWO ENGINE
- HYDRAULICS

- LANDING FAILURES

- SINGLE ENGINE
- FLAPS RETRACTED
- PRIMARY CONTROL
- ELEVATOR LINKAGE
- AILERON LINKAGE
- RUDDEP LINKAGE
- STICK DAMPER
- TRIM
- AIRSPEED INDICATOR
- SINGLE ENGINE & FLAPS UP
- ONE ENGINE AND HYDRAULICS

Figure 22 - S-3 First Flight Training

OPEN DISCUSSION

M.Hacklinger, Germany: Why has the same ω_{nsp} value been specified for the F-14 with and without SAS? This is surprising for an aircraft with a fairly sophisticated flight control system.

R.C.A'Harrah, USA: We felt there was a need for a minimum short period stiffness (stability) of 0.8 rad/sec for satisfactory carrier approach. In view of the considerable controversy on the definition of the SAS-on stability level of another Navy aircraft at the time the F-14 specification was being put together, the decision was to require that the basic (SAS-off) aircraft exhibit the needed stability level. In retrospect, the approach taken was most certainly appropriate.

R.Deque, France: Our experience on simulators including the FSAA at NASA Ames Research Center shows that precise study of the final landing is outside the capability of existing simulators. Your presentation seems to show that you have had different experiences in simulation of carrier landings. Is that correct? Have you explanations for this difference?

R.C.A'Harrah, USA: I quite agree that terminal condition data from landing simulations cannot be considered quantitatively precise. The investigations to which I referred used the results to qualitatively evaluate the influence of configuration refinements but not to determine or revise or be compared with quantitative design criteria.

The difference in the relative level of precision between field landings and carrier landings may be explained by the absence of the characteristically imprecise flare maneuver for the carrier landing.

R.Thorne, UK: Did you use typical service pilots in your evaluations at any stage, or was all the work done with test pilots?

R.C.A'Harrah, USA: All of the evaluations were performed by Contractor and Navy test pilots.

W.Bihrle, Jr, USA: I would like to give a little historical background to the F-14 carrier approach characteristics that Mr A'Harrah has referred to today. The concept of using $\omega_{nsp}^2/n_{z\alpha}$ as a criterion parameter for precision control tasks (such as carrier landings) was conceived almost 18 years ago. A short time before the F-14 proposal, the Navy gave us the opportunity to experimentally verify the validity of the parameter and to develop the numbers for the boundaries that are now in the MIL-F-8785 B Spec. These are the boundaries that many of you have been referring to these last two days. These boundaries, by the way, were developed using an unsophisticated three-degree-of-freedom moving base simulator located at Grumman. The point I wish to make, however, is that Grumman was quite aware of what was required to make a good carrier landing airplane before the award of the F-14 contract,

Now, someone has just raised the question in regard to why the Navy specified the same value for the minimum acceptable ω_{nsp} for both the SAS on and off configurations. Although I cannot speak for the Navy I might justify their decision on past experience within the industry. You see, the criterion parameter of $\omega_{nsp}^2/n_{z\alpha}$ uses the coefficient of the characteristic two degree of freedom equation as a convenient means for actually describing a specific amplitude and time relationship between the anticipatory cue of angular acceleration and the desired steady state aircraft response in load factor.

Black boxes have attempted to duplicate the specified open loop characteristic of ω_{nsp} but in doing so have ended up, in many instances, with a very high order system whose actual output in no way duplicates the desired $\dot{q}$ and $n_{z\alpha}$ amplitudes and time relationships. Use of adaptive autopilots has resulted, therefore, in no noticeable change to the pilot relative to problems associated with flying a sluggish (high inertia or low static margin) configuration. Better written explanations are available for the points I have tried to make.

THE ROLE OF FREE-FLIGHT MODELS IN AIRCRAFT RESEARCH AND DEVELOPMENT

by

R. Fail

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SUMMARY

The special features of free-flight models are discussed. Two examples are given of tests in the field of flight mechanics. Preparations are described for a programme of tests which is about to start at RAE to investigate the low-speed stall and post-stall dynamics of aircraft. Attention is concentrated on the planning of the tests and the instrumentation and control systems in the model. Some details are given of the model handling and retrieval systems.

1 INTRODUCTION

In this paper, discussion is restricted to models which are flown in free air (i.e. not in wind tunnels) and which have no crew on board. Such models have a number of special features, of which perhaps the most important are complete dynamic freedom and the absence of wind tunnel support and wall interference. The scale of the models or the Reynolds number of the tests can be as large as is needed in any particular case and, even when the size approaches full scale, the models will still be much cheaper than piloted aircraft because the structural integrity and systems reliability can be less. Potentially hazardous manoeuvres can be investigated because, again, crew safety need not be taken into account.

Compared with wind tunnel testing, however, free-flight model testing at a reasonably large scale is expensive and time-consuming. As a result of these considerations, free-flight models have been used, and presumably will continue to be used, whenever their special features are needed sufficiently to justify their cost. For example, a great deal of transonic testing had to be done in free flight until transonic wind tunnels were developed. Low speed free-flight models have been used extensively for stability investigations[1,2].

Section 2 of the present paper describes briefly two examples of free-flight model tests in the field of flight mechanics. Section 3 describes preparations being made at RAE for a programme of tests to investigate the low-speed stall and post-stall dynamics of aircraft. Section 4 gives some details of the handling equipment and retrieval systems.

2 EXAMPLES OF FREE-FLIGHT MODEL TESTS

The first example concerns a slender-wing research aircraft (HP 115)[3]. Before this aircraft flew (in the early 60's) it was known, or strongly suspected, that the Dutch roll mode would become divergent at high angles of incidence, and assurance was being sought that the aircraft would be satisfactorily controllable at least up to the conditions needed for the first flights. At the time, the RAE wind tunnel rigs for measuring derivatives[4] were still in the development stage, and there was no guarantee that a valid mathematical model of the slender aircraft could be formulated. It was therefore decided to make free-flight tests of an unpowered $\frac{1}{4}$ scale dynamic model. The model was launched from a helicopter and, during the flight, the angle of incidence was increased slowly and continuously by a simple mechanism which gradually increased the elevator angle. At intervals, the Dutch roll was excited by firing small rockets and the resulting oscillations were recorded. The results[5], shown in Fig.1, provided the required assurance, and later were found to be in excellent agreement with corresponding measurements on the full-scale aircraft.

Another example of the use of free-flight models is provided by RAE measurements of the oscillatory stability derivatives of a slender wing. At the time this investigation started, the only feasible way of obtaining the required information was by the free-flight model technique, and so a series of tests on rocket-boosted models was made. Results were obtained for derivatives due to incidence and sideslip and for most of the important derivatives due to angular velocity[6]. Later, a wind tunnel rig[4] became available and tests on models of the same shape continued both in wind tunnels[7] and in free flight[8], principally to investigate tunnel wall and support interference. The work was further stimulated by the adoption of the model as AGARD Model G[9]. Results for some of the lateral derivatives at near-zero lift over a wide range of speed are shown in Fig.2. The measurements in two wind tunnels and in free flight are in good agreement.

The Reynolds numbers of the free-flight tests are considerably higher than those of the wind tunnel tests but the effects of Reynolds number would be expected to be small on this model which is slender with sharp edges. This seems to be confirmed by the experimental data. No corrections for tunnel interference have been applied since these effects would also be expected to be small in this case. In other circumstances, however, large interference effects can occur; a full account of the interference effects on dynamic measurements has been given by Garner in Ref.10. (See also the first four references therein.)

3 CURRENT FREE-FLIGHT MODEL TESTING AT RAE

The tests being planned at present form part of an investigation of the stall and post-stall dynamics of aircraft at low speeds and altitudes. The reason for this investigation is, basically, a lack of confidence in calculations or simulations of the behaviour. It is necessary to decide how best to model the aerodynamic characteristics, and then to validate the modelling. (A similar investigation, with the same objectives, is being made by NASA[11].) Free-flight tests of dynamically scaled models are an essential and major part of the investigation because they are the only means of providing in a totally realistic manner the required dynamic environment.

If the tests are made at constant (in this case, low) altitude and the model is dynamically scaled (i.e. the Froude number is constant) it can be shown[2] that the Reynolds number is proportional to $(\text{scale})^{3/2}$ and the Mach number to $(\text{scale})^{1/2}$. Testing at representative Reynolds numbers may therefore make demands on model size and speed which are difficult to meet, but this is of paramount importance only if it is required to interpret the model results directly in terms of full scale behaviour. This is desirable but by no means essential to the primary objective of validating mathematical models of aircraft behaviour.

For the RAE tests a ¼ scale model of a fighter type aircraft has been chosen. The Reynolds number is thus 1/8 of the full scale value, which is about the same as will be obtained in the supporting tests in an atmospheric low speed wind tunnel.

As already mentioned, free-flight model testing is relatively expensive and time consuming; it is therefore important to obtain as much useful information as possible from each flight. The planning of the tests is considered to be the most important factor in achieving this, and the proposed scheme is illustrated in Fig.3. We start at the top of the diagram, with the best available aerodynamic data. The dynamic derivatives will be measured on the rig[4] already mentioned in section 2. Static data up to high incidences and some large amplitude dynamic data will be obtained on a special rig which is not yet operational. Some necessary data will, no doubt, have to be estimated. These aerodynamic data, together with measured values of the model inertias[12] and an assumed mathematical model will be used to calculate[13] the model behaviour for a wide range of initial conditions, control inputs etc., and thus provide a 'framework' within which the free-flight tests can be properly planned and the results quickly assessed. Finally, a detailed comparison of the calculations and experiments will be made. The diagram shows the main feedback loops by means of which it is hoped to establish satisfactory aerodynamic data and appropriate mathematical models for future use.

Still with the object of obtaining the maximum amount of useful information, the models will be radio-controlled and comprehensively instrumented. A diagram of the instrumentation and control systems is shown in Fig.4. The main instruments and quantities measured are listed on the left of the diagram; all of these signals will be recorded on magnetic tape using the telemetry and data recording systems already available on the flight-test range. This will facilitate computer analysis of the data.

The model control surfaces will be operated by electrical servo systems designed to have a rapid response (i.e. comparable with the real aircraft systems). It is considered unlikely, however, that it will be possible to 'fly' the model, in the usual sense, by radio control from the ground since the 'pilot' will have no motion cues and the model will be practically out of visual range. It is planned, therefore, to use on-board stabilising systems. Such systems can easily be provided, flexibly, and with negligible weight penalties by connecting appropriate instrument signals to the control-surface servos. The diagram shows some of the interconnections which might be made, for example, when the main interest is in the longitudinal motion. The rudder is used only to maintain zero sideslip by means of on-board systems. The ailerons normally maintain zero bank angle, but a radio signal can bank, and hence turn the model in case this is required by range limitations. The elevator, in this example, is directly controlled from the ground. It is worth pointing out that even though the model is laterally stabilised, information on the overall lateral characteristics may be obtained from the control movements which occur. In other cases, e.g. spin recovery, it will be necessary or desirable to switch off the on-board stabilising systems at an appropriate time and to operate the controls directly from the ground.

The behaviour of the model will be monitored by displaying a suitable selection of the telemetered instrument signals. This display, together with the radio-control transmitter, is installed in a special vehicle (Fig.5). In most cases, the main advantage of radio control is that the test manoeuvre can be initiated when the model motion is free from disturbances due to launch or to gusts, but it will sometimes be desirable to make control inputs at specific stages in a manoeuvre.

4 HANDLING AND RETRIEVAL SYSTEMS

The tests described in the previous section will be made on unpowered models, launched from a helicopter at a height of about 1500 m above a test range and recovered by parachute. Since the models are fairly large (about 3.5 m in length and span) and of considerable mass (nearly 200 kg) some consideration has been given to handling and retrieval systems. Most of these have been tried with a 'mockup' model of about the correct overall dimensions and mass. A handling trolley is shown in Fig.6. This is designed to pick up the model from the ground by means of hydraulic jacks, to provide local manoeuvrability, and to enable the model (with wings removed) to be winched into a caravan for transport (Fig.7). The helicopter lifts the model from the trolley with the jacks in their highest position. When the weight of the model is removed the jacks automatically retract rapidly to minimise the possibility of damage to the model during lift-off.

The model is towed to the launch point on a cable about 50 m long. To improve the stability of the towed model[14] the release device on the end of the cable is ballasted to provide an additional mass of about 100 kg and a drogue parachute is attached to the tail of the model. The drogue is jettisoned automatically when the model is released. After the test period, which lasts about 1 minute, the recovery sequence is initiated by a radio signal, or by a barometric device operating automatically at a height of about 600 m.

The main parachute system (Fig.8) has a mass of about 11 kg and gives a rate of descent of about 7 m/s. The ground impact speed is further reduced by a set of cylindrical air bag shock absorbers[15] which are deployed from the lower surface of the model during the parachute descent (Fig.9). The shock on impact cannot be eliminated by this means but the ground impact speed is roughly halved. The mass of the air bag installation is only about 6 kg and the peak deceleration, during the compression of the bags, is about 12 g.

5 CONCLUDING REMARKS

It is hoped that this paper has shown that unpowered low speed free-flight models have a special role to play at the present stage in the development of the science of flight mechanics. Since flying has not yet started in the current RAE programme, it has been possible to describe only some aspects of the work. Construction of the models is, however, well advanced and most of the systems have been tested. The first flights should take place about the middle of this year.

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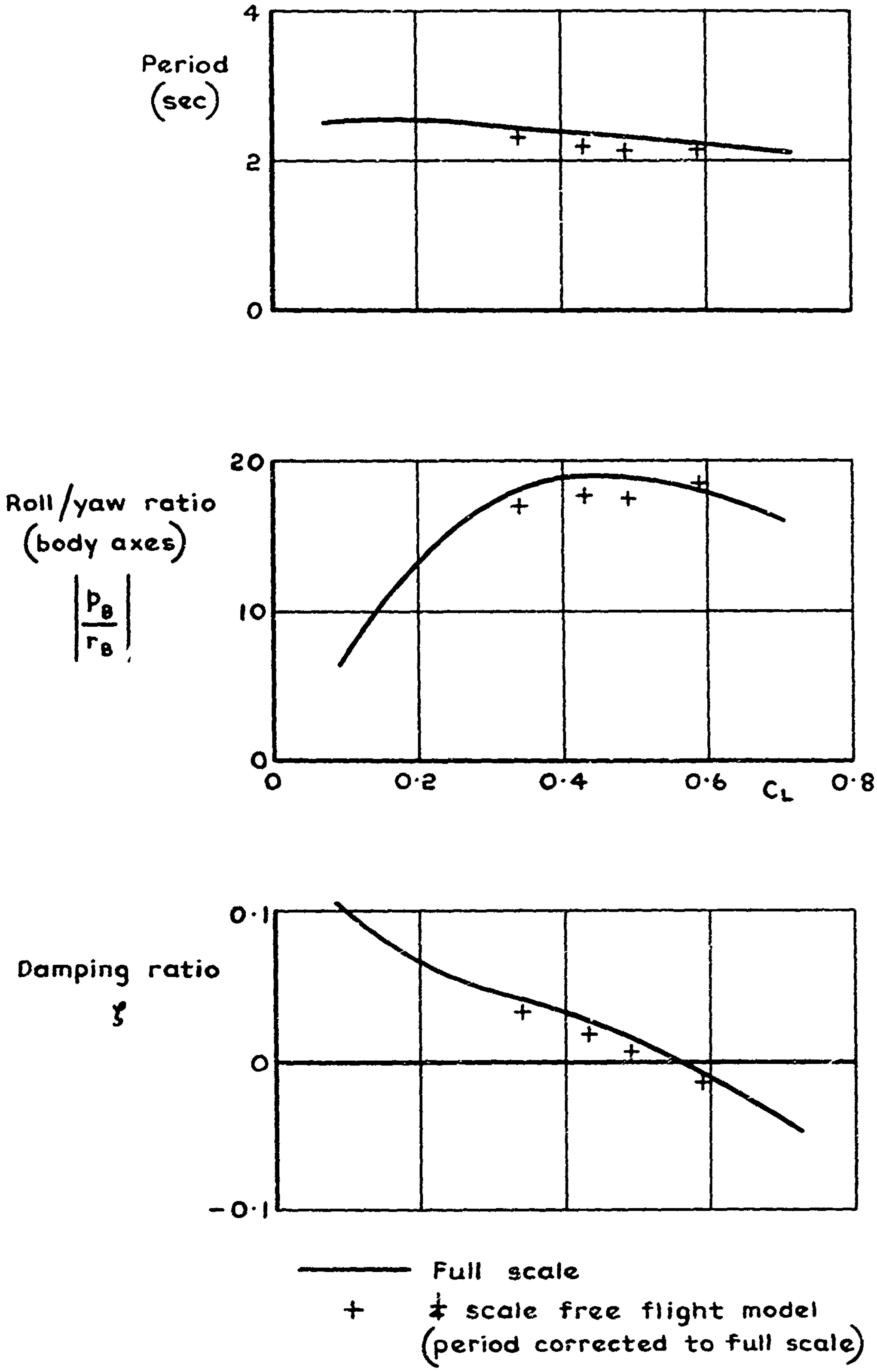


Fig.1 HP115. Dutch roll characteristics

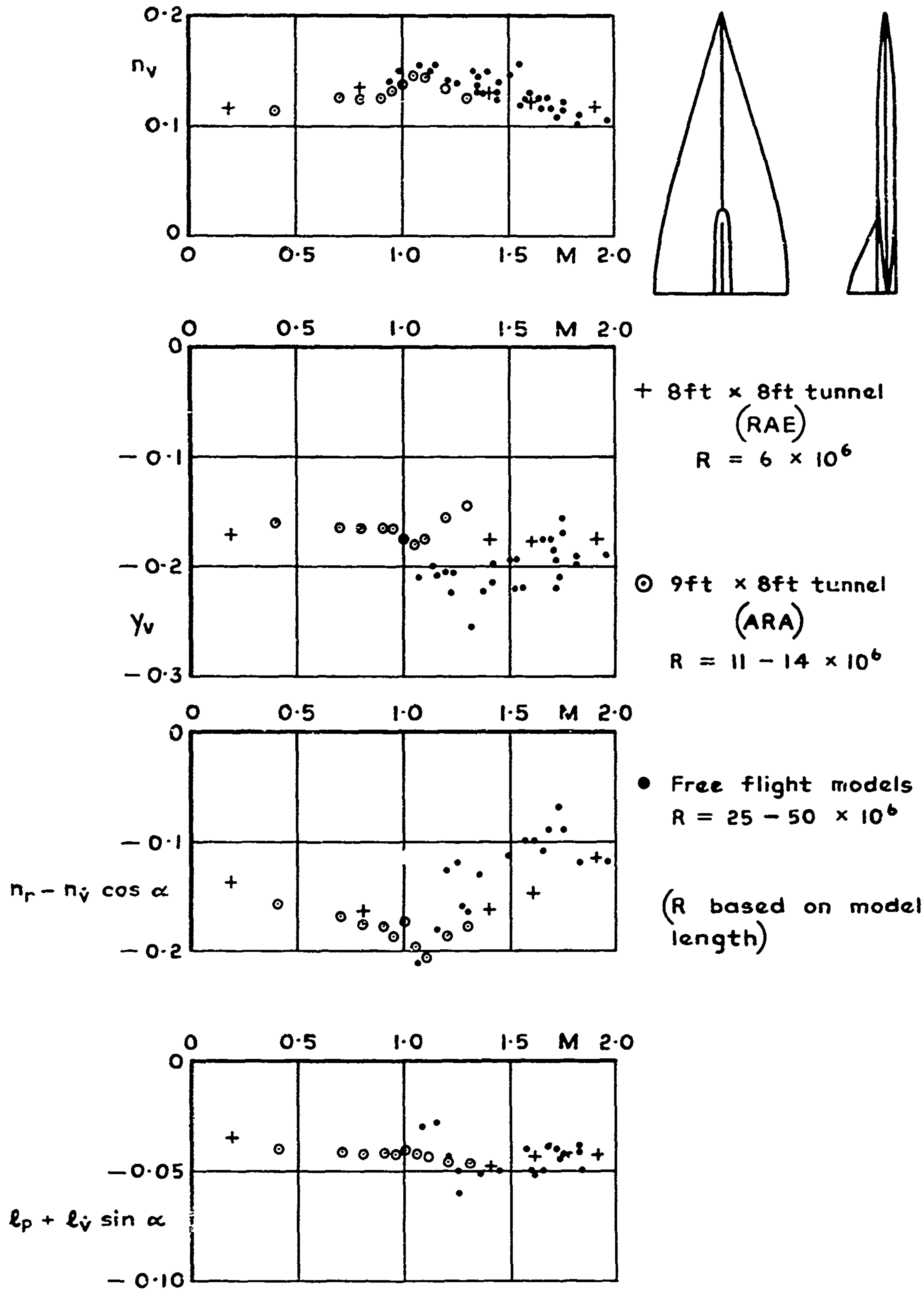


Fig.2 AGARD Model G. Lateral stability characteristics

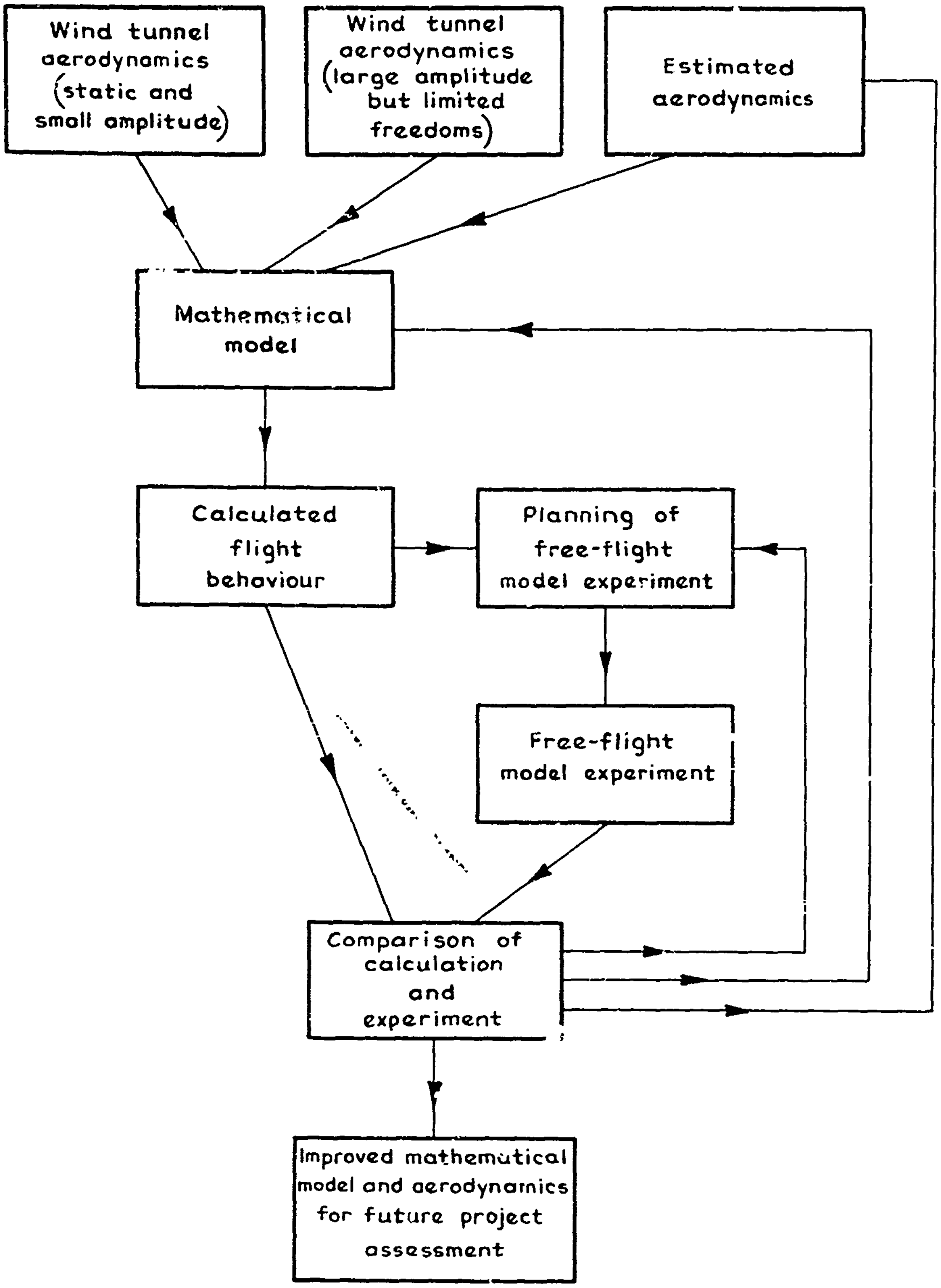


Fig.3 Planning of free-flight experiments

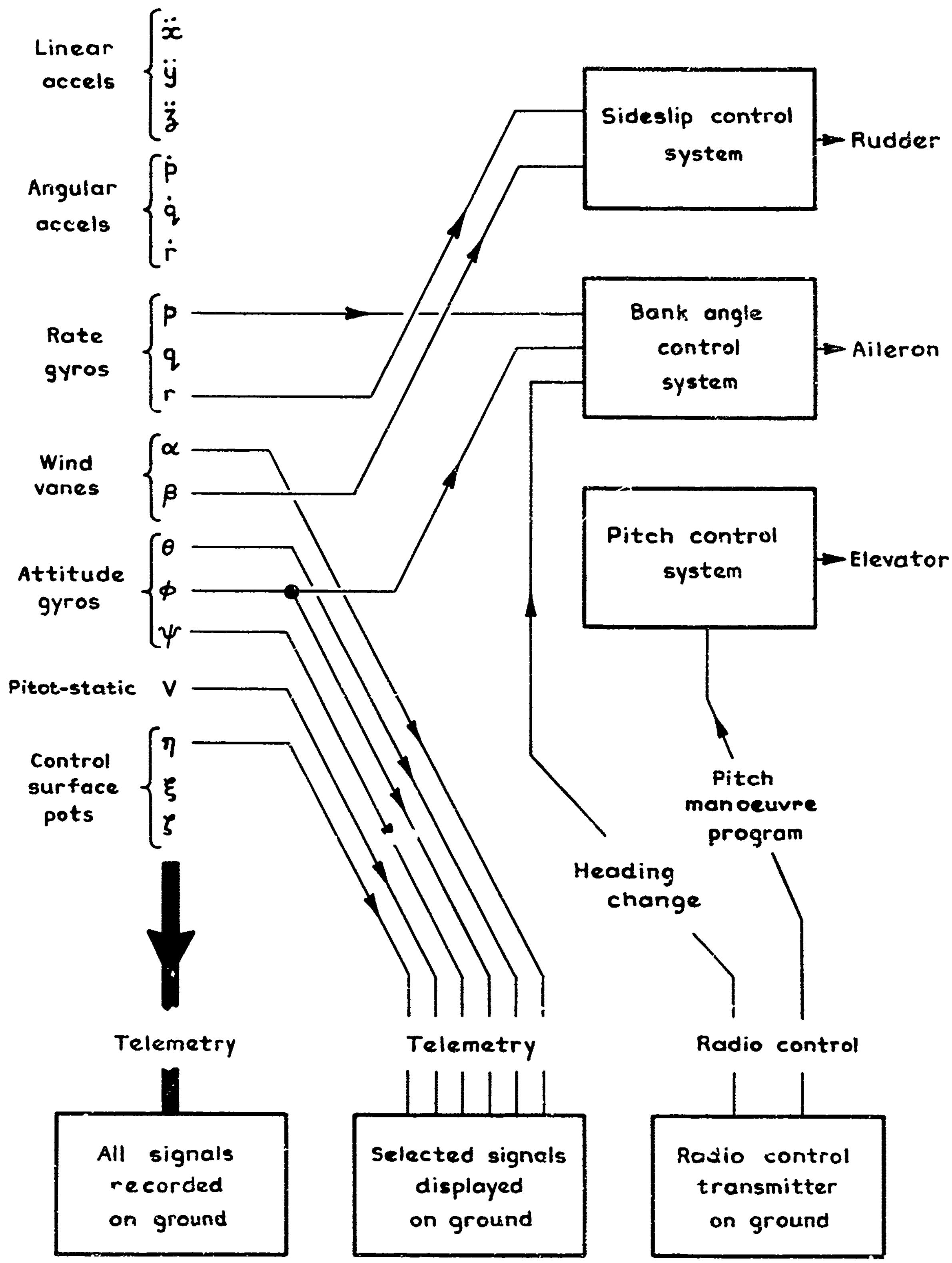


Fig./ Model instrumentation and control systems, 'ıcluding some possible ınterconnections

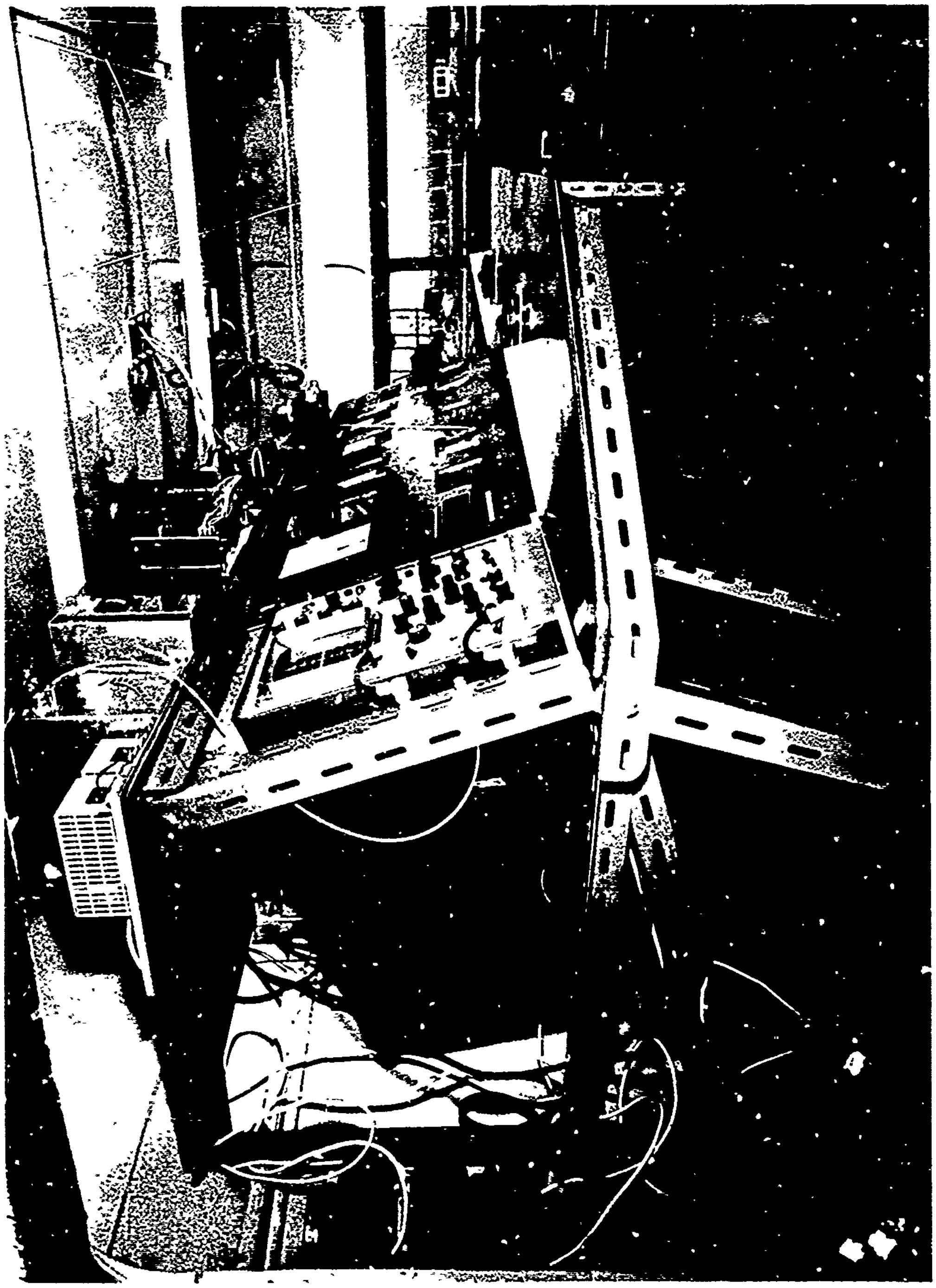
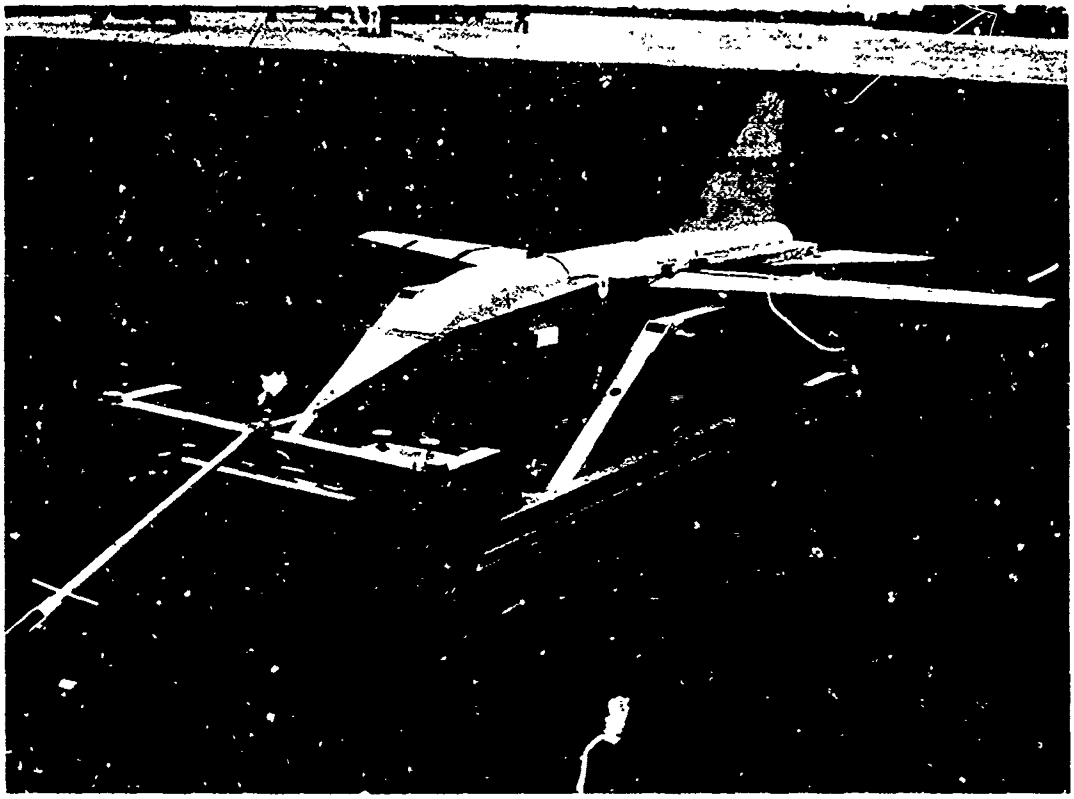


Fig.5 Telemetry display and radio control panels



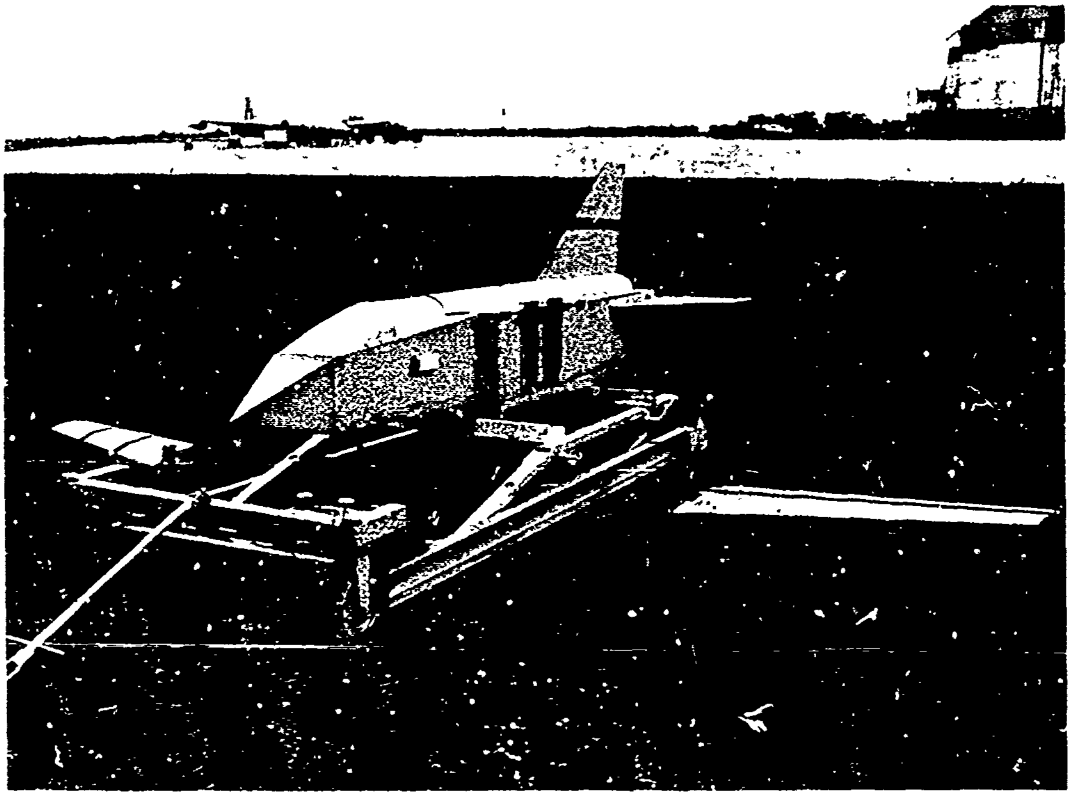


Fig.6 Model handling trolley

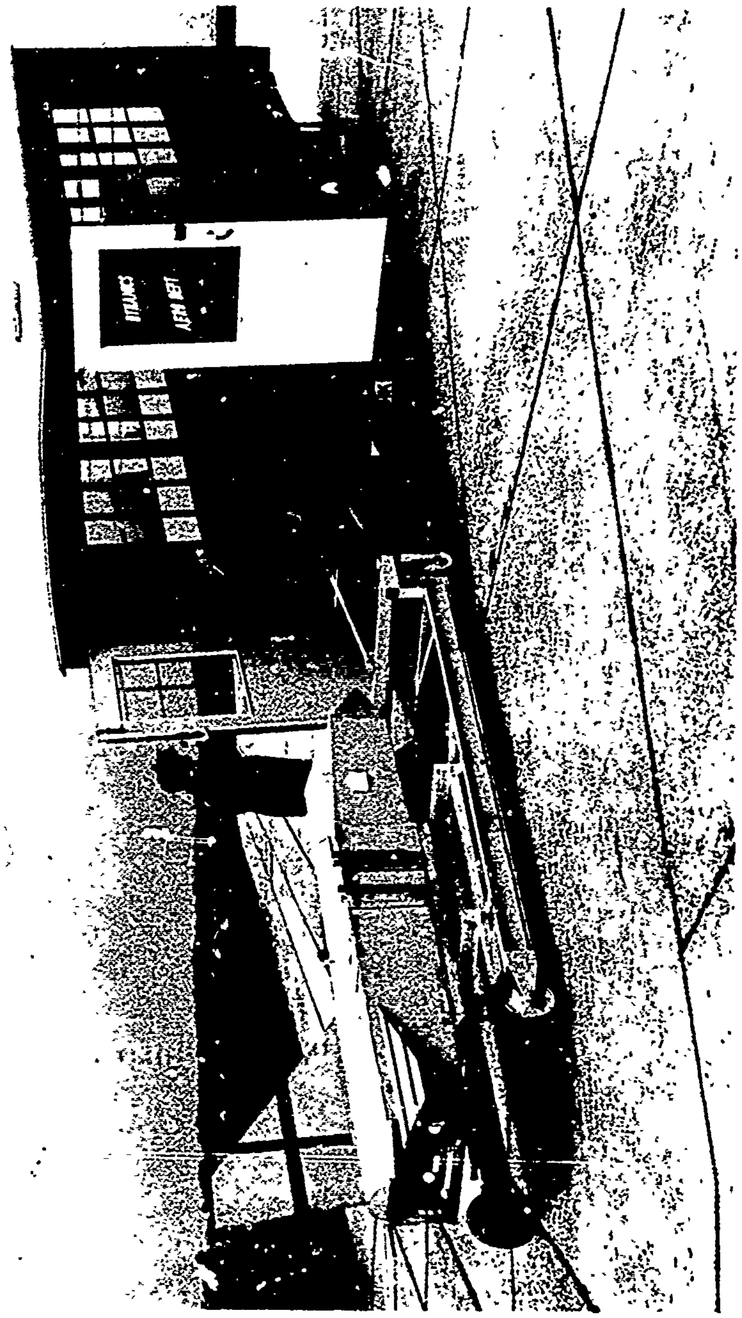


Fig.7 Model transport caravan

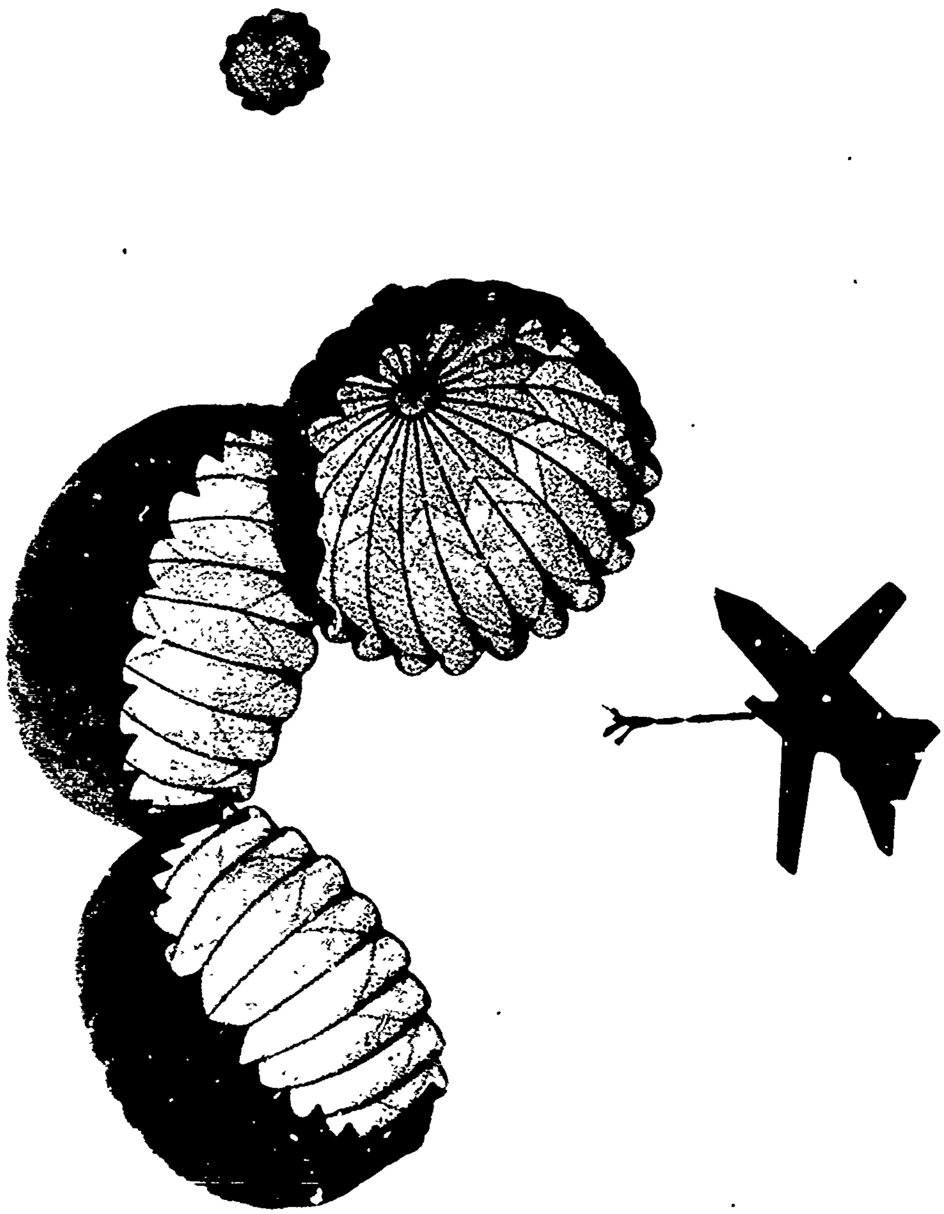


Fig.8 Main parachute system

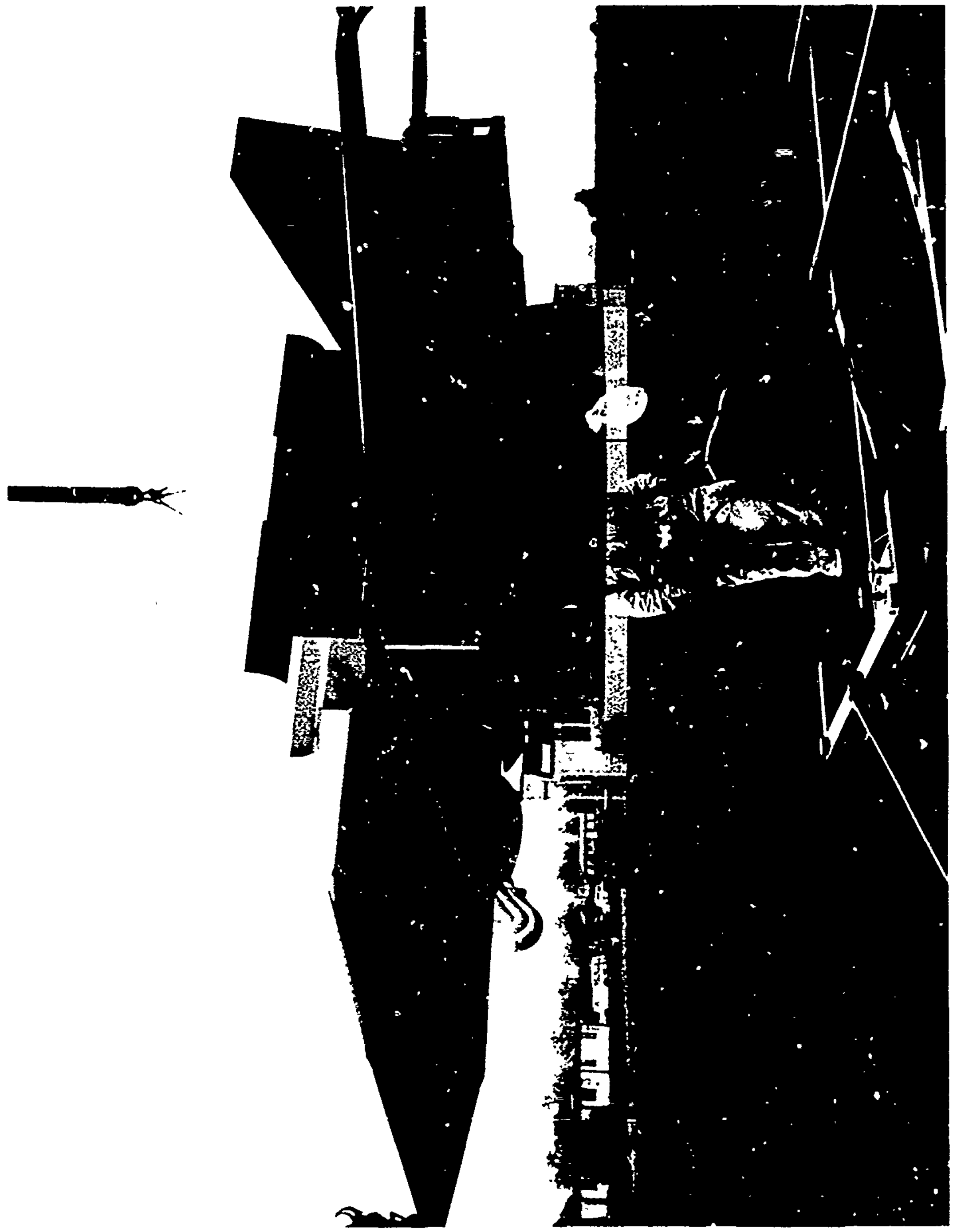


Fig.9 Shock absorbing air bag system

OPEN DISCUSSION

F.Thomas, USA: Have you considered more advanced flight control systems as might be necessary to simulate control configured vehicles which are basically unstable? This would require some computer equipment in your test van.

R.Fail, UK: We have not considered any very advanced flight control systems because we have had no such requirement, but we see no fundamental difficulty in investigating any system by this means.

X.Hafer, Germany: How do you get information of the atmospheric conditions along the flight path?

R.Fail, UK: Comprehensive information on the atmospheric conditions is available, as a service, from the test range.

T.B.Saunders, UK: It was surprising that lack of view of the model had been said to preclude direct radio control. Surely this could be accomplished with the usual quantity of telemetered data suitably displayed.

R.Fail, UK: I may have over-emphasised our opinion that it will be difficult to "fly" the model directly by radio control. Nevertheless we think it will be wise to make provisions for automatic stabilising systems at least for the early flights.

M.Hacklinger, Germany: As Mr Fail has described his programme, he has to undertake the tedious task to obtain aerodynamic derivatives from tunnel tests and program a complete mathematical model for a configuration which will not be flown full scale. Is it not possible to modify the model structure such as to resemble an actual variable geometry aircraft from which then all other data can be applied and for which, later on, correlation between free flight model tests and full scale flight tests can be achieved?

R.Fail, UK: Our main purpose is to validate the mathematical modelling; we are less interested in correlation between model and full scale, which will be affected by differences in Reynolds number and Mach number.

H.Wuennenberg, Germany: Mr Fail, as I learned from your paper, the free flight model testing method is very comfortable but too expensive and time consuming to be used as a tool to get better derivatives for a new project. It would be interesting to know the relation of the accuracy and costs of this method in comparison with dynamic wind tunnel test methods. Did you make a comparison like this on the basis of data from flight test results?

R.Fail, UK: We do not regard these free-flight model tests as an alternative to wind tunnel tests as a means of obtaining derivatives. We have therefore made no estimates of the relative accuracy and cost.

THE EFFECT OF ENGINE FAILURE AT SUPERSONIC SPEEDS ON A SLENDER AIRCRAFT - PREDICTED AND ACTUAL

by

C.S. Leyman - Chief Aerodynamicist (Concorde)
R.L. Scotland - Group Leader, Stability and Control

British Aircraft Corporation Limited
Commercial Aircraft Division

NOMENCLATURE

M	Mach Number
α	Angle of attack (degrees)
β	Angle of sideslip (degrees)
ϕ	Angle of bank (degrees)
p	Rate of roll (degrees per second)
r	Rate of yaw (degrees per second)
δ_p	Aileron angle (degrees)
δ_r	Rudder angle (degrees)
C_{n_β}	Non-dimensional derivative of yawing moment with respect to sideslip
$C_{n_{\delta_r}}$	Non-dimensional derivative of yawing moment with respect to rudder deflection
$C_{n_{EF}}$	Non-dimensional coefficient of yawing moment due to engine failure
C_{l_β}	Non-dimensional derivative of rolling moment with respect to sideslip
$C_{l_{EF}}$	Non-dimensional coefficient of rolling moment due to engine failure
Δn_g	Incremental normal acceleration in g units
N_2	Engine h.p. compressor r.p.m.
ΔC_p	Incremental pressure coefficient
C_o	Reference wing root chord

1. INTRODUCTION

At the time when the design of Concorde began, the available evidence on the problem of engine failure at supersonic speeds on a multi-engined design was sparse, and what little there was, slightly unnerving.

Stories of unexpected variations in critical derivatives and large disturbing moments due to wing flow breakdown in the presence of large intake spill flows, led to a great deal of speculation as to whether such occurrences could ever be tamed to the standard necessary for an aircraft carrying fare-paying passengers.

In consequence of this speculation, the effect of engine failures in cruise has been under study from the very start of the design.

The existence of flight test information now allows the complete cycle of design - simulation - flight test - design feedback to be described and completely allays any fears about excessive aircraft response to engine failure.

2. PREDICTION OF AIRCRAFT STABILITY DERIVATIVES

The prediction of the basic stability derivatives for the rigid aircraft was made directly from wind tunnel tests using conventional techniques. The model conformed to the design cruise shape of wing camber, twist, and dihedral; an anti-distortion allowance was incorporated into the wing shape in th ig-build stage.

The major correction for aeroelasticity in relation to the directional derivatives lies predominantly in the fin and rear fuselage twist and bending due to sideslip. It is therefore important to determine the separate contribution of the fin and rear body ensemble to the total directional stability so that the aeroelastic correction may be applied to this contribution. The complete aircraft derivatives come from wind tunnel tests using a single sting mounting corrected to true rear end geometry by means of subsidiary tests with a twin sting mounting, in which the wing and front fuselage are earthed by rigid supports, attached to the rear of the nacelles, and the rear fuselage is mounted on an internal balance (Fig. 1). These subsidiary tests also provide directly the required contribution of the true geometry rear end and fin ensemble.

The model was tested with a conventional single sting mounting and twice with a twin sting mounting; firstly with the rear end geometry of the single sting test and secondly with the true rear end shape. Simple differences of the second and third tests then gave corrections to be applied to the single sting results. Although this was done primarily to obtain drag corrections, it was possible to utilise this facility to get the lateral forces.

REAR END CORRECTION
USING TWIN STING TECHNIQUE

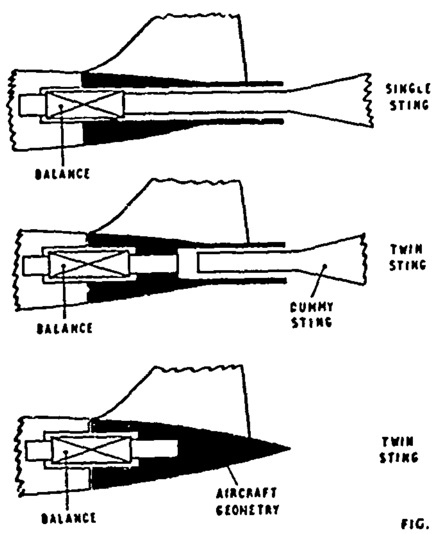


FIG. 1

The aeroelastic calculations use structural influence coefficients at about 150 points on the back end of the structure. The rigid body aerodynamic loading is taken from Wind Tunnel pressure measurements (corrected to integrate up to the measured overall forces) and the aerodynamic loading due to distortion is taken as a linear perturbation about this, calculated by Pines, Dugundji and Neuringer's box method.

The determination of the aeroelastic correction is dependent on a good knowledge of both the structural characteristics and the aerodynamic loading. A typical high Mach number aerodynamic loading distribution obtained from tunnel tests is shown in Fig. 2 and it can be seen that there is very high loading in the region of the leading edge of the fin. Although this peak loading is forward of the flexural axis of the fin itself, it is aft of the point where the rear body may be assumed to be "encastre" with the fairly rigid fuselage in way of the wing rear spar. The net effect is a loss of around 15% in fin efficiency for the cruise condition which gives rise to a loss in total directional stability of about 40% at 2.0M with the reference c.g. position of 50% root chord. A further small loss arises from the effect of forward fuselage bending. The variations of the rigid and flexible derivatives are shown in Fig. 3.

A similar treatment was made in the prediction of the rudder control power derivatives, with particular emphasis also on the hinge moment coefficients, so that the limitations of rudder jack power might be determined.

The aileron control derivatives received separate treatment with corrections, of course, for wing flexibility, but the significant loads induced on the fin by the inboard control were corrected for fin flexibility effects.

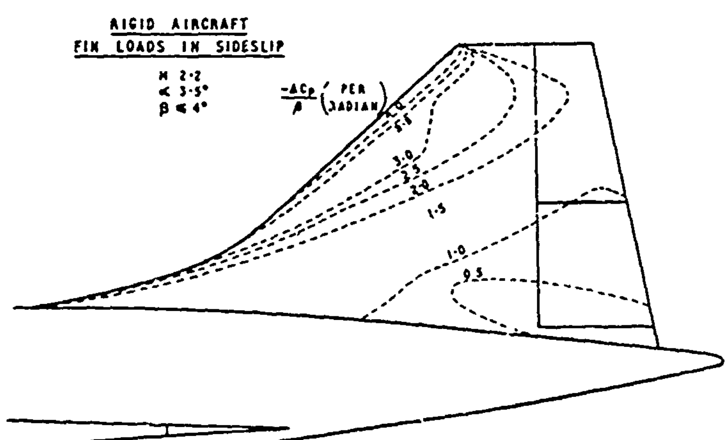


FIG. 2

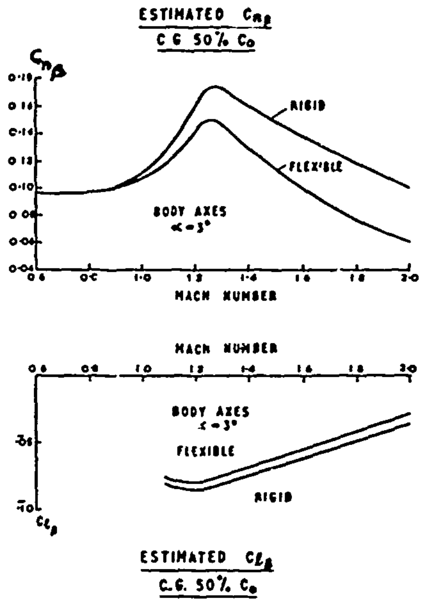


FIG 3

3. PREDICTION OF FORCES AND MOMENTS DUE TO ENGINE FAILURE

The powerplant of Concorde is highly integrated into the overall airframe design. A detailed description of the powerplant design is outside the scope of this paper, but, for the purposes of understanding what follows, some brief description is necessary.

Fig. 4 shows the principal features of powerplant layout for the prototype aircraft.

The important features, so far as this paper is concerned, are the moveable ramps on the upper surface and the dump door mounted in the floor of the intake.

The function of both these items is to regulate the intake spill flow to match the engine demand.

POWERPLANT LAYOUT

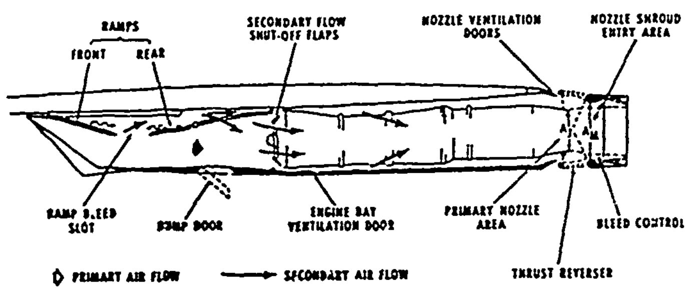


FIG 4

In the case of an engine failure, the intake control system signals both ramp and dump to open at the maximum available rate. The ramp angle above a certain limit is controlled by the dump door opening, so that to some extent the ramp rate is controlled by the rate of dump door movement available. At cruise Mach number, the aerodynamic moment on the dump door is in a favourable (opening) sense right up to the maximum angle required for engine failure. This aids the provisioning of a rapid response system.

In order to economise on actuator weight and size, the design rates for ramp and dump are chosen so as just to avoid intake buzz in the case of engine failure. This means that, during the transient engine rundown time, there is a significant amount of spillage from the front of the intake above that provided for by the increased ramp angle.

The effect of such forespillage was assessed, so far as was possible, by 1/45th scale wind tunnel tests at NLR. Unfortunately, the extent to which such testing can be combined with a correct simulation of ramp angle is limited.

This is due to the lack of secondary flow provisioning on the 1/45th scale model. Without such a bleed, the model intake enters into buzz more readily than does the aircraft intake. Not only are buzz conditions unrepresentative of the real engine failure situation, but also meaningful force measurements are impossible under such conditions. This latter restriction also precludes prior assessment of the effect of engine surge.

The effects of dump door opening were also assessed on the 1/45th scale model. In this case it had been predicted that the lift component of the spill flow momentum would not give rise to significant rolling moments. It was theorised that there would be a compensating suction on the underside of the nacelle immediately aft of the door opening, due to the exiting flow being returned to freestream direction. This proved to be the case and the lifts and drags associated with dump spill are largely those arising from forces on the door itself with little interference.

Besides this 'quasi-static' information, it was necessary to know the engine rundown characteristics. As there had been several Marks of Olympus engine already flown, some assessment could be made right away.

When more information became available, the engine response to a sudden cessation of fuel supply was estimated from an analogue simulation of the engine/intake/intake control system and confirmed by tests in an Altitude Test facility.

Finally, this information was put together to obtain estimates of the disturbing moments arising from cruise engine failure. Fig. 5 shows the calculated variation of C_n and C_l with time for an outer failure at MCP, 2.0M, 50,000 ft.

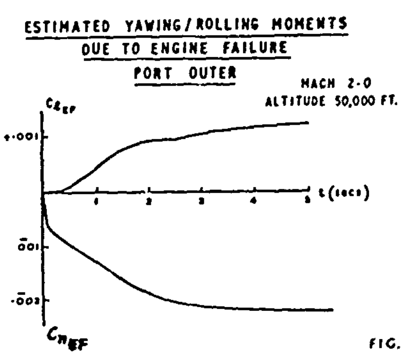


FIG. 5

Points to note are the very sudden initial 'kick' in C_n due to flame out, followed by a more gradual increase as the engine mass flow dies away. The rolling moment (which is in the sense of lifting the dead engine) increases steadily as the engine mass flow diminishes. There is a just perceptible discontinuity at around 2.5 secs., when a change in the prime method of spill (from ramp/dump to dump only) occurs.

4. PRE-FLIGHT SIMULATION EXPERIENCE

Early analogue and fixed base flight simulator results showed that both single and double engine failures appeared to be quite controllable, but there was, however, concern about two particular aspects.

Due to the assumed rolling moment from the dump door of the failed engine and the relatively low roll inertia compared to the yaw inertia, the wing containing the dead engine initially rose until the rolling moment due to sideslip became predominant. There was concern that, with the dead engine rising, there might be a tendency to apply aileron to oppose this initial movement and add to the eventual rolling due to sideslip, or even perhaps to apply wrong rudder.

There was also concern that, should the dump door lift effect not be realised, or should the value of $C_{n\beta}$ be lower than that assumed, the opposing rolling moments could become unbalanced and a rapid roll response result.

Later moving base flight simulator tests, using better-founded derivatives and engine failure effects, in fact showed similar results to the early tests, with no piloting problems. Typical results are shown in Fig. 6 for a single and simultaneous double engine failure. It is to be noted that the values of sideslip achieved were about 2.8° and 5.1° respectively, and that a somewhat oscillatory pilot input gave rise to 20° bank 10 secs. after failure.

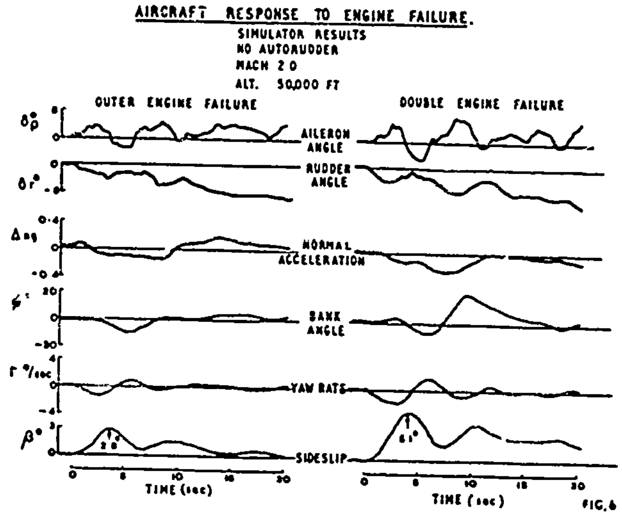


FIG. 6

However, following this flying, results became available from full scale intake/engine tests which showed that there must be a significant limitation of the sideslip to avoid surge. The limitations are shown in Fig. 7. It is seen that, with the basic intake control system, at M = 2 only about 2° of sideslip is allowable. By re-scheduling the intake control and engine control systems for sideslip angles in excess of 1.25°, the intake/engine compatibility in sideslip is improved and the limit is raised to about 3.5° at M = 2.0.

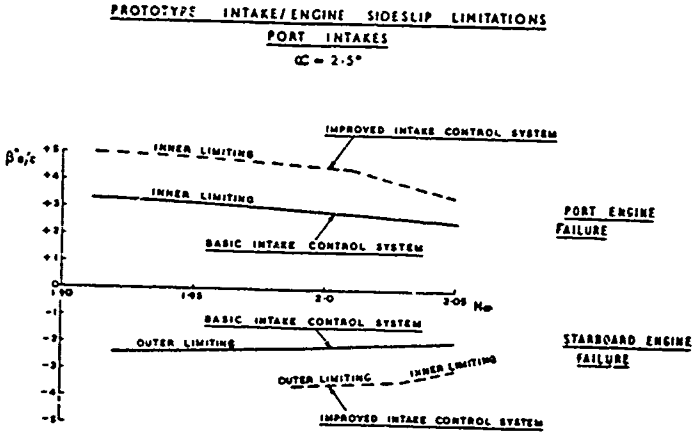


FIG. 7

It was decided that an auto-rudder system was also required, which would apply rudder early, following engine failure, and prevent sideslip building up to large values. The system developed for use in flight testing the prototype was one which sensed h.p. compressor pressure from all engines, and, on detection of an asymmetry across the aircraft above chosen values of both level and rate of change of pressure, triggered a demand for rudder through the autostabiliser system. The demand was for 4° of rudder per pair of engines, with a rise time of sec. and a washout of 40 secs. Effectively this gives compensating rudder applications of about 3° in the case of a single engine failure and about 6° for a simultaneous double failure. In addition to rudder, the system also triggers a change in the engine control laws which gives a reduction in r.p.m. on all the live engines, thus reducing the total yawing moment.

The system was effective in reducing the sideslip obtained, on the flight simulator, to about 1.8° and 2.8° respectively for single and double engine failures, thus meeting the limitations imposed. A typical response to a double engine failure is shown in Fig. 8.

This shows that with the sideslip excursion halved by the use of auto-rudder, the rolling motion is dominated by the rolling moment from the dead engine dump doors, so that the aircraft rolls away from the dead engines.

The response from about 15 secs. onwards demonstrates a P.I.O. which occurred with the rudder jacks saturated (thus losing use of the yaw dampers). This problem was cured by modifying the aileron gearing of the inner elevons so as to reduce the yawing moment due to aileron.

SIMULATOR RESPONSE
TO DOUBLE ENGINE FAILURE

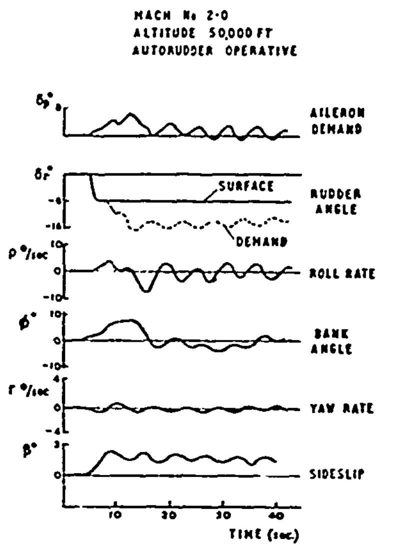


FIG 8

MATCHING OF FLIGHT DATA
RUDDER PULSE

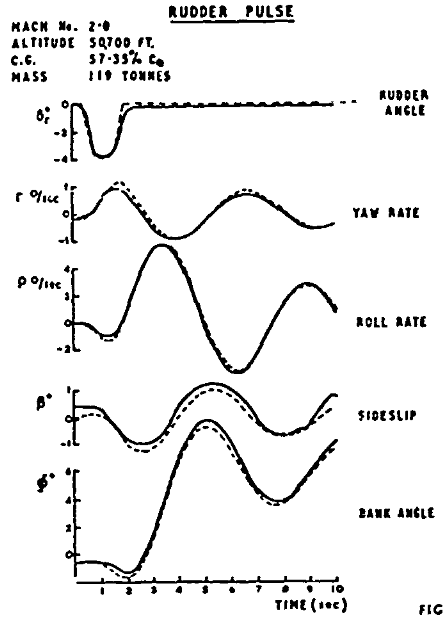


FIG 9

5. FLIGHT TEST RESULTS

In view of the somewhat marginal acceptability of the simulator results, the prototype flight envelope was extended to 2.0M very cautiously, with extensive checks both on the aerodynamic derivatives and the aircraft response to single and double engine failures.

The primary method used for checking the aerodynamic derivatives was an analogue computer 'matching' process comparing actual flight responses to control pulses by overlaying them on computed responses to the same control input. By use of the high speed repetitive operation facility on the computer, good quality matches such as are shown on Fig. 9 could be obtained very quickly.

Fig. 10 shows the results of this work comparing the 'matched' values of C_{n_β}, C_{l_β} with the estimated flexible aircraft values shown in Fig. 3.

Above about 1.7M, the aircraft shows generally similar values to those predicted, although both C_{n_β} and C_{l_β} are slightly higher than the estimated value.

Between about 1.2 and 1.6M there is a significant discrepancy between estimated and matched derivatives, particularly C_{n_β}.

COMPARISON ESTIMATED / FLIGHT C_{n_β}

C.G. 50% C_o

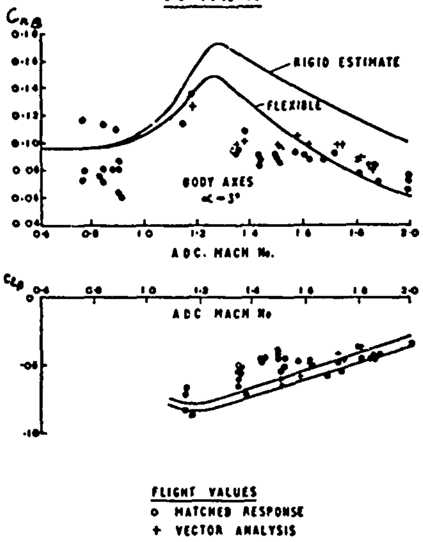


FIG. 10

The reason for this discrepancy has not yet been positively identified. The fin deflections under applied aerodynamic loads have been measured in flight by photographic techniques and the measured deflections agree quite well with those estimated for the conditions. This would seem to indicate that the reason does not lie in the aeroelastic calculations, although of course only a relatively small error in fin efficiency is needed to alter significantly the overall derivative.

At the moment, the most likely reason seems to be the inadequacy of the twin sting technique in this Mach Number range, where shock waves from the sting mountings can reflect on the fin and rear fuselage.

Luckily, the loss in C_{n_β} is unimportant in this region, as the aircraft has more than adequate directional stability, even with this loss.

Fig. 11 shows a similar comparison of rudder power $C_{n_{\delta_r}}$. The aircraft is better than predicted, the difference at 2.0M being very significant in terms of control of engine failure. Here again the reason has not yet been positively identified.

COMPARISON ESTIMATED/FLIGHT $C_{n_{\delta_r}}$

C.G. 50% C_o

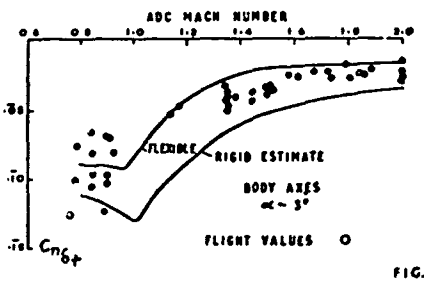


FIG. 11

The aircraft response to a deliberate double engine cut at 2.02 is presented on Fig. 12. It can be seen that the auto-rudder system applied about 6° rudder, which was gradually washed out.

The aircraft began to roll away from the dead engines almost immediately after cutting the engines, but the rolling moment was quickly brought under control by the application of 4° aileron, and the maximum recorded bank angle was only 12°.

The maximum sideslip angle for this double failure was just under 2°, which is comfortably inside the intake limitations.

AIRCRAFT RESPONSE TO DOUBLE ENGINE FAILURE.

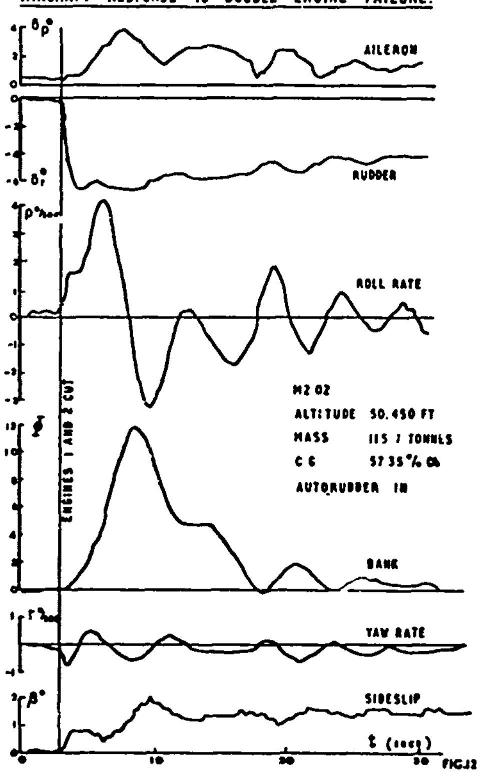


FIG.12

As a point of interest, Fig. 13 shows the response to an unpremeditated double engine surge.

AIRCRAFT RESPONSE TO DOUBLE ENGINE SURGE

LONGITUDINAL RESPONSE

LATERAL RESPONSE

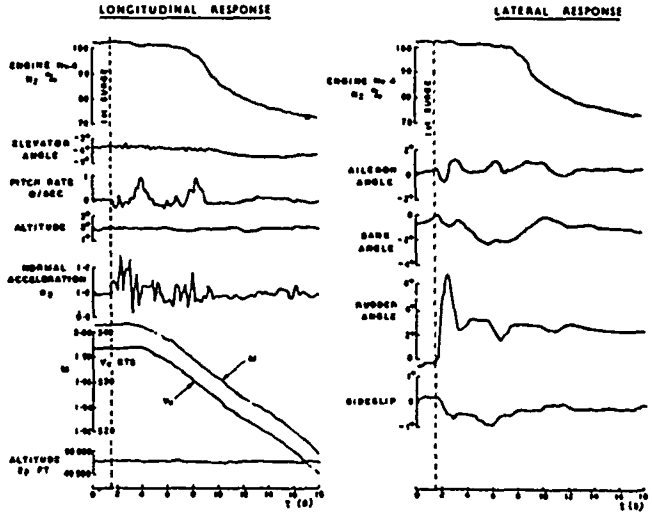


FIG.13

In this case, No. 4 engine surged at 2.0M and caused No. 3 engine to surge also. Neither engine flamed out, but No. 3 suffered damage to the intake ramps. As can be seen from the records, No. 4 was run at high RPM for some time after the first event.

The other traces show quite clearly the efficiency of the automatic systems in controlling such behaviour. The normal acceleration of 1.0g ± 0.2g and the bank angle of only 2° are very modest deviations for such a large perturbation.

In an effort to close the remaining part of the design loop, the derivatives obtained from matching control pulses have been used in conjunction with records of responses to engine failure to compute the yawing and rolling moments associated with failed engines.

Fig. 14 shows the results of this work. There is a distinct trend for the magnitude of $C_{n_{EF}}$ to be greater than estimated at low Mach number and less than estimated at high Mach number. The reverse is true for $C_{\ell_{EF}}$.

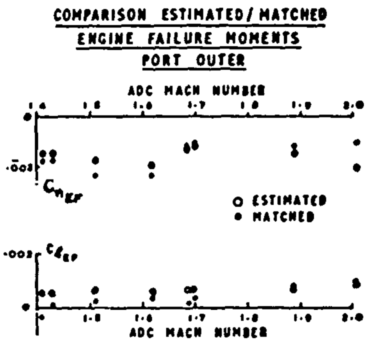


FIG. 14

As explained in Section 3, the principal perturbing forces are those associated with dump door deflection. It is a feature of the powerplant design that, in the case of engine failure, the required dump door opening increases steadily as Mach Number increases. Since there is a more or less steady drift in the estimated/matched comparison, it seems likely that the cause of the discrepancies is associated with the predicted forces on the dump door itself, but this point is yet to be resolved.

6. CONCLUSION

In summary, it may be said that the aircraft is much better behaved than was thought might be the case. In fact, its behaviour in the case of quite severe engine failure disturbances can only be described as innocuous.

This appears to be due to a combination of factors, but principally it is due to better than estimated values of C_{n_β} and $C_{n_{\delta r}}$ plus an efficient auto-rudder system. It is hoped to reduce the complexity of the auto-rudder system for the Production aircraft by using lateral accelerations rather than engine pressures for the failure sensing.

Some unexplained discrepancies remain, but have not been seriously investigated, because of the emphasis put on performance testing. However, it is not expected that the production aircraft will differ from the prototypes in any great degree, so that there is perhaps little reason for urgency in explaining the discrepancies.

ACKNOWLEDGEMENTS

The Author's wish to thank the British Aircraft Corporation (Commercial Aircraft Division) for permission to publish this paper.

OPEN DISCUSSION

W.J.G.Pinsker, UK: Mr Leyman's paper has highlighted a point which I tried to make in my own paper earlier on. This is the inadequacy in many instances of estimating or even obtaining from wind tunnel tests the aircraft derivatives with adequate accuracy. This happens again and again during the development of an aircraft, but once the associated handling problem is solved by some means or other the interest into the causes of the discrepancy disappears and we fail to learn from the problems. I would therefore like to make a plea for substained research into such problems past the point at which commercial pressures diminish. Only by this can we hope to learn to avoid similar disappointments on the next project. Can you please comment on this aspect.

C.S.Leyman, UK: Answered, that Mr Pinsker knows very exactly what is happening to them. They now have moved on to the preproduction airplane and they have their hands rather full with this airplane. So they are not expending very much energy on explaining the prototype results any further. But they should try to explain discrepancies. The worrying flight regime is the low supersonic condition and the reason behind it can be attributed to the twin-string technique in the wind tunnel which is inadequate in this Mach-number region. So, they have probably the explanations where the discrepancies are largest.

E.Obert, Netherlands: Asked, whether the consequences of discrepancies between the calculated derivatives and the derivatives obtained from wind tunnel and flight tests are checked on the load cases used in stress calculations.

C.S.Leyman, UK: Answered, that they do this. Perhaps Mr Scotland can give some further comments.

R.L.Scotland, UK: In connection with the comparison of the derivatives from wind tunnel and flight results, particularly in the transonic region, it is to be noted that the wind tunnel tests were made at $0, \pm 2^\circ, \pm 4^\circ$ and $\pm 6^\circ$ of sideslip, whereas the flight test results more from responses in which the sideslip only reached about 1°. There are non-linear effects, and in conjunction with RAE, we are investigating these by wind tunnel testing at $1/4^\circ$ increments.

The structure has been check-stressed for the loads appropriate to the flight matched values of the derivatives.

R.Deque, France: Commented, that all pilots who experienced a double engine failure at Mach number 2 were very surprised about the insignificance of the response of the aircraft.

W.E.Lamar, USA: Asked, whether one could comment on the type of redundancy they have in the systems and what would happen if one of the pressure transducers failed and the system inadvertently worked when they did not want it to work.

R.Deque, France: The prototype auto rudder system is duplicately monitored. A single failure (sensor, for example) does not affect the behaviour of the aircraft.

CALCUL DES CHARGES INDUITES PAR LA FLEXIBILITE AU COURS D'UNE MANOEUVRE QUELCONQUE

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SOMMAIRE

Si l'introduction de la souplesse avion dans les calculs aérodynamiques a fait apparaître des modifications assez sensibles des qualités de vol, c'est que les champs de pressions induites par cette souplesse étaient eux-mêmes modifiés.

Or, la justification structurale de l'avion doit se faire en avion souple. On se propose, dans cet exposé, de donner un processus de calcul des charges liées à la flexibilité et, pour illustrer ce calcul, de faire une comparaison de ces charges en avion rigide et en avion souple au travers de trois manoeuvres prises parmi celles imposées par les règlements.

INTRODUCTION

Pendant longtemps aérodynamiciens et aéroélasticiens ont suivi des voies parallèles, les uns étudiant les mouvements d'ensemble de l'avion rigide, les autres s'occupant des déformations aussi bien statiques que dynamiques. Et pourtant la frontière entre les deux problèmes est purement artificielle, puisqu'on ne peut concevoir de déformations sans modification des mouvements d'ensemble et réciproquement.

Mais, la taille, le poids, la vitesse des appareils ne cessant d'augmenter, les effets de la flexibilité ont pris une importance telle, qu'aujourd'hui il n'est plus pensable de se contenter de l'avion rigide dans la mécanique du vol.

Le comportement de l'avion souple en vol stationnaire ou sa réponse dynamique à la turbulence fait désormais partie des problèmes aéroélastiques, s'ajoutant à ceux bien connus de flottement, de divergence ou d'inversion de gouvernes. Cependant, si l'aérodynamique de l'avion est remise en cause par la souplesse, c'est bien parce que les forces entrant en jeu sont modifiées tant dans leur intensité que dans leur répartition. Et c'est ce qui explique que la justification structurale doive tenir compte de la flexibilité; tant et si bien que le calcul en avion rigide n'est désormais entrepris que comme comparaison éventuelle.

Il a paru intéressant, pour illustrer ce propos, de faire le parallèle entre l'avion rigide et l'avion souple à travers trois manoeuvres types imposées par les règlements, à savoir : la ressource équilibrée, la manoeuvre contrée de tangage et la manoeuvre de roulis.

CHOIX DE LA METHODE DE CALCUL

Avant de déterminer l'ensemble des charges qui agissent sur l'avion, il est nécessaire d'étudier la manoeuvre elle-même et, pour ce faire, il existe deux voies.

La première, la plus couramment employée, consiste à représenter l'avion par une superposition de ses modes propres englobant les modes rigides sur lesquels on applique une aérodynamique instationnaire ou quasi-stationnaire.

La seconde fait appel aux coefficients d'influence structuraux joints à une théorie aérodynamique stationnaire et c'est cette dernière méthode que nous utilisons plus volontiers bien que l'on puisse, à première vue, lui reprocher son caractère stationnaire.

Mais, aux fréquences d'excitation explorées (entre 0 et 2 hz) ce choix est parfaitement justifiable d'autant que, dans un calcul préliminaire, nous avons comparé les réponses longitudinales de l'avion obtenues à l'aide des deux méthodes. Entre 0 et 2 hz, ces réponses sont pratiquement identiques et ne commencent à diverger qu'au-dessus de cette gamme de fréquences, les déphasages entre excitation et déformation ne pouvant plus alors être négligés.

Cependant, si la méthode dite des coefficients d'influence présente par rapport à celle des modes propres l'avantage de la rapidité et de la simplicité, elle n'en conserve pas moins un certain nombre d'inconvénients tels que la difficulté de se reboucler soit sur le facteur de charge, soit sur les moments de charnière ou l'impossibilité de tenir compte des non linéarités auxquelles on ne peut échapper dès que l'on explore le domaine périphérique.

D'autre part, comme à notre connaissance, il n'existe aucun procédé de pré-sélection rapide des cas dimensionnants, il est absolument indispensable de posséder une programme performant pour explorer tous les cas de calcul.

Ces considérations nous ont amenés à aménager la méthode afin de satisfaire au maximum au double souci de simplicité et rapidité d'exploitation; et dans ces conditions, l'idée première est de conserver le moule mathématique de l'avion rigide en essayant d'introduire les effets aéroélastiques au niveau des coefficients aérodynamiques pour créer une catégorie de " coefficients aérodynamiques apparents ".

EXPOSE DE LA METHODE

Les outils indispensables à l'aéroélasticien peuvent se résumer en deux matrices :

- l'une, structurale liant les forces aux déformations
- l'autre, aérodynamique liant les déformations aux pressions.

La première définit complètement l'avion et contient implicitement toutes ses formes propres. Si sa détermination est parfois laborieuse au stade du projet, il est toujours possible de le mesurer dès que l'avion est construit. Il n'en est pas de même pour la seconde, pour laquelle on doit faire confiance au calcul et à la théorie des surfaces portantes, en se réservant la possibilité d'une vérification et d'un réajustement à la lumière des mesures de soufflerie sur maquette rigide adaptée, toutau moins en ce qui concerne les valeurs globales des gradients de portance et de moment de tangage.

Pour être plus explicite, la matrice aérodynamique $[A]$ relie les déformations angulaires aux points de " contrôle " $\{\alpha_c\}$ avec les coefficients $\{a_{ij}\}$ du polynôme de pression $p(\bar{\xi},\bar{\eta})$ choisi. Le passage aux forces induites sur les points d'intégration se fait par l'intermédiaire d'une matrice poids $[W]$; ce qui se traduit par :

$$\{a_{ij}\} = [A]\{\alpha_c\}$$

$$\{f_i\} = [W][A]\{\alpha_c\}$$

$$p(\bar{\xi},\bar{\eta}) = \frac{8\bar{q}L}{b_r b}\sqrt{1-\bar{\eta}^2}\Big[\sum_j a_{0j}\,\bar{\eta}^j\sqrt{\frac{1-\bar{\xi}}{1+\bar{\xi}}} + \sum_j a_{1j}\,\bar{\eta}^j\sqrt{1-\bar{\xi}^2} + \sum_j a_{2j}\,\bar{\eta}^j\,\bar{\xi}\sqrt{1-\bar{\xi}^2}$$

$$+ \sum_j a_{3j}\,\bar{\eta}^j\,\bar{\xi}^2\sqrt{1-\bar{\xi}^2} + \sum_j a_{4j}\,\bar{\eta}^j\,\bar{\xi}^3\sqrt{1-\bar{\xi}^2}\Big]$$

Dès lors, il est aisé d'exprimer les déformations angulaires $\{\alpha_c\}$ en tous les points de contrôle au cours de n'importe laquelle des manoeuvres. Ces déformations sont des fonctions des paramètres de vol comme le braquage des élevons, l'incidence, le facteur de charge ...... et si nous reportons l'expression de ces déformations dans les équations de la manoeuvre, nous faisons apparaître de nouveaux coefficients aérodynamiques.

En prenant, par exemple, le cas simple d'une ressource équilibrée dont les équations se déduisent de celles de l'avion rigide par adjonction des forces et des moments induits par la souplesse et en écrivant que les déformations angulaires provoquent des variations d'incidence et de braquage, nous obtenons de nouvelles équations semblables à celles de l'avion rigide telle l'équation des forces :

$$\frac{nP}{\bar{q}S} = [C'_{z i R} + \Delta C'_{z i s}][i + \Delta i] + [C'_{z\beta R} + \Delta C'_{z\beta s}][\beta + \Delta\beta]$$

$$+ [C_{z0R} - \Delta C_{z0s}] + \Delta C'_{zns}\,n + [C'_{z q \frac{l}{V} R} + \Delta C'_{z q \frac{l}{V} s}]\,q\frac{l}{V}$$

où chacun des coefficients est la somme du coefficient de soufflerie et de la correction apportée par la flexibilité. C'est ainsi qu'apparaissent les coefficients aérodynamiques apparents.

Donc, grâce à ce schéma, il est loisible d'étudier la manoeuvre sans rien changer au processus de résolution de l'avion rigide, sinon le mode d'interpolation des coefficients eux-mêmes. En effet, la création de ces coefficients est malgré tout assez longue et laborieuse en raison du nombre de paramètres dont ils dépendent (Mach, pression dynamique, poids, centrage) et les fichiers aérodynamiques ainsi constitués sont, par voie de conséquence, relativement volumineux, mais leur exploitation peut entièrement se mécaniser dès que l'on se fixe le mode d'interpolation en fonction des divers paramètres.

Ainsi, tous les coefficients aérodynamiques avion rigide, y compris les coefficients de moments de charnière des gouvernes, trouvent leur homologue en avion souple tant en régime symétrique qu'en régime antisymétrique.

Par conséquent, le calcul des charges auxquelles est soumis l'avion au cours d'une manoeuvre quelconque se trouve ainsi considérablement simplifié. En effet, dès que l'on connaît les paramètres de vol tenant bien sûr compte de la souplesse, il n'est besoin que de leur associer les répartitions unitaires correspondantes. Cependant, il subsiste une dernière difficulté : ces répartitions unitaires étant mesurées en soufflerie sur maquette rigide, il est nécessaire de leur ajouter les charges induites par les variations de cambrure et de vrillage; autrement dit, de créer des répartitions en accord avec les coefficients aérodynamiques apparents.

Or, il s'est avéré impossible ou du moins impensable de déterminer ces charges par un calcul préliminaire pour en dresser des catalogues de base, car pour couvrir tout le domaine de vol, il faudrait constituer un fichier d'une ampleur démesurée.

Pour éviter cet inconvénient, nous calculons ces charges à postériori, c'est-à-dire dès que la manoeuvre est étudiée et que le cas de calcul est complètement défini. Pour ce faire, il suffit de remonter aux déformations angulaires locales et grâce à la matrice aérodynamique, expliciter les coefficients du polynôme de pression que nous pouvons alors évaluer sur n'importe quel point de la surface portante .

Pour reprendre l'exemple de la ressource équilibrée, après le calcul de l'incidence i, du braquage β, de la vitesse de tangage q, on exprime :

$$\{\alpha_c\} = \bar{q}[B]^{-1}[C][W][A]\{1\}i + \bar{q}[B]^{-1}[C]\{F_\beta\}\beta + \bar{q}[B]^{-1}[C][W][A]\{\sigma_c\}q\frac{l}{V}$$

$$+ \bar{q}[B]^{-1}[C]\{F_0\} - [B]^{-1}\{\delta\alpha\} + [B]^{-1}[C]\{P\}n$$

$$[B]^{-1} = \left[[1] - \bar{q}[C][W][A]\right]^{-1}$$

$$\{\sigma_{ij}\} = [A]\{\alpha_c\}$$

En résumé, la détermination des charges agissant sur l'avion déformable, se décompose en trois phases :

1) étude de la manoeuvre à l'aide des coefficients aérodynamiques apparents et extraction des paramètres de vol pour le point de calcul retenu

2) création des charges de l'avion souple, c'est-à-dire combinaison des charges mesurées en soufflerie sur maquette rigide pondérées par les paramètres de vol

3) création des charges exclusivement induites par la variation de vrillage et de cambrure.

Grâce à ce processus, les effets de la souplesse n'empruntent à la théorie qu'une part réduite, les non linéarités s'introduisent assez aisément dans le calcul, et point essentiel, nous conservons un moule mathématique permettant une exploitation suffisamment rapide.

EXEMPLES DE CALCUL

Nous donnerons maintenant trois exemples appliqués à un avion supersonique et qui entrent tous trois dans les exigences du règlement. Ces trois types de manoeuvres ont été effectuées en régime subsonique ($M = 0,65$ $Z = 5000$ ft) et en régime supersonique ($M = 1,74$ $Z = 43\ 000$ ft) pour les deux avions rigide et souple.

1) Ressource équilibrée à $n = 2,5$

Si en subsonique l'effet de la souplesse ne joue que très peu sur les angles de braquage, par contre à $M = 1,74$, pour un facteur de charge de 2,5, il faut 5° à cabrer contre 2°4 dans le même sens à l'avion rigide. L'incidence n'est guère affectée dans les deux cas (1 à 2 dixième de degré).

Pour comparer les charges induites sur la voilure par la ressource équilibrée, il est difficile dans le cadre de cet exposé de présenter une grille de calcul; aussi, pour essayer d'imager les répartitions, avons-nous cru bon de tracer l'évolution en envergure des forces et des moments de tangage intégrés sur des tranches parallèles aux nervures et ce pour la voilure exclusivement.

La lecture de ces diagrammes (planches 5 et 6) oblige à une conclusion fort simple : entre l'avion rigide et l'avion souple, il n'existe pas de différence très notable ni en force ni en moment. Autrement dit, les répartitions de charges sur les deux avions sont assez semblables malgré le gros écart de braquage du cas supersonique.

Cependant, en poussant plus avant l'examen des charges, les petits écarts mentionnés se traduisent par une légère surcharge du bord d'attaque compensée par un allègement des parties elevonnées, surtout l'élevon externe dans le cas à $M = 1,74$ (constatation logique si on rappelle la faible efficacité de braquage dans cette zone).

2) Manoeuvre contrée de tangage

Il s'agit au cours de cette manoeuvre d'atteindre le facteur de charge $n = 2,5$ en imposant une loi de braquage sinusoïdale dont la période aussi faible que possible est conditionnée par la saturation des servodynes, le point de calcul retenu étant celui où le facteur de charge passe par le maximum.

A $M = 0,65$ (planche 1) les réponses des deux avions présentent une nette similitude alors que les braquages de fréquence identique diffèrent un peu par l'amplitude.

A $M = 1,74$ l'avion souple est plus long à répondre (presque 0,5 seconde de retard pour $n = 2,5$) et ce au prix d'une amplitude de braquage plus grande associée à une fréquence plus faible (planche 2).

Dans ces deux manoeuvres comme dans les deux précédentes on est amené aux mêmes conclusions : l'évolution des charges en envergure ne laisse apparaître que très peu de différence entre les deux avions; et si la zone arrière de l'avion souple a encore tendance à s'alléger (phénomène plus visible dans le cas supersonique), le bord d'attaque se surcharge toujours mais sur la voilure externe seulement.

3) Manoeuvre de roulis

En partant du facteur de charge $n = 1,67$ l'avion entame une manoeuvre de roulis en braquant ses élevons à la vitesse maximum jusqu'à butée ou saturation des servodynes, puis maintient son braquage et dès que l'assiette latérale atteint 60°, contrebraque jusqu'à une valeur opposée moitié du braquage maximum de la première phase pour conserver cette nouvelle valeur. Lorsque l'assiette passe par son maximum, on ramène le braquage à zéro.

L'instant retenu pour le calcul est celui où l'avion est à 60° d'assiette latérale.

Dans les deux cas étudiés, l'avion est allé jusqu'à saturation de la servocommande de l'élevon médian. On peut voir sur les planches 3 et 4, que l'avion souple nécessite un braquage plus élevé et que sa réponse présente un net retard sur celle de l'avion rigide.

Si nous considérons les charges (planches 9 et 10) induites par ces manoeuvres, nous ne pouvons que constater un accroissement en passant de l'avion rigide à l'avion souple. Mis à part le bout d'aile, toutes les tranches sont très nettement surchargées, en particulier dans le cas subsonique, ce qui peut s'expliquer par les grands braquages atteints par l'élevon médian. Et, si le bord d'attaque, surtout dans la zone centrale de voilure est toujours sensible aux déformations (et toujours dans le même sens), la partie arrière de voilure, du moins les élevons externes, présente également un surcroît de charge due à la souplesse.

Pour essayer d'illustrer le phénomène, nous avons tracé pour la manoeuvre de roulis supersonique, une carte des charges de l'avion souple en pourcentage des charges correspondantes avion rigide.

A l'exception des zones immédiatement en avant des élevons externes, toute la voilure est surchargée en moyenne de 10 % et ce pourcentage augmente en s'approchant du bord d'attaque.

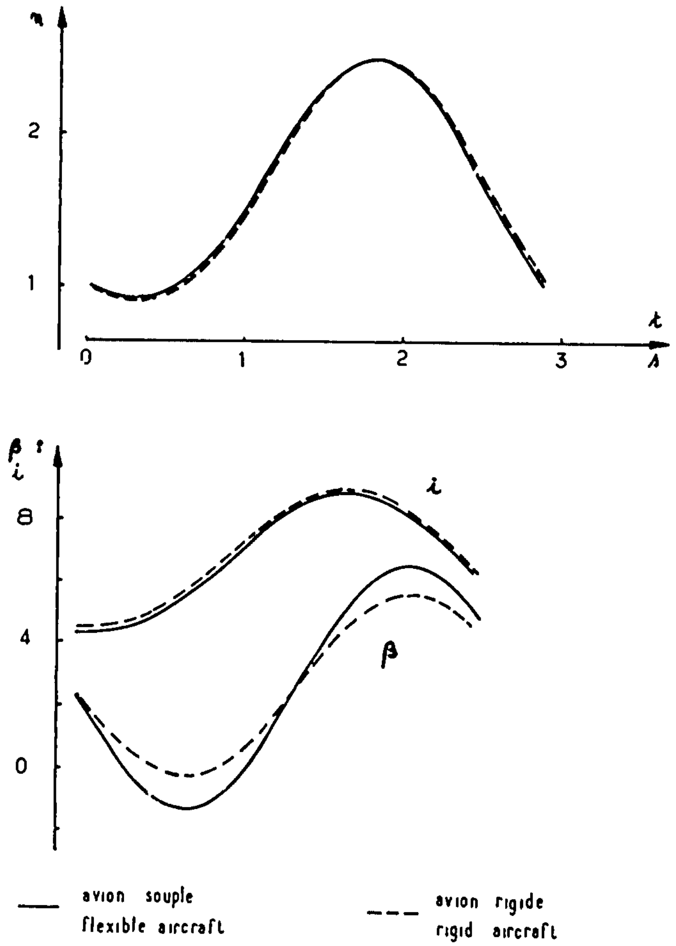
CONCLUSION

Ce court exposé n'a aucunement la prétention de formuler de loi générale ni de donner une méthode universelle à suivre. Les conclusions ne s'appliquent qu'à un avion supersonique de type delta et encore ne couvrent pas toutes les manoeuvres que les règlements peuvent exiger de cet avion. Mais, à la lumière de tous les calculs effectués, il est possible d'affirmer qu'à facteur de charge équivalent les avions souple et rigide ne diffèrent que très peu dans leur répartition de charges à telle enseigne qu'une étude rapide du domaine périphérique peut se faire en avion rigide quitte à affiner le calcul en tenant compte de la souplesse uniquement dans les cas déterminants.

Cependant, lorsque le but à atteindre est un des paramètres de vol comme l'assiette latérale par exemple, il n'est plus question de négliger l'influence des déformations.

En résumé, si la souplesse des surfaces portantes induit d'importantes modifications sur les qualités de vol de l'avion, son action sur les charges dans les manoeuvres de justification de la structure est moins évidente mais non négligeable.

Quoiqu'il en soit le processus de calcul adopté n'apporte pas de gêne considérable et l'étude de l'avion souple peut aisément se substituer à l'étude de l'avion rigide.

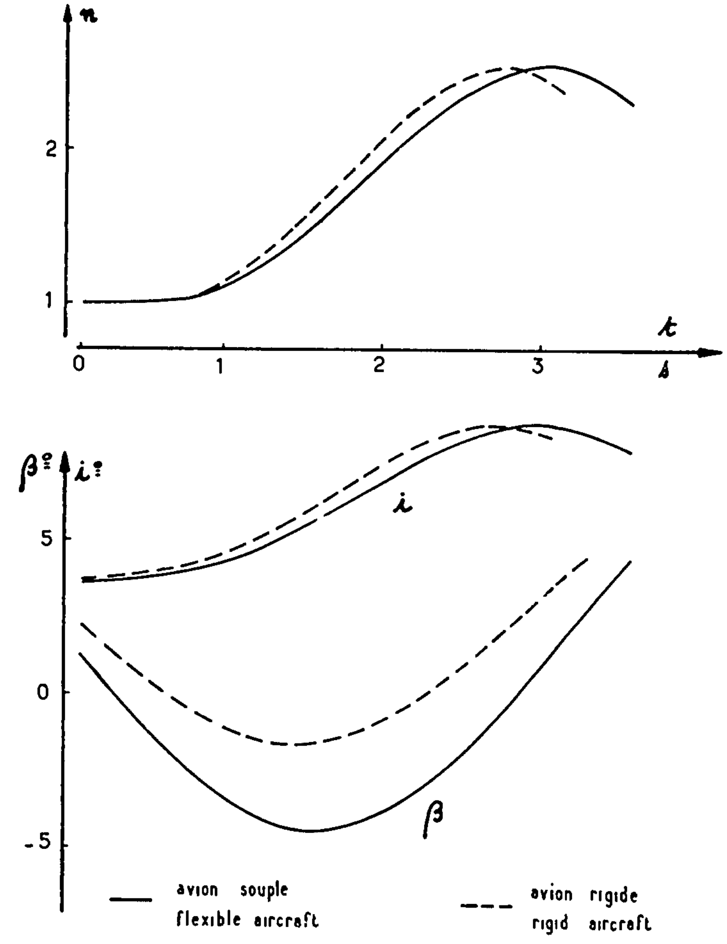


MANOEUVRE CONTREE DE TANGAGE

M = 0.656 Z = 5 000 ft.

CHECKED MANOEUVRE

fig 1

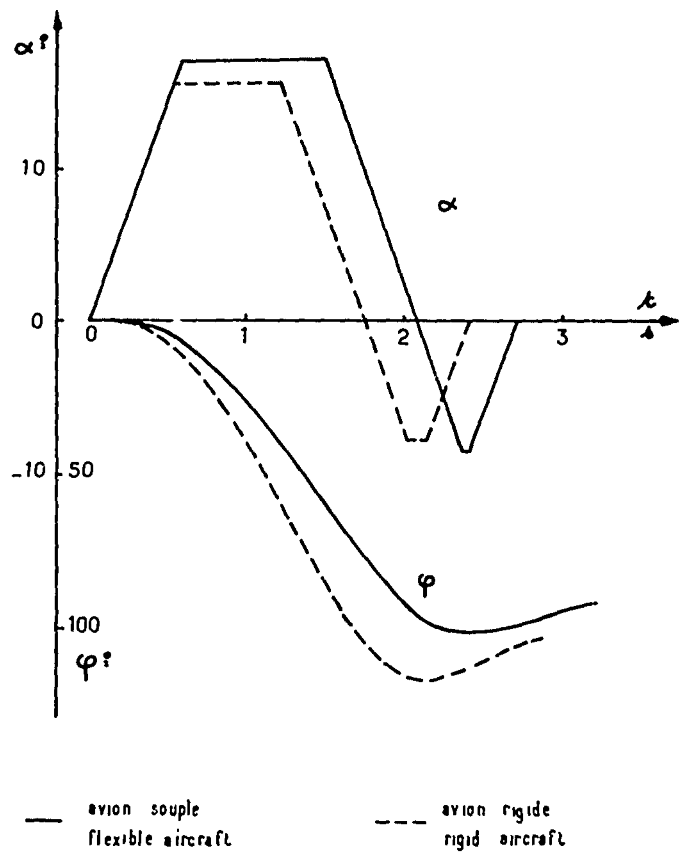


MANOEUVRE CONTREE DE TANGAGE

M = 1.743 Z = 43 700 ft.

CHECKED MANOEUVRE

fig 2

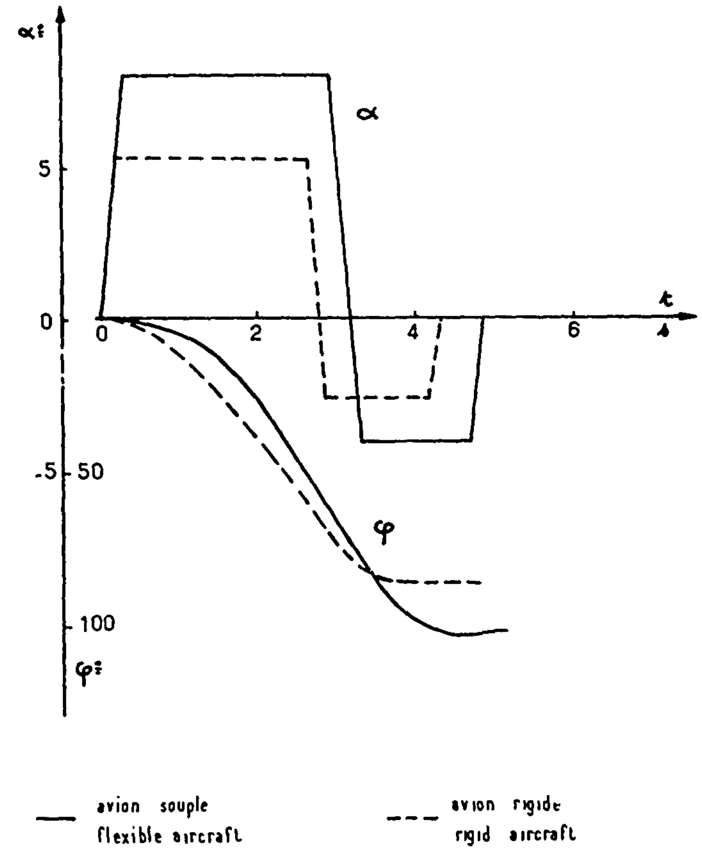


MANOEUVRE DE ROULIS

M = 0.656 Z = 5000 ft.

ROLLING MANOEUVRE

fig.3

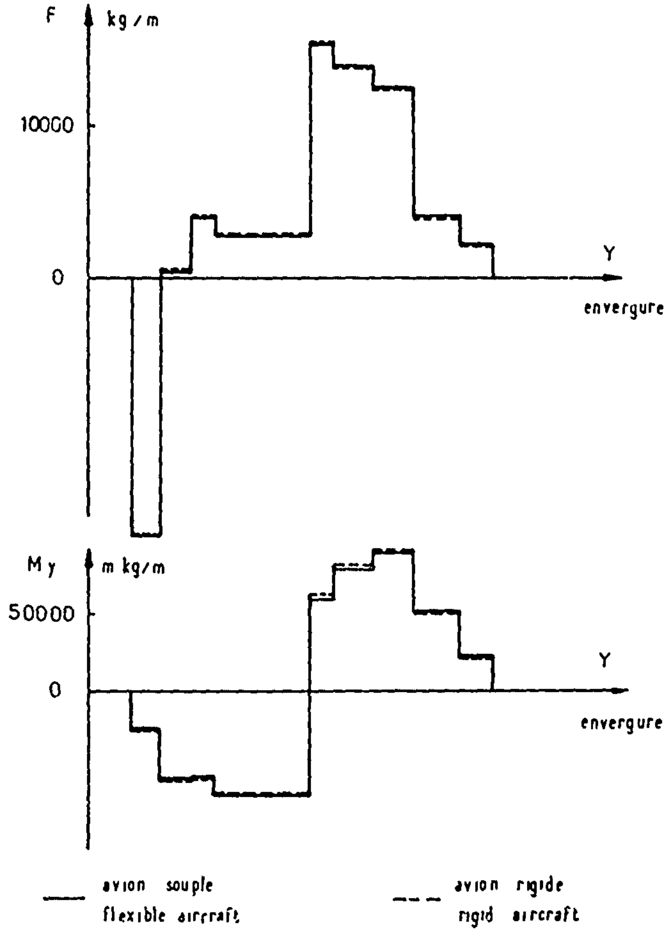


MANOEUVRE DE ROULIS

M = 1.743 Z = 43 700 ft.

ROLLING MANOEUVRE

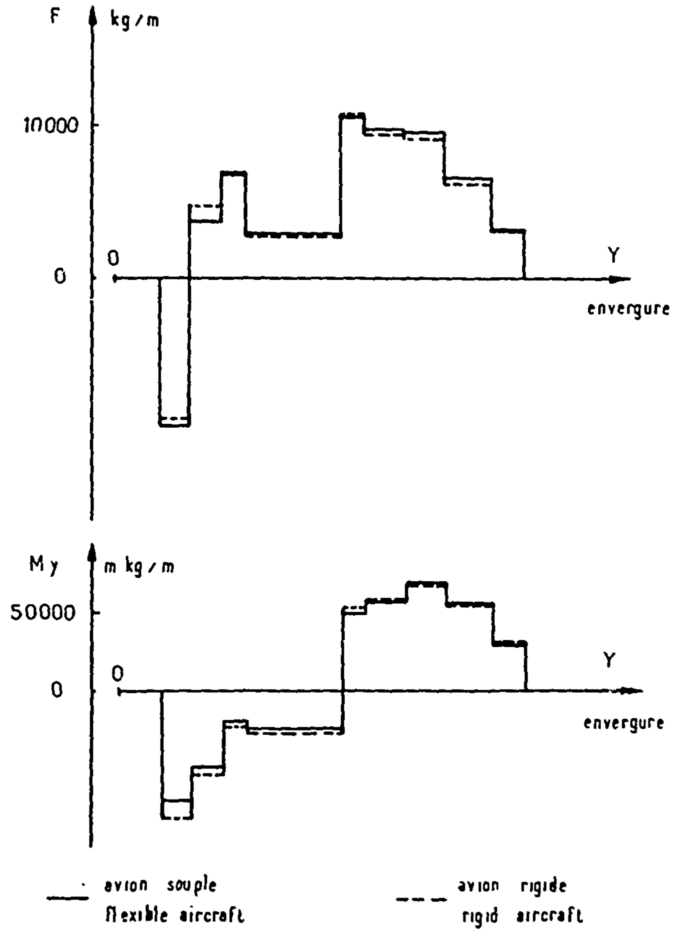
fig.4



RESSOURCE EQUILIBREE
M = 0.656 Z = 5000 ft

FORCES ET MOMENTS PAR TRANCHES

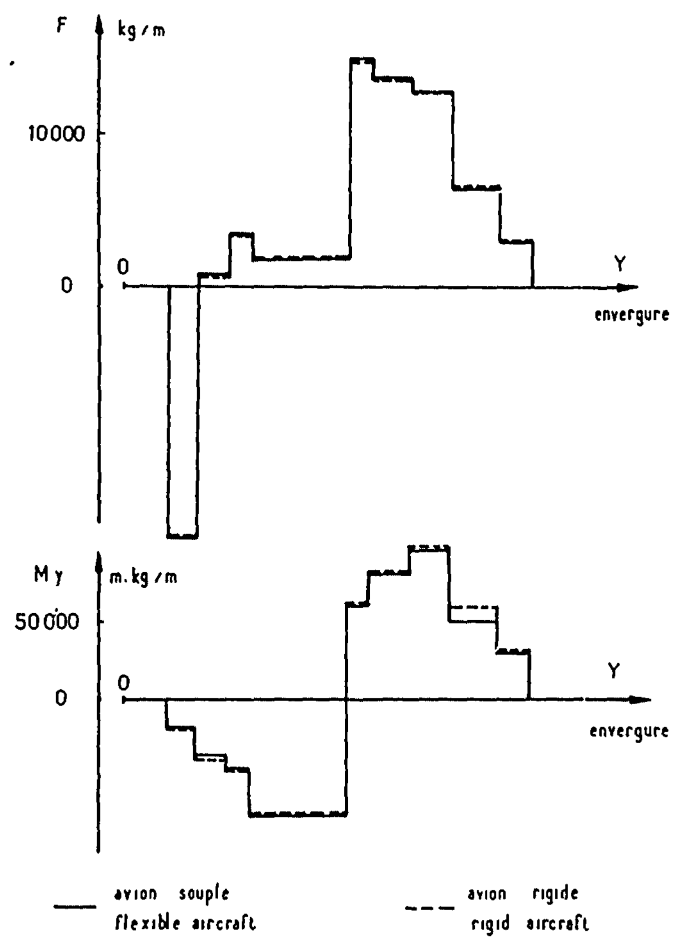
fig 5



RESSOURCE EQUILIBREE
M = 1.743 Z = 43 700 ft.

FORCES ET MOMENTS PAR TRANCHES

fig 6

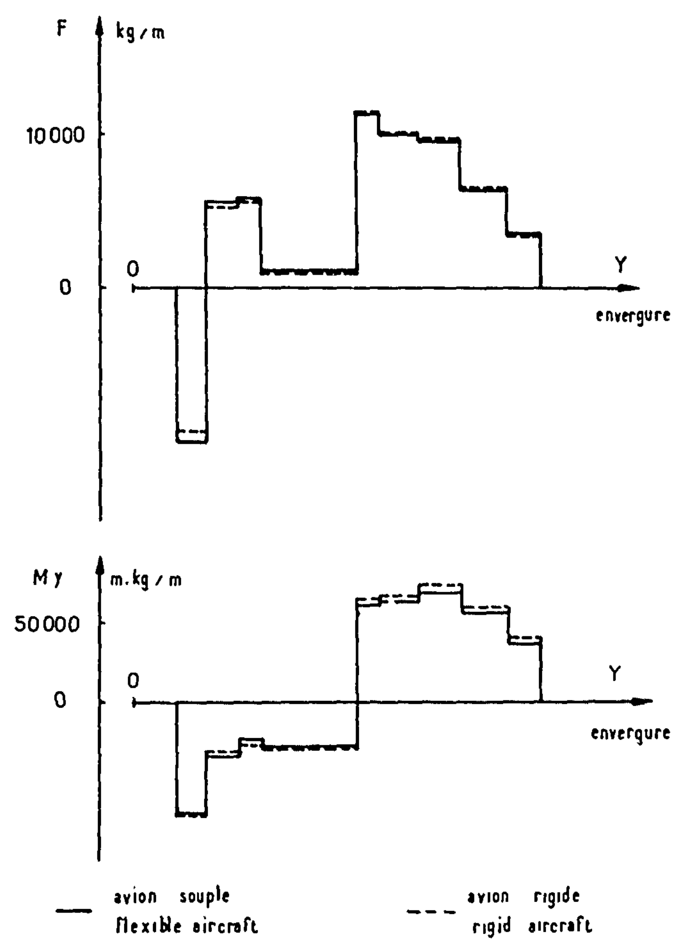


MANOEUVRE CONTREE DE TANGAGE

M = 0.656   Z = 5000 ft.

FORCES ET MOMENTS PAR TRANCHES

fig 7

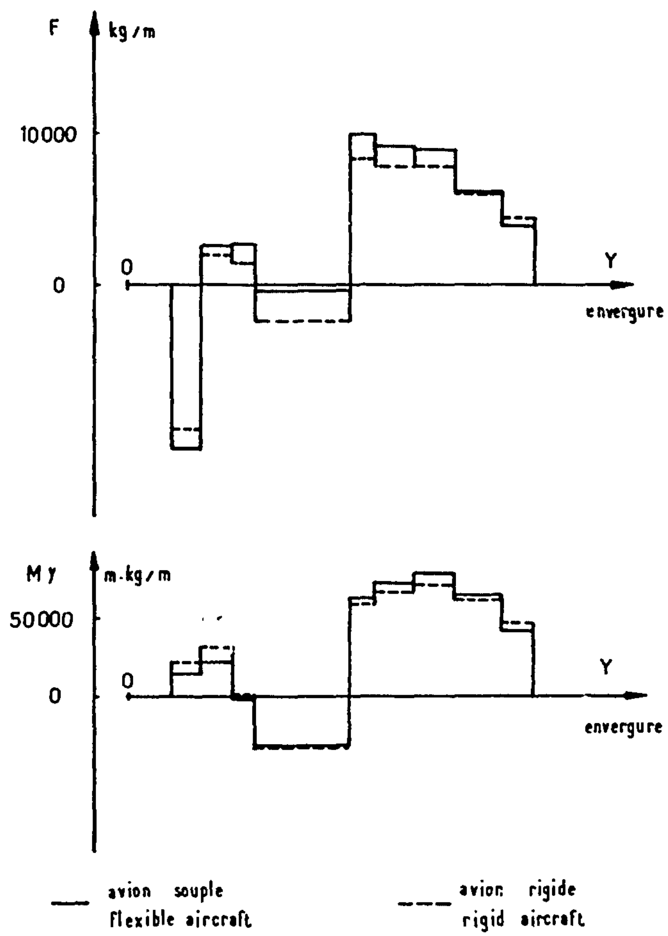


MANOEUVRE CONTREE DE TANGAGE

M = 1.743   Z = 43700 ft.

FORCES ET MOMENTS PAR TRANCHES

fig 8

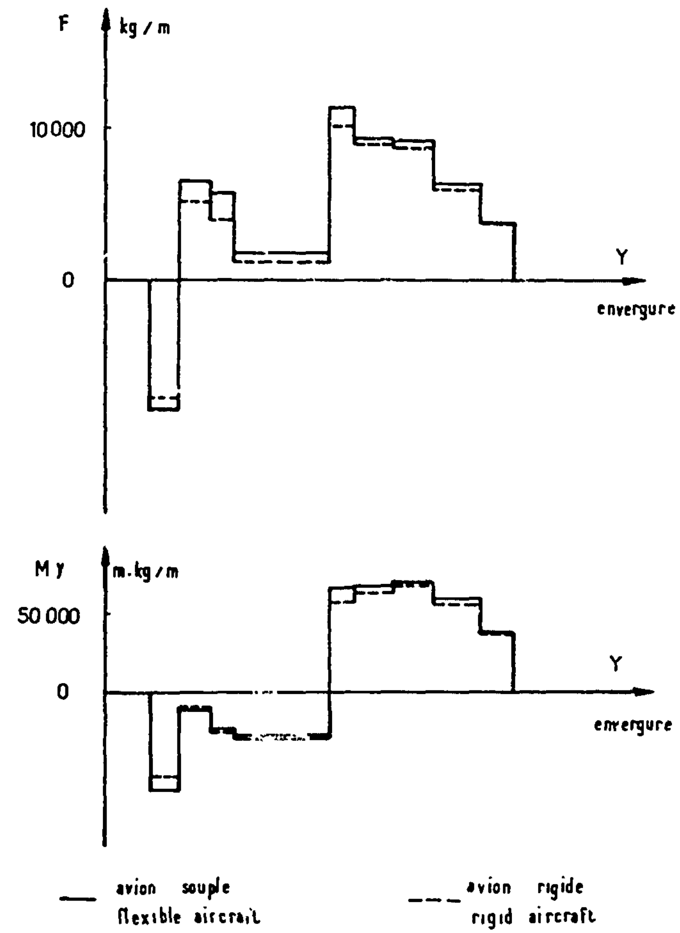


MANOEUVRE DE ROULIS

$M = 0.656$ $Z = 5000$ ft.

FORCES ET MOMENTS PAR TRANCHES

fig. 9



MANOEUVRE DE ROULIS

$M = 1.743$ $Z = 43700$ ft.

FORCES ET MOMENTS PAR TRANCHES

fig. 10

ACTIVE CONTROL OF AEROELASTIC RESPONSE

by

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SUMMARY

Conceptual and wind-tunnel programs leading to the development of technology for applying active controls to the suppression of flutter indicate that these methods may prove to be a powerful tool in providing required safety margins for flutter in future high-performance supersonic aircraft. The nature of flutter considerations in the design of the United States SST prototype aircraft has been described as an example of the type of application where active flutter suppression shows promise. Although this new technology is emerging, several years of additional development will be required in order to bring the technology to a complete state of readiness, particularly for civil applications.

INTRODUCTION

As designers seek the ultimate in efficiency for large, supersonic aircraft, aeroelastic considerations tend to play a dominant role in the development of such aircraft. Unfortunately, the complexity of the field of aeroelasticity involving interactions of a large number of parameters makes it difficult to assess the full impact of aeroelastic considerations early during the basic configuration layout. Thus, the requirements of other disciplines such as aerodynamics, propulsion, and strength, which are more amenable to early, accurate definition, tend to set the framework within which the aeroelastician must operate in evolving a satisfactory overall design. The aeroelastician has many needs which would help in the evolution of a more nearly optimum system. One of these needs, of course, is the ability to perform accurate and more timely analyses in the early layout phase, so that aeroelastic considerations could influence the selection of an optimum system at an earlier point in the design program. Another need is to bring to a state of readiness the emerging technology of active control of aeroelastic response.

The application of active controls for the suppression of flutter is one such emerging technology which, if fully developed, shows promise of providing attractive alternatives to the aeroelastician's standard tools for solving the flutter problem. This paper will be concerned with some of the activities at NASA Langley directed toward bringing this technology to a better state of readiness. As an example of the kind of need for which this technology could be applied in the future, flutter considerations relative to the United States SST prototype design are described, along with a brief assessment of our flutter analytical capability which, of course, is also pertinent to the analysis of active controls for flutter suppression.

FLUTTER CONSIDERATIONS IN THE U.S. SST DESIGN

Turner and Bartley (Ref. 1) of the Boeing Company have recently given an excellent description of the flutter prevention program employed during the design of the American SST prototype aircraft. Some of the highlights of their paper are repeated here.

Figure 1 illustrates the extent of the flutter problem encountered in the design of the American prototype airplane. This figure compares the flutter boundary calculated for the airplane with a structure designed just to have adequate strength without additional stiffness to the design dive speed envelope of the airplane. In the transonic range, it can be seen that the flutter speed is only about 80% of the design dive speed. Thus, it can be seen that substantial improvements in the flutter speed over that obtained by the strength design structure was required. The American SST configuration is characterized by a long, slender fuselage, thin wings of moderate aspect ratio, and aft-mounted engines. These characteristics, however, are typical of other large, supersonic aircraft. The feature of the American SST which is believed to be primarily responsible for the relatively low flutter speed of the strength design configuration is the fact that the engines for the SST employed three-dimensional inlets leading to a podded, beam-supported installation. Other large, supersonic aircraft with aft-mounted engines employ two-dimensional inlets with a relatively stiff inlet-duct system which minimizes the influence of the large mass of the engines in participating in the flutter mode. In the American SST the critical flutter mode always involved large motions of the outboard engine nacelle. Studies of configuration without engines indicated much higher flutter speeds. Such a configuration might have substantial advantages from the point of view of noise and pollution; however, a supersonic glider has little economic viability.

The flutter boundary shown in Figure 1 is, of course, calculated and a question might be raised regarding the accuracy of such flutter calculations. During the course of the SST prototype development, a variety of transonic and supersonic flutter model programs were conducted which permitted assessment of the adequacy of analytical methods used. One such assessment is shown in Figure 2 which compares experimental flutter boundaries obtained on two different models, one in the transonic range and the other in the supersonic range, with corresponding calculated flutter speeds. This figure serves to illustrate the current state of development of various unsteady aerodynamic theories. In the subsonic range, the flutter speeds calculated employing the kernel function method agree adequately with the experimentally determined values. In the transonic range, there is no calculated result simply because there is no transonic unsteady aerodynamic theory developed to the point where it is useful for routine flutter calculations used in the design of an aircraft. In the supersonic range, the standard tool used in flutter prevention design activities in the United States is the supersonic box method and it can be seen that the accuracy with which it predicts flutter results leaves much to be desired.

The results of Figure 2 prompt some general remarks concerning the state of development of unsteady aerodynamic theory. In the subsonic range, in addition to the kernel function method illustrated, recent years have seen the development of the doublet-lattice method which seems to be equal in accuracy to the more classical kernel function method, but also exhibits versatility in application to a wide variety of complex configurations (Ref. 2). Although there are several approaches to the transonic unsteady aerodynamic problem, most of them either lack the ring of physical reality or remain too cumbersome for routine numerical application in the design process. The transonic unsteady aerodynamic field remains a fruitful area for research. In the supersonic range several promising approaches exist, and it can be hoped that practical design tools employing improvements over the supersonic box method will become available in the near future.

Successful exploitation of the emerging technology for flutter suppression by active controls also requires improvements in unsteady aerodynamic theory, in that design methods for active controls require accurate representation of the unsteady control effectiveness and hinge moments. These and other aspects of the development of active control technology are described below.

AN AERODYNAMIC ENERGY CONCEPT OF FLUTTER SUPPRESSION

One of the needs which must be met if we are to bring to fruition the promise of advanced active control systems in improving the efficiency and safety of aircraft is a better merging of the fields of controls theory and aeroelasticity. One recent contribution to this need from the aeroelastician's point of view is the development of an aerodynamic energy concept by Eliahu Nissim of the Technion, Israel. This work (Ref. 3) was performed while Nissim was at the NASA Langley Research Center as a National Research Council Research Associate. Nissim's concepts have been explained in the recent Fifth Theodore Von Karman Memorial Lecture, by I. E. Garrick (Ref. 4) which also reviews other current aeroelastic topics of interest. The description below is essentially excerpted from Garrick's excellent lecture.

During flutter, energy must be transferred from the airstream into the airplane system. This statement may be put in another way; namely, necessary and sufficient for flutter prevention is the circumstance that all conceivable, allowable oscillatory motions will require positive work to be done by the aircraft on the airstream. To understand Nissim's aerodynamic energy approach, consider oscillatory motion first for the special case of two degrees of freedom; bending (h) and torsion or pitch (α).

We may write for the rate at which work (W) is done against the aerodynamic force (F) and moment (M_α) acting

$$\dot{W} = F\dot{h} + M_\alpha \dot{\alpha}$$

Now $\dot{W}$ is zero at zero airspeed, becomes positive with increase in airspeed and then goes to zero again at the flutter speed beyond which it becomes negative. Expressed in terms of the average work per cycle of oscillation $\bar{W}$ and in terms of the maximum amplitudes h_o and α_o, a quadratic expression results

$$\bar{W} = B_1 h_o^2 + B_2 \alpha_o^2 + B_3 \alpha_o h_o$$

where the B's are purely aerodynamic terms, B_1 represents the damping in bending, B_2 damping in torsion, and B_3 is a cross-coupling aerodynamic damping, a function of the phase difference between h and α.

Consider now n degrees of freedom given by generalized coordinates q defined by a column matrix of order n

$$\left\{q\right\} = \left\{q_R + iq_I\right\} e^{i\omega t} = \left\{q_o\right\} e^{i\omega t}$$

The work per cycle done by the motion, as shown by Nissim is

$$\bar{W} = C \left\lfloor \bar{q}_o \right\rfloor [U] \left\{q_o\right\}$$

where C is a constant, q_o is the complex amplitude of q (hence includes the phase), $\bar{q}_o$ is its row conjugate and $[U]$ is a square matrix of order n of purely aerodynamic origin, and is in fact given by

$$[U] = \left[- \left(A_I + A_I^T\right) + i \left(A_R - A_R^T\right)\right]$$

where $A_R + i A_I$ represent the aerodynamic terms in the n equations of motion, and superscript T represents a matrix transpose. It is significant that $[U]$ is a Hermitian matrix (i.e., $U = \bar{U}^T$), and therefore it possesses real eigenvalues which can be expressed as a diagonal matrix $\lceil \lambda_i \rfloor$. Making further use of properties of the solution of the characteristic equation of $[U]$, Nissim shows that the energy per cycle $\bar{W}$ can be written as a principal quadratic form, that is, without coupling terms, as

$$\bar{W} = C\left[\lambda_1 \left(\xi_{R1}^2 + \xi_{I1}^2\right) + \lambda_2 \left(\xi_{R2}^2 + \xi_{I2}^2\right) + \ . \ . \ . + \lambda_n \left(\xi_{Rn}^2 + \xi_{In}^2\right)\right]$$

where the ξ's are new modal coordinates, obtained from the q's. The λ's, as stated must be real, and if they can be determined (by aerodynamic means through change of U) so that they are all positive then $\bar{W}$ will be positive, and flutter of the assumed system is then not possible.

To apply this theory concretely to active flutter suppression Nissim uses two-dimensional subsonic oscillatory aerodynamics and considers a system having four degrees of freedom — bending h, torsion α, a leading-edge control surface motion β, and that of a trailing-edge control surface δ. The leading- and trailing-edge controls are related to the h and α degrees of freedom by a "control law" as determined by, say, two independent sensors; for example, accelerometers or gyros. A numerical study was made to determine this control relationship so that over the reduced frequency range of interest the lower algebraic value of the two λ's associated with the problem becomes and remains positive. This then assures that $\bar{W}$ will be positive, thus preventing the occurrence of flutter. One may regard the suppression as achieved through the aerodynamic decoupling of the sensitive modes, thereby greatly improving the effective damping. Another physical interpretation that can be placed on Nissim's approach is to point out the necessity of two independent controllers (leading-edge and trailing-edge surfaces) in order to control the two types of motion involved in flutter, namely, bending and pitch. As might be expected, and as discussed by Nissim, various practical compromises may need to be introduced to reduce undesirable sensitivity and to meet other requirements.

SOME PRACTICAL APPLICATIONS OF THE AERODYNAMIC ENERGY FLUTTER SUPPRESSION SYSTEM

In order to explore some of the practical aspects of application of the aerodynamic energy flutter suppression concept to a real aircraft design, the Boeing/Wichita Company, under a contract to the Langley Research Center, performed a brief study of flutter suppression systems applied to an early mathematical model of the American SST. Some of the results of this study are illustrated in Figure 3. The sketch on the right indicates the three pairs of leading edge-trailing edge controls considered, located at stations referred to as outboard, midspan, and inboard. Motion was sensed by two accelerometers located near the hinge line of each of the controls. The control law used to close the loop was that indicated by Nissim's aerodynamic energy analysis. As indicated in the figure, the three different flutter suppression systems yielded increases in flutter speed ranging from 11% for the outboard section, up to 28% for the midspan section. Another calculation was made with both the midspan and inboard sections activated. The results of this analysis are shown in the root locus diagram to the left of Figure 3. The analysis included 10 elastic modes plus the rigid body motions of pitch and plunge. However, only the two elastic modes involved in flutter are shown in the diagram. For the basic airplane without flutter suppression, the third elastic mode becomes unstable at a speed of 422 knots. The fourth mode becomes unstable at just a slightly higher speed. When both the midspan and inboard sections are activated, the roots remain in the stable portion of the diagram, even when the speed is increased to the value corresponding to sea-level flight at a Mach number of 0.9. Thus, the two systems working together produced an increase in flutter speed of something in excess of 40%.

MODEL PROGRAMS FOR STUDYING FLUTTER SUPPRESSION

In order to further develop the technology needed to bring flutter suppression to use in aircraft design, a program has been initiated at the Langley Research Center to develop and apply wind-tunnel modeling techniques for studying flutter suppression systems in the critical transonic range where suitable aerodynamic theories do not exist. Although this program has not yet reached the point where active flutter suppression has been demonstrated in the wind tunnel, substantial progress has been made, and the following indicates the status and plans for that program.

One of the models being used in this program is illustrated in the photograph of Figure 4. The model is a simplified representation of the American SST. It has the gross geometric characteristics and structural characteristics such that its flutter mode is similar to that of more accurately scaled models and its flutter speed is within a few percent of these model results. The model is equipped with a single set of leading edge-trailing edge control devices similar to the midspan flutter suppression control system described previously and illustrated in Figure 3.

Boeing/Wichita, under a contract to the Langley Research Center, is providing general support for this model program in the area of controls implementation and analysis. The controls are actuated by specially designed hydraulic actuators of a wiper-vane type. A photograph of the model showing one of the actuators installed is shown in Figure 5. Each actuator weighs about 2 ounces and has essentially flat frequency response well above the flutter frequency of the model.

As mentioned previously, closed-loop operation of the system has not yet been accomplished, although it is hoped to reach this stage by summer or early fall of 1972. In order to aid in the design of the model control actuators in the flutter suppression system, the model has been installed in the Langley transonic dynamics tunnel for measurements of the hinge moments on the leading-edge and trailing-edge controls. Prior to the measurement program, estimates of hinge moments by a variety of methods produced a wide range of values. A comparison of the measured static hinge moments with values calculated after the measurements had been made is shown in Figure 6. The agreement between measured and calculated results for the leading-edge control is surprisingly good, and the lack of substantial variation through the transonic Mach number range is encouraging. The lack of agreement between measured and calculated values for the trailing-edge control is not surprising, being typical of the usual inability of nonviscous theory to predict this type of detailed aerodynamic behavior. The aerodynamic theory used for the subsonic predictions is doublet-lattice, while that employed for the calculation at a Mach number of 1.2 is the simple piston theory. This appears to be another case of piston theory yielding a useful answer even though the physical flow conditions violate the underlying assumptions of the theory. Although comparisons of control effectiveness coefficients are not available, it is hoped that the aerodynamic treatment employed in the control systems analysis will adequately represent this feature of the system.

OTHER FLUTTER SUPPRESSION PROGRAMS

In addition to the model program employing the simplified representation of the American SST design, the Langley Research Center is engaged in a cooperative program with the U.S. Air Force Flight Dynamics Laboratory to study flutter suppression on a model of the Boeing B-52 bomber. This model program is being conducted in conjunction with, and coordinated with, a flight research program. The B-52 flight and model program will include studies of other types of active control systems other than flutter suppression. A general outlook on the prospects of advanced active control systems for applications to civil aircraft is given in Reference 5. Some types of advanced active control systems have been demonstrated and incorporated in United States military aircraft. The ability to take full advantage of the promise of the concept of flutter suppression, however, probably requires several more years of technology development. The needed technology development is in several areas, such as development of criteria for use of active control in providing the flutter safety margin, development of new hardware concepts that will not only provide the necessary safety aspects of reliability, but also the logistics and economic requirements associated with routine airline operations. Perhaps, as another new technology, flutter suppression will first be applied to military vehicles, providing operational confidence before it is incorporated in civil aircraft. That this technology is evolving is indicated by the fact that an entire session was devoted to this subject at the most recent Joint Automatic Control Conference. The papers presented in this session are listed as References 6 through 9.

CONCLUDING REMARKS

Conceptual and wind-tunnel programs leading to the development of technology for applying active controls to the suppression of flutter indicate that these methods may prove to be a powerful tool in providing required safety margins for flutter in future high-performance supersonic aircraft. The nature of flutter considerations in the design of the United States SST prototype aircraft has been described as an example of the type of application where active flutter suppression shows promise. Although this new technology is emerging, several years of additional development will be required in order to bring the technology to a complete state of readiness, particularly for civil applications.

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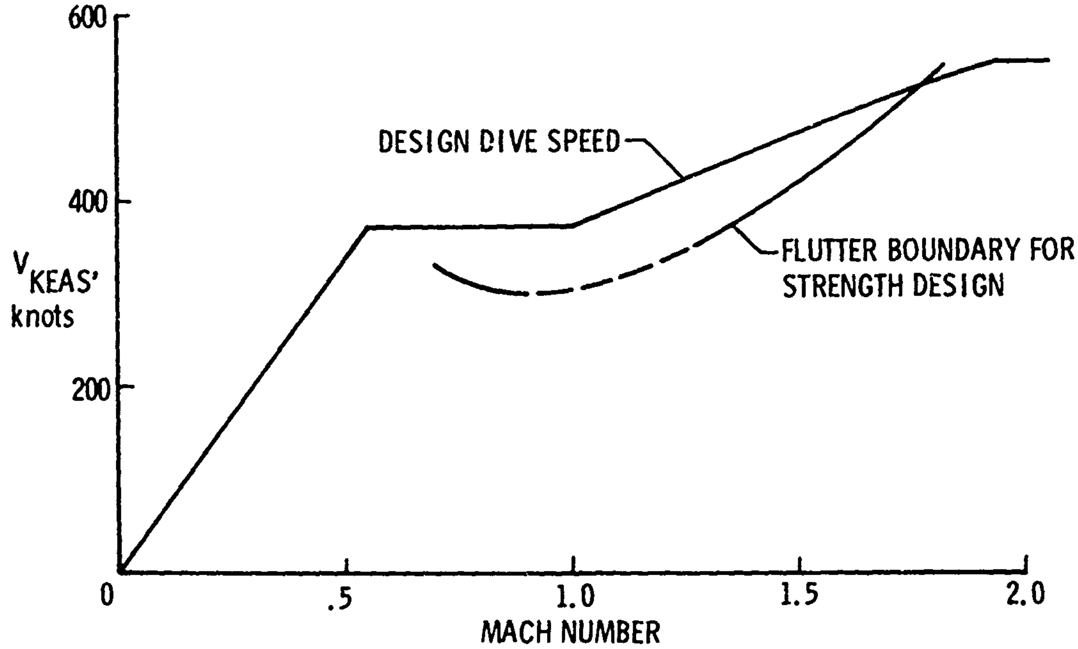

Fig.1 Flutter prevention considerations for the US SST

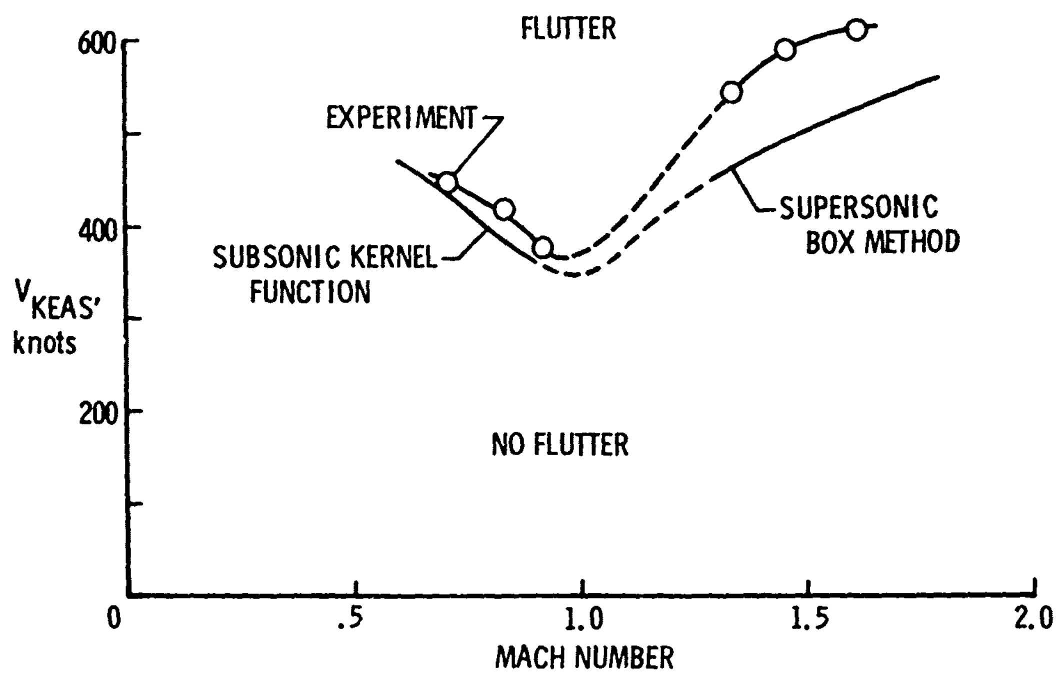

Fig.2 Comparison of measured and calculated SST model flutter boundaries

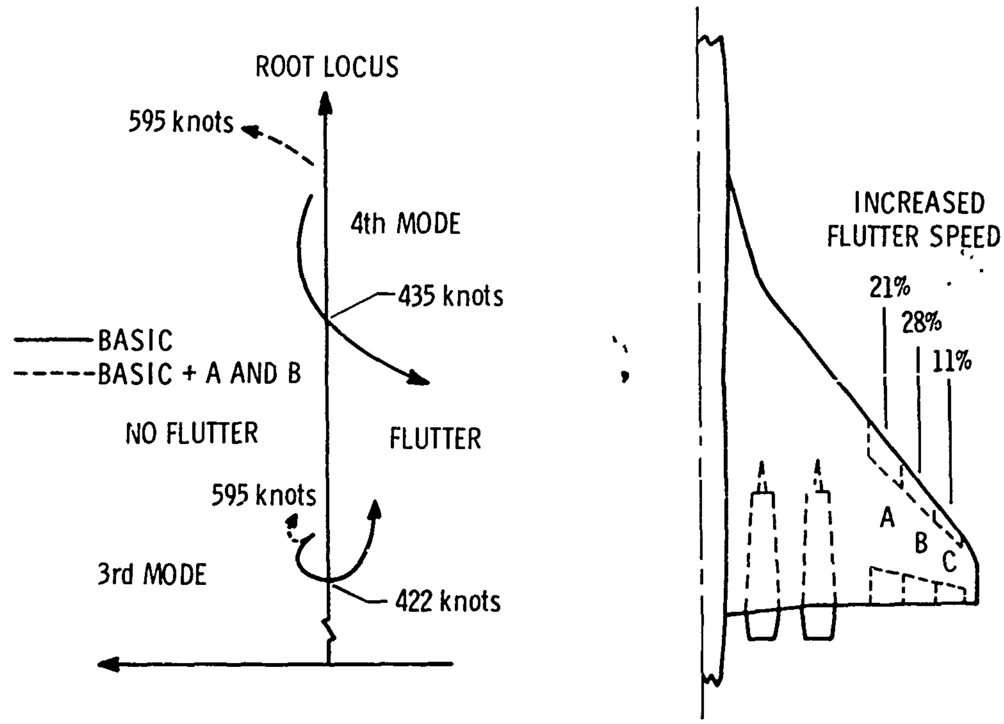

Fig.3 Calculated effectiveness of aerodynamic energy concept flutter suppression systems

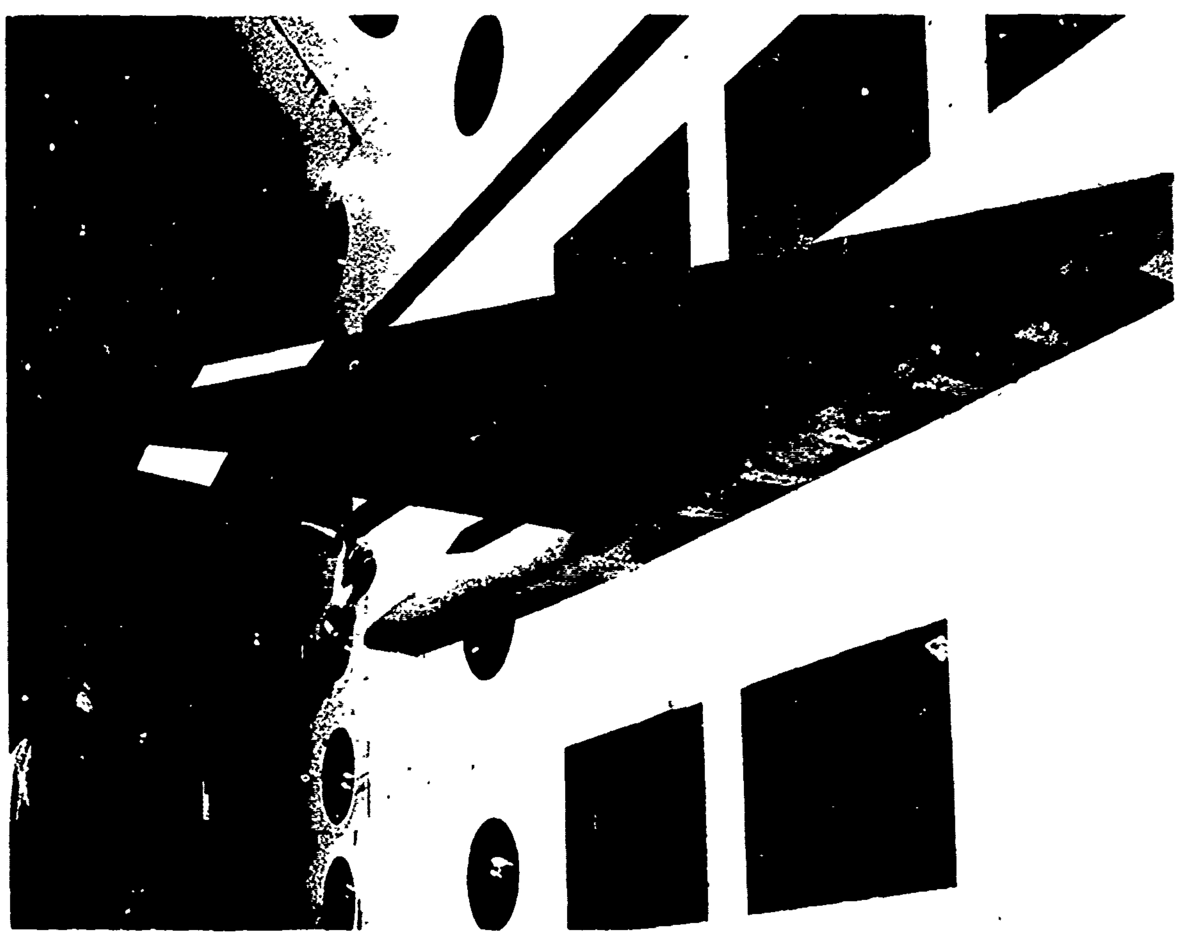

Fig.4 Flutter suppression research model

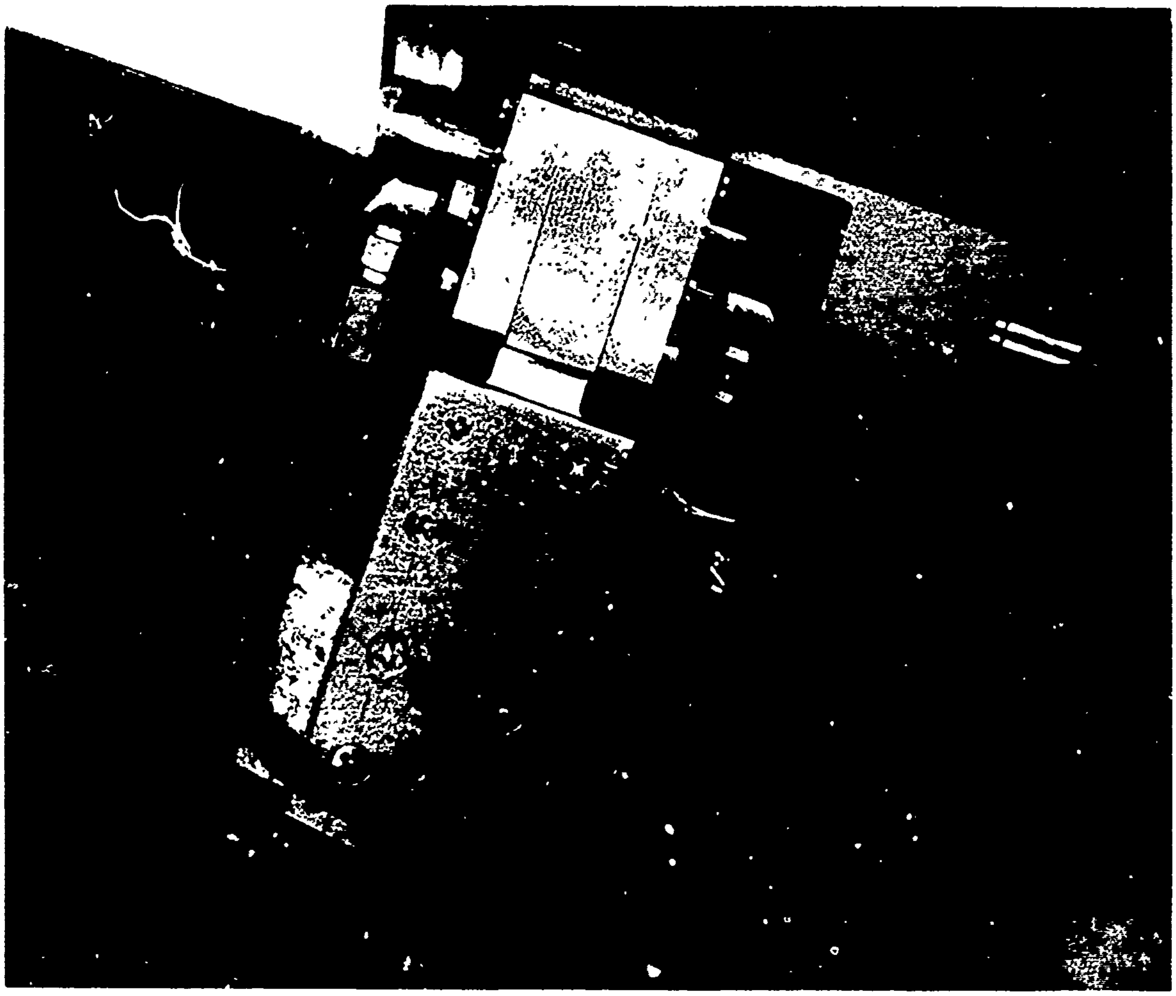

Fig.5 Flutter suppression research model control actuator

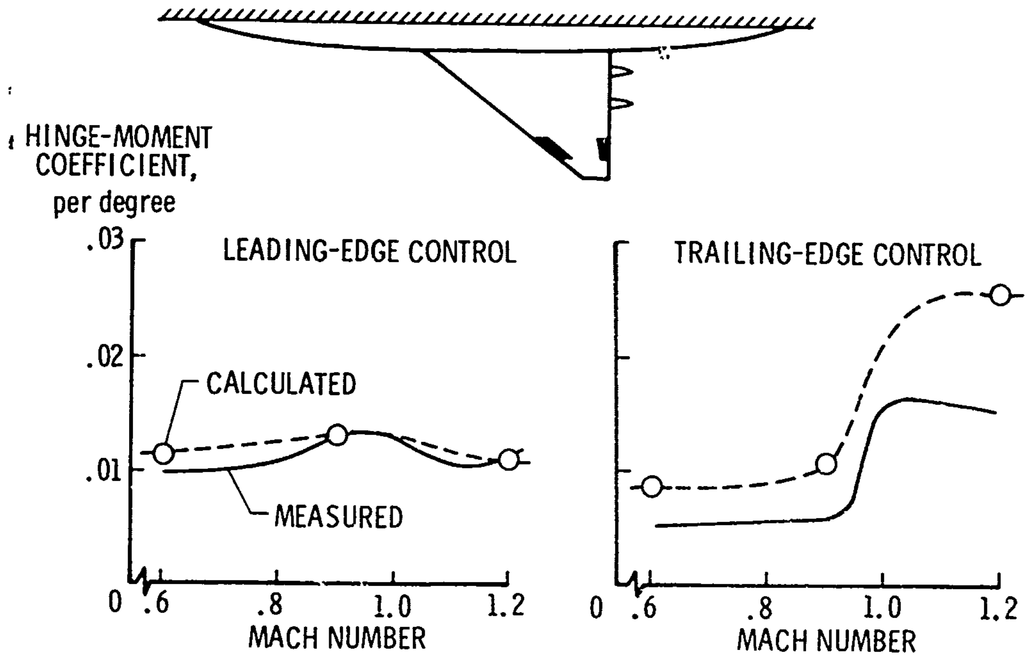

Fig.6 Comparison of measured and calculated static hinge moment coefficients

OPEN DISCUSSION

W.T.Hamilton, USA: Comment: The US-SST Program considered active control flutter suppression but did not have time in the schedule for the long extensive control system development required to get adequate reliability in the system. Consequently, a conservative criterion was selected to decide whether to include active flutter prevention. It was believed that a reliable, relatively simple system could be developed, within the time schedule, which would be consistant with the more conservative criterion.

W.J.G.Pinsker, UK: I have recently become interested in CCV and the first thing we tried to do was to get an idea of the applications where CCV brings the greatest returns. From admittedly superficial studies it appeared that active flutter control showed the least attractive return in terms of weight saved by the time one allows for the extra system hardware. I would like Mr Rainey to tell us whether he sees the principal attractions of active flutter control in structural efficiency or rather in ease of design and development.

A.G.Rainey, USA: The potential benefit of any of the control functions normally included in the phrase "CCV" is configuration dependent, that is, depends on the characteristics of the particular airplane design to which they are being applied. Of course, some airplane designs do not suffer a flutter penalty and, consequently, flutter suppression would have no payoff at all for such a design. In other cases flutter suppression would be needed in order to take full advantage of the application of other control concepts, such as maneuver load control, for reducing the basic strength requirement of the design. The use of active flutter control is envisioned as a means of providing improved structural efficiency and, as a matter of fact, its application would probably complicate, rather than ease, the design and development effort.

D.J.Walker, UK: Presumably the technique depends on a preknowledge of the flutter modes. How is this compatible with the strike role of a military aircraft for which different modes can flutter depending on wing store load?

A.G.Rainey, USA: One of the advantages of the aerodynamic energy concept for flutter suppression is that this approach leads to a relative independence of the control system to changing structural characteristics such as fuel distribution or store loadings. Nissim's analysis of the aerodynamic energy concept shows that this type of control system, regardless of modal changes, cannot cause a decrease in the speed of instability and can only go to zero effectiveness in the event that the flutter mode changes such that the response at the control station is zero. The probability of this happening in a real design seems remote, but would have to be examined during the design process. If two or more control stations were used in the design, it seems very unlikely that a condition of zero effectiveness could be found, and our very limited experience indicates that such a control could be very effective. The practical application of this concept to aircraft design requires considerably more effort than we have had time to put into it.

PREDICTION OF AEROELASTIC HINGE MOMENT EFFECTS ON STABILITY AND CONTROLLABILITY

by

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SUMMARY

Stability and control characteristics have been affected by aeroelastic deflections for many years. Some early difficulties are discussed and examples are shown that have resulted in control problems and aircraft operating limitations. In certain cases, it may not be necessary to give design consideration to the aeroelastic hinge moment case since other factors control the configuration development. This is discussed and some particular examples are presented to illustrate that the elastic hinge moment problem was not a major factor. Methods of predicting aeroelastic hinge moments are reviewed and some of the problems that arise by the use of these methods are discussed. A recently developed program for analysis of structural deformations is described which may be used to analyze many aeroelastic problems. The control system designer must consider the problems of aeroelasticity when sizing and locating actuators and determining control surface shapes, locations, and deflections.

When speaking of aeroelasticity, it is very difficult to speak only of hinge moments and not about some of the other problems of aeroelasticity. The various aspects cannot be separated because they are so interrelated and consideration must be given to control surface effectiveness, aeroelastic characteristics of the lifting surfaces and the torsional and bending stiffnesses of the fuselage. This paper will mention not only hinge moments but also other problems and characteristics that are related to control surface problems.

There is a classic case which shows how the aeroelastic characteristics can limit the usefulness, or flight envelope, of an airplane. This airplane was a swept wing bomber with outboard ailerons for roll control. The control system was not an irreversible system so the control forces increased with dynamic pressure for a given aileron deflection. Figure 1 shows the aerodynamic forces generated at two airspeeds for a constant aileron control force. At the higher speed, due to increased hinge moment, the aileron deflection is less than at the lower speed. The wing twist generated by the deflected aileron produces a force which opposes the aileron roll force. The decreasing aileron deflection prevents the roll force from the control from increasing while the opposing roll force from a given amount of wing twist increases as the dynamic pressure increases. Eventually, the two forces become equal and opposite and this condition is referred to as the reversal speed. Figure 2. The roll capability decreases as the reversal speed is approached. At speeds greater than the reversal speed, the airplane will roll in the direction opposite to that intended by the pilot. This places a very real operational limit on the airplane because some speed less than the reversal speed must be used to assure some minimum roll capability.

For the bomber mentioned above, the roll reversal speed was appreciably less than either the design limit speed or the level flight power limit shown in Figure 3. The roll reversal speed determined the maximum usable speed of this airplane at low altitudes.

Similar control losses have occurred on other aircraft with trailing edge surfaces. Rudder and elevator control powers have been reduced due to aeroelastic deflections caused from a deflection of the control surface. Such losses, if unexpected or not properly accounted for during the design process can materially reduce the effectiveness of the airplane by either decreasing the maneuverability of the aircraft or by reducing the operating envelope at high dynamic pressures. In addition to these limitations, there are also effects on structural design and control system design. If it is essential that normal operation occur at high dynamic pressures then the structural properties must be sufficient to provide a base from which the control system will be able to function. The control system must be able to provide the power or the hinge moment capability to move the particular surface of concern at high dynamic pressures and frequently at conditions quite different from the rigid conditions that are seen in normal design layout drawings and on conventional wind tunnel models. Both major components of the hinge moments may be different from that seen in the rigid case. The angle of attack of the control surface may be appreciably different due to surface deflections in flight. This is especially true of control surfaces such as ailerons which are on a wing. A highly swept wing will undergo a large amount of bending due to lift forces and will also have some torsional deflections. Some large aircraft have experienced wing tip deflections of the order of 15 to 20 feet. The angle of attack of the control surfaces, if located well outboard, is hardly the same as it was in the undeflected case.

The other component of the hinge moment, the surface deflection, also undergoes some change. There are Mach number effects on control surfaces which change their effectiveness. For a given airplane response of roll rate, roll acceleration, pitch rate or pitch acceleration, the surface deflection will differ as a function of dynamic pressure. The airplane structure will deflect and absorb some of the control power without corresponding motion about the airplane center of gravity. Additional control power in the form of surface deflection must be applied to obtain the required response. The hinge moment predictions should account, in some manner, for the aeroelastic effects on angle of attack and also the required surface deflection to enable the system designer to prepare a control system that will properly control the airplane throughout its flight envelope.

There are several methods that are used to try to predict the aeroelastic effects on stability, controllability, and hinge moments. All of them require the best possible structural definition of the airframe for accuracy. Structural definition is usually one of the last things that is determined with good confidence during the design because it changes so much as more and more is known about the airplane. There usually are from three to five structural design iterations due to loads, stress analysis, weight changes and weight distribution changes. The control system must be committed to design before the last structural design iterations occur and frequently the control system characteristics play a part in the later design changes.

Therefore, the knowledge that is attained about flexible structural characteristics that influence the control designer must be flavored with some added factor to account for the possibility that the actual design may be less stiff (somehow it always seems to be less stiff) than the early predictions or data indicate.

Techniques are in use to determine aeroelastic characteristics. There are two well known wind tunnel test methods in use. One of these methods involves the use of flexible models. These models are built with a rigid fuselage section in the vicinity of the balance. The forward and aft sections of the fuselage and the wing and control and stabilizing surfaces are flexible so that they will distort or move under the dynamic pressures encountered in the wind tunnel. The model structure is scaled to represent the bending and torsional stiffness of the actual airplane as known at the time. The model is then tested and force, moment, pressure distribution, and strain gauge data may be obtained. These data will show deflections of the basic structure. If control surfaces are included on the model then surface effectiveness and hinge moment data can be obtained.

This type of testing has received varying amounts of acceptance. The model must be scaled to account for the effects of weight and inertia as well as the air load effects. For model structure, these scaling difficulties mean that a fairly narrow speed range, or dynamic pressure range, can be simulated with each model. This may require the use of several very expensive models to complete a thorough test or to account for the various conditions of interest. The accuracy of the structural scaling is affected by the amount of deflection encountered so that this also may lead to inaccuracies. Although testing of this type is done, there is enough concern over the final accuracy of the results to look for better methods of obtaining aeroelastic and hinge moment data.

Another technique also involves wind tunnel testing. With this method, rigid models are used. The rigid models are built to a shape that is meant to represent the shape of the airplane, or a portion of the airplane, at some flight condition. These tests may be conducted like any rigid model tests and the normal force and moment data obtained. There are some disadvantages to this method. Each deflected model can be made to represent only one flight condition; therefore, several models must be built to include even a portion of the flight cases of interest. This type of testing is very expensive because of the number of models that must be built and this type of model is very expensive. It is never known for sure if the shape prescribed for the model is the correct shape so that some judgement and prediction must be used. The data can be easily obtained and used, however, since it is rigid body data. It is felt that there must be great concern over aeroelastic effect and a good and complete knowledge of structural characteristics to justify the use of this method because of its cost.

Still another way to predict the aeroelastic effects on the structure and the control system is analytical. This method was developed by the Boeing Company under a NASA contract. The method is called FLEXSTAB and it is a digital computer program that solves the elastic airplane residual flexibility equations of motion. The program is intended to evaluate those parameters which affect the stability and control and flight control design of an airplane. The program represents mathematically the interactions of the structure (by stiffness or flexibility), the aerodynamics (steady or quasi-steady), and the mass and the distribution of the mass of an arbitrary configuration. The aerodynamic inputs may be determined analytically and later refined by the use of experimental data.

The Air Force Flight Dynamics Laboratory has obtained the FLEXSTAB program. The Lab is in the process of adding subroutines to the program to expand the capabilities. The items being added permit the analysis of the flight control system to study flying qualities and ride qualities.

When fully implemented, it is intended to use the improved FLEXSTAB program to analyze and determine the actual in-the-air shape of the airplane at a number of flight conditions. Knowledge of the actual shape and not the jig shape or the one-g shape of the airplane is useful to performance engineers as well as structures and flight dynamics engineers. The program will also determine stability derivatives, control derivatives, and hinge moments. This program is not limited to a few flight conditions but can investigate many conditions throughout the flight envelope since the input data of airplane geometry and structural characteristics will be the same.

The program is also intended to be used to analyze the control systems of large elastic airplanes in order to determine as soon as possible during design what basic control characteristics are required. Effects of changed structure can be quickly determined once the new input data are prepared. The FLEXSTAB program is faster and cheaper than the methods which require the construction of models. It is also more versatile and can examine many more flight conditions and can readily evaluate changes. The program is relatively new and unproven against actual airplane data results and it must be used and tried against known results to establish the confidence necessary to use it for design and development purposes.

There is a strong body of opinion in the United States that feels, at least for some types of aircraft, it is possible, by careful design, to build an airplane without a complicated and expensive test program to determine aeroelastic characteristics and aeroelastic effects on control systems.

Many things have already been done by designers to reduce control effectiveness losses due to aeroelasticity. When the roll rate losses mentioned early in the paper were discovered, the designers moved the ailerons. Many aircraft now have ailerons located well inboard away from the more flexible and moving tip location. Mid-span ailerons are common on many aircraft now in the world's military and commercial fleets. Spoilers have become a common lateral control surface and when they are located on or close to the wing's torsional axis produce little or no elastic distortions to the wing. Determination of the hinge moments of such surfaces closely resemble the calculations required for the rigid airplane.

Pitch control surfaces have reduced the problem of aeroelastic losses by the use of single all-movable units. These surfaces do certainly deflect under air loads but the effect on hinge moment calculations and effectiveness has been to greatly reduce the problem of predicting these characteristics.

In some cases, at least, the rudder and vertical tail stiffnesses are determined from flutter requirements which require more stiffness and structure than the requirements for control power. For these cases, again, the hinge moment determination problem has been reduced for the surface is stiffer and more rigid than the minimum required for control.

Another design improvement has been used for trailing edge surfaces which has also reduced the aeroelastic problem. That is the use of segmented surfaces and multiple actuators. When a long surface is broken up into four or more segments then the overall distortions are reduced since the loads are more evenly distributed and inboard loads are not carried to outboard hinge locations.

In addition to these design improvements, control system designers usually added some extra moment capability to actuator design to allow for lack of precision of hinge moment calculations. This is why it is felt that complex and expensive methods of testing may not be necessary in all cases to determine aeroelastic and hinge moment characteristics. If the FLEXSTAB analytical method is successful, and it will be several years before we know if it is or not for we must have flight test data to establish true confidence, then perhaps the need for many of the present aeroelastic wind tunnel tests may diminish.

Each airplane program should closely examine its requirements before embarking on a large aeroelastic test program. Current system design capabilities may allow the use of control surfaces with relatively little loss from aeroelasticity and hence permit the determination of hinge moments with an accuracy approaching that of a rigid system. Elastic effects must continue to be determined but these effects may be diminished when compared to past problems.

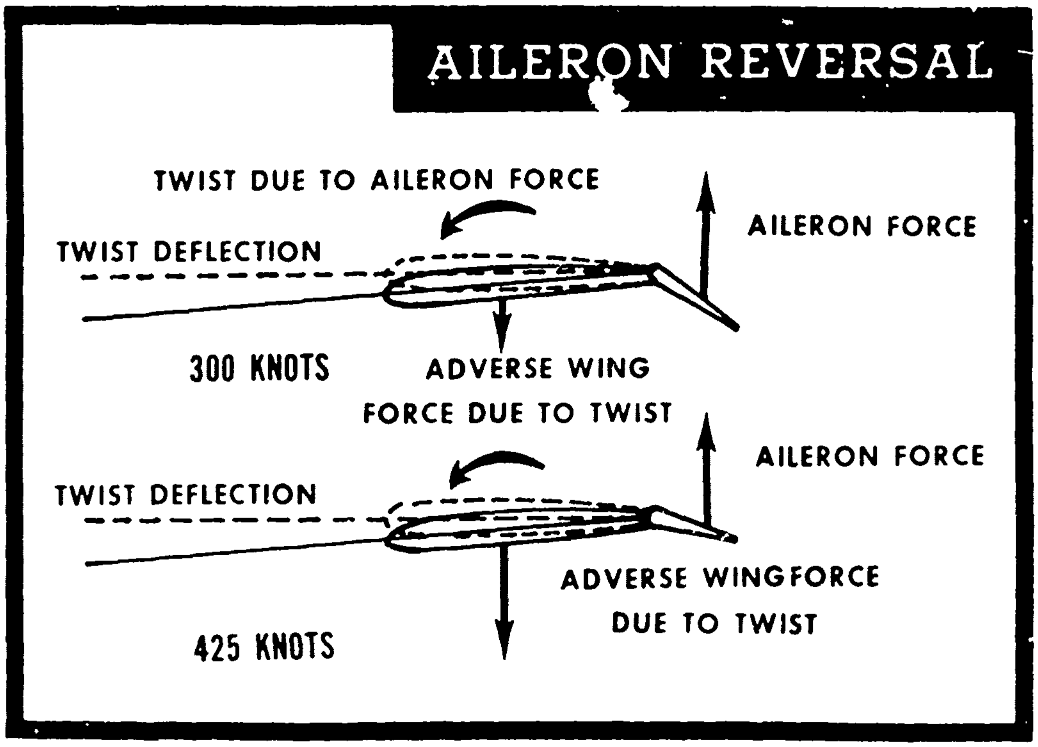

FIGURE 1

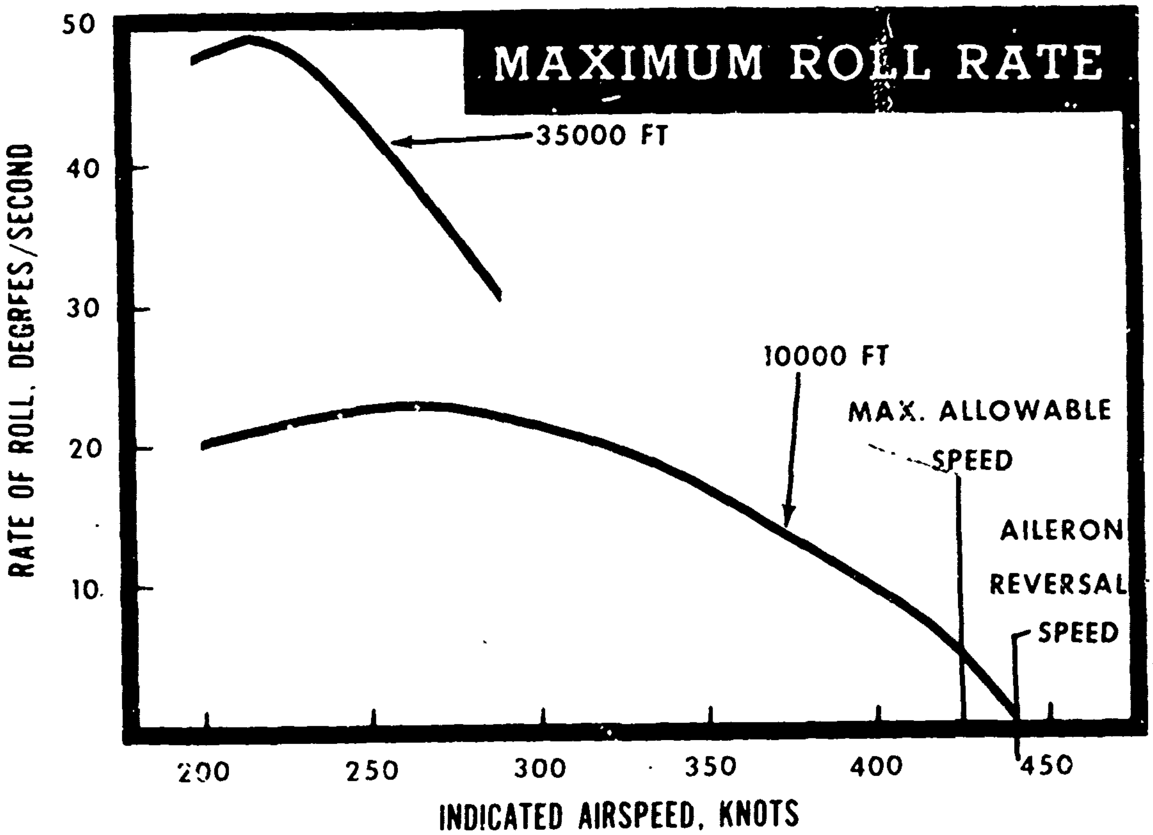

FIGURE 2

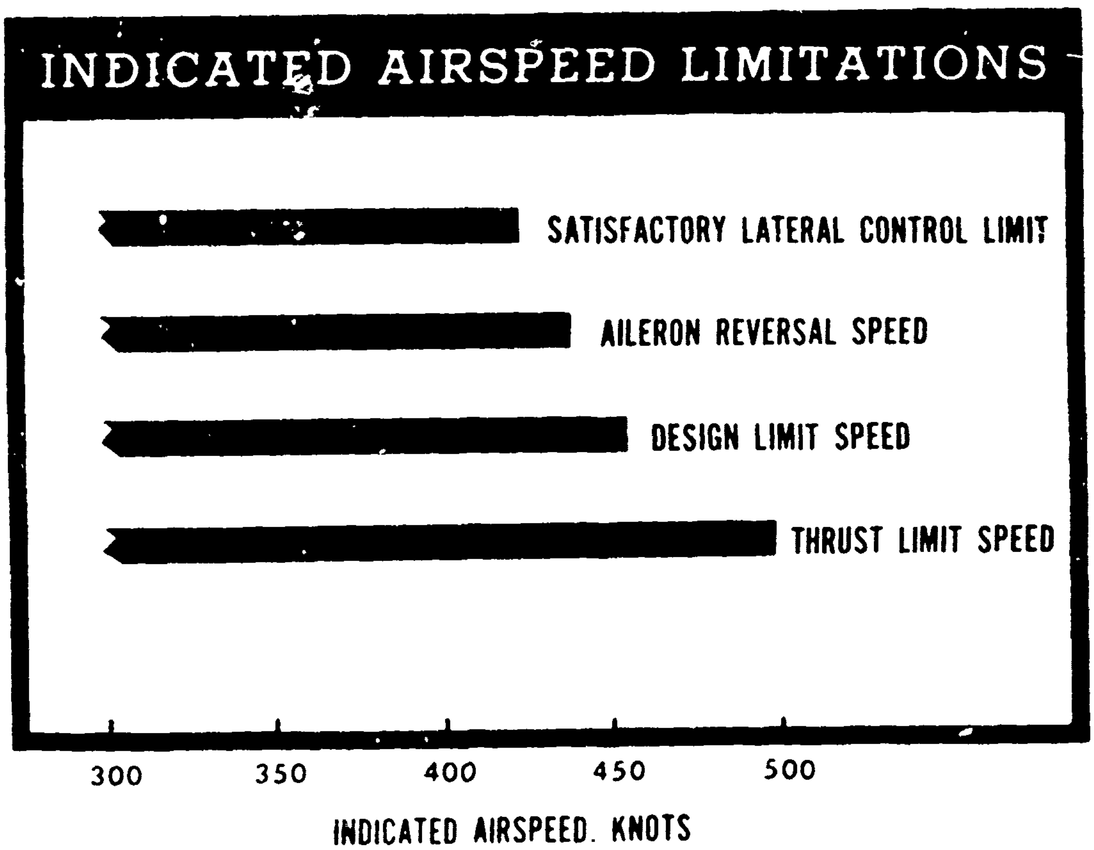

FIGURE 3

OPEN DISCUSSION

M.Hacklinger, Germany: I would like to offer a somewhat provocative comparison of the lecture of Mr Carlson with that of Mr Rainey: In my opinion, it is less dangerous from the airworthiness point of view to fly an aircraft beyond its aileron reversal speed – that is scheduling roll control with dynamic pressure – than to fly beyond the natural flutter speed of the main surfaces. In the first case, if you lose the artificial device, you have still a chance to neutralize the aileron and fly on rudder only; in the second case, if you lose action of the anti-flutter flaps, you are dead.

J.Carlson, USA: M.Hacklinger is correct, for it certainly is more dangerous to fly beyond the flutter speed than to fly beyond the aileron reversal speed. Even so, it still can be quite dangerous to fly beyond the aileron reversal speed. In that case, the pilot does not have the capability to fly the airplane according to his training and he may command a right roll in an emergency and the airplane will roll left. The results in this case may also be catastrophic if terrain following is being flown.

CONSIDERATIONS ON THE MANUAL FLIGHT CONTROL DESIGN OF A MILITARY TRANSPORT AIRCRAFT.

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SUMMARY

Large transport aircraft provided with manual control are still designed and flown nowadays, able to fulfill the present demanding Handling Qualities requirements.
This paper gives approach criteria to design a manual control system with reference to the typical characteristics of a military transport aircraft.
A guide to the choice of the manual control parameter comes out: practical problems mainly concerning the non-linear hinge moments behaviour and the control force scatter with the flight conditions are pointed out.
The peculiar aspects of matching manual operated ailerons with hydraulically driven spoilers are featured, and practical methods to design spring-tab control surface are dealt.

SYMBOLS

a	airplane lift-curve slope
a_t	tail lift-curve slope
a_2	tail lift-curve derivative respect to elevator angle
A_t	tail aspect ratio
b	span of airplane
b_o	hinge moment coefficient for zero incidence and zero surface angle
b_1	hinge moment coefficient derivative respect to incidence
b_2	hinge moment coefficient derivative respect to surface angle
b_3	hinge moment coefficient derivative respect to tab angle
$\bar{c}$	lenght of mean aerodynamic chord
$\bar{c}_a$	mean aileron chord
$\bar{c}_e$	mean elevator chord
c_3	tab hinge moment coefficient derivative respect to tab surface
C_{h_a}	aileron hinge moment coefficient
C_{h_e}	elevator hinge moment coefficient
C_{h_t}	tab hinge moment coefficient
C_L	lift coefficient
$C_{l_{\delta_a}}$	aileron effectiveness
$C_{l_{\delta_t}}$	rolling-moment coefficient derivative respect to tab angle
$C_{l_{\delta_{sp}}}$	spoiler effectiveness
C_{l_p}	roll damping coefficient
C_m	pitching moment coefficient
C_{m_δ}	elevator effectiveness
e	wheel radius
F	stick force, longitudinal control

F_p	wheel force, lateral control
g	gravity acceleration
G	elevator gearing
h	c.g. position, fraction of mean chord
h_n	neutral point of airplane, fraction of mean chord
h_{n_p}	neutral point of the wing-body combination, fraction of mean chord
h'_n	neutral point, stick free
h_m	manoeuvre point, stick fixed
h'_m	manoeuvre point, stick free
I_y	moment of inertia, pitch axis
k	parameter accounting for the effective position of the aileron
K	spring tab stiffness
l_p	distance between c.g. and pilot station
m	spring tab connecting ratio
M	mass of airplane
n	normal load factor
n/α, n_{z_α}	normal acceleration change per unit change in angle of attack
p	roll rate
$\bar{q}$	dynamic pressure
S	wing area
S_a	area of aileron aft of hinge line
S_e	area of elevator aft of hinge line
S_t	area of tail
V	true air speed
V_H	horizontal tail volume
W	aircraft weight
α	angle of attack
α_t	tail angle of attack
δ_a	angle of aileron
δ_a^*	angle of aileron when the torsion bar is not twisted
δ_e	angle of elevator
δ_t	angle of tab
δ_p	angle of the wheel
ε	angle of downwash
ρ	air density
ω_n, $\omega_{n_{sp}}$	short-period frequency

Subscripts:

L	left
R	right
i	initial
req.	required
obt.	obtained
f	final

1. INTRODUCTION

This paper deals with the experience gained in the design of the flight control system of the G 222 (Fig. 1), a Military Transport Aircraft of 26500 kg max gross weight, 9000 kg max payload, powered by 2 turboprop engines G.E. T 64 - P 4 D (2 x 3400 SHP) in the production version.
In the conception of this aircraft, a basic manual control system was considered advantageous because of its high reliability, low production cost and maintenance requirements.
While the longitudinal control is fully manual, the lateral control is basically designed on manual ailerons augmented by hydraulically powered spoilers, also able to operate as lift dumpers, in order to negotiate the small ailerons size with a large span flap system, still retaining outstanding roll control power in the full flight envelope.
The big size of the rudder, due to the requirements of very low minimum control speeds, has suggested the adoption of its fully powered operation.
The development of a manual control system for an aircraft like the G 222, is clearly more troublesome than the development of a powered one, because its design is strongly affected by the control free aircraft statics and dynamics in addition to the control fixed ones.
The hinge moments of the control surfaces, which play an important role in the control free aircraft behaviour, are usually of difficult prediction, even taking advantage of wind tunnel tests, because of their dependance on the chord pressure distribution and therefore on the Reynolds number.
Besides the typical non-linearities of the hinge moments versus control deflection and incidence, require a long time expense in order to reduce their magnitude and their negative effects on the aircraft handling qualities. As a consequence the setting up of the manual control system can be succesfully achieved only with a process of continuous alternate theoretical and experimental evaluation and checks, to be carried out also along the flight tests.

2. LONGITUDINAL CONTROL

- Apart from the stick-free dynamic stability, the difference between a manual and a powered control system can be brought to light by considering the response to a pilot stick maneuver.
  The ratio between the final and the initial stick force per pilot load factor in a step control maneuver is computed by taking the initial load factor at the pilot station:

$$\Delta n_i = \frac{\bar{q} S \bar{c} C m_\delta}{I_y} \cdot \frac{l_p}{g} \cdot \Delta\delta_e$$

  related to the initial pilot stick force:

$$\Delta F_i = G\, b_2\, \bar{q}\, S_e\, \bar{c}_e \cdot \Delta\delta_e$$

  which gives the initial stick force per "g", or the inverse of sensitivity:

$$\left(\frac{\Delta F}{\Delta n}\right)_i = G \frac{S_e \bar{c}_e}{S \bar{c}} \frac{I_y\, g}{l_p} \frac{b_2}{C m_\delta}$$

  The final stick force per "g" is given by the well known expression:

$$\left(\frac{\Delta F}{\Delta n}\right)_f = G \frac{W}{S} S_e \bar{c}_e \frac{b_2}{C m_\delta} (h-h'_m)$$

  Hence the ratio between the final and the initial stick force per "g" becomes:

$$\frac{(\Delta F/\Delta n)_f}{(\Delta F/\Delta n)_i} = \frac{h-h'_m}{I_y} \frac{W\, l_p\, \bar{c}}{g} = \frac{h-h'_m}{h-h_m} \frac{\omega^2_n}{n_{z\alpha}} \qquad (2,1)$$

  On the ground that the ratio $\frac{h-h'_m}{h-h_m}$ may be as low as 0,5 ÷ 0,8 , the manual controlled aircraft should be provided with higher control parameters than those required for powered controlled aircraft in order to maintain a satisfactory balance between the control force per "g" and the initial sensitivity.
  In fact the control force due to a typical negative b_1 in connection with the aircraft α response, is similar to that of an alleviating bob-weight and results in a lagged final load factor following constant control force application: by increasing the aircraft stability higher elevator deflections per α are required and thus control forces due to α are less important than those due to δ_e.
  Despite the above considerations are not fully relevant to a transport aircraft not requiring tight maneuvers, the G 222 longitudinal stability has been designed great in reference to $n_{z\alpha}$ (Fig. 2), what has turned out useful also in the aerial delivery maneuvers.

- A synthesis of the criteria defining the required aircraft stability and control level and the corresponding size of the horizontal tail can be drawn in dealing with the center of gravity limits versus the horizontal tail volume.
  One criterion requires that control force per "g" are within a defined range depending on aircraft category and limit load factor:

$$F_{n1} \leq \frac{\Delta F}{\Delta n} \leq F_{n2}$$

  By bringing out in the expression of $\Delta F/\Delta n$ the main design parameters of the horizontal tail it is obtained:

$$\frac{\Delta F}{\Delta n} \simeq G \frac{(S_e/S_t)^2}{d\alpha_t/d\delta_e} \cdot \left(\frac{\bar{c}}{l_t}\right)^{3/2} \frac{W\sqrt{S}}{\sqrt{A_t}} \frac{b_2}{a_t} \sqrt{V_H}\, (h-h'_m) \qquad (2,2)$$

  where the control-free maneuver margin is given by :

$$h - h'_m = h - h_{np} + h_{np} - h'_m$$

  the distance between the control-free maneuver point and the neutral point of the tail-off aircraft being proportional to the horizontal tail volume V_H and to the tail lift slope a_t :

$$h'_m - h_{np} = a_t V_H \left[\frac{1-d\varepsilon/d\alpha}{a} + \frac{\rho S l_t}{2M}\right]\left(1 - \frac{b_1}{b_2}\frac{d\alpha_t}{d\delta_e}\right) \qquad (2,3)$$

Fig. 3 illustrates the behaviour of the forward and rearward center of gravity limits, in compliance with the control force per "g" requirements, computed versus the horizontal tail volume with typical data of a transport aircraft. In the background of the same figure a network of constant elevator per "g" curves are traced, computed according to the following formulation:

$$\frac{\Delta\delta_e}{\Delta n} = \frac{C_L}{V_H \cdot a_t \, d\alpha_t/d\,\delta_e}(h-h_m) = -\frac{C_L}{a_2}\left[\frac{h-h_{np}}{V_H} + \frac{a_t}{a}\left(1-\frac{d\varepsilon}{d\alpha}\right)\right] \qquad (2,4)$$

The forward center of gravity positions are limited further on by the need to balance the aircraft in the extreme flight envelope conditions: the stall in ground effect or the aircraft rotation in take off are usually the most critical cases. Such limitations are superimposed on the Fig. 3, which at the end can address the choice of the horizontal tail size and configuration.

-A forward shift of the center of gravity limits is obtained by reducing the tail volume, unless the maximum control power doesn't impair the aircraft balance at $C_{L\,max}$.

-The range of the center of gravity limits may be increased mainly by raising the horizontal tail aspect ratio or by lowering the elevator hinge moment to control deflection derivative.
The Fig. 4 features the disadvantage of reducing the tail aspect ratio from 6,5 down to 4,5, which would shrink the CG range of the 25% - 30% for high V_H, while the CG forward limit due to A/C balance at $C_{L\,max}$ doesn't depend directly on the tail aspect ratio.
On the other hand a too large aspect ratio tail must be avoided owing to its sensitivity to the large angle of attack variations occurring behind the wing in extending the flaps.

-The setting of the hinge moment to control deflection derivative b_2 must be seen in combination with the derivative b_1 of the hinge moment to angle of attack, since the parameter $(1-\frac{b_1}{b_2}\frac{d\alpha_t}{d\delta_e})$ is an index of the elevator floating.. From a theoretical standpoint the most attractive solutions should be those leading to b_1 near zero, which would cut out the elevator floating, leaving therefore the "stick-free" neutral point about equal to the "stick-fixed" one and allowing a reduction in the horizontal tail size (10% in the Fig. 5) while keeping the same C.G. range. In the reality such solutions have a main drawback in the large hinge moment non-linearities matched with the horns or the very large hinge line set-back, which are the typical control configurations allowing almost complete b_1 balance (Fig. 6). Owing to the difficulty in managing control force non-linearities it is advisable to avoid these configurations, therefore retaining controls with resonable negative b_1, the main responsible for the difference between a manual and a powered control system.

-It is worthwhile to discuss more in depth, the aspects of the tail and elevator design able to affect the hinge moment behaviour, which is certainly the most delicate problem in a manual control design. The G 222 has been provided with an aerodynamic balanced elevator through a set back hinge line and a geared tab, able to cut the unbalanced hinge moment down to 1/6 or 1/8. A servo tab was discarded owing to possible flow separation difficulties at high incidences or elevator deflections and a spring tab was considered unsuitable because it would have stressed the existing discrepancy in the G 222 control forces per "g" at high and low airspeeds.

-In order to increase the C.G. limits range (see Eq. 2, 2), the total elevator to horizontal tail area ratio should be kept as low as allowed by the control power requirements. Then the alternative in the design of a geared tab elevator is between a small overhang type aerodynamic balance supplemented by a large chord tab and a more important aerodynamic balance coupled to a small tab. The effectiveness-hinge moment ratio, the elevator floating tendency, the occurence and the magnitude of non-linearities are the main criteria to guide the choice. In the Fig. 7 the two dimensional elevator effectiveness $d\alpha_t/d\,\delta_e$ and the floating tendency $\frac{d\alpha_t}{d\delta_e}\frac{b_1}{b_2}$ are plotted, for the two above mentioned elevator configurations, as a function of a two-dimensional control force gradient parameter $\bar{c}_e^{\,2}\, b_2/(d\alpha_t/d\,\delta_e)$, varying with the geared tab ratio. The elevator provided with overhang appears as a better solution because of a higher elevator effectiveness and mostly because of a lower floating tendency. A different conclusion should be drawn for a spring tab, owing to its capability to oppose the elevator floating. As said before, the limit to an increase in the overhang extension is due to the relevant non-linearities occurring at the large elevator deflections, because of premature stall phenomena or of hinge moment reversals (Ref. 1)

-Apparently the $\frac{\Delta F}{\Delta n}$ expression suggests the idea that in a manual control system the control force per "g" are constant thoughout the airspeed range: unfortunately that is not the case because of the remarkable differences in the aerodynamic derivatives with the angle of attack, the thrust coefficient and the aircraft configuration. A first impression of the flight condition effect can be caught by the exam of the Fig. 8 showing an increase of the original G 222 control force per "g" gradient at low speed, what besides is allowed by the Military Specifications.
The reason of such a behaviour stays in the increasing aircraft stability at high angle of attack, due to the horizontal tail position in the wing wake, and in the typical non-linearity of the hinge moment to elevator gradient, increasing with the elevator deflection.

- An idea of the hinge moment curve shapes is given by Fig. 9 where, moreover, the optimum balance range of the hinge moment versus elevator angle is seen to shift to more negative elevator deflections as the angle of attack increases.

- The camber of an elevator elliptic nose can be used to control the hinge moment curve shape: a positive nose camber tends to shift the optimum balance range toward negative elevator deflections, leading to an alleviation of control force gradient at low speeds and to the opposite at high speed. (Fig. 10).

- Similar effects can be obtained connecting control column and elevator through a non-linear linkage, designed in a way to oppose the unpleasant non-linear characteristics of the aerodynamic control.

- By optimizing a suitable set of vortex generators, fitted on the tailplane in order to remove local flow separations at large elevator deflections, outstanding improvements on the linearity of the elevator hinge moment characteristics may result, as it is shown in Fig. 10.
  A remarkable increase on the elevator effectiveness can take place as well owing to the airflow improvement (Fig. 11), and thus the control force versus load factor behaviour can take a double advantage from the adoption of vortex generators.

3. LATERAL CONTROL

3.1. General remarks

- The late military specifications of U.S.A. are certainly an useful guide in the definition of lateral manoeuvrability requirements. According to MIL-F-8785 B, the roll performances of a transport aircraft shall allow the achievement of 45 degrees bank angle in 1.9 sec. in cruise and 30 degrees in 1,8 sec. during the approach, following an abrupt imput on the control wheel.
  In the same time the maximum control force shall never be greater than 50 lb in all flight phases except in approach, landing and take-off, when the maximum control force shall be halved to allow piloting with one hand. In order to define the maximum allowable pilot work, the wheel throw necessary to meet the roll performance requirements is limited to 60 degrees or to 80 degrees for completely manual systems.
  The fulfilment of such severe requirements for a military transport aircraft like the G 222 depends on the possibility to find a satisfactory trade-off between low control forces and high control power.
  From the point of view of the roll performances, the need of high roll power, expecially at low speed, has been met adding hydraulically driven spoilers to the manual control ailerons.
  In fact the spoilers are particularly effective when the flaps are completely lowered, as in the approach and landing configuration.
  The same spoiler surfaces are usefully available as lift-dumpers after touch-down.

- From the point of view of the aircraft roll response to the pilot control input, attention must be paid to the spoiler to aileron operating connection; a typical inconvenience is a sudden change of response sensitivity occurring at the spoiler deflection start, particularly when the flaps are down.
  It is therefore advisable to avoid large dead zones in the spoiler operation and the corresponding low response sensitivity at control wheel position around the zero.
  The adoption of spoilers, matched to manual ailerons in such a way, results in peculiar problems of designing the aileron aerodynamic balance, which are in connection with the typical behaviour of the spoiler effectiveness changing remarkably with the flaps deflection.
  In the case of the G 222 a convenient solution to said problem was found in the installation of ailerons provided with spring tabs, as discussed below.

3.2. Geread-tab design

The geared-tab is the most attractive device for the balance of the hinge moments, due to its easiness of design and construction.
Significant reductions of b_2 can be achieved with geared-tab, while the corresponding reduction of b_1 is usually small or negligible.
Let's see in more detail the design of a geared-tab for the ailerons similar to those of the G 222, acting in parallel to the spoilers.
Supposing that the roll manoeuvre will concern only one degree of freedom, the equilibrium of roll moments during steady state rolling is:

$$C_{l_{\delta_a}} \delta_a + C_{l_{\delta_t}} \delta_t + C_{l_{\delta_{sp}}} \delta_{sp} + C_{l_p} \left(\frac{pb}{2V}\right) = 0 \tag{3.1}$$

The wing elix angle ($\frac{pb}{2V}$) can be easily evaluated from the required time to 45 or 30 degrees bank.

Rearranging Eq. 3.1. the aileron angle required for the specified manoeuvre as function of the geared-tab ratio is obtained:

$$\delta_{a_{req.}} = -\left(\frac{pb}{2V}\right) \frac{C_{l_p}}{C_{l_{\delta_a}}} \frac{1}{\left(1 + \frac{C_{l_{\delta_t}}}{C_{l_{\delta_a}}} \frac{\delta_t}{\delta_a} + \frac{C_{l_{\delta_{sp}}}}{C_{l_{\delta_a}}} \frac{\delta_{sp}}{\delta_a}\right)} \tag{3.2}$$

Fig. 12 shows the required aileron angle versus the tab gear ratio, not accounting for the control force requirements, for the extreme flight conditions: cruise high speed and landing with flaps fully extended. The aileron requirements appear to be about the same because the lowered flaps in the low speed condition grow remarkably the spoiler control power.
For the same two flight conditions it is also defined the maximum allowable pilot work, which shall equate the hinge moment work of the two ailerons.
An expression for the required hinge moment of the single aileron can be given by:

$$C_{h\,a_{req.}} = \frac{e\,F_p\;\delta_p}{2\,\bar{q}\,S_a\,\bar{C}_a\;\delta_{a_{req.}}} \tag{3.3}$$

Clearly $C_{h_{a_{req.}}}$ is a function of the tab gear ratio as well as $\delta_{a_{req.}}$ is, and shall equalize the actual obtained hinge moment coefficient:

$$C_{h_{a_{obt.}}} = b_o + \left[b_1\,k\,\left(\frac{pb}{2V}\right) / \delta_a + b_2 + b_3\,\frac{\delta_t}{\delta_a} \right]\,\delta_a \tag{3.4}$$

Fig. 13 shows $C_{h_{a_{req.}}}$ and $C_{h_{a\,obt.}}$ versus the tab gear ratio for the two different flight conditions; at low speed $C_{h_{a\,obt.}}$ is remarkably less than at high speed, because of the value of $\left(\frac{pb}{2V}\right) / \delta_a$ which is increased by the improved spoiler effectiveness with flaps deflected.
It is clearly seen that at high speed a tab-gear ratio higher than about 0.6 shall be chosen, but this same value should cause the reversal of the aileron hinge moment at low speed.
The disappointing conclusion is that a tab gear ratio variable with the speed should be made available and therefore a conventional geared tab can't be used.
The results here presented are typical of aircraft having outstanding spoiler control power in relation to the aileron one.
The problem has been solved, for the G 222, providing the ailerons with spring-tabs; the other solution of fitting very low b_1 ailerons to the aircraft was considered risky for the difficulty of reducing the associated non-linearities and for the required long time development.

3.3 The "spring-tab" balance design

Fig. 14 shows the outline of a control surface with a simple spring-tab device, whose working is well known (Ref. 5) and which has been taken as reference in this paper.
The equilibrium equations of the complete lateral control system are the following:

$$e\,F_p\;\delta_p = \bar{q}\,S_a\,\bar{c}_a \left[C_{h_{a_R}}\,\delta^*_{a_R} + C_{h_{a_L}}\,\delta^*_{a_L} \right] \tag{3.5}$$

$$\delta_{t_R} = -\,m\,(\delta^*_{a_R} - \delta_{a_R}) \qquad\qquad \delta_{t_L} = -\,m\,(\delta^*_{a_L} - \delta_{a_L}) \tag{3.6}$$

$$\bar{q}\;\frac{K}{S_a\;\bar{c}_a}\;(\delta_{a_R} - \delta^*_{a_R}) = C_{h_{a_R}} + m\,C_{h_{t_R}} \tag{3.7}$$

$$\frac{K}{\bar{q}\;S_a\;\bar{c}_a}\;(\delta_{a_L} - \delta^*_{a_L}) = C_{h_{a_L}} + m\,C_{h_{t_L}} \tag{3.8}$$

where K is the stiffness of the torsion bar and

$$m = \left(\frac{\partial\delta_t}{\partial\delta_a}\right)_{\delta^*_a = 0} = -\left(\frac{\partial\delta_t}{\partial\delta^*_a}\right)_{\delta_a = 0} \tag{3.9}$$

is the connecting ratio of the spring tab.
The main point in the spring-tab design is the determination of the two parameters K and m in order to meet the requirements in terms of roll control effectiveness and of peak control force.

- As a first approach, let's see the spring-tab design in a very simplified manner just to show clearly the peculiar characteristics of this device.
  The simplifying assumptions are the following:
  a) linearity of aileron hinge moment
  b) neglecting of tab hinge moment about its hinge line
  c) equality of right and left required aileron angle (as absolute value).
  The required aileron angle, as function of the ratio between tab and aileron, is still that of fig. 12 like in the geared-tab design.

The results are shown in fig. 15 for high and low speed respectively.
Pilot control force and stiffness K have been plotted as a function of the ratio between tab and aileron for several values of the m parameter.
Choosing the maximum allowable control force and wheel throw in the high speed case, it is possible to select several couples of K and m fulfilling the roll response as well as the control force requirements. The corresponding value of the tab/aileron ratio is always very close to the value necessary in the geared-tab design.
At low speed, for the same couple m and K, the pilot control force is much lower, due to the stronger effect of spoiler wich is hydraulically operated, but doesn't change sign like in the geared-tab solution. Really the spring-tab device is self-adaptive to the hinge moment change and the tab/aileron ratio varies according to the tab unbalanced hinge moment curve, thus avoiding to result in control force reversal.

- Removing the previous simplifications, the complete set of equilibrium equations shall be solved. This is possible either in graphical way or using digital iterative computations; for dynamic investigations, the simulation in an analog or digital computer shall be carried out.
  Fig. 16 shows the graphical calculation of the obtained left aileron and tab angles corresponding to 10° required aileron when the local incidence is that attained during positive steady-state roll.
  It has been assumed that the tab hinge moment about its own hinge line can be expressed as:

$$C_{h_{t_L}} \simeq c_3 \, \delta_{t_L} \qquad (3.10)$$

therefore accounting for the Eq. 3.6, equilibrium equation (3.8) becomes:

$$\lambda(\delta_{a_L} - \delta^*_{a_L}) = C_{h_{a_L}} \qquad (3.11)$$

where:

$$\lambda = \frac{K}{\bar{q}\, S_a \, \bar{c}_a} - m^2 \, c_3 \qquad (3.12)$$

Equivalent graphical construction to the Fig. 16 can be easily arranged in the case the tab hinge moment C_{h_t} is available from wind tunnel tests as a function of the angles of incidence and aileron as well as of the tab angle.
Making use of such kind of calculations, and assuming a given linkage between wheel and ailerons, we can obtain the results of Figs. 17 ÷ 20 where the pilot control force and corresponding rolling moment coefficient due to ailerons and tabs are shown as function of the wheel angle for the imposed steady-state roll-rate and for various configurations of m and K.
The same plots cover the roll moment coefficient required from ailerons whose amount comes down at increasing wheel angles because of the associated spoiler contribution to roll control power.
Equating available and required roll moment coefficient, the pilot control force can be shown (see Fig. 21 - 22) as function of the parameters m and K for the required value of the steady state roll rate at low and high speed flight conditions.

- While such plots stress the sensitivity of the lateral handling characteristics to the mechanical connection of the spring tab, similar diagrams can be made up to investigate the effect of the spring tab aerodynamic balance on the control force per roll rate behaviour with the aircraft speed.
  While the lack of spring tab aerodynamic balance should tend theoretically to increase the control force to roll rate gradient at high speed, the experience has shown that the measured spring tab hinge moment may be very different from the theoretical assumptions and may reverse the mentioned predicted influence of the tab aerodynamic balance.

- Anyhow the choice of the spring tab mechanical and aerodynamic design parameters shall be done not only from the handling qualities standpoint but accounting also for the flutter limitations.
  This is an argument out of the aim of the present paper: it may be worthwhile just to mention that the aerodynamic balance and the connecting ratio of the spring tab can be the parameters affecting the flutter speed more than the torsion bar stiffness.

4. CONCLUSION REMARKS

Some of the problems which are at the basis of a manual control design have been discussed with reference to the flight characteristics of a military transport aircraft like the G 222.
With regard to the longitudinal control, when fitting a geared tab balance elevator, the following optimization design criteria have been shown:

1) large horizontal tail aspect ratio
2) low b_1
3) low elevator to horizontal tail ratio
4) large overhang elevator balance

Said criteria must be considered as trends to be taken with the reserves discussed in the paper, mainly concerning the problem of non-linearities, which is to be coped with by a careful aerodynamic design and, if required, by an ingenious mechanical non-linear linkage.
Some peculiar aspects of the control characteristics of manual operated ailerons augmented by powered spoilers are dealt with.
Geared tab balanced ailerons, designed to meet control force requirements at high speeds, may come across to hinge moments reversals in flaps down flight conditions.
The spring tab is a convenient solution to this problem: design methods have been given and influence of the spring tab mechanical control parameters have been shown.

5. REFERENCE

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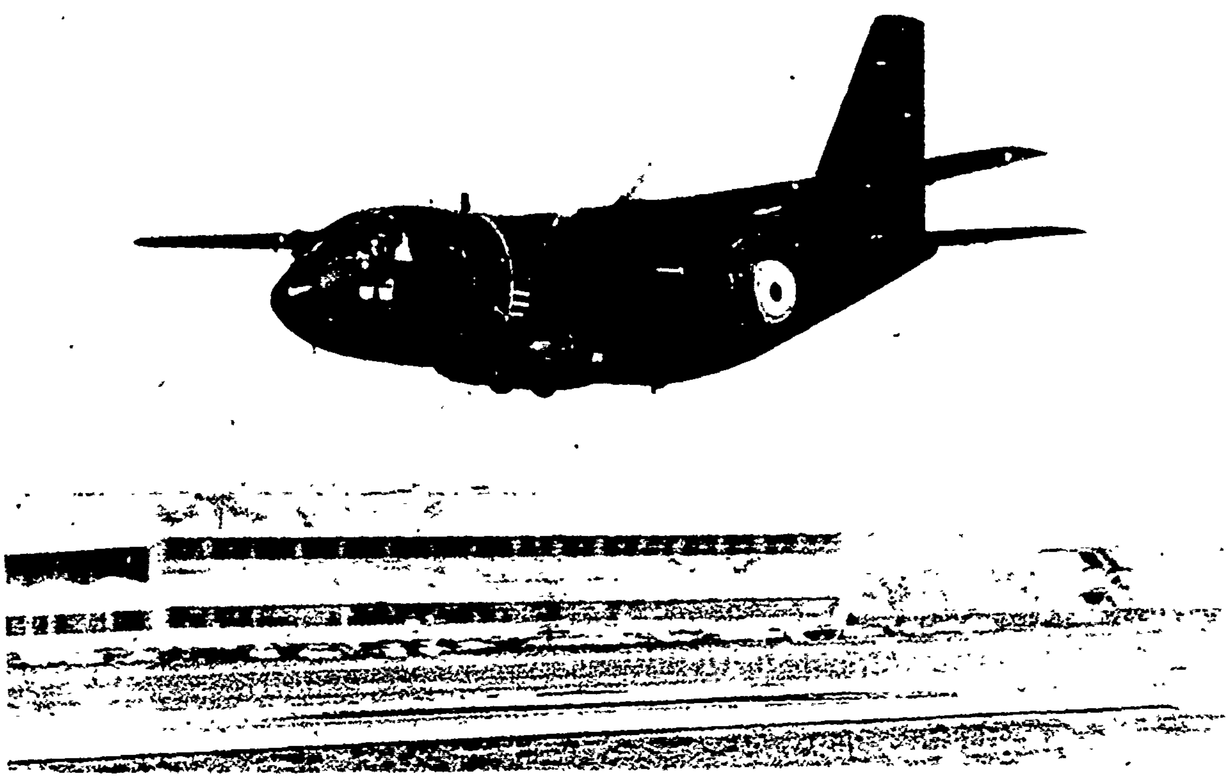

FIG. 1- G 222 AIRPLANE

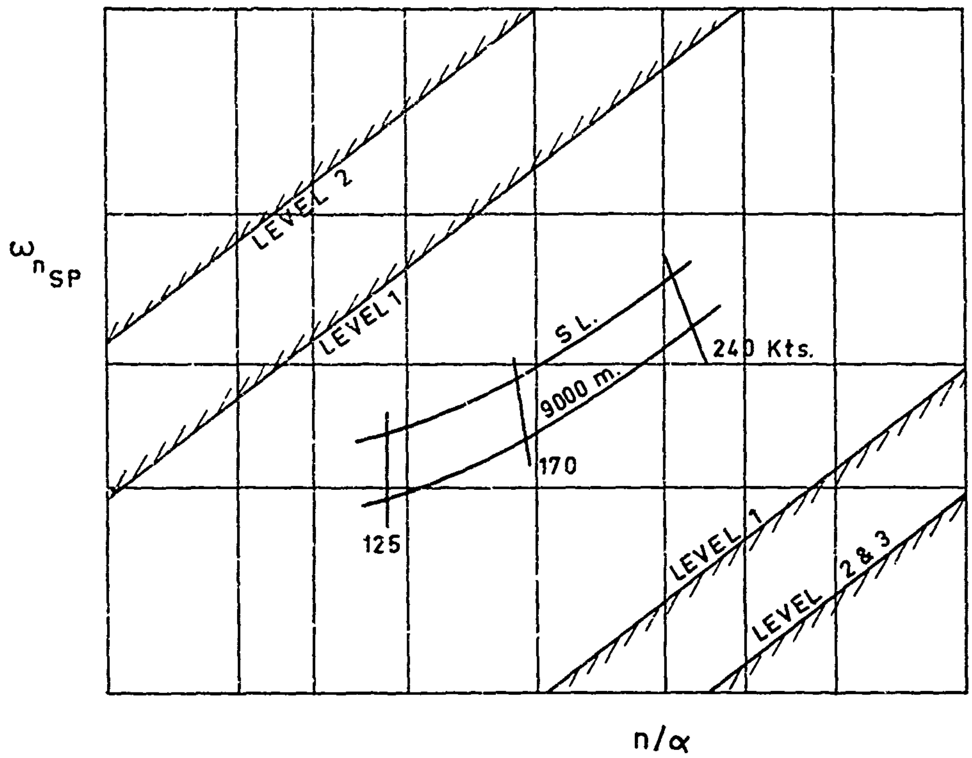

FIG. 2 - G.222 SHORT-PERIOD CHARACTERISTICS (CRUISE CONF. - CONTROL FIXED)

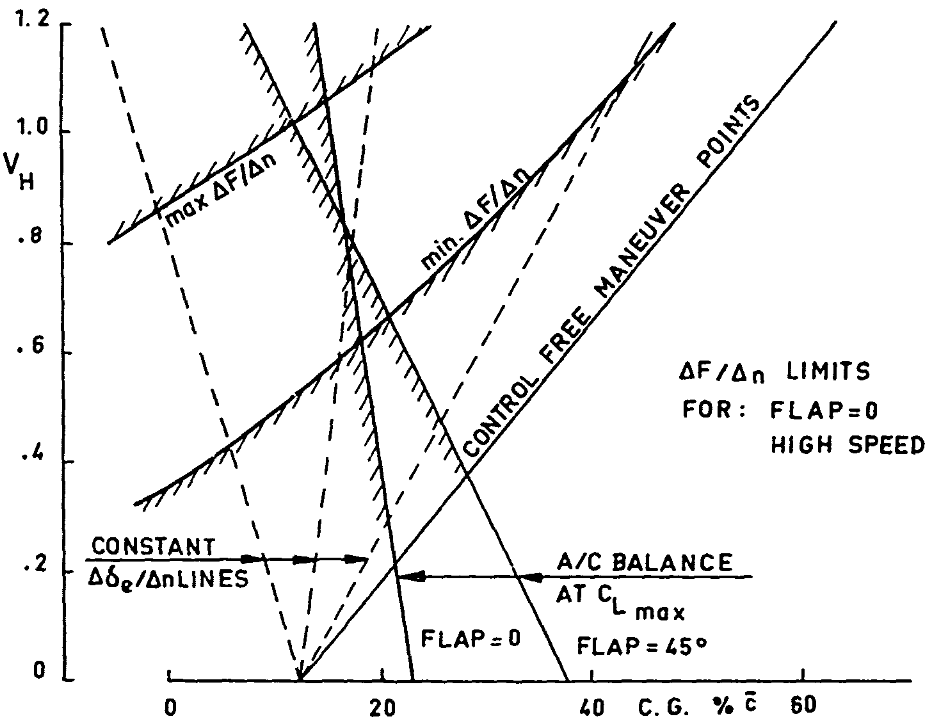

FIG. 3 - CENTER OF GRAVITY LIMITS VERSUS HORIZONTAL TAIL VOLUME

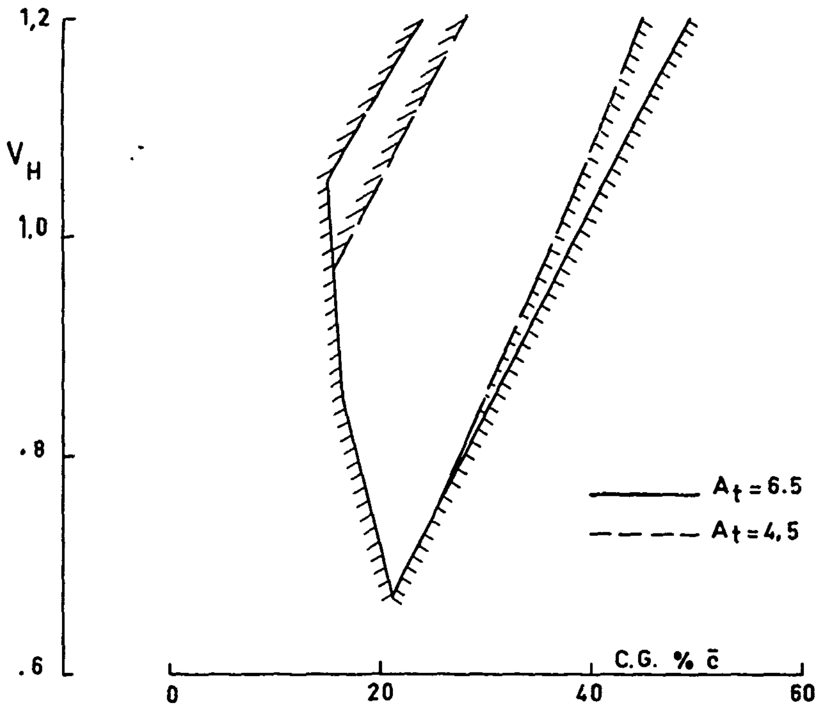

FIG. 4 - INFLUENCE OF HORIZONTAL TAIL ASPECT RATIO ON CENTER OF GRAVITY LIMITS

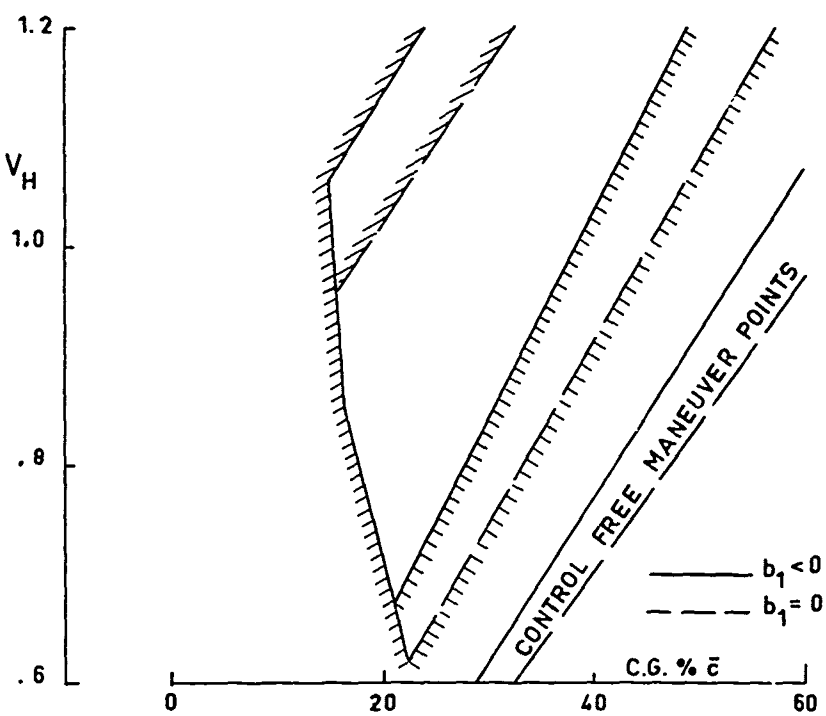

FIG. 5 - INFLUENCE OF b_1 ON CENTER OF GRAVITY LIMITS

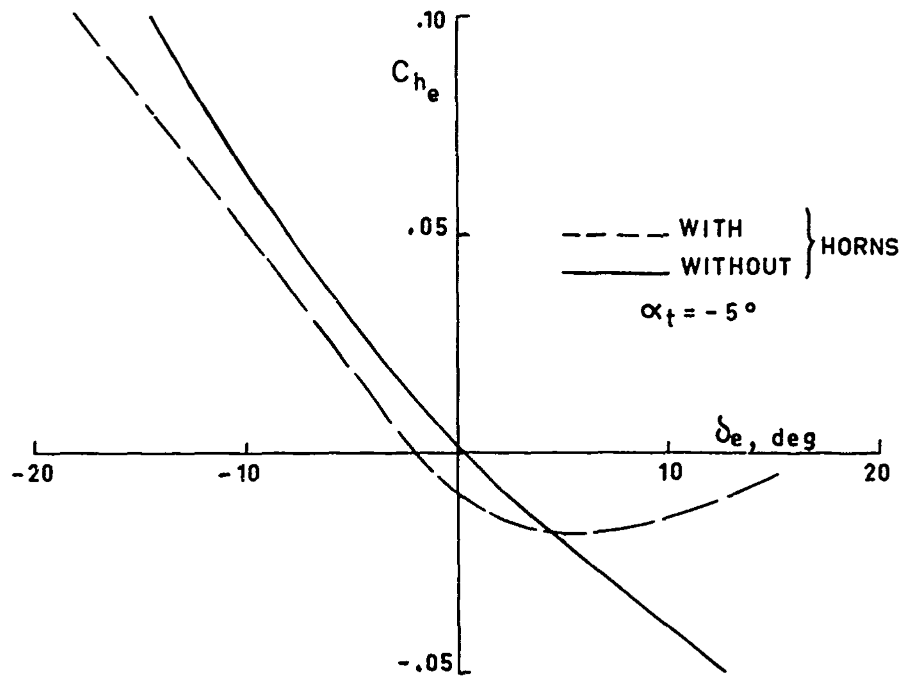

FIG. 6 - TYPICAL EFFECT OF HORNS ON ELEVATOR HINGE MOMENT BEHAVIOUR

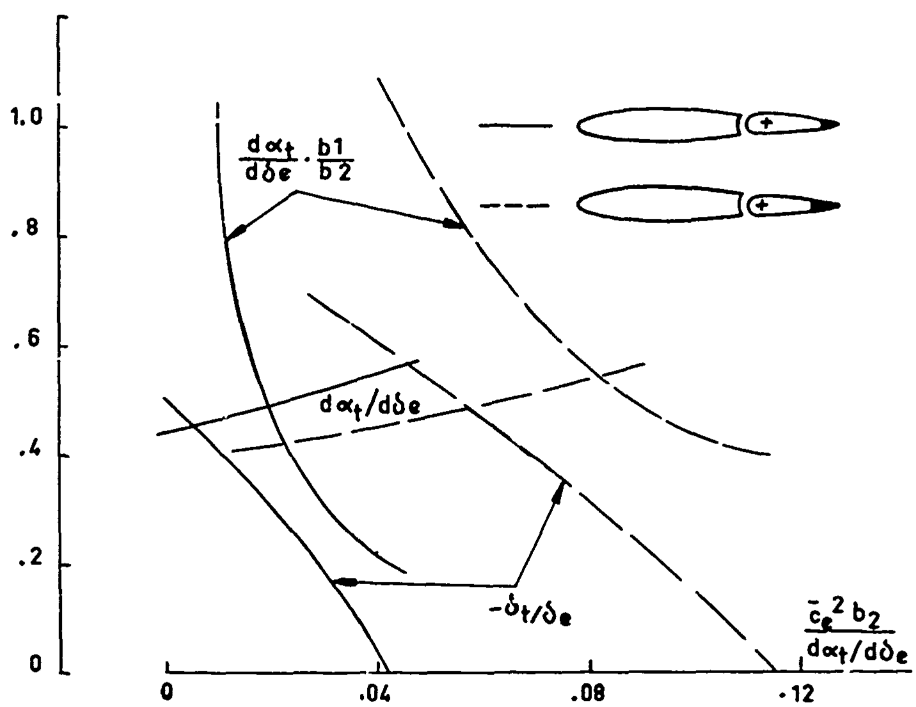

FIG.7 - TWO-DIMENSIONAL CONFIGURATION INFLUENCE ON ELEVATOR EFFECTIVENESS AND FLOATING TENDENCY

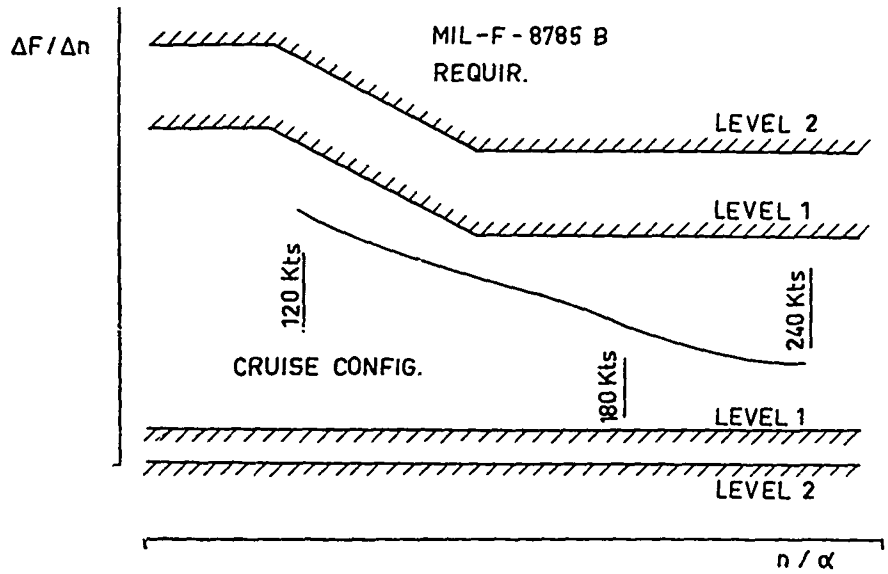

FIG. 8 -CONTROL FORCE GRADIENT VARIATIONS WITH SPEED

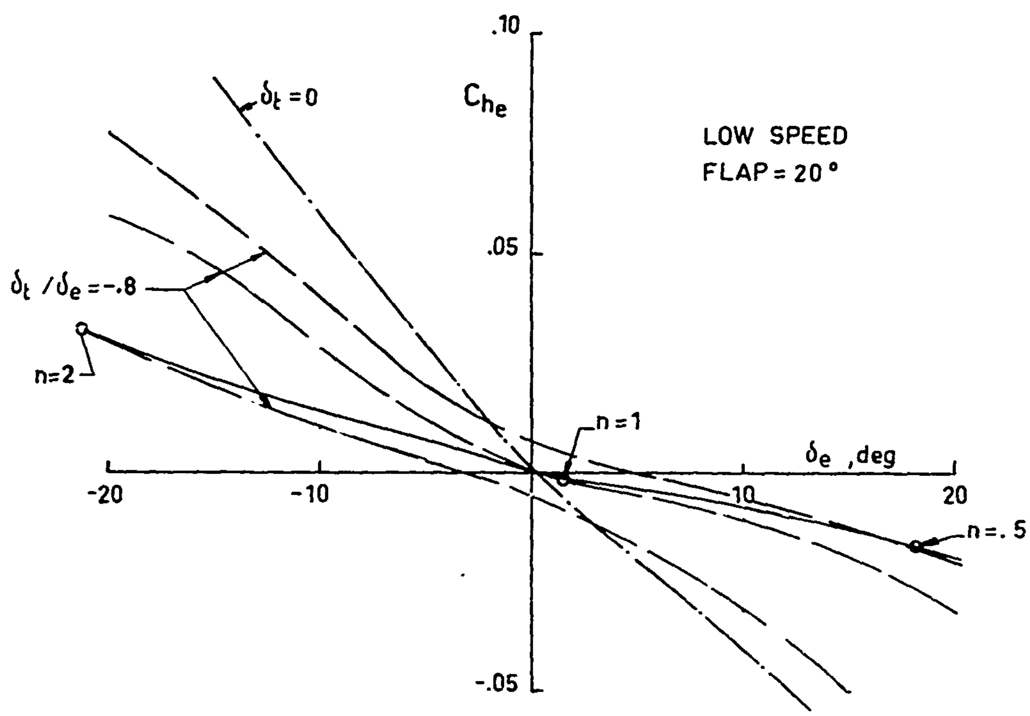

FIG. 9 - TYPICAL ELEVATOR HINGE MOMENT CHARACTERISTICS

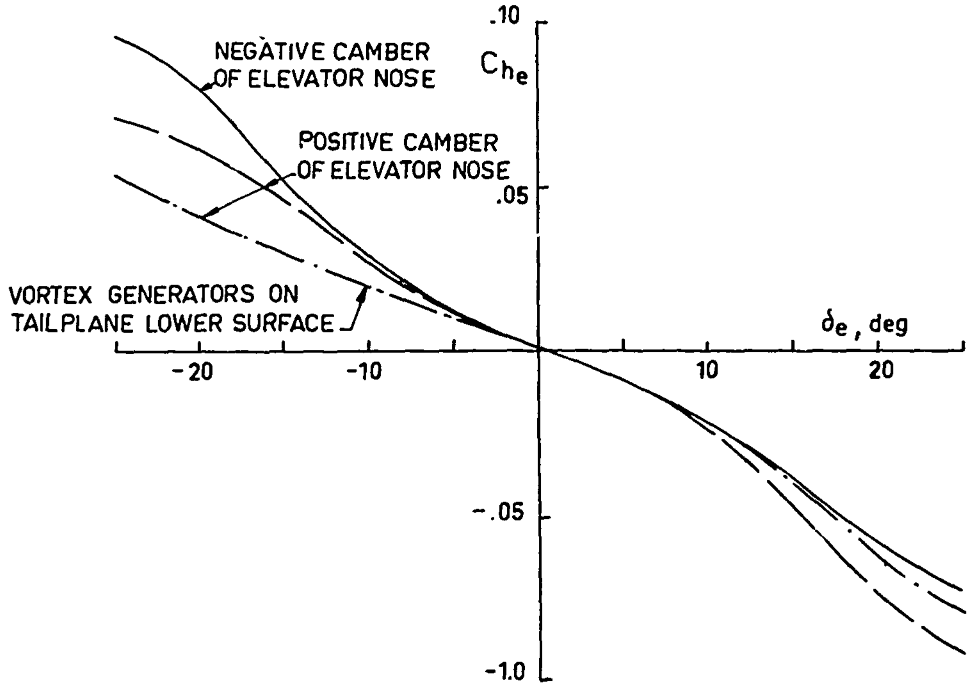

FIG. 10 - INFLUENCE OF ELEVATOR NOSE CAMBER AND VORTEX GENERATORS ON HINGE MOMENT BEHAVIOUR

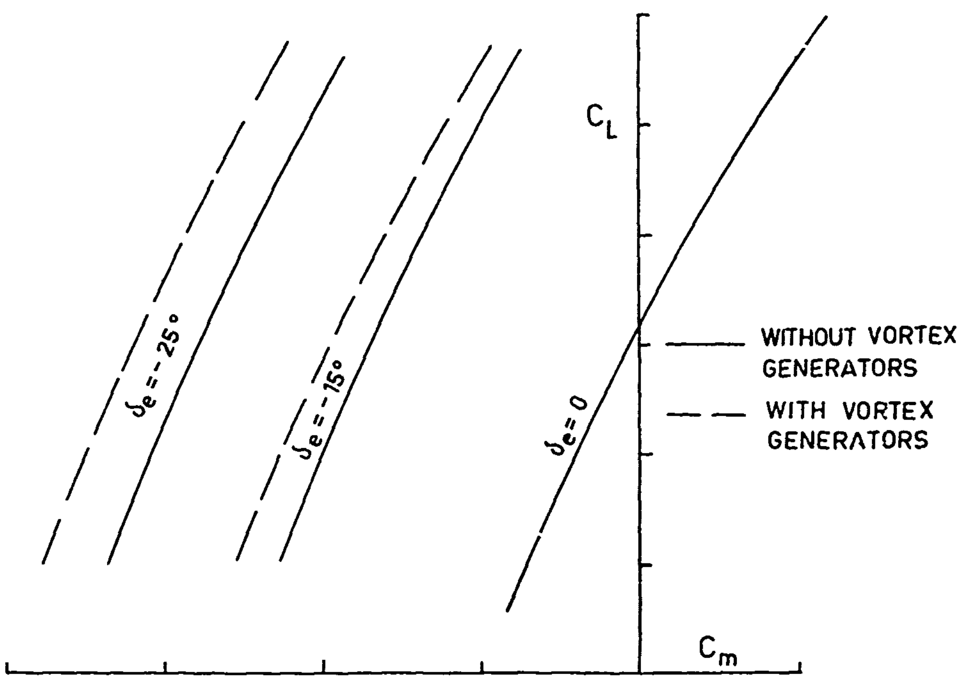

FIG.11 - EFFECT OF VORTEX GENERATORS UNDER TAILPLANE ON ELEVATOR EFFECTIVENESS

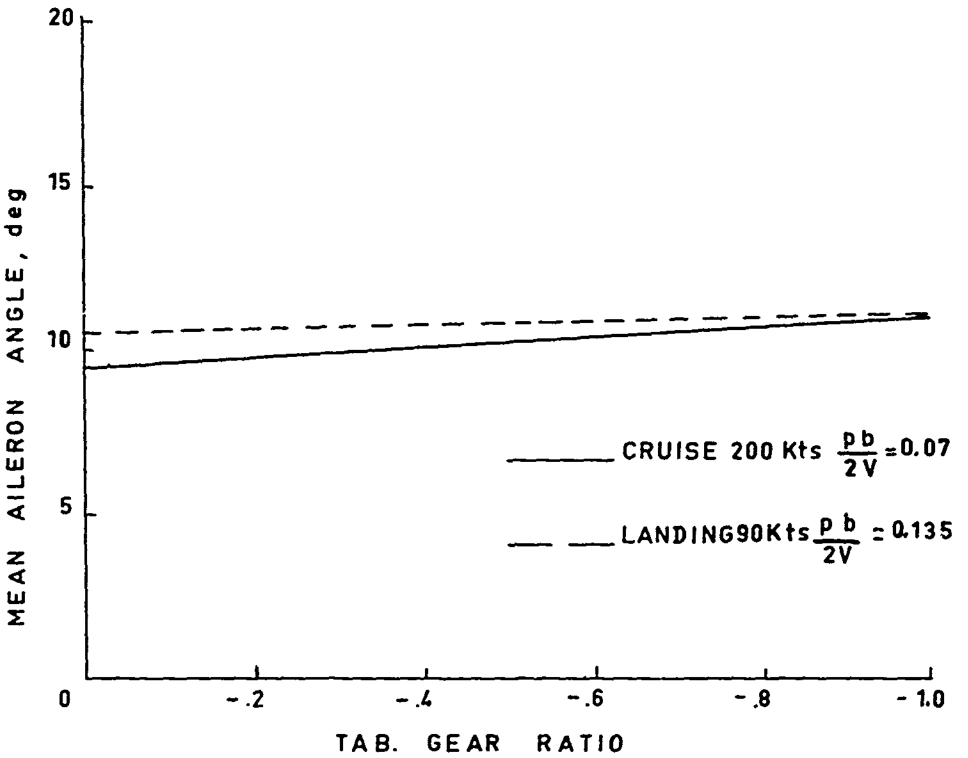

FIG. 12 - REQUIRED AILERON ANGLE VERSUS TAB GEAR RATIO

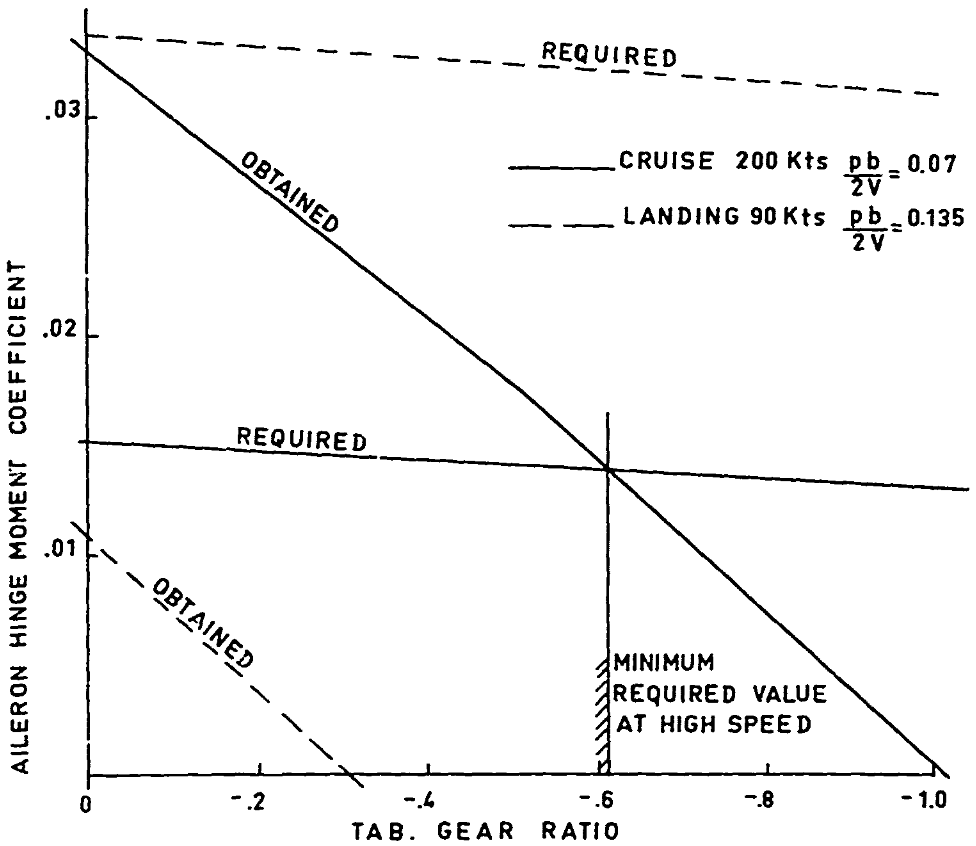

FIG. 13 - REQUIRED AND OBTAINED AILERON HINGE MOMENT VERSUS TAB GEAR RATIO.

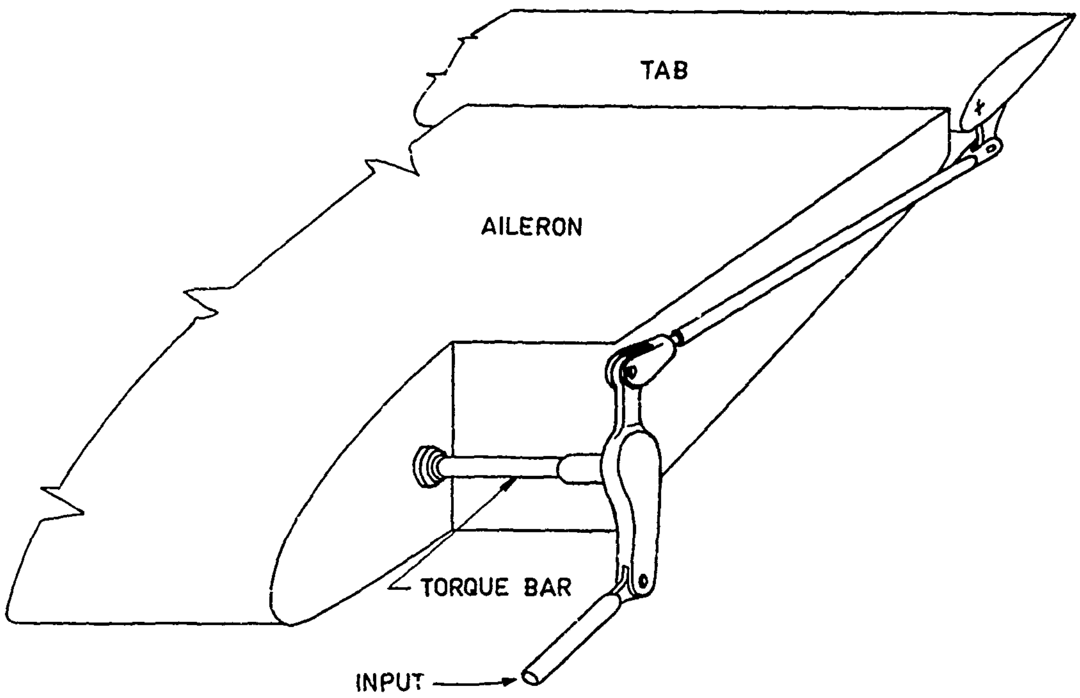

FIG.14- SKETCH OF AILERON SPRING-TAB

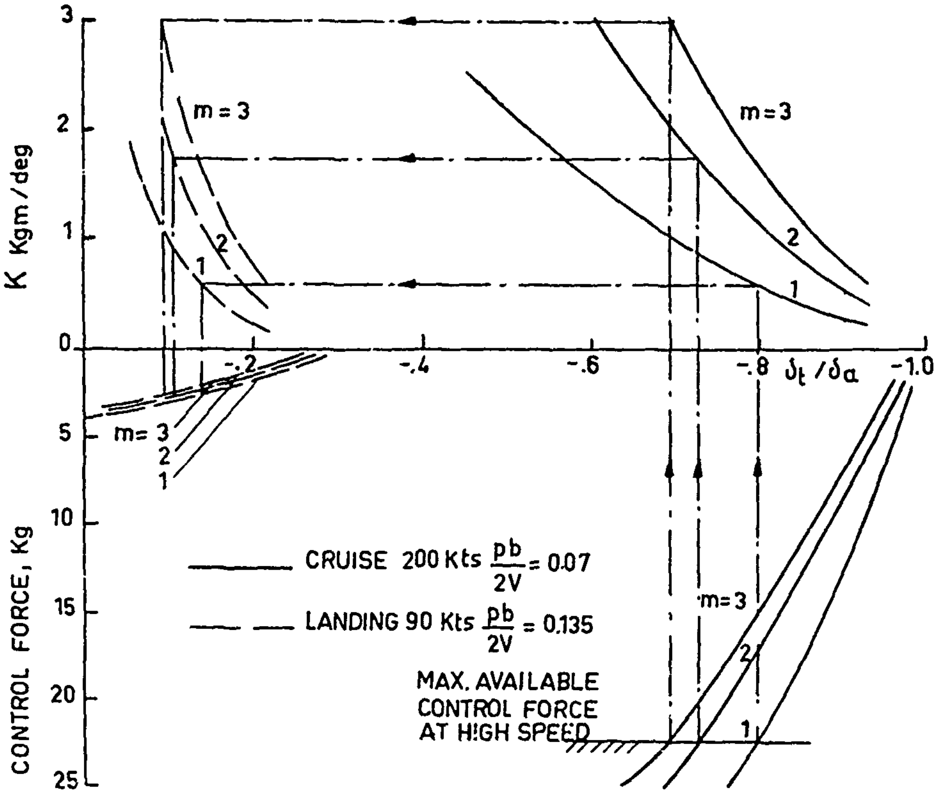

FIG.15 - DEFINITION OF AILERON SPRING-TAB CHARACTERISTICS
SIMPLIFIED CALCULATIONS

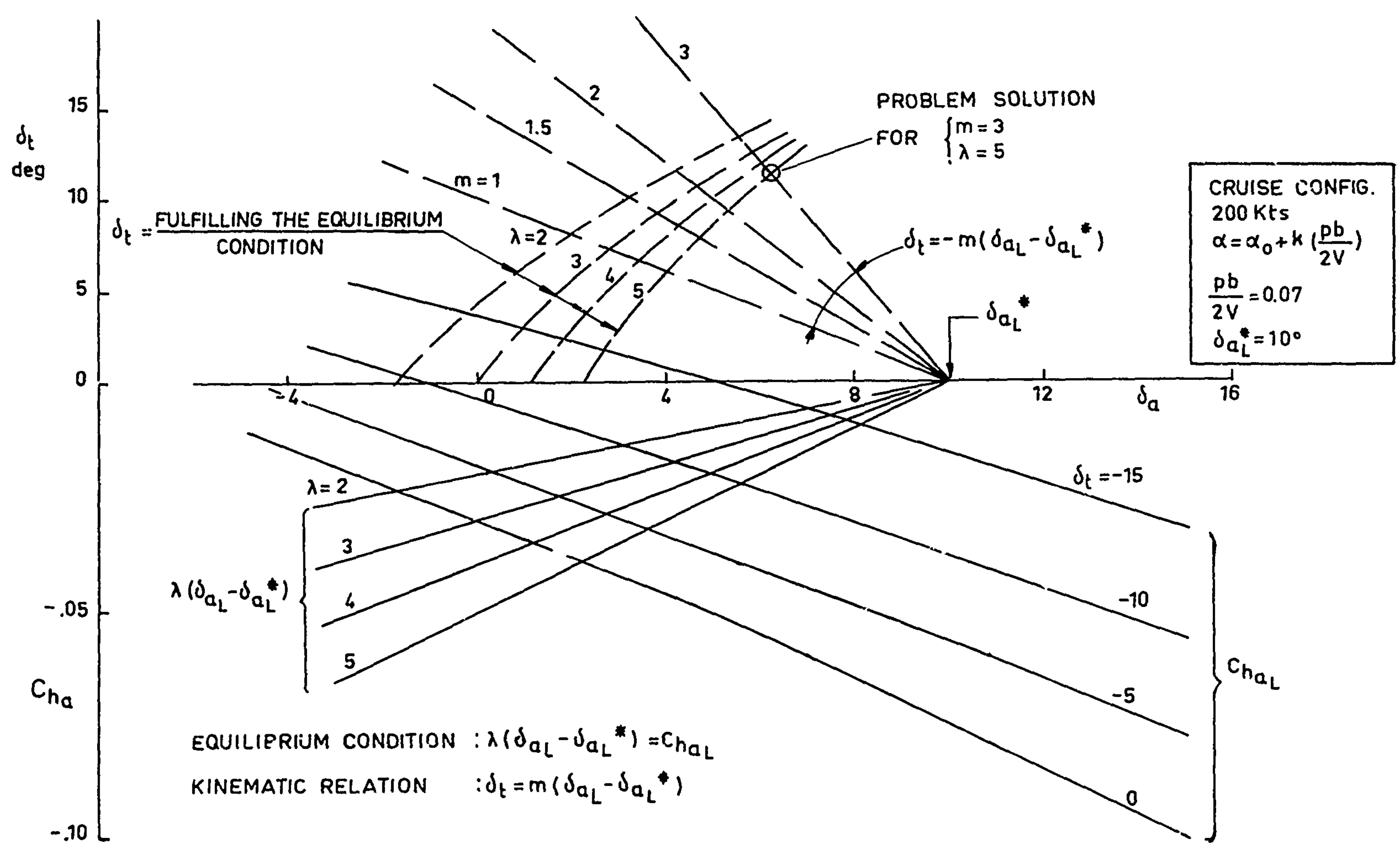

FIG. 16- GRAPHICAL CALCULATION OF OBTAINED AILERON AND SPRING-TAB ANGLES

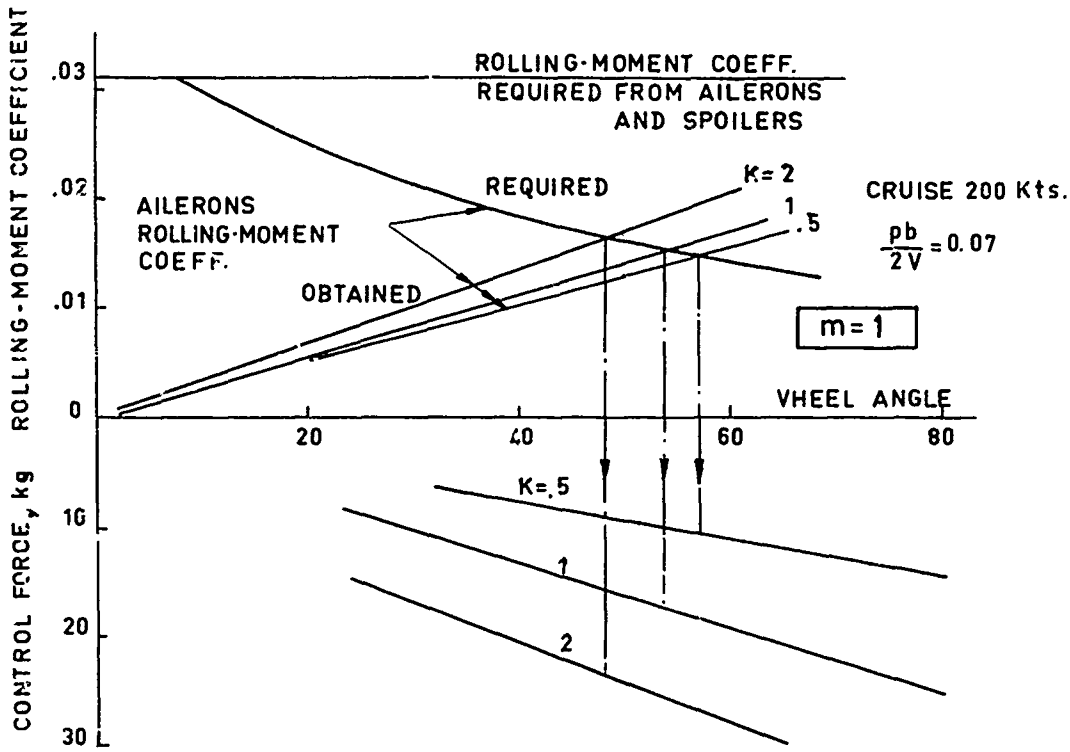

FIG. 17 - DEFINITION OF AILERON SPRING-TAB CHARACTERISTICS.

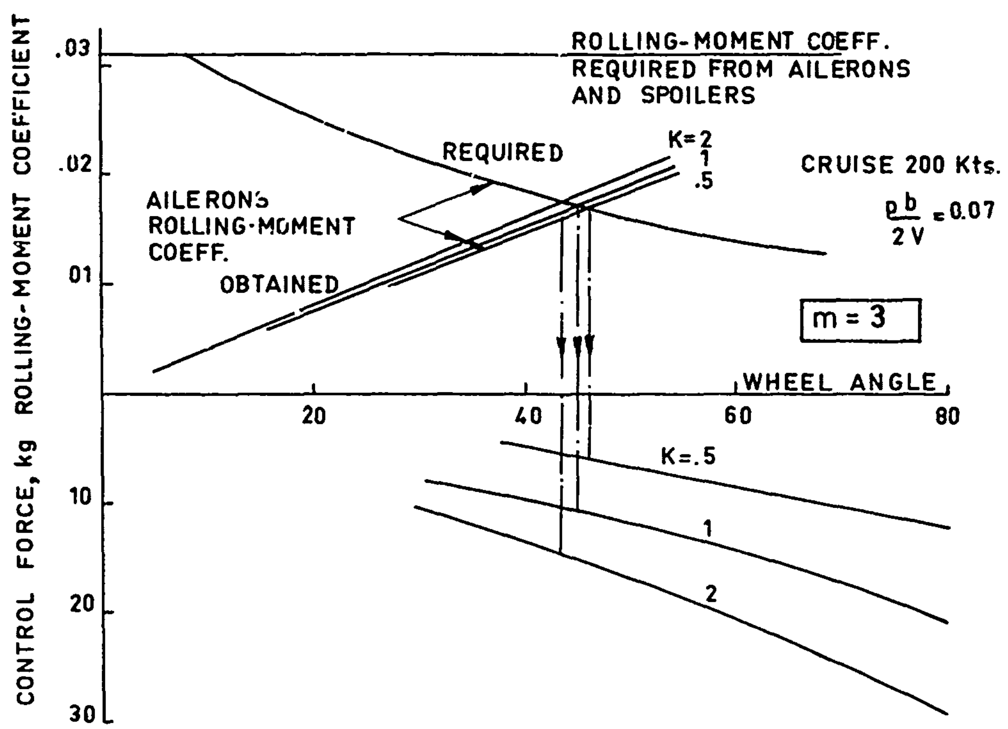

FIG.18 - DEFINITION OF AILERON SPRING-TAB CHARACTERISTICS

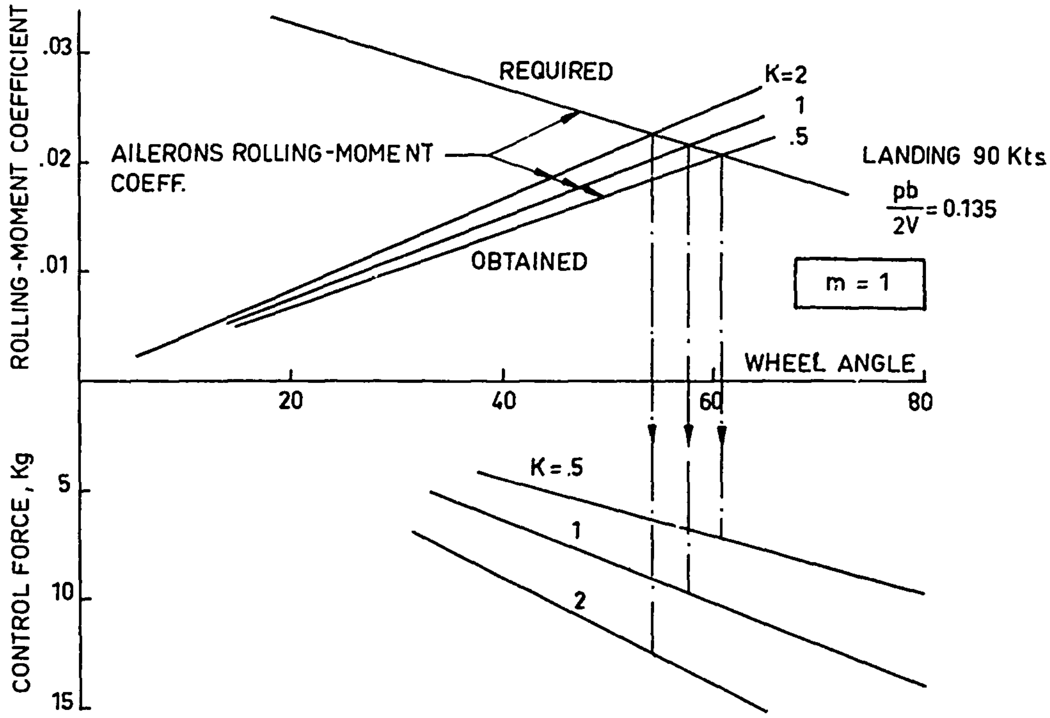

FIG. 19-DEFINITION OF AILERON SPRING-TAB CHARACTERISTICS

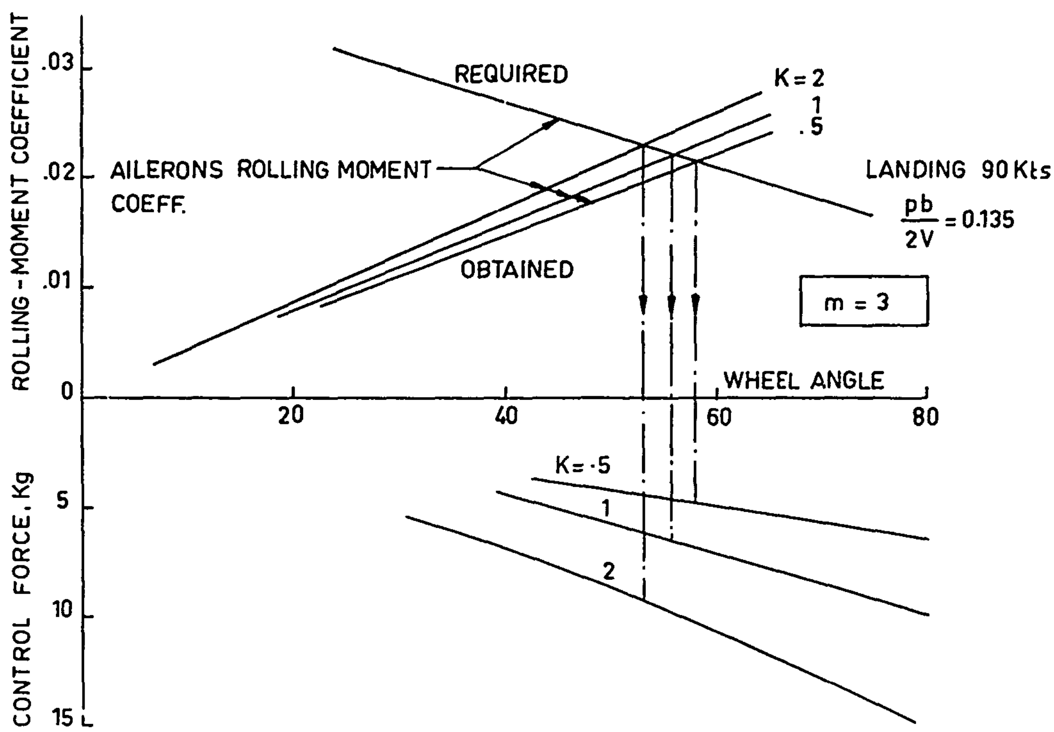

FIG.20- DEFINITION OF AILERON SPRING-TAB CHARACTERISTICS

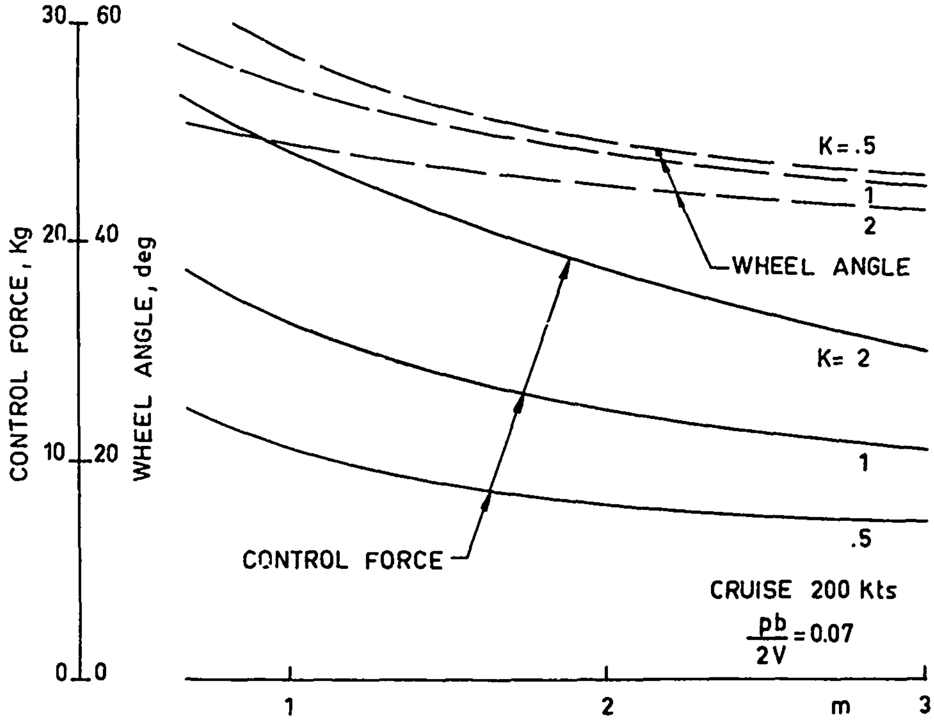

FIG. 21- SYNTESIS OF AILERON SPRING-TAB TYPICAL CHARACTERISTICS

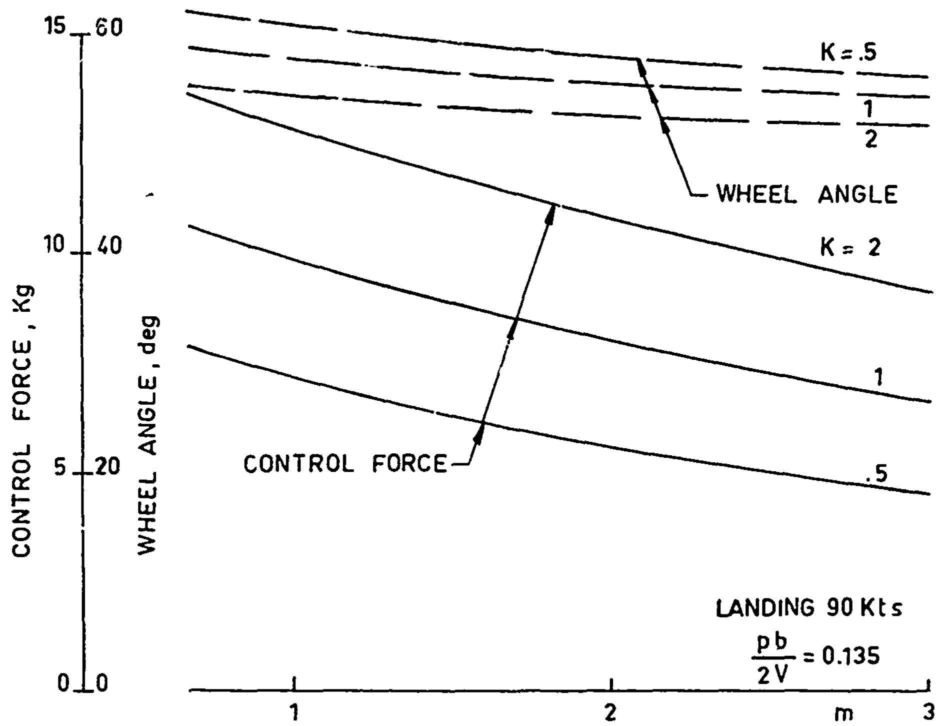

FIG. 22-SYNTESIS OF AILERON SPRING-TAB TYPICAL CHARACTERISTICS

OPEN DISCUSSION

R.L.Schoenman, USA: What was the motivation for selecting a manual control system design rather than all-powered flight controls? Was it development or operating costs rather than battle damage?

A.Filisetti, Italy: The main reasons for selecting a manual control system in the G222 are to be found in the advantages of reliability, maintainability and low production cost of that system in comparison with the all-powered flight controls. On the contrary it must be said that the development cost of a manual flight control system is higher than that of a powered one because of the larger amount of wind tunnel and flight tests and analysis required to tune the control surfaces' aerodynamic characteristics.

As for battle damage, the G222 is provided with duplicated mechanical control lines designed in compliance with the Military Specifications.

J.F.Renaudie, France: Have you had any vibration problems with the spring tab system?

A.Filisetti, Italy: No problems of vibrations had to be faced in the testing of the spring-tab system. In some flight conditions a small amount of oscillation, not felt by the pilot, was observed in the aileron and tab surface records, at extreme aileron deflections, because local flow random spearation excited the spring tab system. Owing to its low occurrence, low frequency and small amplitude, this phenomenon was not considered a problem.

Th.Schuringa, Netherlands: Did you need to make many adjustments of the aileron controls, including the spring tab characteristics and torsion bar torque, to arrive at satisfactory control forces? What about control in the stall?

A.Filisetti, Italy: Following the first flight trials of the G222 the aileron controls were modified through the addition of a gear mode to the existing trim tabs and through a change of the spring stiffness and tab connecting ratio, within the constraints given by the flutter requirements. These changes were made in order to reduce the lateral control forces to a level allowing the pilot to easily control the aircraft with one hand in the approach and landing phases. The resulting control forces and control power relationship are in compliance with the new MIL-F-8785 B requirements.

Lateral control remains fully effective through the stall and is considered satisfactory by test pilots in this condition.

POWERED CONTROLS, INFLUENCE ON STABILITY AND MANEUVERABILITY

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SUMMARY

The Powered Controls do have an important influence on the dynamic and static behaviour of modern high performance aircraft, especially with regard to full authority Command and Stability Augmentation Systems (CSAS). The actuators dynamics form a part of the overall control loop and when incorrectly specified they will limit performance. This applies especially when the possibilities of improving the Stability and Maneuverability by interconnections in the various axes are considered. An example for a modern fighter type aircraft will be demonstrated.

Future high performance aircraft will need more sophisticated flight control systems in order to make use of for instance artificial stability to improve the aircrafts performance especially the specific excesspower. The design trend for these systems is the fly by wire Command and Stability Augmentation System (CSAS). Not only on the reliability side, but also on the performance side the actuator is at present the weakest link when designing such a system.

The study conducted to show the influence of the powered controls on stability and maneuverab ility was based on a delta wing aircraft with at some points of the flight envelope marginal static stability. The control surfaces are a rolling tail for pitch and roll control (in this paper called taileron) and a rudder for yaw control.

The longitudinal control system schematic is given in Fig. 1. The stick does have a conventional Feel and Trim system. The pilot's command is sensed by an electrical pick off, fed to a multiplier which schedules the command with dynamic pressure ($P_t - P_s$) in order to give good stickforce per g relationship. A prefilter then shapes this command signal.

The aircraft motion is sensed by a rate gyro. The rate signal is fed through a structural filter and a control filter to the summing point where the commanded and sensed signal are compared. The error signal then is fed via a multiplier which schedules the error signal with dynamic pressure ($P_t - P_s$) to the servo loop. The actuator loop is an electrical feed back position system. In order to compare the augmented aircraft with the unaugmented a direct link between stick and actuator loop can be switched in as an alternative. The $P_t - P_s$ scheduler in this link is used to adapt the stickforce per g to the flight condition.

The Lateral Control System (Fig. 2) is basically the same for roll control. The YAW Control System is in sofar different, as only a damper is being used with the pedal command being a direct link. Also cross-feeds from roll to yaw are being used to improve turn entry, turn coordination and the rapid rolling characteristics of the augmented aircraft.

The actuator assumed for this study is a three stage actuator with the feed back potentiometer fitted directly on the actuator itself, so that the mounting structures elasticity is not within the control loop. The correct representation for this servo system is (Fig. 3):

$$H(s) = \frac{y_3(s)}{u_i(s)} = \frac{1/K_f}{(1 + a_1 s + a_{2(\omega, y_{20})} \cdot s^2 + a_{3(\omega)} s^3)\,(1 + \frac{1}{K_{oi}} s)}$$

with

$y_3(s)$	=	Surface motion related to actuator stroke
$u_i(s)$	=	Command signal
K_f	=	Feedback gain
y_{20}	=	Max. actuator stroke when sinusoidally excited
K_{oi}	=	Gain of open loop second stage (innerloop) = 5 m sec
a_2	=	$f(\omega, \lvert y_{20} \rvert)$ influence of Complex Surface mass, internal hydraulic damper etc.
a_3	=	$f(\omega)$ influence of hydraulic spring etc.

The term $\frac{1}{1+s/K_{oi}}$ can be omitted, since the second stage gain K_{oi} is very high, so that $\frac{1}{K_{oi}}$ is in the order of 0,005 [sec] or [5 m sec] and compared with the other factors does add very little.

So the remaining transferfunction is :

$$H_s = \frac{1/K_f}{(1+a_1 s+a_{2(\omega, y20)} s^2+a_{3(\omega)} s^3)}$$

This can be split and normalized into:

$$G_s = \frac{1}{(1+T_1 s)(1+\frac{2\zeta}{\omega_n} s+\frac{s^2}{\omega_n^2})}$$

T_1 in this representation is a function of ω_n and K_{oa}, the overall loop gain, as well as of ζ. When ω_n is larger than 30 and $\zeta = 0,5$ then T_1 becomes rather small and can be omitted on initial investigations. Its influence will be shown later.

Using this representation with T_1 omitted and $\zeta = 0,5$, $\omega_n = 30$ the transferfunction becomes

$$G_{Act} = \frac{1}{1+\frac{2 \cdot 0,5}{30} s+\frac{1}{30^2} s^2}$$

Now the loop gain for the longitudinal axis can be established. Using the boundaries from MIL-F-8785B for ζ_{SP}, a minimum value of $\zeta_{SP} = 0,35$ for Cat A and C and $_{SP} = 0,3$ for Cat B flight phases has to be reached.

The two flight cases shown in Fig. 4 do have identical dynamic pressure of 54 KN/m^2 and are M=0,8 Alt = 0 and M=1,5 Alt = 36000ft.

Assuming that the actuator branch shall not go via $\zeta_{min} = 0,25$ a max. loop gain for M=0,8 Alt=0 could be 0,6 $\left[{}^{\circ}\eta \ /{}^{\circ}/_{sec}\ q\right]$, since the actuator branch does go to the right. Using this same gain for the M=1,5 and Alt=36000ft case, since only a P_t-P_s scheduler is used, the short period would be underdamped. In this case an additional height scheduler for Mach No correction could improve the situation, but at the same time adding complexity to the system.

Taking this thus established loop gain of 0,6, the actuator transferfunction second order was varied, keeping $\zeta=0,5$ and varying ω_n from 15 to 60. On Fig. 5 two actuator branches can be seen with the first one for M=0,8 Alt=0 crossing $\zeta=0,25$ border for low ω_n and going slightly to the left for higher ω_n. The second branch is rather well damped and at much higher frequencies. The short period roots are not very largely influenced. The same tendencies exist for M=1,5 and Alt=36000 ft. only that $\zeta_{min} = 0,3$ is never crossed, but the basic short period always at the borderline.

Since the second order approximation can be assumed to be rather pessimistic, the third order representation was now used. The natural frequency of the actuator was kept at $\omega_n = 60$, also the same overall loop gain was used. This is shown in Fig. 5 too. There is an immediate improvement on the actuator roots when comparing the cases for $\zeta = 0,5$. But a strong influence can be seen as ζ is varied. Even so the basic low frequency short period is not strongly influenced, the high frequency branch of the actuator, which represents a superimposed higher frequency oscillation, does tend to go unstable for both flight cases.

Since the elasticity of the adjacend structure and the nonlinearities were not included in these calculations the actual damping ratio for the actuator with respect to the surface will be lower than the actuator damping by itself. Therefore a very thorough check on the actuator damping is necessary.

In Fig. 6 the time histories are given for M=0,8 Alt=0 for the second order assumption and the third order assumption. It can be clearly seen, that from the handling point of view the left hand system is rather poor due to the superimposed oscillation.

Turning to the Lateral Control System the needed performance of the Taileron Actuator was kept to be

$$G_{Act}\ \text{Taileron} = \frac{1}{1+0,03s} \cdot \frac{1}{1+\frac{2\cdot 0,4}{60}s+\frac{1}{60^2}s^2}$$

Initially again a second order assumption for the rudder actuator was made. As can be seen from Fig. 7, the dutch roll is improved considerably when introducing the control system and the influence of the actuator on the basic dutch roll roots is rather low. But again the roots stemming from the actuator tend to go marginally stable for low ω_n. It is also interesting to note, that at higher ω_n the roots are pushed into the stable region.

When using the third order assumption, the improvement is not quite so explicit. But equivalent to the longitudinal axis, the variation of ζ also shows, that for the closed loop at low ζ the roots show a low damped higher order oscillation superimposed.

Conclusion

The actuator performance does have a major influence on the stability of high performance aircraft and can dominantly influence the overall design of Fly-By-Wire Command and Stability Augmentation Systems. As soon as the stability aspects are properly met also the manoeuvrability of the aircraft is satisfied. Using crossfeed methods in the lateral axis does improve the damping and can at the same time boost the manoeuvrability of the aircraft considerably. Not shown in this report is the influence of the actuator on automatic flight modes, but since additional outer loops are being closed which basically are relying on the augmentation system the influence will not be too large, when the basic system is sound.

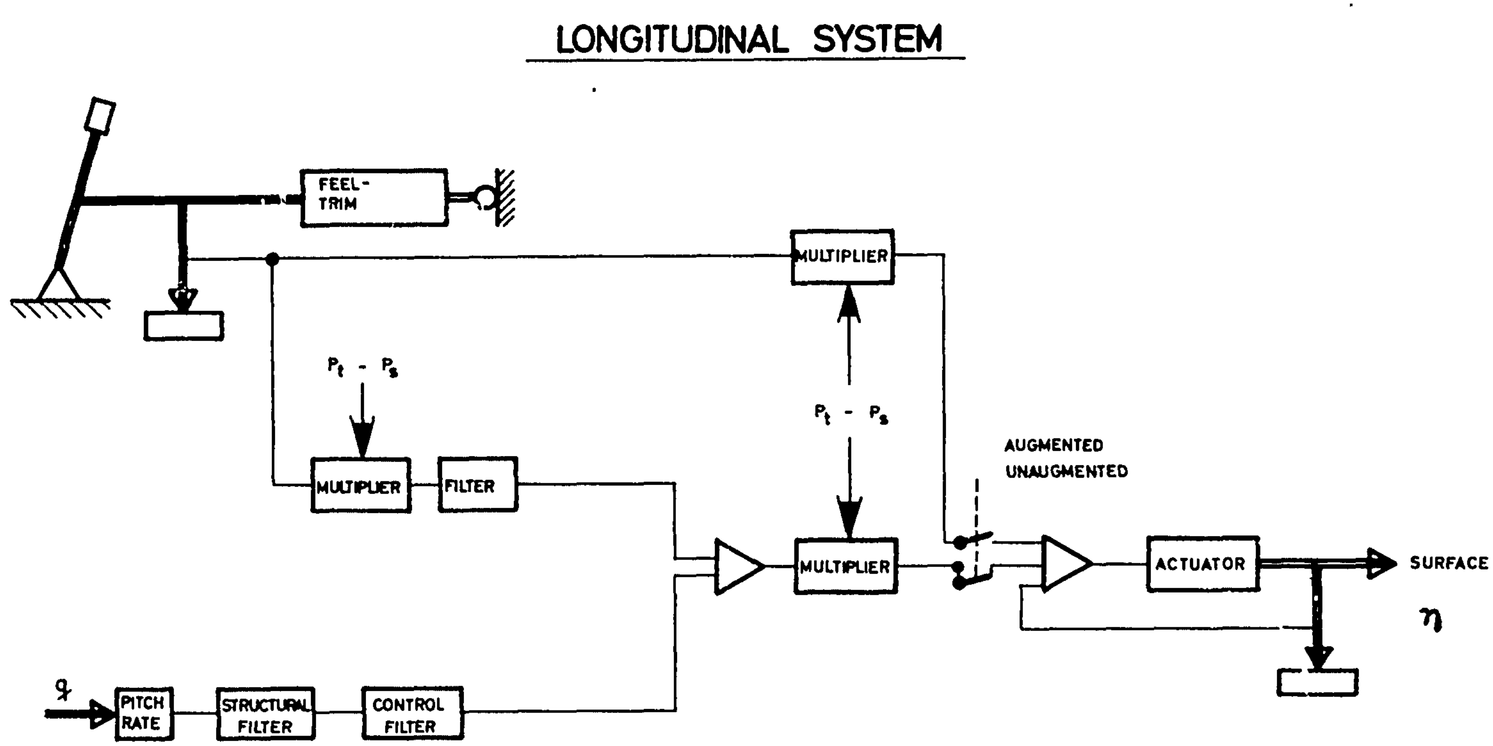

LONGITUDINAL SYSTEM
FEEL-TRIM
MULTIPLIER
$P_t - P_s$
MULTIPLIER
FILTER
$P_t - P_s$
MULTIPLIER
AUGMENTED
UNAUGMENTED
ACTUATOR
SURFACE
η
q
PITCH RATE
STRUCTURAL FILTER
CONTROL FILTER

FIG.1

LATERAL SYSTEM

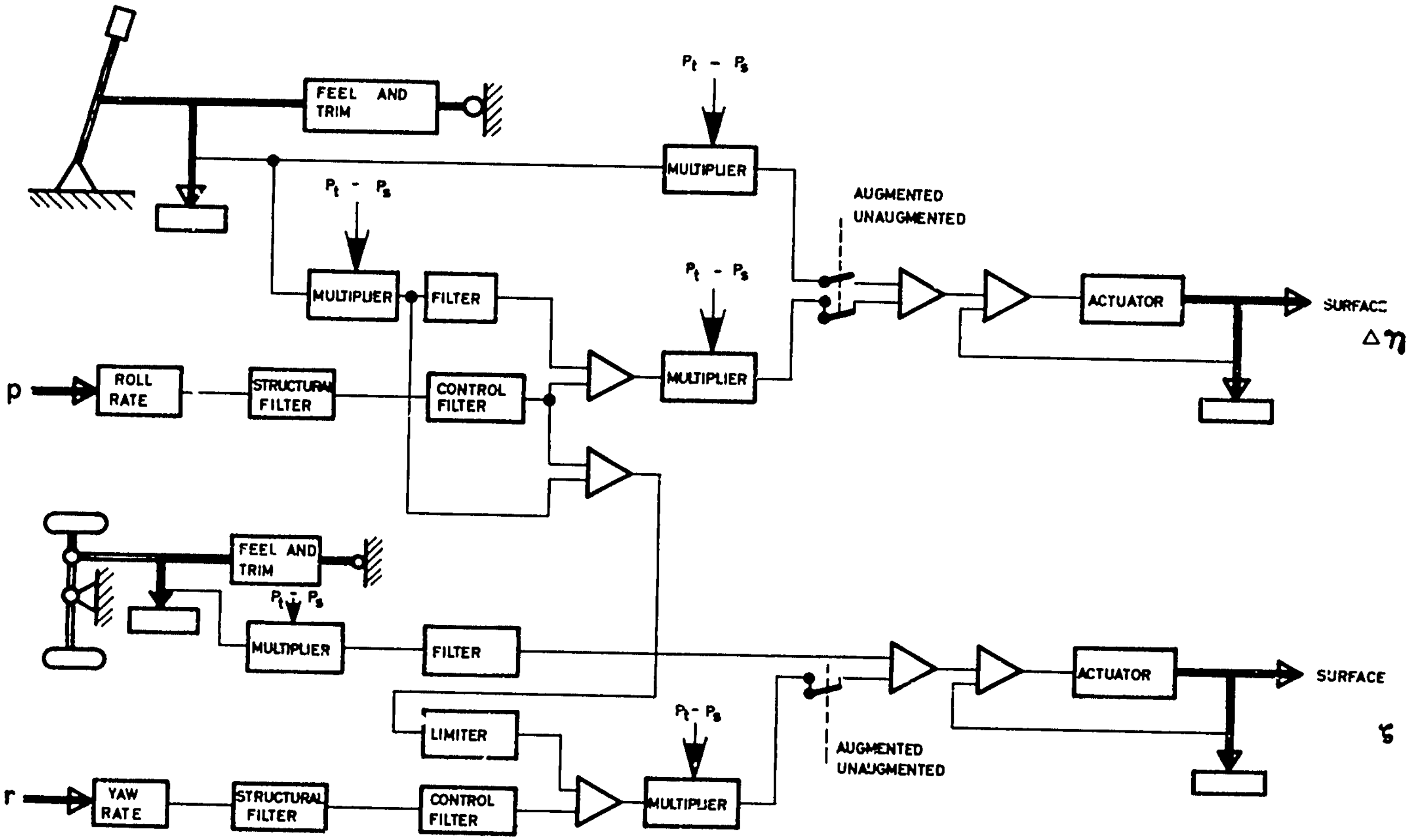


FIG. 2

$$H(s) = \frac{y_3\ (s)}{u_i\ (s)} = \frac{1/K_f}{(1+a_1 s+a_{2(\omega, y_{20})} \cdot s^2+a_{3(\omega)} s^3)(1+\frac{1}{Koi}\ s)}$$

$$H_s = \frac{1/K_f}{(1+a_1 s+a_{2(\omega, y_{20})} s^2+a_{3(\omega)} s^3)}$$

$$G_s = \frac{1}{(1+T_1 s)\ (1+\frac{2\zeta}{\omega_n}\ s+\frac{s^2}{\omega_n^2}}$$

FIG. 3

ROOT LOCUS PLOTS FOR TWO FLIGHT CONDITIONS WITH SAME DYNAMIC PRESSURE OF = 54 KN/m²

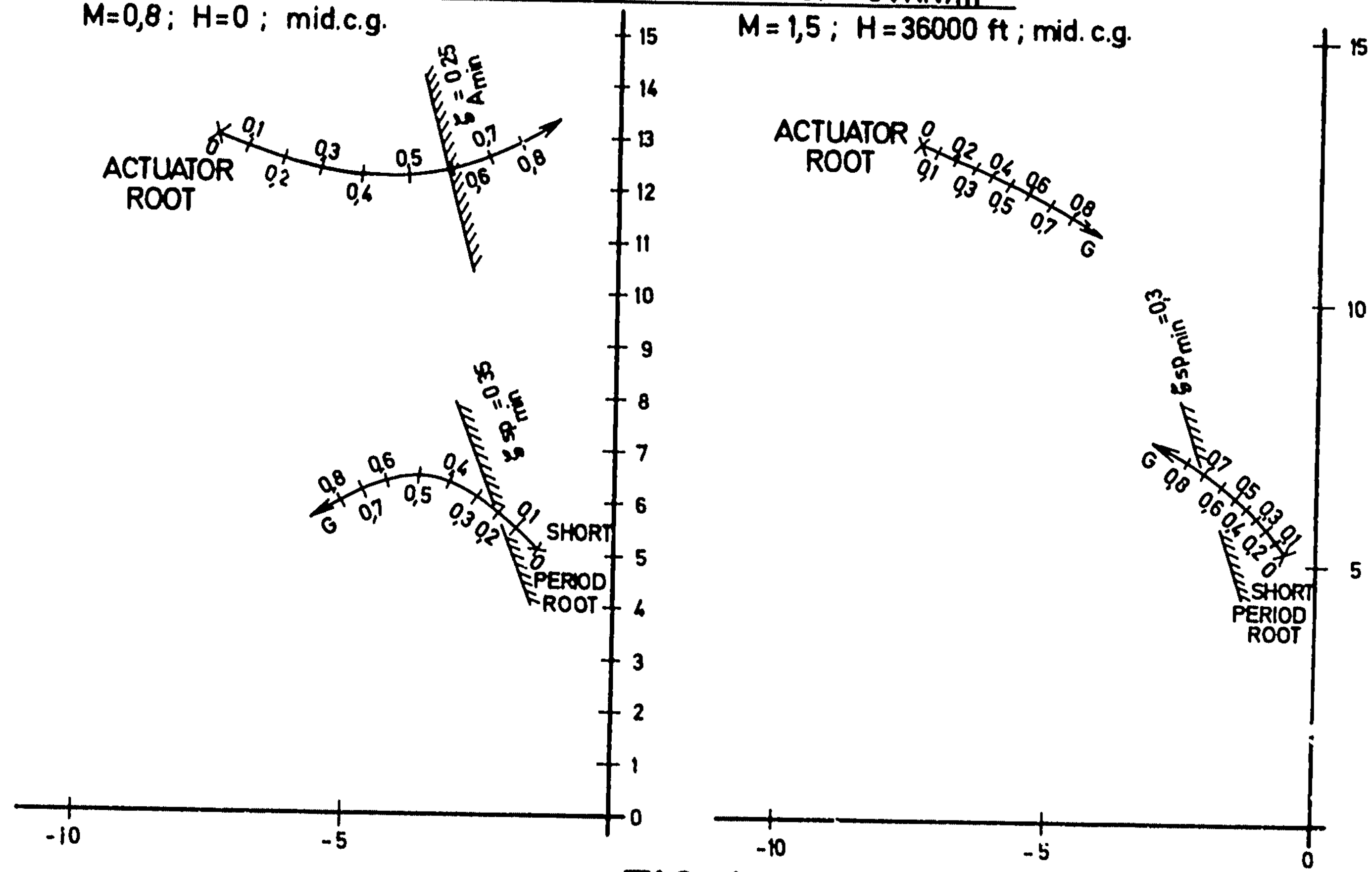


FIG. 4

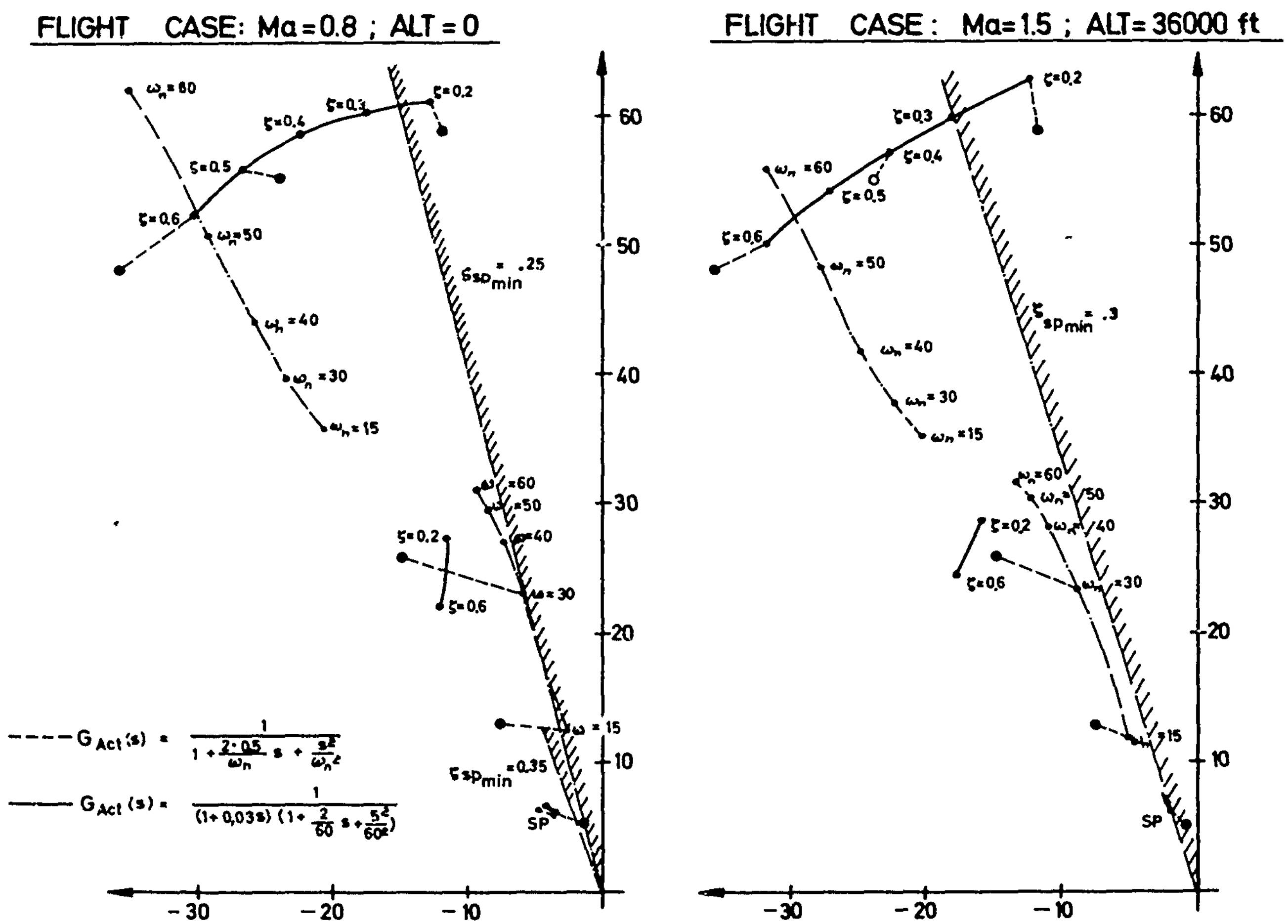

FLIGHT CASE: Ma=0.8 ; ALT=0
FLIGHT CASE: Ma=1.5 ; ALT=36000 ft
$\omega_n = 60$
$\omega_n = 50$
$\omega_n = 40$
$\omega_n = 30$
$\omega_n = 15$
$\zeta = 0.2$
$\zeta = 0.3$
$\zeta = 0.4$
$\zeta = 0.5$
$\zeta = 0.6$
$\zeta_{sp_{min}} = .25$
$\zeta_{sp_{min}} = 0.35$
$\zeta_{sp_{min}} = .3$
SP
$G_{Act}(s) = \frac{1}{1 + \frac{2 \cdot 0.5}{\omega_n} s + \frac{s^2}{\omega_n^2}}$
$G_{Act}(s) = \frac{1}{(1+0{,}03s)\left(1+\frac{2}{60}s+\frac{s^2}{60^2}\right)}$
60
50
40
30
20
10
-30
-20
-10

FIG. 5

NORMALIZED c^* n_z AND q TIME RESPONSE. FLIGHT CASE M=0.8
ALT = 0

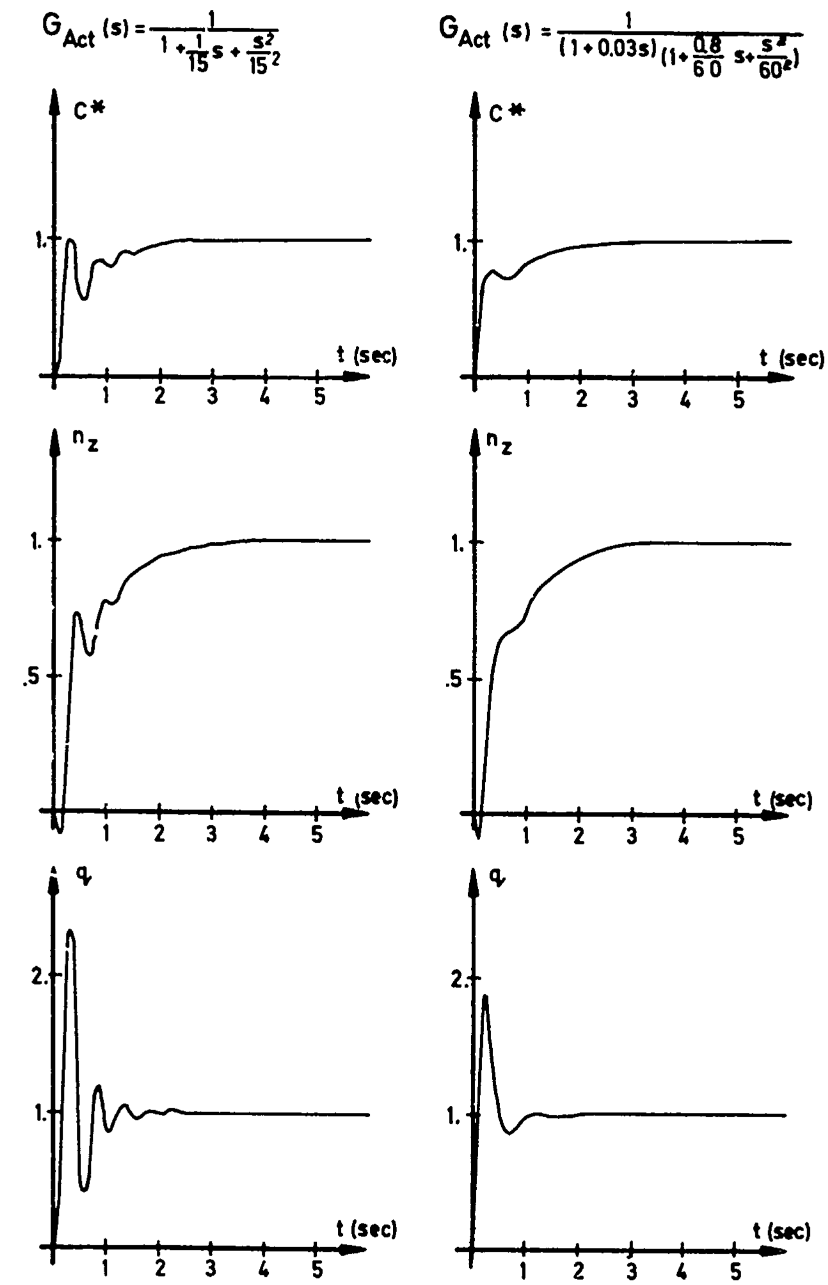


FIG. 6

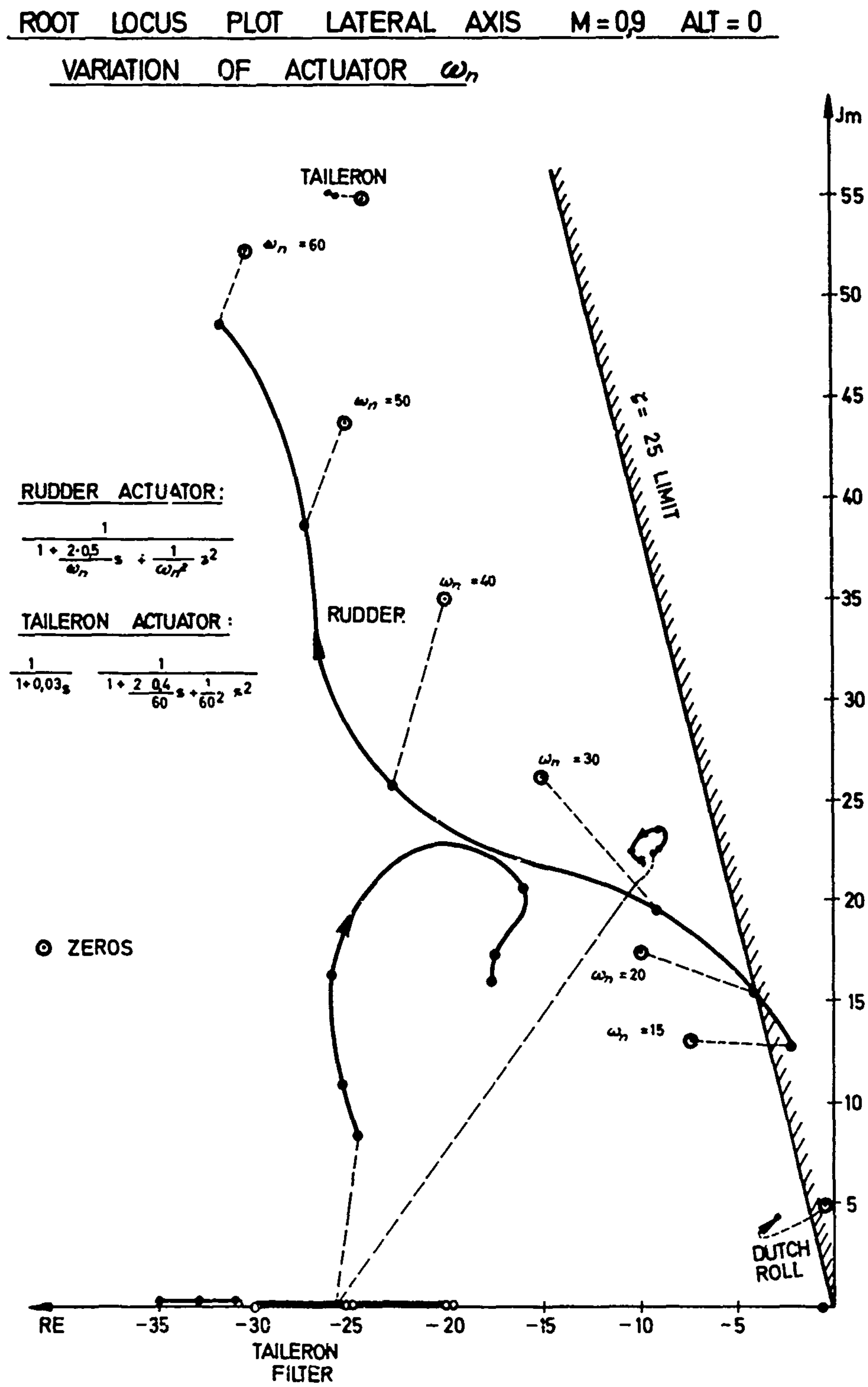

ROOT LOCUS PLOT LATERAL AXIS M = 0,9 ALT = 0
VARIATION OF ACTUATOR ωn
Jm
TAILERON
ωn = 60
ωn = 50
ωn = 40
ωn = 30
ωn = 20
ωn = 15
ζ = 25 LIMIT
RUDDER ACTUATOR:
TAILERON ACTUATOR:
RUDDER
ZEROS
DUTCH ROLL
TAILERON FILTER
RE
-35
-30
-25
-20
-15
-10
-5
5
10
15
20
25
30
35
40
45
50
55

FIG. 7a

ROOT LOCUS PLOT LATERAL AXIS M = 0.9 ; ALT = 0

VARIATION OF DAMPING OF THE ACTUATOR

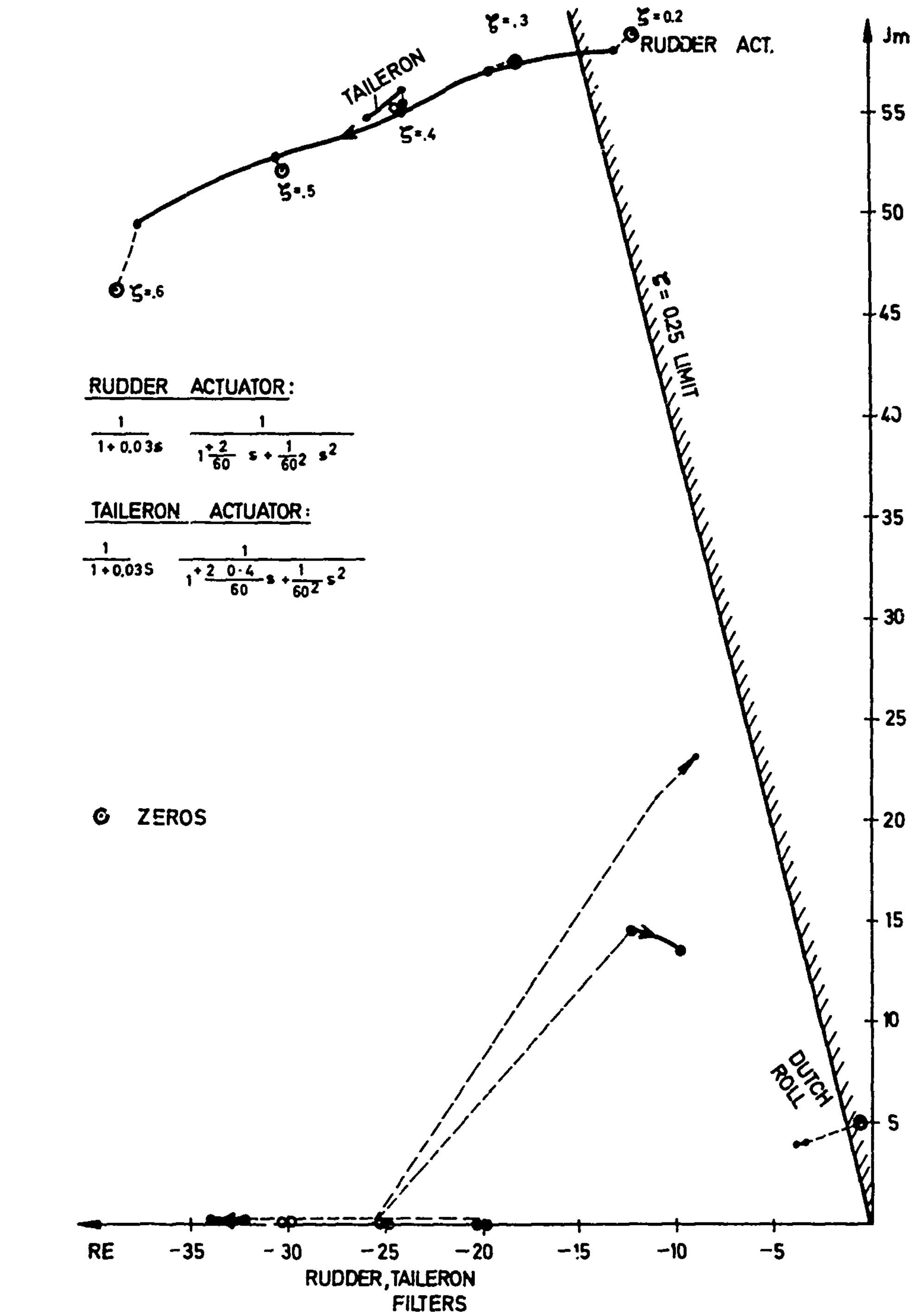


FIG. 7b

OPEN DISCUSSION

R.L.Schoenman, USA: While Mr Kissel discussed the effects of powered controls on stability and maneuverability for fighter aircraft, I would like to point out that these effects are even more noticeable for large, flexible transport aircraft such as an SST. Although not pointed out in the paper, the coupling effects of fuselage bending, flutter, and actuator response may restrict the performance achievable from the stability augmentation system or automatic flight control system.

FLY-BY-WIRE AND ARTIFICIAL STABILIZATION DESIGN

by R. L. Schoenman
The Boeing Company
Seattle, Washington USA

SUMMARY

Performance demands for aircraft which operate over a broad flight envelope such as VTOL, STOL, and SST have forced implementation of artificial stabilization to correct serious stability and control deficiencies. For this class of aircraft safety-of-flight is dependent on the integrity of these systems, and has resulted in the development of redundant system designs. The vehicle configuration process is dependent on timely solution to flight controls problems, and as such this area has become a critical and integral part of the configuration effort. The conventional SAS design approach is compared to that recommended for those vehicles which require augmentation for safety-of-flight. The impact of system redundancy on maintainability and operating costs is also discussed. Benefits of and justification for fly-by-wire control systems are related to vehicles which require augmentation. A different systems approach is proposed which features integration of flight critical functions, and use of digital computation to simplify overall system complexity and improve maintainability.

INTRODUCTION

Over the past twenty-five years tremendous advances have occurred in aircraft technology. Significant increases in speed and range have made possible meetings such as this with relatively little loss of time and comfort for the air traveler. In addition, advances made in military vehicle design allow operation over wide flight envelopes. Aircraft now have VTOL takeoff and landing capability and yet are capable of supersonic flight. The helicopter has become a practical machine, and numerous experimental vehicles have been built and tested which are able to explore expanded flight regimes or perform special missions. These improvements in performance, at both ends of the flight spectrum, have been made possible primarily by technology advancements in three major areas: aerodynamics, structures, and propulsion.

In the past stability augmentation and flight control technology, while keeping pace with these advancements, have not provided the major impetus. Serious demands, for the sake of improving performance, were not levied against the flight control design organizations, and they were not necessarily consulted concerning matters which affected the configuration development. Often this organization was called upon late in a program to correct deficiencies not identified as the design had progressed.

Electronic systems had not advanced to the point where size, weight, cost, and reliability permitted serious consideration as a primary means of aircraft control. Since there was no requirement to rely on artificial stability systems to improve performance, minimal attention was focused on the allocation of resources for flight control research. It was not until the advent of manned space exploration that serious consideration was given to this technology. These programs, in conjunction with missile system development, have brought about a remarkable advancement in electronic components and system technology.

More recently, the approach to vehicle design has been changing, and this has been brought about by two factors:

1) The desire to obtain more performance from the vehicle than can be accomplished by improvement in aerodynamics, structures, or propulsion technology alone.

2) The acceptance of electronic systems to augment vehicles with otherwise unacceptable dynamic characteristics.

A much broader flight spectrum is now being examined. The desire for high cruise speeds to improve productivity is in conflict with the requirement for low takeoff and landing speeds. Noise requirements are also having a significant impact on the airframe configuration. As the flight envelope is expanded, it has become increasingly difficult to achieve satisfactory unaugmented characteristics without unacceptable performance penalties. Consequently, the role of the flight control system designer becomes of increasing importance in the airplane design cycle. For example, stability augmentation systems may be required for safety-of-flight rather than used as a complementary system.

Significant advances in flight control technology are now resulting from increased vehicle performance requirements, i.e., range, speed, size, or even utilization. Size has also had an impact on flight control design for commercial aircraft. For example, it has been common practice to rely on mechanical reversion where the pilots controls were directly connected to the control surfaces by cables or rods. Hydraulic power was used to boost or power the surfaces when required. The control surface hinge moments could be made small enough by balance devices or aerodynamic tabs to allow direct mechanical operation in an emergency situation. While some airframe manufacturers stepped up to the challenge of all powered controls with no mechanical reversion, such as was implemented on the Comet and Caravelle, it has been the size of the vehicle which has established a firm requirement for the redundant flight control system with no mechanical reversion.

Aircraft utilization is becoming more and more significant. Airline economics are affected by schedule reliability which, in turn, is related to at least two factors: (1) airspace and terminal area congestion and (2) weather. The impact of all-weather operation is exemplified by the installation of automatic landing systems as basic equipment on the latest generation of commercial aircraft. Automatic landing system requirements have had a significant impact on the resulting flight control system configuration. The changing air traffic control environment will undoubtedly have a similar impact on system design resulting in a requirement for more automation in the flight deck controls and displays area. Airframe designers are now relying more on stability augmentation and flight control technology to achieve additional performance benefits. From past experience, this approach appears to be attractive because of the gains which have been realized for minimum penalties in weight, size, and cost. It remains to be seen if this trend will continue as flight safety demands become greater with an associated increase in redundancy and system complexity.

STABILITY AUGMENTATION SYSTEMS DESIGN FOR NON-CRITICAL FLIGHT CONFIGURATIONS

The remarks which follow relate primarily to commercial vehicles with which the author is most familiar. However, flight control systems differences between military and commercial vehicles are not striking. Different requirements are placed on the design such as survivability with battle damage, but in general the trends which will be discussed are generally applicable to military vehicles as well.

Artificial stabilization is not a new technology. Stability augmentation systems have been in use for the past fifteen or twenty years, but what is new is the impact that it is now beginning to have on vehicle configuration. Probably the most significant difference between the past and present is the approach to systems design. It has been conventional practice to design vehicles according to stability, control, and handling qualities requirements (MIL Spec 8785, or FAR 25, for example). The designer endeavored to meet the requirements as best possible without relying on augmentation devices. Inherent aircraft stability was provided by locating the C.G. ahead of the static neutral point with enough margin to demonstrate speed stability. Vertical fin size was chosen to be compatible with engine-out requirements, crosswind landing, or directional stability.

The "tuck" effect at near sonic speeds was tailored to be mild. The vehicle met most of these handling quality requirements and generally was acceptable from a safety-of-flight standpoint. In those areas where improvements were needed, single channel systems were considered such as yaw dampers, Mach trim, and stall warning devices. If the configuration exhibited characteristics better than predicted, selected systems were deleted. On the other hand, if the characteristics were worse than anticipated, redundancy was increased or additional systems were installed. For instance, a stick-pushing function may have been implemented in addition to a dual stick-shaker if the pitch-up characteristics were unsatisfactory. The necessary technology existed and handling qualities improvements, not safety, was the primary reason for installing these systems. These undesirable characteristics were generally mild with respect to divergence rates, were low in frequency, or occurred only infrequently in extreme corners of the flight envelope, and could be handled adequately by a skilled pilot. The certifying agency might allow operation of the vehicle with the augmentation system inoperative, depending on the seriousness of the deficiency. In a number of cases the system did not appear on the minimum equipment list required for dispatch.

The Boeing 707, for instance, can be dispatched with its single yaw damper inoperative. There are cases, however, when the flight envelope is restricted by the certifying agency. A good example of this is the Boeing 727. This airplane is fitted with dual yaw dampers. Each yaw damper independently drives one segment of a two-segment rudder. Either yaw damper provides adequate Dutch roll augmentation if the other is inoperative. The 727 exhibits unstable (divergent oscillatory) characteristics at high gross weights and high altitudes and speeds with both yaw dampers disengaged. This is considered to be unsatisfactory by the FAA, and therefore flight operating restrictions are imposed. Both yaw dampers must be operable to remain in the normal operating envelope. Should one fail, a reduction in speed and altitude to a restricted envelope is required where the unaugmented characteristics are acceptable to protect against the effects of a second yaw damper failure. This restriction in flight envelope is illustrated in Figure 1.

The philosophy supporting this requirement is that the pilot should not be exposed to a situation where the airplane is considered to exhibit marginal handling qualities. In the event of the first failure of the dual yaw damper system, the pilot must divert to a new and safer envelope. Since the two yaw dampers are independent and drive two separate surfaces there should be no common failure mode between channels. The probability that the second system will fail before the airplane reaches a reduced speed and altitude is considered to be extremely remote. Diversion, while a safe procedure, is undesirable from an operational standpoint since it may result in schedule delay. One might conclude that a triply redundant system would be a better choice from this standpoint. More will be said about this subject in a later section.

Stability augmentation systems in a number of past designs have required only minimum authority in terms of equivalent surface deflection, and as such have not been critical from a hardover failure standpoint. When a single channel system drives a separate surface or a separate segment of a surface with no failure isolation employed, it must be demonstrated that a hardover or oscillatory failure equivalent to full SAS capability will not cause a safety problem from either a handling qualities standpoint or from excess structural loading. When limited authority augmentation systems are used, and especially when these systems drive split surfaces, this failure condition can generally be met. The 727 system shown in Figure 2 illustrates this point.

With two yaw dampers operating a single failure, either hardover or oscillatory, can be accommodated anywhere within the normal flight envelope. With a single yaw damper operative it must be demonstrated that the loss of this remaining system will result in a safe recovery and continued safe flight. When surfaces are not split the problem is more difficult. The reliability of the augmentation systems is important if a change in flight envelope or diversion is necessary after the first failure. Fortunately, the 727 system is not complex with a mean time between failure (MTBF) of approximately 2000 hours. Based on this MTBF and an average utilization of 2000 hrs/year per aircraft, a diversion only once every 12 months would be necessary.

While there may be no uniformity with respect to the system designs or operating requirements between classes of aircraft being considered, one can conclude that stability augmentation requirements have been met with relative ease with the technology available, and that safety implications have not been a major constraint. The impact on system cost and weight have been minimal, and the handling qualities benefits resulting from these devices have been significant.

STABILITY AUGMENTATION SYSTEMS DESIGN FOR CRITICAL FLIGHT CONFIGURATIONS

The previous discussion examined past practice associated with airframe designs that utilized augmentation to correct deficient handling qualities characteristics. The alternate approach is to configure the vehicle to take full advantage of the benefits of "augmentation". Augmentation, as used in this context, is not restricted to improvement of handling qualities only, but in a broader sense improvement of a variety of characteristics such as ride qualities, handling qualities, flutter suppression, load alleviation, etc., in other words, the application to any area in which a performance benefit can be gained. The terminology "Control Configurated Vehicle (CCV)" is presently used to describe a vehicle of this type. The basic difference in this approach is that the configuration of the vehicle is dependent on the impact that the control system makes on the design. The flight control system with its associated augmentation features then becomes as important as structure with respect to flight safety. This concept has a major influence on how the vehicle is configured as shown in Figure 3.

The upper diagram shows the conventional design cycle where the main iterative loops are structures, aerodynamics, and power plant. Longitudinal balance and tail sizing is done according to conventional standards, and the flight control design is started once a configuration is established. Minor adjustments in the configuration are expected and do occur after the system is firmed up. In contrast, the lower diagram shows the required approach for a CCV design. The major impact is that flight control trade studies must be performed in a timely manner and be iterated along with the other major technologies. This is a new method of operation and may require some adjustment in organizational structure to assure integration of aerodynamic stability and flight control personnel with other organizations responsible for development of the configuration.

The emphasis on incorporation of flight control technology as an integral part of the design process does not necessarily mean that ride qualities, flutter suppression, or load alleviation, etc., will be incorporated in a particular design. The successful application of this technology is highly configuration dependent. For instance, the wing of an airplane may be strength designed and not critical from a flutter standpoint. That is to say, no additional weight is required in the wing to meet flutter requirements. A flutter suppression system would not be required for this design. On the other hand, if the wing is critical with respect to maneuver loads some weight may be saved by developing a maneuver load alleviating system. There is the added possibility that if enough weight can be removed from the wing by application of gust load or maneuver load alleviation, the wing will then become flutter critical at which time a flutter suppression system may be desirable. Since these studies are highly configuration dependent, they must be initiated during the configuration cycle with adequate time allowed to obtain meaningful answers. The tendency in the past has been to ignore or minimize this requirement. This is not meant to imply that basic research in this area is not appropriate, but that while research establishes the basic technology the application of the technology must be made on each individual configuration. In all of these areas the resulting system is associated in some manner with safety, i.e., either stability, flutter, loads, etc., and therefore the matter of system redundancy must be considered.

With the present state-of-the-art in hydraulic and electronic components adequate safety cannot be guaranteed for flight critical items on a single channel basis. The probability of failure of a critical system must be extremely remote (extremely improbable) inferring that total system failure rate must be in the order of 1×10^{-9} or 1×10^{-10} failures per flight hour. Electro-hydraulic systems on a single channel basis demonstrate 1×10^{-3} to 1×10^{-4} failures per flight hour which falls far short of this goal. Redundancy therefore is required to make up this difference, which leads to additional complexity. The number of redundant channels required depends to a great extent on the means of mechanizing the system. When limited authority systems are implemented using separate surfaces, and when failure transients are not severe, a two-channel system qualifies as being fail operational with some performance degradation evident after the first failure. No in-line monitoring is required, and the system is in its simplest form.

Flight critical augmentation systems may require substantial amounts of surface authority. This is especially true for example in the longitudinal axes if the airplane is unstable (CG behind the maneuver point). Full authority will be required in order to guarantee stable characteristics throughout the flight envelope. If this full authority redundant system drives a single surface, such as a slab horizontal tail for instance, then force voting of the augmentation channels will be required to limit failure transients to an acceptable level. Figures 4 and 5 illustrate two possible configurations.

In Figure 4 the SAS channels are shown, each driving separate surfaces. These surfaces could be four elevators or wing trailing edge surfaces used for longitudinal control. Each SAS channel independently controls one surface, and if one channel fails one-fourth of the system authority is lost. The system is subject to transients due to failures unless some type of in-line or cross-channel monitoring is utilized. If reduced authority and system transients can be tolerated, this system has the advantage of simplicity and less susceptibility to nuisance failures. If augmentation requirements are not demanding the system can be considered to be fail-operational after three failures and will demonstrate high reliability. This type of configuration is used in the Concorde longitudinal control system.

In contrast, Figure 5 illustrates a system where longitudinal control is accomplished with a single large horizontal slab tail powered by four hydraulic actuators. Because of the sensitivity of this surface, failure transients due to augmentation system failures must be minimized. The outputs of all four channels are therefore bussed together mechanically resulting in a force voting system. The bus also insures that inputs to the main hydraulic actuators are synchronized within acceptable limits. This type of system is fail-operational after two failures for the following reason: when all channels are engaged, a failure can occur in any one of the channels and the system will continue to operate in a normal manner. Since the channels are force voted the three good channels will outvote the fourth. If the failed channel is disengaged, and a failure should subsequently occur in one of the three remaining channels, the system is still operable. Assume now the second failure is cleared by disengagement of the second faulty channel. Two channels remain and the system is considered to be fail passive, that is, a third failure will not cause a significant transient to propagate into the control surface. However, this failure will render the system inoperative, and if the function is required for safety-of-flight then control will be lost. Three failures must occur, however, before the system is inoperable, that is, the system ceases to operate when three out of four channels have failed. Note that it takes one more channel to provide the fail-operational capability than the system shown in Figure 4 which has separate surface controls; however, no significant transients are propagated to the control surface when a failure occurs. This is the most significant advantage of the force summed system and is a very necessary feature for a high authority system when applied to an unstable vehicle.

A four-channel system has been used as an example, since the probability of failure of three out of four channels in any one flight can be shown to be extremely remote (extremely improbable) using currently achievable MTBF's on a per channel basis. Proper design will assure that the channels are independent, that failures do not cascade, occur in a random fashion, and are cleared in a reasonable period of time. It is interesting to note that a three-channel system using independent surfaces also meets this requirement if failure transients can be tolerated and sufficient authority is available from any one of the three surfaces.

The previous discussion illustrates the effect on redundancy and complexity when safety-of-flight is a factor. System isolation, to prevent cross-channel failure propagation, is probably the most difficult requirement to insure.

Redundance alone is not sufficient in insuring a safe system design. The mechanization of the single channel elements which make up the redundancy system must also be scrutinized carefully from a reliability standpoint.

The US-SST was intentionally balanced to be longitudinally unstable in the subsonic flight regime, and incorporated a four-channel augmentation system. Very rigid requirements were instituted to maximize reliability on a per-channel basis. Weight and size were not considered to be of major importance. Some of the design features incorporated were:

(1) no cooling air required, thus making the system independent of the aircraft's cooling system,
(2) physical separation of electronic channels to reduce susceptibility to damage of more than one unit,
(3) special separation of wiring and use of protective coating on wiring to reduce failures,
(4) special highly reliable connectors for all system connections,
(5) no air data inputs required,
(6) single pitch rate sensor per channel for augmentation, and
(7) circuit components of proven reliability.

Figure 6 is a photograph of a laboratory test unit constructed during the early phases of the US-SST program to demonstrate these principles. Two units accumulated 48,500 laboratory test hours over a three year period without a failure. The rate gyro on one unit failed at 25,500 hours but to date no electronic component failures have occurred. The housing was made of cast aluminum and heat was conducted from the housing to surrounding structure.

These are but a few of the design features incorporated to insure high reliability on a per-channel basis.

Redundancy must be consistent throughout the system. This does not necessarily mean that the same level of redundancy be used throughout. Figure 7 illustrates this point.

Where failure rates are low, dual systems may be justified as is usually the case with mechanical devices. Hydraulic and electronic portions of the system may need to be triplicated or quadruplicated to achieve the required reliability. Each portion must be consistent, however, with regard to the overall failure rate to be achieved. Of course the series element with the lowest MTBF tends to control the total, and too much redundancy in non-critical areas results in a poor design.

The role of augmentation may change in a flight critical application from a stability augmentor (SAS) with feedback loops only, such as a yaw damper, to a system which has both feed forward and feed back loops as shown in Figure 8.

This type of system, called a Command Augmentation System (CAS), incorporates both the pilots control sensors as well as feedback sensors and allows shaping or filtering to be accomplished in both paths. While the feedback path or paths provide the desired stability characteristics, the feed forward loops provide the proper response and feel forces. Adjustments can be made in both paths to optimize these characteristics.

The electronic system then becomes the primary means of control since it couples the pilot through his controls to the aircraft's control surfaces. If both feed forward and feed back paths are required for safety-of-flight a cable system is not justified except for a backup during a prototype flight test phase.

FLY-BY-WIRE

Consider first fly-by-wire applications to vehicles which exhibit safe although possibly not desirable handling qualities. For this case there is no safety-of-flight requirement for augmentation and the justification for a fly-by-wire system must be based on other factors such as weight, cost, development time, survivability due to battle damage, etc. The technology to implement a fly-by-wire system is available, and in fact has been demonstrated. Several "pseudo" fly-by-wire systems have been developed and tested. These systems utilize electronic systems for the primary means of control, but have a backup mechanical system. The Concorde flight control system is a typical example of this type of configuration.

Fly-by-wire research is presently being sponsored by the United States Air Force in the 680J Survivable Flight Control System program. A fly-by-wire control system is being installed in a F4 fighter and will be flight tested soon. Figure 9 shows some representative electronic components from this program. NASA-FRC is also actively pursuing fly-by-wire research.

The major impetus for this program was the reduction in aircraft loss due to battle damage of critical systems. Fly-by-wire designs have the advantage of multiple signal paths with better isolation between paths than do cables, push rods, and hydraulic lines.

Fly-by-wire techniques should be used, however, only when an advantage can be demonstrated. The weight saving due to elimination of cables, pulleys, bell cranks, etc., is sometimes offset by multi-channel wiring, extra electrohydraulic servo valves, more complex power control unit design, and the attendant monitoring and failure detection circuitry. Each design must be evaluated fairly to determine the best choice. Studies run on the lateral control system of the US-SST indicated that a cable system in the wing was nearly as light as a fly-by-wire approach and less complicated. On the other hand, weight savings were shown in the pitch axis for fly-by-wire control versus a cable system. The pitch axis required multi-channel command augmentation for safety-of-flight and drove the control surface directly. Since the cable system ran parallel to the CAS wiring, it was additional weight. This was not the case in the roll axis where augmentation signals drove a central control unit mounted in the body with a cable system running to the wing surface actuators. As in the case of CCV, fly-by-wire applications need to be scrutinized carefully to determine the resulting benefits.

Electric throttle control has been used for years and is presently being used on the Concorde. Studies on the US-SST did not show a distinct advantage with regard to weight, but control forces and position resolution were improved significantly.

For those designs using mechanical backup in conjunction with a primary electronic control system, the problem of engagement and disengagement of the cable system must be considered. If the controls are not centered when reversion is made to the mechanical control, a transient will result. If this problem is serious it can be alleviated by continuously synchronizing the declutched mechanical input to the electronically controlled surface. This type of system is shown diagramatically in Figure 10 and requires a synchronizing loop with motor drive to minimize the error between the cable system and surface position. An alternate means of accomplishing the same effect is to have the mechanical control engaged and operable at all times but to negate its effects by a feed back loop to the stability augmentation actuator. This approach is also shown in Figure 10. The latter scheme includes a SAS actuator offload into the trim system to maintain near zero output during steady-state conditions. Since the operating point is near zero the transient is minimized when the SAS actuators are de-energized and locked to center.

Since the fly-by-wire systems depend on both electrical and hydraulic power for operation, system redundancy and isolation are as important to overall performance and integrity as is the CAS itself. Susceptibility of the electric power system to lightning strikes, battle damage, etc., must be considered. Isolated electrical power sources rather than parallelled generators on a single bus, backed up by an emergency battery or bus, will probably be normal practice.

SYNTHESIS OF CRITICAL FLIGHT STABILITY AUGMENTATION SYSTEMS

As pointed out previously, when a vehicle is deliberately configured to take maximum advantage of control technology to enhance performance, the role of the flight control engineer changes markedly. Some of these changes are discussed below.

First of all, in this new environment the flight control organization becomes a major element in the configuration team and as such must be responsive to the schedules and rapidly changing demands. The configuration job cannot proceed unless decisions are forthcoming regarding the impact of flight controls. The organization must be staffed to define and execute the necessary trade studies in a timely manner. Very close coordination is required between those skilled in the stability and control fields and the flight control analyst. This may appear rather elementary, but a review of industry practice indicates that there has not been effective integration from an organizational standpoint. Progress is being retarded because of this.

The next major requirement for developing systems of this nature is to prepare detailed design criteria covering the important aspects of the particular task. General performance criteria, as found in MIL-F-8785 and FAR 25 can be used as a baseline where applicable. Since this is a new area, requiring a change in philosophy and closer coordination between the various technology groups, documentation of design criteria is absolutely essential. The procedure used in preparing and maintaining this documentation must be flexible so that changes can be made quickly. Changes in the criteria must be expected as studies progress and the system is synthesized. Proposed changes in criteria must be reviewed on the management level to properly weigh the impact that these changes will make on the program as well as the design, (i.e., schedule, cost, etc.). The criteria document must be accepted by all members of the organization as a necessary instrument for design. This criteria document should cover all technical fields related to the flight controls design effort.

In addition, a document is required defining the reliability requirements for the total flight control system that is compatible with the airframe design. Although one may question the authenticity of the individual MTBF on a component basis, reliability analysis on a system level to determine levels of redundancy and to insure consistency throughout the system is a valuable tool when working in an area where there is a low experience level. The reliability of peripheral systems such as electric or hydraulic power is also very important with regard to safety. Digital computer programs are presently available whereby complex systems can be analyzed on an overnight basis and changes to the system recommended. Programs of this nature are essential to allow quick turnaround needed during the design phase.

Another area of importance is that of aerodynamic stability and control data and how it is processed. For those vehicle designs which do not rely on augmentation for safety-of-flight, a system synthesis can be made generally by examination of a few flight conditions (10 or less) using linearized aerodynamic stability coefficients representing particular conditions of the flight envelope. As long as safety is not considered a factor, off nominal conditions in the flight envelope are not generally examined during the synthesis phase but are checked during flight test. This is not the case when the vehicle is dependent on augmentation for safety-of-flight, for example, an airplane balanced to be inherently unstable. The number of flight conditions examined becomes most important. If the number of flight conditions increases five or ten fold, the generation of aerodynamic stability derivatives becomes serious. Again, some method of automatically extracting the necessary information from basic data has to be developed. If the vehicle configuration is such that the effects of structural flexibility is an important factor, then a mathematical model of the airplane structural characteristics must be formulated for a representative number of critical flight conditions, and these must be compatible with the aerodynamic data. Saturation effects such as surface rate and authority limits, as well as aerodynamic non-linearities, must be analyzed in detail.

Digital computer programs have been developed which will allow a complete synthesis of a CAS or SAS for as many as 100 flight conditions in any one axis to be analyzed on a one-to-two day turnaround basis. The effects of non-linearities in either the basic aerodynamic data or control system must be carefully analyzed in order to insure that limit cycle situations do not exist for those vehicles which have inherently unstable characteristics.

Finally, the role of flight simulation is expanded. It becomes a primary synthesis tool as well as an analysis or evaluation device. It is important that the simulator be operable during the early phases of the design cycle. This places emphasis on early receipt of aerodynamic data, including structural and aeroelastic effects. A coordinated plan must be developed which will support simulator evaluations. The simulator is a valuable tool for both generating and evaluating meaningful criteria, and it provides a device to make evaluations on an integrated basis.

SYSTEM MAINTENANCE

As performance requirements for aircraft become more demanding, more dependence is placed on artificial stabilization devices to correct serious handling quality deficiencies. Since single channel systems cannot provide the safety levels required, multi-channel systems result which, of course, add to the complexity and to overall system cost. In addition to first-cost, the additional burden of maintaining more equipment increases operating costs. These costs must be examined in relation to the worth of the vehicle being designed. Whereas a system costing initially $0.5 million may be justified for a $20 million airplane, the performance benefits must be appreciable to justify that same system cost for a 5 million dollar airplane.

Maintenance of complex redundant systems is becoming a problem as evidenced by the fail-operational automatic landing systems that are presently being used in commercial service. Approximately 30% of electronic flight control systems removed from commercial airplanes because of suspected failure are unconfirmed at the maintenance center. This is due to inadequate on-board test equipment combined with the very short turnaround time allowable to diagnose the problem. Although this situation is costly to the airlines, it is not crippling because in most cases the airplane may be dispatched to the next destination with the system inoperative. However, when the system is required for safety-of-flight, and the airplane must remain grounded until the failure is corrected, the penalty in terms of schedule delay may become prohibitive. Better methods of failure identification than are presently being used in service today must be devised to make redundant system operation cost effective. The trend over the past ten years in the commercial airline industry has been to keep functions separated. For instance, today there are separate components for air data, inertial navigation, radar altimeters, autopilots by axis, auto throttles, stability augmentation systems by axis, etc. Each function has its own "black box" or "boxes", and while this is a practical approach when the functions are independent and not integrated, this approach becomes very expensive and unwieldy as more sophistication and redundancy is required. An integrated system test capability is difficult if not impossible to achieve. The use of analog circuitry in these systems also makes testing more difficult because of component tolerances. With systems distributed among many "black boxes" the test circuitry and interface wiring to accomplish the maintenance function becomes a significant percentage of the total system complexity.

Fortunately, there are solutions to this problem. The first involves a change in system organization and the second a change in computational methods. These two areas will be covered in detail in the next section.

FUTURE TRENDS

A new approach to the design of complex redundant systems is needed. First-cost must be reasonable and maintenance cost per function must be reduced below today's level. One proposed method of accomplishing this is to integrate the functions considered to be safety-of-flight into one group and those not required for safety-of-flight into another. This had the advantage that systems with a like requirement for redundancy can be treated as a group.

An example is shown in Figure 11 illustrating a proposed approach to the problem for an airplane which requires stability augmentation, thrust management, and automatic landing, and navigation capabilities. Note that while functions and axes are integrated, channel separation is maintained to insure safety with respect to computational failures. Note also that such functions as stability augmentation, control augmentation, automatic landing, etc., which have similar requirements for redundancy are considered together as one group, while such functions as horizontal and vertical navigation, maintenance computation, system test, etc., which could operate single channel are considered as an alternate group.

Once the functions have been grouped the type of computation must be examined. The use of digital computation is proposed for three reasons. First, the problems associated with system tolerances for redundant operation is much reduced. Second, pre-flight and maintenance testing can be performed more expeditiously. Third, changes to control laws and logic functions are accomplished more readily with digital computation as compared to analog computation.

Since the digital computer can perform many functions on a time shared basis, less numbers of computers are required. This has the added benefit that the amount of interface wiring is considerably reduced. Figure 12 shows an example of a typical automatic flight control system in use today requiring eleven "black boxes" of five different types. This can be reduced to six boxes of two types if a digital system were implemented.

Preliminary studies of this concept indicate that initial cost will be less than a system configured in the conventional manner, and that the cost of maintaining the system should be considerably reduced. In addition, the cost of developing and certifying the system should be reduced considerably because software, rather than hardware, changes will predominate. The use of digital computation does introduce the new problem of certification for commercial aircraft systems and in-service protection of software. This problem requires careful consideration leading to the development and implementation of a well thought out software control plan if the digital computer is controlling a high authority system.

CONCLUSION

In summary, several conslusions are evident.

1) The desire to improve performance and operate over more extreme ranges of the flight envelope will result in more serious consideration of augmentation to make the vehicle acceptable from a safety-of-flight standpoint.

2) The method of configurating such a vehicle will change with flight controls playing a fundamental rather than supporting role in vehicle configuration.

3) Safety-of-flight requirements result in more sophistication with regard to system function and a substantially higher level of redundancy then commonly in use today.

4) Fly-by-wire and Control Configurated Vehicle (CCV) concepts, while providing substantial potential benefits, need to be examined for each vehicle application for the benefits are configuration dependent.

5) Maintenance of sophisticated redundant systems will be a major problem to the user unless more adequate means of system checkout are devised.

6) A new approach is required to reduce the number of "black boxes" and attendant interface wiring which will result from redundancy. This can be accomplished by integration according to requirements for safety-of-flight rather than by function or axis. Safety-of-flight is insured by channel separation.

7) Digital rather than analog computation offers substantial benefits with regard to: performance during redundant operation, automated system checkout which enhances maintenance capability, and flexibility for change during development and pre-production phases.

8) Integration as described above will result in a reduced first cost and less operating cost as compared to the approach utilized in common practice today.

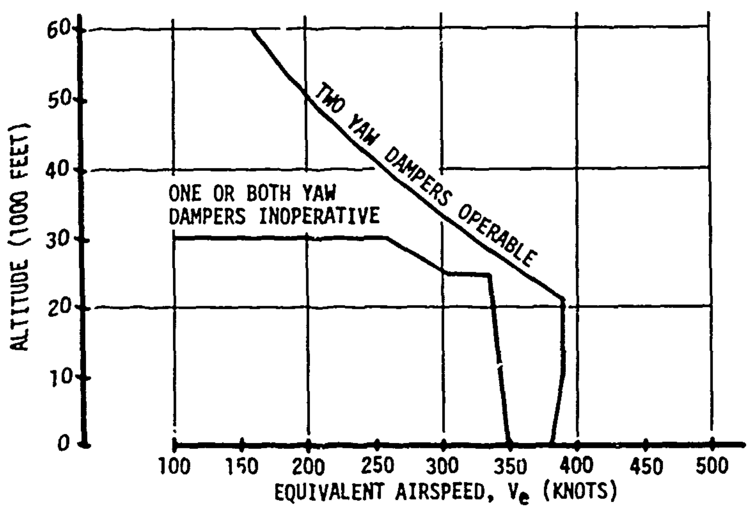


Figure 1 727 Operating Restrictions With One Yaw Damper Failed

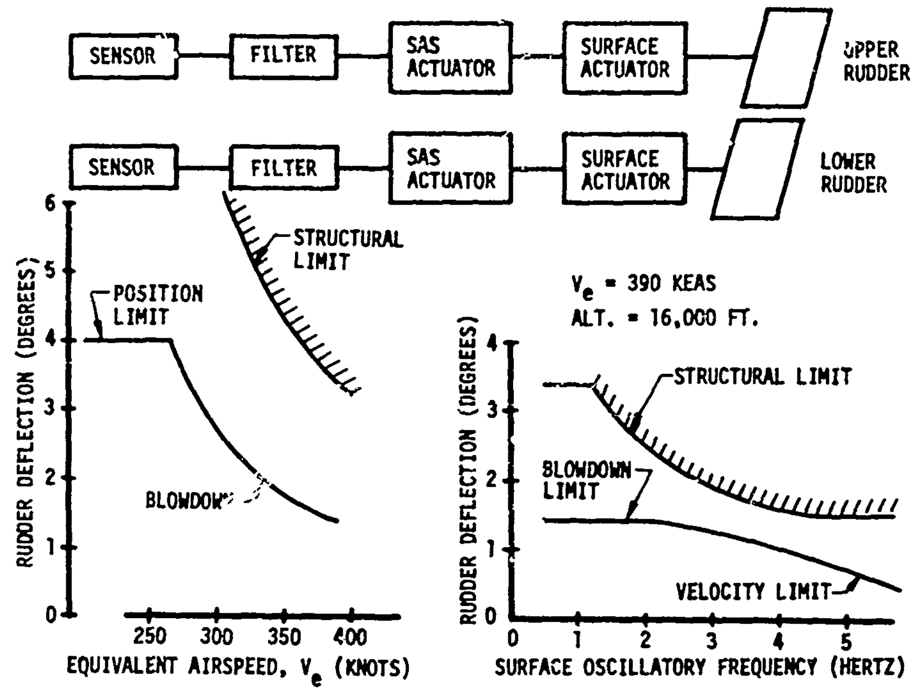


Figure 2 Yaw Damper Authority

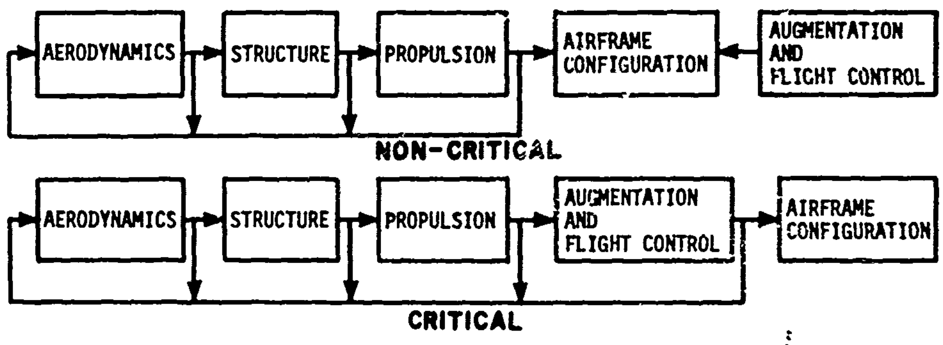


Figure 3 Design Cycle For Non-Critical And Critical Flight Control System Designs

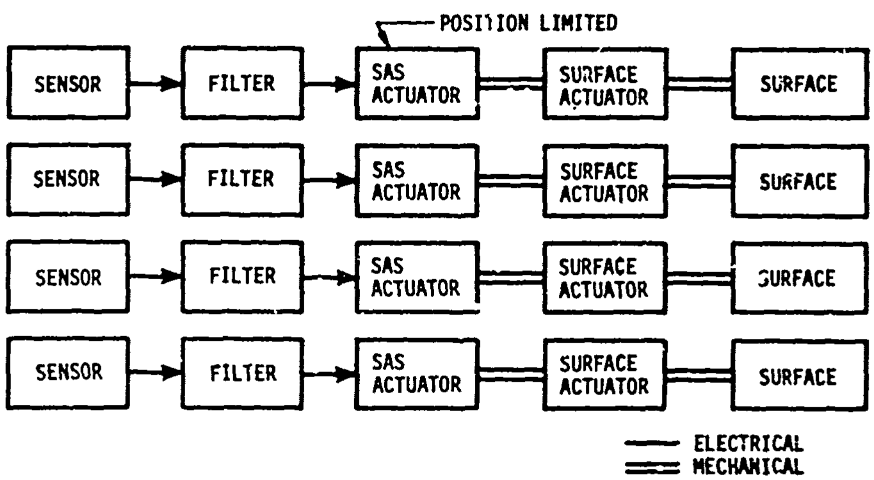


Figure 4 System Mechanization Using Split Control Surfaces

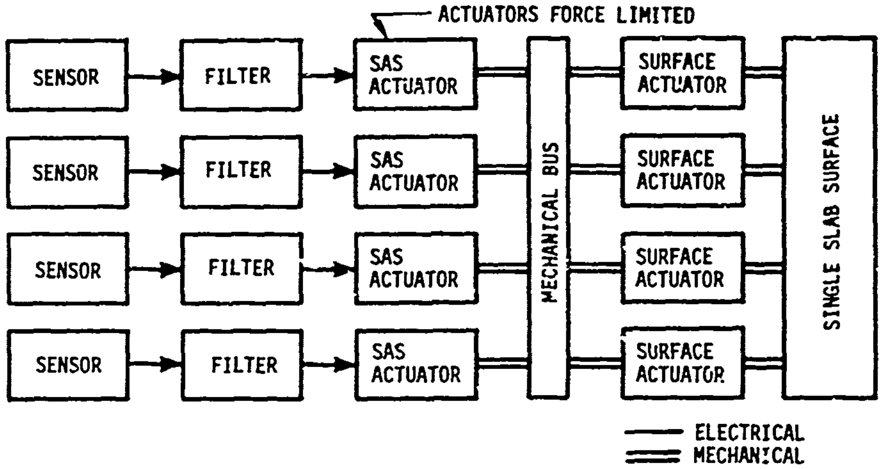


Figure 5 System Mechanization Using Single Control Surfaces

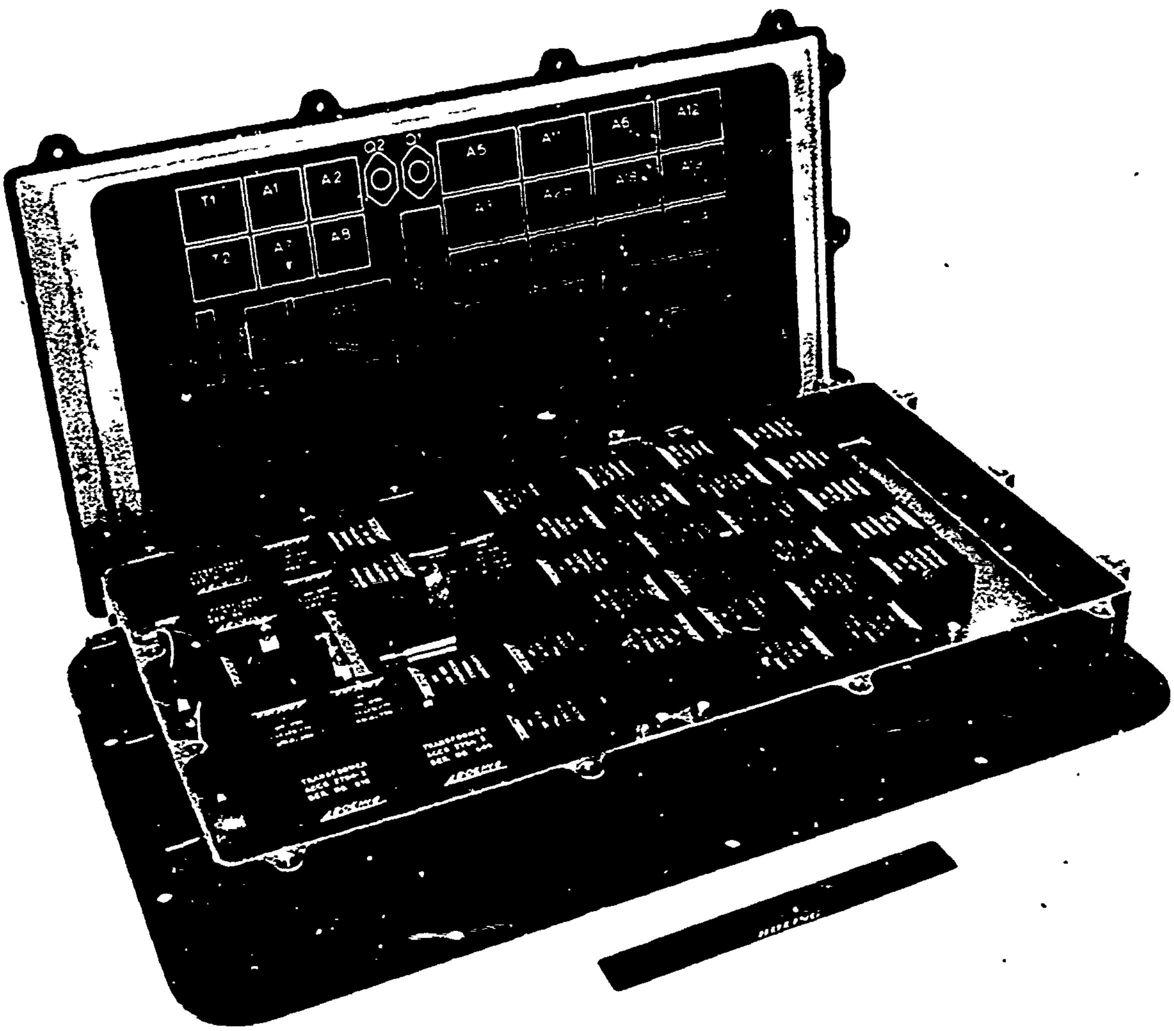


Figure 6 Electronic System Design For Flight Critical Applications

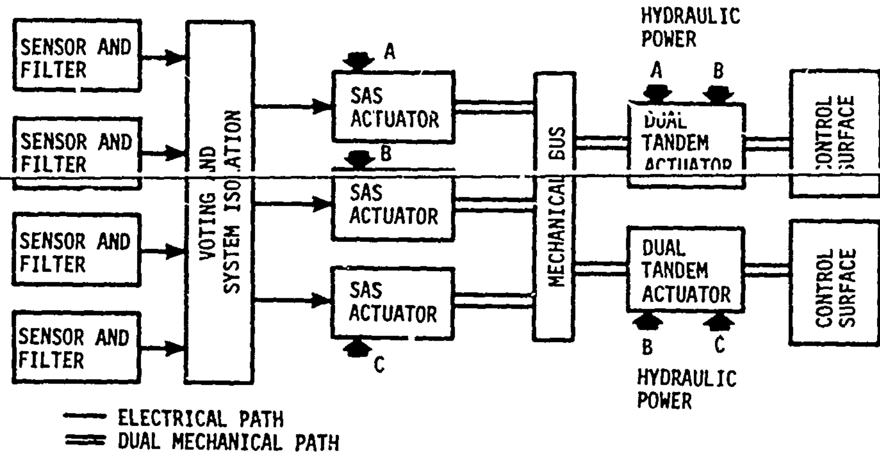


Figure 7 Redundancy Levels For Flight Critical Applications

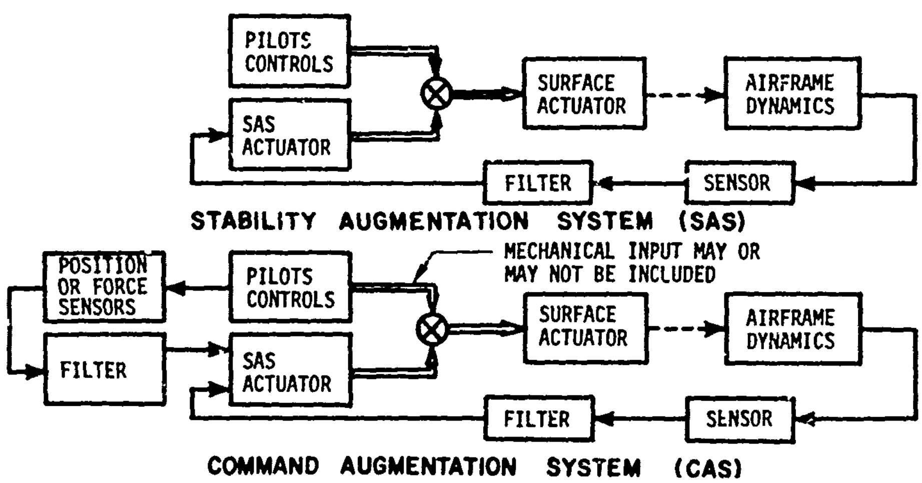
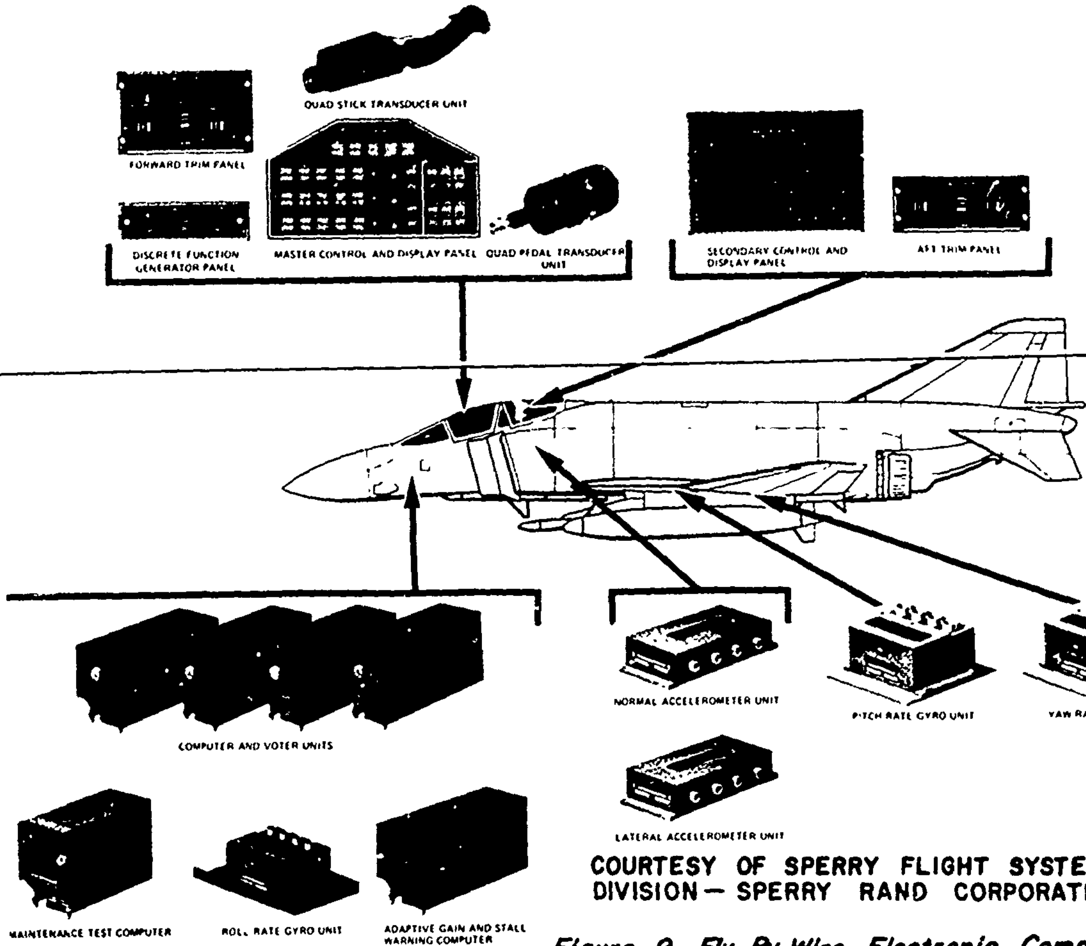
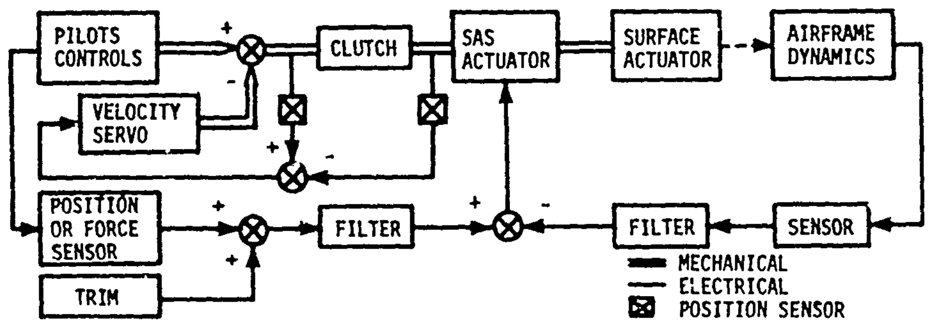


Figure 8 Stability Augmentation And Command Augmentation Systems

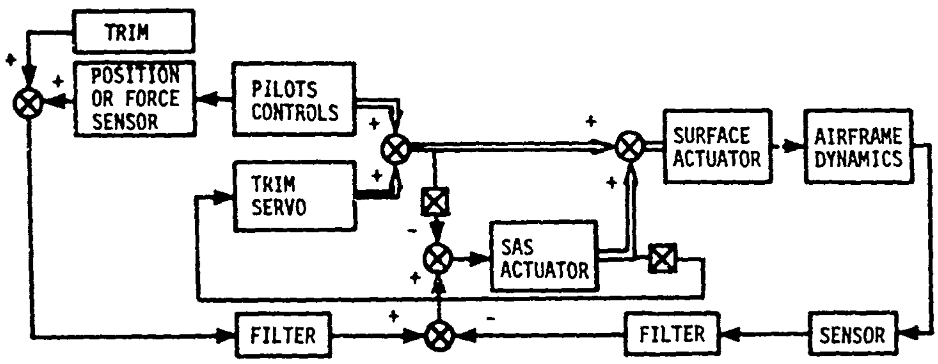


COURTESY OF SPERRY FLIGHT SYSTEMS DIVISION — SPERRY RAND CORPORATION

Figure 9 Fly-By-Wire Electronic Components



CLUTCHED SYSTEM WITH SYNCHRONIZATION
(Clutch Locks And SAS Actuator Depressurized On Reversion)



MECHANICAL SYSTEM WITH TRIM FOLLOWUP
(SAS Actuator Centers On Reversion)

Figure 10 Two Methods Of Reducing Transients During Reversion To Mechanical Control

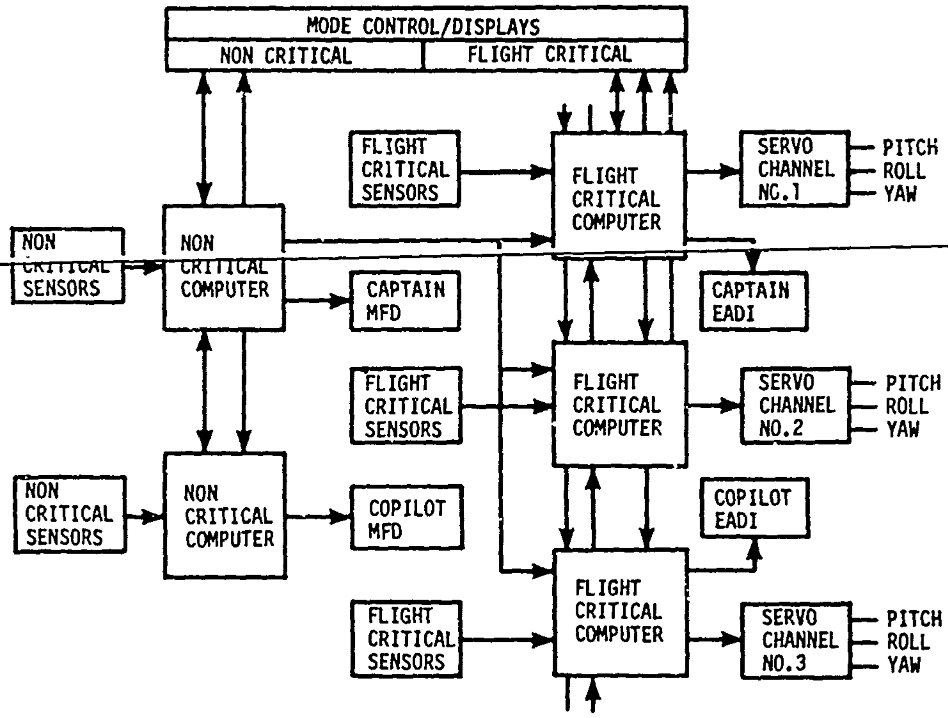


Figure 11 System Organization By Critical And Non-Critical Navigation, Guidance And Control Functions

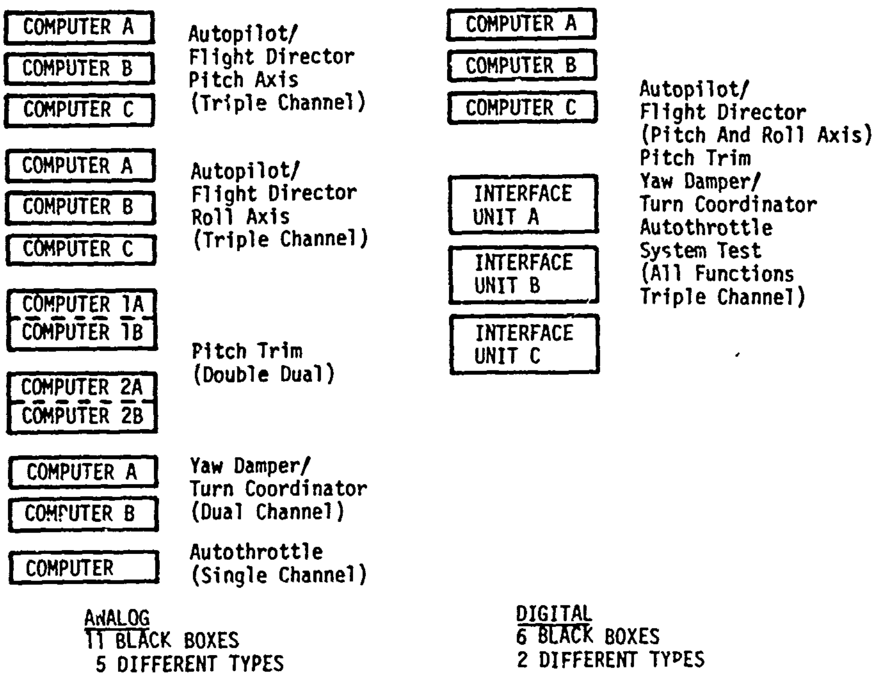


Figure 12 Comparison Of Hardware Complexity For A Typical Automatic Flight Control System

OPEN DISCUSSION

R.Onken, Germany: As I have learned, the US efforts on digital fly-by-wire flight control, including primary control is concentrated on the flight test program carried out by NASA Flight Research Center on the basis of a single channel digital system with a multi-channel analog back-up. I would like to know whether there is any flight test program running for a multichannel digital control system in the US in the near future.

R.L.Schoenman, USA: NASA Flight Research Center plans to flight test a dual digital system with a three-channel backup at the end of 1973. The computers used in this installation will not be the Apollo guidance computers, but a new class which is reprogrammable and specially chosen for this task. The choice of computers has not been specified at this date.

R.Deque, France: Was Boeing considering the possibility of dispatching the SST with one of the four channels failed?

R.L.Schoenman, USA: Not initially. After the reliability of the system has been proven this decision would have been reviewed with the possibility that the SST could be dispatched with three of the four channels of augmentation in any one axis operable.

PILOT WORKLOAD - A CONCEPTUAL MODEL

by

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SUMMARY

A conceptual model is put forward for the study of these situations when, some of the crew, some of the time are unable to complete satisfactorily, some of their tasks.

A case is made for more realistic simulations of the more difficult tasks.

1 INTRODUCTION

Satiated with endless discussions and conferences on man-machine interactions, workload, the interface, systems analysis etc., I have, in my own defence, developed a conceptual model which has helped me to look at the problem in general terms. I have also found the model useful for assessing the potential merit of a particular piece of research both at the planning stage and also after execution. The original idea for the model stems from numerous discussions with Professor Cumming of Monash University who had developed a similar approach to help him study the capabilities and limitations of car drivers.

When studying the overall effectiveness of systems the ultimate criteria which should be used in any evaluation are performance, safety, convenience, comfort and cost not necessarily in that order. It is extremely difficult if not impossible for the engineer to translate the dependent variables usually measured in Human Factor experiments for example, time on the job, errors, learning, psychophysical thresholds and physiological measures, into such criteria. Indeed, despite the large literature, we have only just reached a stage where we can begin to evaluate in engineering terms the consequences of a single stress on the human operator and his performance, and this only applies to certain stresses such as the effects of high altitude.

To look deeply at the effects of compound stresses we shall need to extend our research techniques and methodology, and many established laboratory practices will have to be discarded. The model which follows establishes the need for more specifically aimed research rather than more of the general studies so popular in this field.

2 PERFORMANCE AND SAFETY

Convenience, cost and comfort are not discussed in this paper, for the sake of brevity, although they are most important items, which must always be considered in the ultimate evaluation of a system's effectiveness. In military and civil aviation, improved safety has been vigorously sought and the spectacular improvement achieved during the last twenty years has not reduced the desire for further improvement. In military aviation the performance of the weapon system, including the man, is probably the dominant criterion. What follows relates directly to both safety and performance and is directed towards the exploration of those occasions where some of the crews, some of the time, cannot satisfactorily perform their task. I think that this is the most important area for study if we wish to explore in any depth the effect of workload on the Human Factor contribution to accidents and aborted or unsuccessful missions.

Human Factor errors can be loosely classified into three groups. The first comprises those primarily arising from inattention on the part of the operator. These do not occur very frequently in aviation. The second group includes those situations in which the operator does something stupid or misconstrues the situation, though he was not unduly stressed at the time. These I will call blunders. The third group includes all those situations in which some overloading of the operator is present.

3 BLUNDERS

AMA Majendie characterised blunders as being indeterminate in their magnitude and in no sense related to normal human performance. He also said that the individual making a blunder may find it extremely difficult to recognise it as such, even after prompting.

We try to avoid the consequences of blunders by eliminating single unmonitored human operator functions, but this is difficult in modern strike aircraft. We can, of course, transfer certain unmonitored human operator functions to black boxes and we can supply black boxes with built in redundancy, but this increases the overall cost. At present it is impossible to assess the overall cost of blunders and until we can estimate the incidence of blunders in practical situations, where training and procedures are designed to minimise their occurrence, argument is likely to be fruitless. Where blunders are known to have occurred it is, of course, most important to investigate the matter, to try and determine why they occurred. In some cases, it will be found that blunders have been induced because of bad design features in the operator's workspace or the operating procedures used.

4 OVERLOADING

The remaining human error is that resulting from overloading of the human operator, and this is an area that should be amenable to scientific study. The rest of this paper presents a model for studying those situations where overloading of crew can lead to disaster or aborted and unsuccessful missions.

Overloading exists when the crew of an aircraft cannot cope adequately with the situation in hand. It can arise from a number of causes:

(a) The basic task is too difficult for a human operator to comprehend or execute in the operational environment.

(b) The difficulty of the basic task has been aggravated by previous events such as failure of part of the system.

(c) The training of the operator has been inadequate. This may be because of cumbersome and unnecessary procedures.

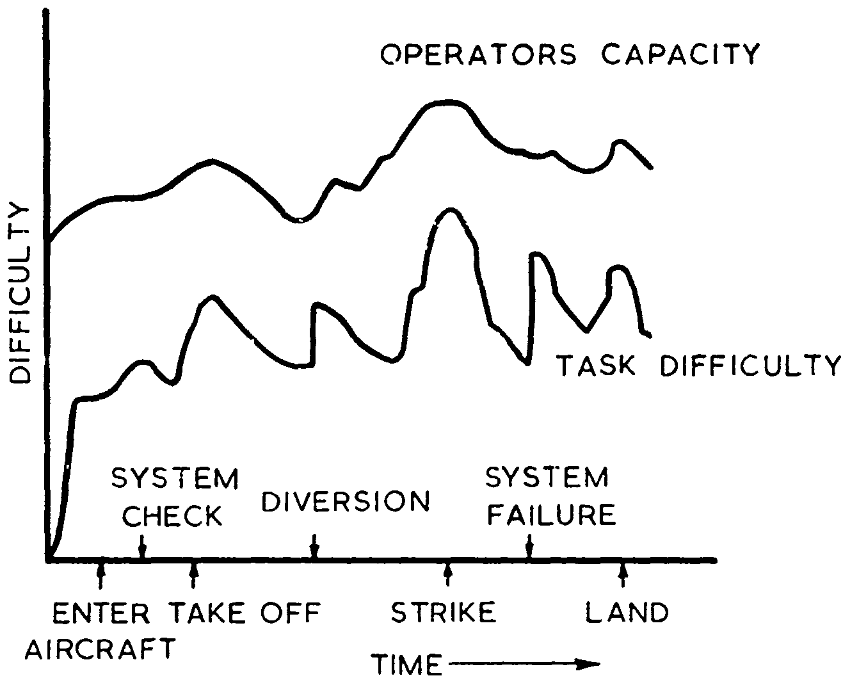
(d) The capacity of the operator is below standard. This can rise from poor selection of aircrew but even good crews have their bad days.

(e) The crew select the wrong order of priorities and deal with the wrong thing first.

(f) The task is distributed incorrectly amongst crew members.

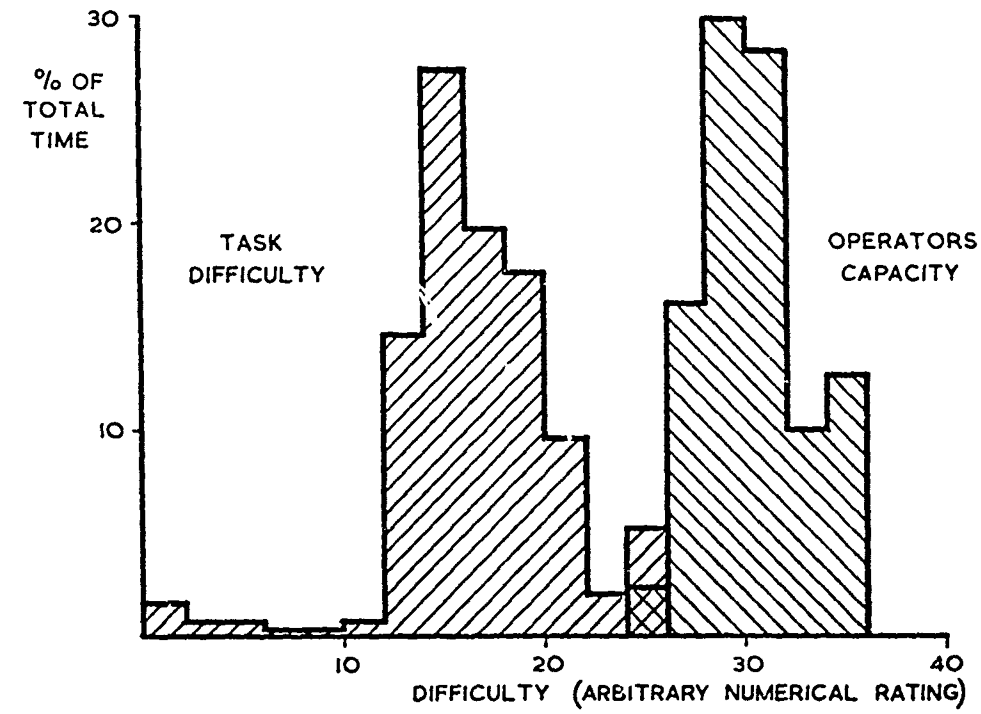
(g) Interference between sub-tasks.

Suppose it is possible to define a continuum which can be used both to assess task difficulty and the capacity of crews to cope with such tasks. It should then be possible to study the variation in task difficulty during a particular flight and to plot it against elapsed time. In a similar way the maximum capacity of the crew could also be plotted and the resulting plot might conceivably look something like that given below. Operator's capacity, in this case, is defined as the maximum ability one could expect from the operator at a given moment.



This illustration attempts to show that the difficulty of the task will vary over a wide range, during a particular flight, and will be aggravated by unplanned incidents *en route*, e.g., diversion to a new target or systems failure. It also shows that the maximum instantaneous useable capacity of the crew will vary with time and that motivation may play a part in enabling the crew to cope with the more difficult situations. If this motivation has been impaired by events before or during the flight dangerous situations will arise or an abortive mission will result. In the above situation the crew were able to cope with their task, in other situations or with other crews this may not be achieved.

Supposing for example, that the system failure had been only a little more difficult to deal with, or that the crew had relaxed a little more after the strike; this failure might easily have resulted in disaster. The only way this can be effectively discussed is on a statistical basis since one is looking for comparatively rare events. At least they should be rare otherwise a non-viable situation exists. The following illustration shows the histogram which can be deduced from the above flight history giving the difficulty an arbitrary numerical rating. This histogram shows the proportion of the total flight time that task or the operator's maximum capacity lay within a given level of difficulty.



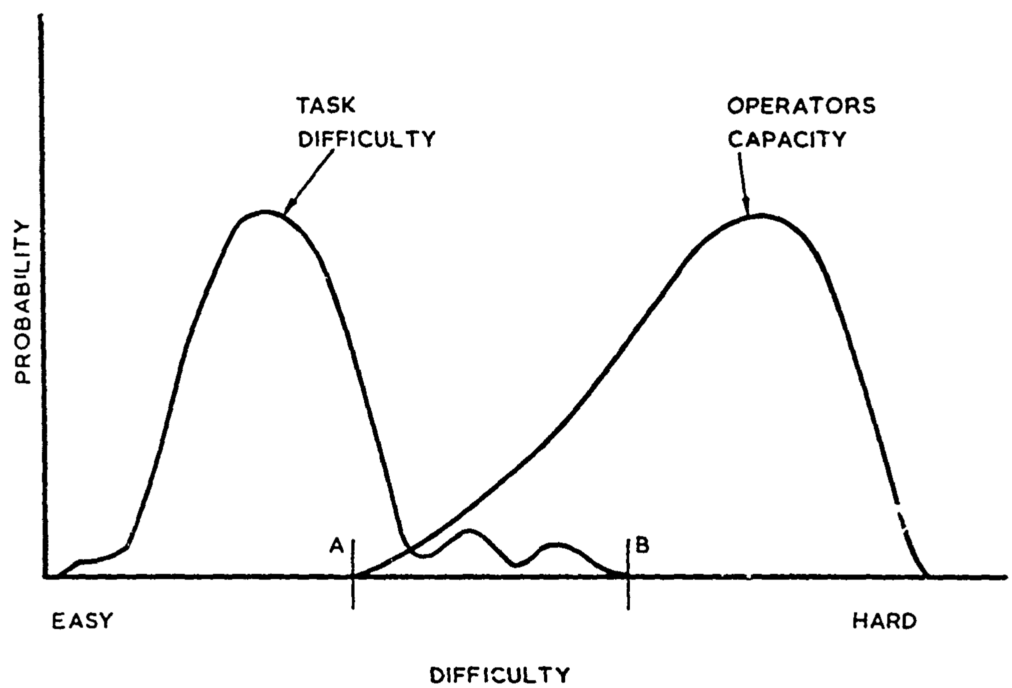
5 THE 'RARE EVENT'

Suppose that histograms can be drawn for a large number of sorties showing:

(a) The proportion of the total time spent on tasks of a given level of difficulty.

(b) The proportion of the total time that the crew can cope adequately with all tasks up to a given level of difficulty, but cannot cope with any tasks of a greater difficulty.

These histograms become probability distributions if very small divisions of difficulty are used and might look as follows for many sorties. The left hand curve shows the probability of task difficulty and the right hand curve the probability that crews can cope with tasks up to a given level of difficulty, but not harder tasks.



The probability distribution for task difficulty shows two small bumps at the hard end or right hand side of the scale. These indicate the relatively small amount of the total time when the task is difficult and the even smaller time when it is extremely difficult. There is some evidence from physiological studies, if one is prepared to assume that these give a measure of task difficulty, that such bumps do occur in the statistical distribution.

The left hand side of the histogram on the right hand side which shows the maximum capacity one can expect from the operators includes poor crews, good crews on their bad days and crews whose capacity at any given time has been seriously reduced by recent events. Most crews most of the time, can cope with all probable conditions, but a few crews all of the time and probably most crews some of the time cannot cope with the most exacting conditions. Accidents may be due to blunders in which case there is no typical pattern. But if a typical pattern of human error can be traced in incidents, accidents or aborts then it is clear that these occurred in situations, where critical overloading existed, that is in the area A-B of the last illustration. I must point out that since many sorties are included the area A-B does not represent the proportion of the total time where difficulty exceeds capacity. It does however, indicate the area where it may occur.

6 DISCUSSION

At first sight it would appear that the conceptual model put forward in the last illustration is self defeating because the only obvious solution is to send trained observers into battle to recount all that happens onto a crash resistant recorder. However, it is believed that a significant step forward in improving the efficiency and safety of both military and civil aircraft could be achieved if the right sort of research programme was mounted. This is discussed in general terms below, but first I would like to record the lessons I have learnt from this model. Perhaps the first point I should make is, that research which would enable us to draw, with more precision the illustrations I have just described is likely to be fruitless. I doubt if we will ever be able to measure task difficulty and operator capacity, on the same scale. I have put forward a conceptual model only and, as such, I believe it has some value, if only to indicate that a lot of previous work on stress has no application to real life situations.

If we could remove the more difficult tasks we would obviously reduce the overloading. I believe that many of these tasks could be removed or reduced without undue cost or penalty. I have read many accident reports in detail, and in many cases, fatal accidents have occurred because of stupidities, stupidities in the basic aircraft design, stupidities in the procedures used, or stupidities by the crew.

We have found from experience, that an impartial investigation of any cockpit and its environment will reveal anomalies which just should not be there. I believe a thorough attack aimed at removing or reducing the more difficult tasks would pay handsome dividents. I am equally convinced that laboratory experiments on the effects of compound stresses are unlikely to yield practical improvements for many years to come.

7 PROPOSED RESEARCH PROGRAMME

We first need operational field surveys and accident and incident report analyses which will help to define the cockpit tasks and environment, and pin point those situations and sequences of events which can lead to operating difficulties. Analysis of problems encountered during training could also be a valuable source of information. We must then develop realistic simulations of these difficult environments and situations and confront operational crews with them to establish what happens and the results of possible cockpit modifications. It is difficult to mount a good simulation and to obtain typical operational crews but it must be done if we are to improve the present situation.

Our primary aim is to adjust the machine or system to be within man's capabilities and limitations. It may well be that crews can eventually be selected, by certain characteristics, to reduce the area A-B in the last diagram, but at present our main aim should be towards correcting the deficiencies of the machine. In general terms we should be able to alleviate or remove the overload condition by one or more of the following means.

(a) Improvements to the cockpit environment.

(b) Altering the nature of the tasks.

(c) Reliably automating tasks.

(d) Reallocating tasks amongst crew members.

(e) Reallocating tasks with respect to time.

If we gain sufficient insight into particular problems we might reduce the probability of blunders as well.

If one accepts the argument put forward in this paper there would appear to be a number of areas in which basic research is required, in addition to the work described above.

First we must improve our techniques for assessing task difficulty and the capacity of operators to cope with such tasks. Physiological and psychological measurements have so far proved unable to do more than point to these situations which should be explored in more detail. We need a quantitative measure of task difficulty and different tasks have to be measured on a common scale. Until we can develop such techniques which must in the end rely on detailed tasks and skills analysis we are limited in our ability to pin point those situations and sequences of events which can lead to operating difficulties.

I think we will also need to use adaptive techniques to investigate some of the tasks in the cockpit to see how near they approach the capacity limit.

If complex environment factors are shown to be important from the operational studies, we must develop techniques for selecting subjects who are highly sensitive to such factors, e.g. noise, vibration or heat.

OPEN DISCUSSION

U.Kirchhoff, Germany: Have you any suggestion on how to measure task difficulty and operator's capacity and how to scale them?

R.G.Thorne, UK: The whole gist of my paper is that our preoccupation with exact measures of workload or operator's capacity has led us to ignore those sequences of events which lead to situations where some of the crews, some of the time, cannot cope adequately with some of their tasks. At the RAE we do not try to measure "workload" or "capacity" but we are developing techniques for effective time and motion study of the crew at work.

W.J.G.Pinsker, UK: In your Figure 2 you do not distinguish between latent and actual capacity. Surely this distinction must be observed, since increased task difficulty must be expected to arouse the pilot to energize his full capacity.

Mr Thorne traces all the problems he has discussed ultimately to human stupidity. I am not sure if he holds out any promise of improving man in this respect. If so we can look forward to the golden age. But I fear that history has proved abundantly that human stupidity is an invariant property.

R.G.Thorne, UK: I did not distinguish between latent and actual capacity because it does not affect the main argument in my paper. In Figure 1 the operator's capacity available at any time is shown, latent capacity would be above this. Motivation will undoubtedly play a large part in determining the capacity available at any given moment and this I have tried to illustrate in Figure 1.

THE ROLE OF THEORETICAL STUDIES OF FLIGHT DYNAMICS IN RELATION TO FLIGHT TESTING

by

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SUMMARY

It is argued that calculations have an important role to play in the planning of, the conduct of and the analysis of flight tests. This is especially true of those areas of flight testing which involve manoeuvres near limiting flight conditions and hence potentially hazardous. In this case the safety is of paramount importance. Other directions in which pre-flight calculations are invaluable are in arriving at a clearer definition of the objective of a test, the recording and instrumentation requirements. Interpretation of the test results can call for theoretical studies wherein much more detail is provided to aid analysis.

Provided the aerodynamic forces acting on the aircraft can be adequately and reliably represented there is no inherent difficulty about undertaking the solution of the equations of motion in an appropriate number of degrees of freedom with the aid of available digital computers. The real problem, therefore, lies in reducing to a manageable form the output of such computer studies and thereby achieving a deeper understanding of and a more ready interpretation of the results. As an illustrative example, the longitudinal motion of an aircraft involving an extended angle-of-attack range is considered.

Not all calculations need to be so complex in nature and certain trends can already be established on the basis of linearized equations of motion, as demonstrated by other examples considered which relate to lateral motion.

SYMBOLS

b	wing span
c_0	wing root chord
C_D	drag coefficient = drag/$\frac{1}{2}\rho V^2 S$
C_L	lift coefficient = lift/$\frac{1}{2}\rho V^2 S$
C_ℓ	rolling moment coefficient = rolling moment/$\frac{1}{2}\rho V^2 Sb$
C_m	pitching moment coefficient = pitching moment/$\frac{1}{2}\rho V^2 Sc_0$
C_n	yawing moment coefficient = yawing moment/$\frac{1}{2}\rho V^2 Sb$
C_{mq}	normalized coefficient derivative due to rate of pitch $\left[\dfrac{\partial C_m}{\partial\left(\dfrac{qc_0}{V}\right)}\right]$
$C_{m\dot{\alpha}}$	normalized coefficient derivative due to rate of change of the angle of attack $\left[\dfrac{\partial C_m}{\partial\left(\dfrac{\dot{\alpha}c_0}{V}\right)}\right]$
d	pitching moment arm of the thrust
F	engine thrust
h	height
I_y	aircraft's moment of inertia in pitch
k	damping index of Dutch-roll mode
k_1	damping index of roll subsidence mode
k_2	damping index of spiral mode
ℓ_p	non-dimensional derivative of rolling moment due to rate of roll
ℓ_r	non-dimensional derivative of rolling moment due to rate of yaw
ℓ_v	non-dimensional derivative of rolling moment due to sideslip
m	aircraft mass
n_p	non-dimensional derivative of yawing moment due to rate of roll
n_r	non-dimensional derivative of yawing moment due to rate of yaw
n_v	non-dimensional derivative of yawing moment due to sideslip
p	rate of roll
q	rate of pitch
r	rate of yaw
S	wing area
t	time
V	aircrafts' velocity relative to the air
V_{eas}	equivalent airspeed
α	angle of attack (or incidence)
β	angle of sideslip
γ	angle of climb
η	total elevon deflection

Θ inclination (pitch attitude)
ν frequency of Dutch-roll oscillation
ρ air density
ω_d undamped frequency of Dutch-roll oscillation

1 INTRODUCTION

That calculation is the thread from which is woven the fabric of rational design would, we venture to think, find ready and wide acceptance. Indeed there has been in recent years a rapid growth in the part played by calculation methods in the design of aircraft to yield specified performance levels. Calculation methods have not tended to play such a dominant part in designing for satisfactory handling qualities. This is to some extent related to the fact that there has been, if anything, a decrease in the attention given to the aerodynamics under dynamic conditions at a basic level, both theoretically and experimentally. Very few experimental investigations of a systematic nature seem to be made. However, in this paper we are not concerned with the place of calculations at all stages of the design procedure, but rather with the important role they can play in the planning of, the conduct of and the analysis of flight tests.

There has been during recent years a marked increase in the amount of data recorded during flying operations of all sorts, but it is arguable whether, alongside this recording explosion, there has been a willingness to embark on equally comprehensive pre-flight calculations relevant to the kind of questions the flight tests are intended to answer. In what follows we consider the possible underlying reasons for this position. Does this apparent reluctance to undertake theoretical studies of various aspects of the aircraft's dynamics stem from a lack of confidence in the results? Is this in turn a result of an uncertainty concerning the validity of the aerodynamic framework to be adopted in a given problem? As will be clear from the discussion of the examples chosen to illustrate the basic philosophy, there is curiously nearly always some lack of aerodynamic data even if we set aside the more difficult fundamental question concerning the mathematical modelling of certain dynamic problems. Again we may ask whether this reflects a lack of demand rather than any fundamental difficulty in the acquisition of such data. Nevertheless there are a number of reasons for some disquiet as will be seen when we discuss this aspect more fully later.

If we can rest assured that the aerodynamic forces acting on the aircraft during a certain motion can be adequately and reliably represented by some particular mathematical formulation there is no inherent difficulty in obtaining solutions, with the aid of modern computing equipment, to any dynamic problem we may pose. Rather the difficulty will centre around the definition of the problem, a matter closely related to the planning of the flight test. Just as the planning of the flight test relates to the purpose of test so in the same way we must frame some questions to be answered by our calculations. The difference, and it may be argued, the advantage that the conduct of the theoretical study has over the flight test lies in the fact that the former is likely to be fairly broadly based and look behind and beyond the requirements as at present laid down rather than being closely linked to them. If the results of the more-broadly based theoretical studies are available they provide, with the requirement, a firmer basis on which to plan the flight test, in which the requirement guides rather than dictates what is done. In this way we can move toward a more narrowly defined, yet realistic objective for the flight test.

Of paramount importance in the conduct of any flight test is the safety of the crew and the integrity of the aircraft structure. It is also important to the successful outcome of the tests to make a sound decision as to what variables to record during the flight, whilst a related consideration is the anticipation of the range of these variables and other characteristics such as rate of change, frequency content etc. Finally, however complete the flight data may be its interpretation and analysis may still be challenging. Properly designed a programme of computations can be an invaluable aid in coping with all these aspects of flight testing.

Although the main argument applies across the whole spectrum of flight tests we shall illustrate the role which we envisage calculations playing in the development of a design by consideration of areas of particular interest at the present time. These relate to aircraft handling qualities in flight conditions near to the edge of the flight envelope.

2 BACKGROUND KNOWLEDGE PROVIDED BY CALCULATIONS

As indicated in the introduction the main function of the pre-flight theoretical study is to provide background knowledge against which decisions on various aspects of the flight testing can be more confidently made. Before considering some of the difficulties which may distract from the natural desire to provide this background information and the problems that these considerations in turn pose we look in a little more detail at the question of how calculations can help.

2.1 Objective of test

During the development of an aircraft design the objective of a given flight test is nearly always related to a requirement set down by some authority. It is right and proper that this should generally be the case, but it must always be borne in mind that the requirement is the outcome of past experience in the main and that in the rapidly changing scene of aircraft design the relevance of a particular requirement may be open to some question. The remarks apply more specifically to the detail rather than the broad basis of a requirement, since the latter should be much more permanent in character. However, the calculations we see being undertaken in relation to flight testing would naturally have this broader basis. In this way some of the doubts, which may arise as to the applicability of a requirement, are resolved. Hence it should be possible to move toward a more narrowly defined objective for the test.

2.2 Safety

We have already mentioned the importance that attaches to the safety of the crew and the integrity of the aircraft during any flight test. Certain tests and flight conditions, such as those described later, increase the degree of hazard. Insofar as there is increased hazard so does the need to investigate the problems beforehand become more important. Although we are concerned here specifically with the role of calculations in this respect other means such as free-flight model tests and simulator studies are equally important. The degree of hazard and the influence of various factors on this can be assessed from a suitable series of calculations. In some circumstances the pilot's action may be dictated by a need to perform in a programmed fashion rather than in response to certain motior cues. These considerations distinguish between the cases which are better investigated by analytic means and those for which the simulator is real tool.

The possibility of loss of control of the aircraft may be rated so high in some tests as to justify the installation of supplementary means by which control of the aircraft may be recovered. For such tests it becomes necessary to try to anticipate any troubles that could arise in the operation of such devices. Some guidance must be given to the pilot as regards the indications for operation of the device and as to the appropriate instant at which it may be advisable to dispense with it.

2.3 Recording requirements

It is necessary, for a number of reasons, to obtain a quantitative record of the aircraft behaviour during test. Seldom is it possible to record all quantities of interest and a selection of the more important must be made. Here again the calculated behaviour of the aircraft provides a rational basis for making such a selection.

2.4 Instrumentation

The recording of the various parameters imposes demands on the instrumentation to be installed in the aircraft. In order to specify the instruments precisely it is necessary to know the expected range of a particular variable, its frequency characteristics, if any, and so on. This type of information can be readily made available from an analytic study.

Similar considerations may be involved in the specification of some of the pilot's flight display instruments. For example, it may be desirable to provide instruments additional to, or alternative to, the standard instruments to ease the pilot's task.

2.5 Interpretation and analysis

The availability of the wealth of detailed information that it is possible to extract from computational studies of the aircraft's dynamics render the task of interpretation and analysis of flight records easier. If a successful correlation can be achieved between the actual and predicted behaviour the influence of various design features may be recognised. Such a state of affairs is unlikely to pertain at the initial stages of the flight testing. However, it is envisaged that a progressive upgrading of the mathematical modelling of the aircraft would take place as tests proceed. The recognition of the design features responsible for any shortcomings in the aircraft's handling qualities points the way to the sort of measure that is likely to alleviate the problem.

It may not always be easy to identify certain features of the motion as arising from inputs due to the pilot rather than from other causes. By investigating the aircraft's response to a variety of inputs calculations enable us to distinguish between these effects. For aircraft motion of a non-linear character these remarks are especially true.

Thus we see the analytical study of the aircraft dynamics occupying the position indicated in Fig.1 during the development phase of the aircraft design process. As implied by the flow diagram the results of the calculations input into the definition of the flight test directly and also possibly indirectly through free-flight model tests and/or simulator studies. In a research context either of these two techniques may also supply the ultimate objective of the analytic studies. However, in the present context we would see these techniques as reinforcing the direct use of the calculations in planning the flight test. Also shown are the probable feedback loops from the flight tests as well as certain subsidiary ones.

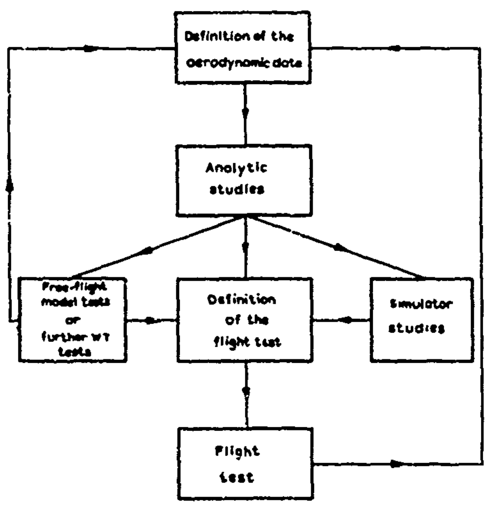


Fig.1 Analytic studies in relation to other stages of the development of an aircraft design

3 AERODYNAMIC FRAMEWORK

As is clear from Fig.1 the whole of the process just outlined depends totally and critically upon our ability to provide an adequate and reliable framework for the aerodynamic content of the problem under investigation.

Let us begin by considering the sort of aerodynamic data that are usually available to the aircraft designer at the stage at which he might contemplate the type of analytic study being advocated here. Conventional 'static' wind-tunnel tests yielding force and moment coefficients, including the effect of control surface deflection, with respect to all three axes, as functions of angle of attack and sideslip are almost certain to be available. Only two questions are likely to arise? Do the data cover a sufficiently wide range of motivator deflection, angle of attack and angle of sideslip? If the test conditions fall appreciably short of the full-scale conditions can we obtain an indication of probable Reynolds number effects from some other source?

Additionally the measurement of oscillatory derivatives over small amplitudes and at low frequency parameter values is becoming more common, although not yet the matter of routine it should perhaps be. Such measurements form the basis for a quasi-steady treatment of the aerodynamic terms arising in the equations of motion. For small amplitude motions and questions of linearized stability the usual derivative formulation is adopted. In certain problems, for example, when higher frequency structural modes of motion are involved, it is necessary to evaluate the derivatives at the appropriate frequency parameter.

For aircraft motions involving large excursions in either the angle of attack or the angle of sideslip or both the non-linear nature of the 'static' aerodynamic forces and moments require that the aerodynamic data be expressed as functions of these two variables. Similarly the contributions of other variables although expressed in a derivative form must be generalized by allowing the derivative to be a function of the angles of attack and sideslip. This formulation still falls within the quasi-steady assumptions in that the aerodynamic terms are solely determined by the current values of the motion variables.

Whilst there exist at present no well-established grounds for suspecting the adequacy of this aerodynamic framework for our present purpose, it is necessary to be mindful that there are circumstances when it is likely to be inadequate. Many of the flight conditions to be investigated imply passage, sometimes repeatedly, through critical flow conditions, for example the neighbourhood of the low-speed stall or shock-induced separation. Furthermore the rate of change in the angle of attack and/or in the angle of sideslip may be large. Such considerations expose the need for further aerodynamic research into the nature of the airflow under the conditions mentioned and into the effect on the forces acting on the aircraft.

As far as present experience goes it suggests that these doubts, which surround the adequacy and reliability of the aerodynamic framework normally used, are not sufficient in themselves to dissuade the aircraft designer from undertaking a wide ranging programme of pre-flight calculations of the aircraft's dynamical behaviour. As long as they exist they rather indicate a need for progressive development during the flight tests, which on the one hand takes account of the nature of the doubts and on the other provides the information, on which a modification of the original aerodynamic data may be based to bring the predicted and observed behaviour into line.

4 SOME EXAMPLES

Perhaps the best way to bring out some of the points we have just discussed is to consider some examples. The first concerns the longitudinal motion of an aircraft over an extended angle-of-attack range during a decelerating manoeuvre at low speed, where the objective of the flight test is to determine and demonstrate limiting flight conditions specifically in terms of a minimum speed. Our second example is not so specific and covers a whole range of problems being concerned with the lateral characteristics of combat aircraft and the way these affect the handling qualities at high angles of attack and over a range of Mach numbers.

4.1 Longitudinal motion over an extended angle-of-attack range

For the purpose of illustrating the role of the analytic study in this context we take as the subject aircraft a slender-wing tailless configuration. It is sufficient for our purpose to give only a brief outline of the calculations with a few isolated numerical results.

The variation of the lift, drag and pitching moment coefficients with the angle of attack and the elevon angle for the configuration in question were available from wind-tunnel tests over the angle-of-attack range 0° to 35° and the elevon angle range -15° to 15°. Some additional data, of a somewhat more uncertain nature, were available for even larger angles of attack. On the basis of these later test results the aforementioned wind-tunnel data were extrapolated over the range 0 to 90° in the angle of attack (see Figs.2, 3 and 4).

For the quasi-steady aerodynamic framework adopted for the analysis the contributions to the forces and moments due to the angle of attack and the elevon angle are assumed to be polynomial functions of these variables, which reproduce as faithfully as possible the wind-tunnel data. To complete the aerodynamic data we require the contributions to the forces and moments due to the rate of change in the angle of attack and the rate of pitch. These are assumed to be adequately represented by coefficient derivatives and the force derivatives are neglected. Furthermore due to absence of experimental data the moment derivatives were estimated and assumed invariant with the angle of attack. An added reason for making this latter assumption is the large proportion of damping in pitch provided by the autostabiliser.

Certain features of the aerodynamic characteristics of the aircraft are of interest, whilst some have an important bearing on the aircraft's behaviour. In the neighbourhood of an angle of attack of about 25° the lift, drag and pitching moment all show evidence of some abrupt change in the character of the flow around the wing, possibly some change in the vortex pattern associated with wings of this type. It is also seen from Fig.3 that, as is well-known, the lift coefficient increases gradually up to angles of attack of the order of 37°. The drag coefficient increases rapidly between angles of attack of 20° and 35°.

The manoeuvre to be studied is a deceleration from the equilibrium conditions defined by level flight at an angle of attack of 12° at a given altitude. To execute the manoeuvre two inputs are assumed (1) a reduction in thrust to give a specified initial deceleration, (2) application of elevon in the nose-up sense to give increased angle of attack.

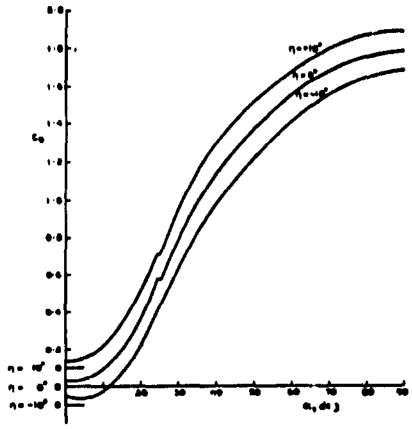


Fig.2 Drag characteristics assumed for the slender-wing aircraft

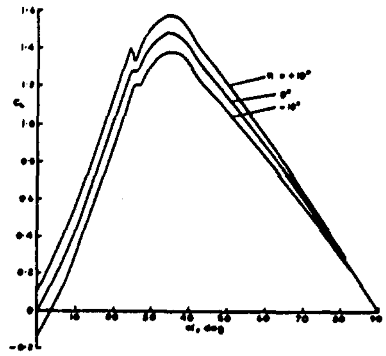


Fig.3 Lift characteristics assumed for the slender-wing tailless aircraft

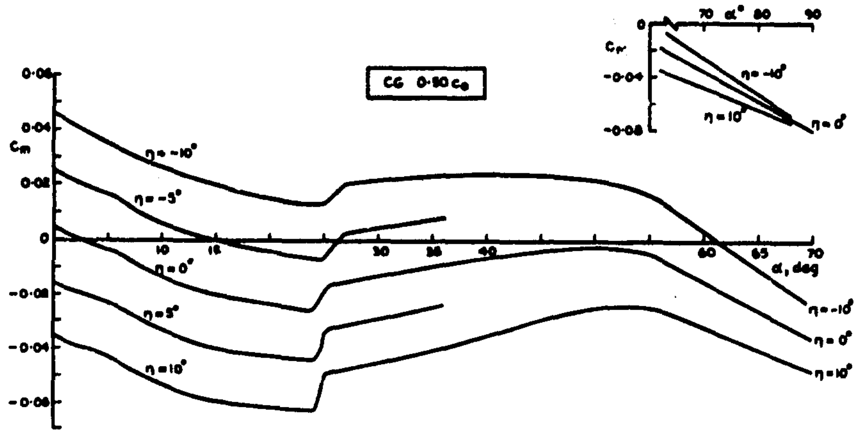


Fig.4 Pitching moment characteristics assumed for the slender-wing aircraft

If we resolve the forces along and normal to the flight path the equations of motion of the aircraft can be written

$$m\dot{V} = F\cos\alpha - \tfrac{1}{2}\rho V^2 S C_D(\alpha,\eta) - mg\sin\gamma$$

$$mV\dot{\gamma} = F\sin\alpha + \tfrac{1}{2}\rho V^2 S C_L(\alpha,\eta) - mg\cos\gamma$$

$$I_y\dot{q} = \tfrac{1}{2}\rho V^2 S c_0\left\{C_m(\alpha,\eta) + C_{mq}\frac{qc_0}{V} + C_{m\dot{\alpha}}\frac{\dot{\alpha}c_0}{V}\right\} + Fd \quad .$$

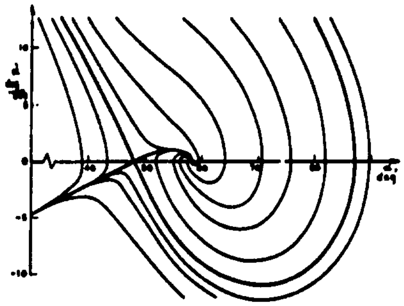
We have additionally the kinematic relationships

$$q = \dot{\theta}$$

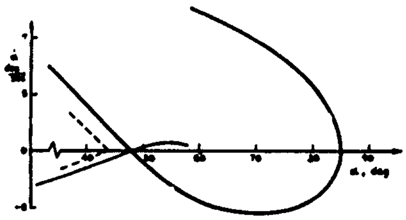
$$\theta = \alpha + \gamma \quad .$$

Here the elevon angle consists of two parts, η_p, the pilot's input and η_s, the contribution of the pitch autostabiliser. For the present investigation the autostabiliser is downgraded to a simple gearing, but its authority is limited to ±4°.

By varying the amount and rates of application of elevon in the nose-up relation sense a wide range of dynamic conditions can be generated, from which an attempt may be made to recover by application of effectively full-down elevon. To aid in the determination of the coverage of these calculations the single degree-of-freedom motion with freedom to pitch only is examined. The results of calculations which refer to this last motion, are best presented in a phase-plane plot, see Fig.5. A typical set of responses of the aircraft with full three degrees of freedom is displayed in Fig.6.



a Speed corresponding to level flight at $\alpha = 12^\circ$



b Separatrix curves for speed of unstable steady state ($\alpha = 48^\circ$)

Fig.5a & b Longitudinal motion with freedom to pitch only

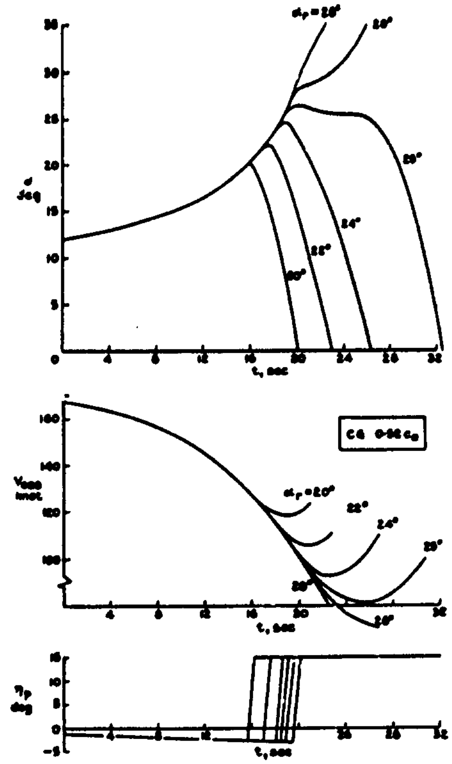


Fig.6 Longitudinal motions to large angle-of-attack conditions and attempted recovery (WT data)

Besides varying the conditions in terms of angle of attack and its rate of change for a given position of the aircraft's centre of gravity the effects of a number of factors were assessed. If we define as the recovery angle of attack that angle of attack at which the elevon starts to move towards its fully-down position we may summarize the coverage of the calculations by listing the various effects studied. These are,

(1) the effect of different recovery angles of attack

(2) different rates and times of application of recovery control

(3) different rates of approach to a given recovery angle of attack

(4) changes in thrust level during recovery

(5) changes in the inertial properties of the aircraft

(6) different locations of the centre of gravity

(7) effect of some modifications to the pitching moment characteristics

(8) manoeuvres with autostabiliser active and inactive.

The emphasis in the above items is on the influence various factors have on the ease or even the possibility of recovery. In the event of a successful recovery another question naturally arises and this concerns the ease or otherwise with which the pilot can restore the aircraft to some level, or perhaps climbing, steady flight condition well removed from the critical range in the angle of attack without incurring any risk from overshoot into negative angles of attack.

It is not proposed that we discuss these items of work severally and they are listed solely to serve as an indication of the breadth of the sort of analytic study we have in mind.

During preliminary flight testing it was found that the elevon angles required to trim the aircraft in level flight over a range of the angle of attack did not agree with those predicted on the basis of the wind-tunnel test data. An adjustment of the curve of pitching moment coefficient against angle of attack is necessary to bring the calculated values into line with measured values. The adjustment to be made can be represented with sufficient accuracy by the following increments in the pitching moment coefficient,

$$\Delta C_m = -0.04167\alpha^r \quad \text{for} \quad 0 < \alpha < 11^\circ$$

and

$$\Delta C_m = -0.008 \quad \text{for} \quad \alpha > 11^\circ \; .$$

With the continuation of flight testing information became available on the aircraft's response during a manoeuvre of the kind described above and in the course of which the angle of attack reaches a maximum value of nearly 18°. Fig.7 shows a comparison of the test results for this manoeuvre with those obtained by calculation for initial conditions closely approximating those of the flight tests, but using the unmodified wind-tunnel test results. The discrepancy in the elevon angles is immediately evident and, of course, the angle of attack attains the chosen recovery angle of attack of 17.5° rather earlier in the calculated response. These differences are to be expected in the light of the values of the elevon angles to trim.

We next modify the pitching moment characteristics as outlined above and recalculate approximating closely to the flight initial conditions in all but the angle of attack. To adjust to the precise initial conditions would require a considerable amount of recalculation, which scarcely seems to be justified. Also the elevon input was not reproduced in all its detail and consequently the computed histories of the angles of attack and inclination are somewhat smoother than the measured ones. However, the agreement is fairly good in general and justifies the use of the modified pitching moment characteristics (Fig.8a and b).

We now consider the effect of the changes in the shape of the pitching moment curve on the ability to recover from a high angle-of-attack flight condition by direct use of elevon alone. In the calculations whose results are displayed in Fig.9 we once more have a fixed entry into the high angle of attack and a number of different recovery angles of attack.

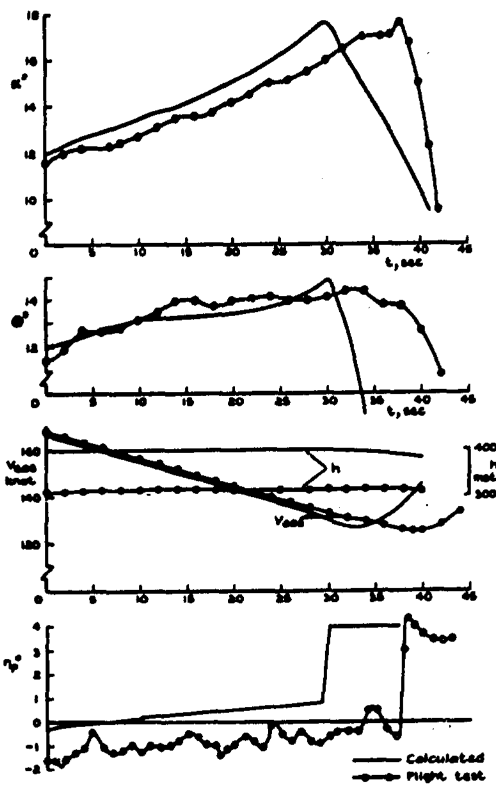


Fig.7 Comparison of calculated motion (WT data) with flight test results.

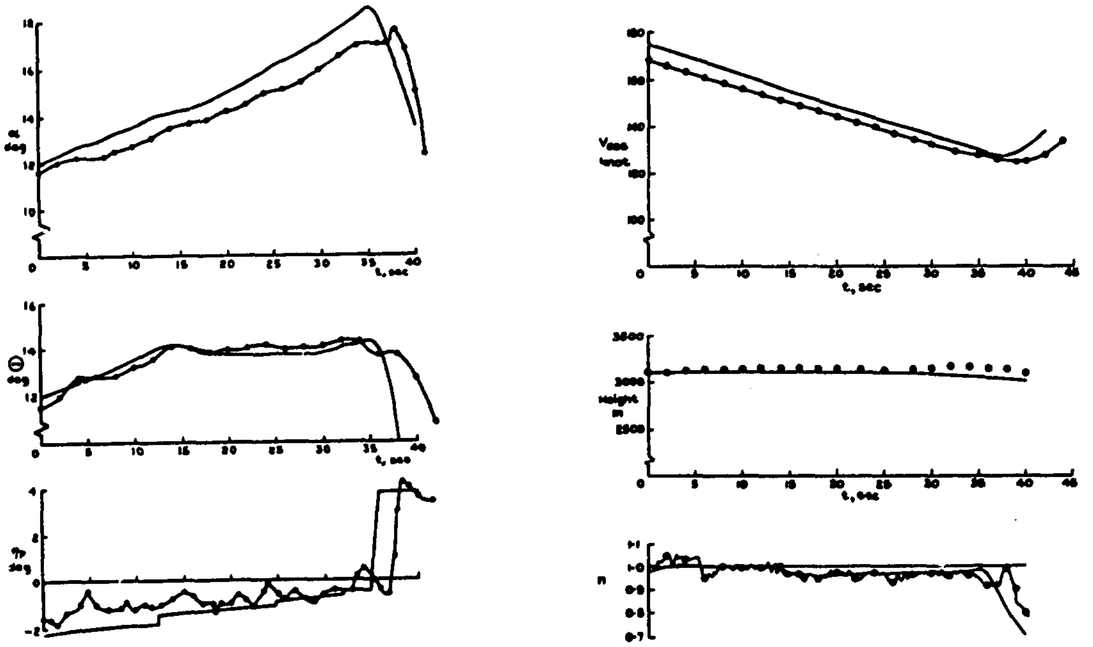


Fig.8a & b Comparison of measured and calculated aircraft response

The values of the latter for which the angle of attack continues to increase after down elevon is applied and held are somewhat higher than those corresponding to the pitching moment characteristics as determined by the wind tunnel tests in spite of the further aft position of the centre of gravity assumed in these later calculations.

A feature of these results (and those of Fig.8) which is worthy of comment is the fact that as compared with the responses shown in Fig.6 there is little rotation of the aircraft until angles of attack in excess of 22° are reached. This is in the first place due to the very gentle nature of the nose-up elevon input and to the fact that at zero elevon angle the pitching moment becomes positive beyond an angle of attack just slightly less than the above value. The increased deceleration associated with the rapid

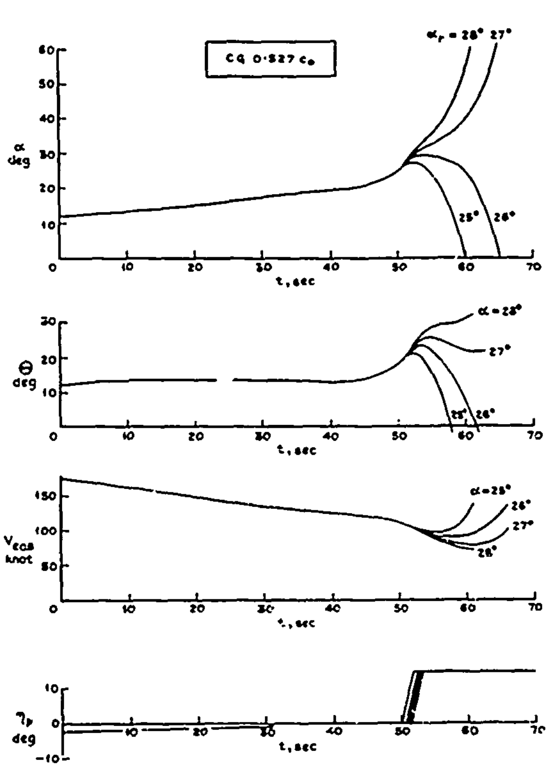


Fig.9 Response of aircraft with modified pitching moment characteristics to elevon input shown

increase in the drag coefficient with increase in the angle of attack in the range 20° to 35°, approximately, also contributes in part to the trends shown by the curves. Once rotation in the nose-up sense begins to grow both the angle of attack and the attitude angle increase at about the same rate. These features of the motion provide the pilot with useful cues. Such motion cues can be obscured or even absent for an aircraft with an abrupt and conventional stall.

On the basis of a number of calculations of this kind and their analysis it is possible to define in some degree the boundary or the threshold between the normal recovery motions and the others. This in turn enables us to arrive at a reasonable basis for flight test procedures and to design, where necessary, a protective automatic recovery system to guard against inadvertent excursions to too large an angle of attack. A number of different protective automatic systems have been employed ranging from warning devices like stick 'shakers', through simple stick 'pusher' systems to fairly complex systems in which the system is activated by a combination of angle of attack and rate of pitch with the control being centralised as lower angles of attack are reached. These have usually fallen short of being truly recovery systems and have sometimes limited the manoeuvring boundary by more than perhaps is strictly needed for safety. The design philosophy of automatic systems for protection in the large angle motion in the six degrees of freedom is a matter which merits further study. However, further discussion of even the simpler longitudinal motion lies outside the scope of this paper, but studies so far suggest that the angle of attack and its rate of change are the principal motion variables involved in the definition of the threshold between the 'preventative' recoveries and the very large angle motions ('superstall' and 'bounce recover')[1]. Accordingly a well-designed automatic system would make use of this information for triggering a recovery control action and enabling as much as possible of the potential of the aircraft to be exploited.

At some stage in the flight testing of the aircraft and the selected automatic recovery system, if fitted, the maximum angle of attack will move outward towards the boundary. It may be considered that the degree of hazard involved warrants the fitting of an independent recovery device under the direct control of the pilot. Let us suppose that the choice falls on a parachute, which usually exerts an appreciable nose-down pitching moment on the aircraft at very large angles of attack.

Some questions arise in relation to the use of such auxiliary systems. For example,

(1) How efficient is the device in promoting recovery?

(2) What criterion should the pilot use for taking the decision to deploy the parachute?

(3) Having deployed the parachute how does he decide that the aircraft is sufficiently under control to permit him to release the parachute?

To provide guidance on such matters it is necessary to calculate the effect on the behaviour of the aircraft during an attempted recovery of varying the instant at which the parachute is deployed. Examples of the results of calculations of this kind are displayed in Fig.10. The curves corresponding to recovery under the action of the elevon, with the automatic recovery system inactive, and for an assumed recovery angle of attack (i.e. angle of attack at which recovery action is initiated) of 26° show that there is time lapse of about 11 seconds before the angle of attack returns to 20°. Deployment of the parachute at 52 seconds (about a second later than the instant when down elevon is applied) cuts this time lapse to about 6 seconds. Earlier deployment of the parachute at 51 seconds (approximately at the same time as down elevon is applied) speeds up recovery even further and brings the time interval between deployment of the parachute and the return to an angle of attack of 20° down to 5 seconds. The nose-down rotation is faster and becomes progressively more so with earlier use of the parachute and should provide the pilot with a good indication that all is progressing according to plan. It is seen from Fig.10 that the equivalent airspeed is hardly affected for the first few seconds due to the fact that the parachute drag just more than offsets the reduction in the basic drag of the aircraft as the angle of attack is brought below its value without parachute. There can be little doubt that the angle of attack and its rate of change are the appropriate criteria for the pilot to use in deciding when to augment recovery action by use of the parachute. Thus as a minimum requirement the pilot must be given a reasonably accurate angle-of-attack indicator with an open scale, so that rate of change can be judged. The exact procedure to be followed can only be determined from a more exhaustive investigation and consideration of the properties of the instrumentation available. It would, however, be expected on the basis of these limited calculations to follow a pattern such as - if the angle of attack reaches some value (say 23° to 24°) and is still increasing maximum nose-down elevon should be applied and if subsequently the trend in the angle of attack is such that its value is likely to exceed 25° the parachute should be deployed.

Let us suppose that for some reason or another not only is the application of fully-down elevon delayed, but that a further delay occurs before deployment of the parachute. These conditions are such that without the parachute the aircraft would continue to rotate nose-up and to lose speed in spite of the maximum down elevon. Consequently very large angles of attack would be reached if the motion were allowed to continue, see Fig.11. Even under the combined action of the elevon and the parachute, which is assumed to be deployed at 53 seconds, recovery is slow and the speed remains low for some while.

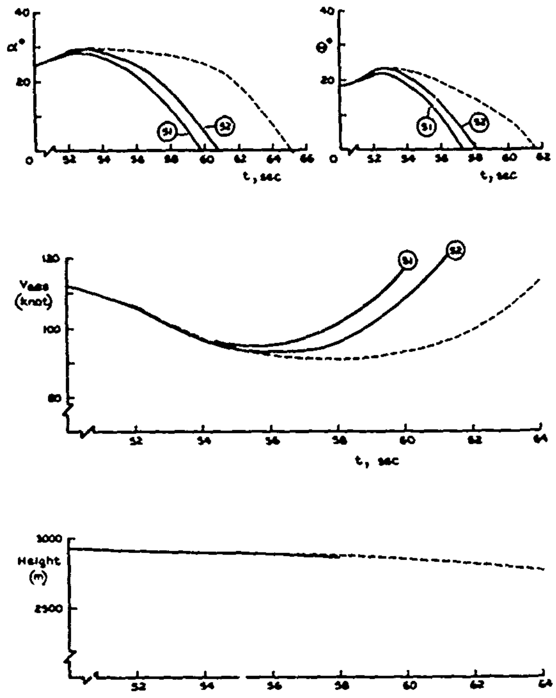


Fig.10 The effect on the recovery motion of deployment of a parachute at the instant indicated ($\alpha_r = 26^0$)

After such an experience a pilot is likely to view with some anxiety the prospect of deciding when to rid the aircraft of its parachute. We consider first a straightforward jettisoning of the parachute as soon as some prescribed angle of attack is reached during recovery and the two values of 22° and 30° are assumed in the calculations, the results of which are shown in Fig.12. At both angles of attack the rate of change is now negative and for the larger angle of attack the aircraft's attitude is level, whilst it is 10° nose-down for the smaller angle of attack. If the decision to release is delayed until the angle of attack reduces to 22° the consequent effect on the motion is hardly perceptible in the angles of attack and inclination and only very slight in the speed variation. A larger effect is apparent (see Fig.12) when the parachute is released after the angle of attack has reduced to 30°, but the speed changes are still small. There is hardly any change in the flight path during the time covered by the calculations. We may conclude that the timing of the release of the parachute is not critical. An early release tends to reduce the overswing to large negative inclination angles. Provided the stick is kept in its forward position until the angle of attack falls to some angle well removed from critical conditions, about 15°, say, there should be no trouble. On the other hand if the angle of attack is allowed to reduce to 22° before release of the parachute the subsequent return to some reasonable steady state should be straightforward, but would require more reversed control to counteract the nose-down rate of pitch that would have developed during recovery.

An alternative procedure would be to reverse control with the parachute still attached and to release it only when the aircraft has returned to a more normal altitude.

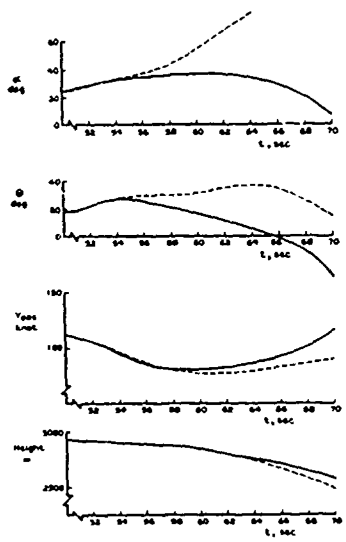


Fig.11 Late recovery action ($\alpha_r = 28^0$ and parachute deployed at 53 seconds)

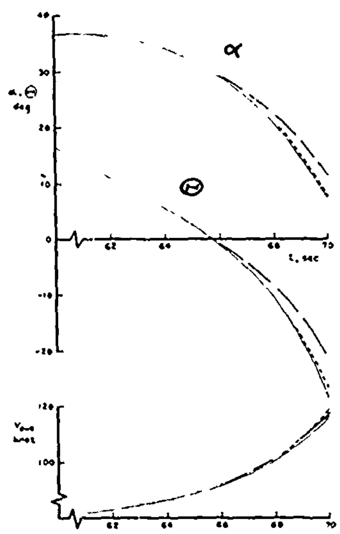


Fig.12 Release of parachute at $\alpha = 22^0$ and 30^0

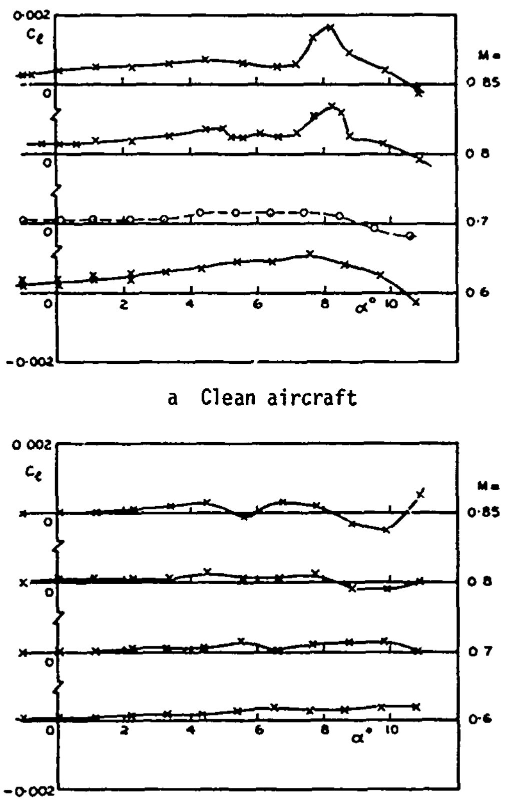
4.2 Lateral characteristics of combat aircraft at high angles of attack

In contrast to the markedly non-linear dynamics just discussed we now turn to a theoretical study, recently undertaken by the second author into the lateral handling qualities of combat aircraft at low and transonic speeds and at high angles of attack based on a linearized treatment of the dynamics. The study is in fact retrospective in nature, but is still illustrative of the type of study we are advocating.

It is well known that most combat aircraft suffer from some form of handling deficiencies at high angles of attack (arising basically from localised stall-separation of the flow at low speed and shock-induced separations at transonic speeds) and that different aircraft types are deficient in different ways[7].

It is not always clear that there is a unique interpretation of some of the terms used to describe the shortcomings in the handling qualities and that these in their turn can be ascribed to certain identifiable aerodynamic characteristics.

At the higher angles of attack associated with the deterioration in handling qualities the aerodynamic forces and moments are non-linear in character and strongly dependent on the angle of attack. Furthermore there may be aerodynamic inputs of a more random nature which can give rise to aircraft response. The rolling moments measured for seemingly symmetrical conditions of flight for one of the aircraft used in the present analysis is a case in point, see Fig.13. Such rolling moments could be a mechanism for 'wing drop' or a trigger for 'wing rock', if a poorly damped Dutch-roll mode of motion exists for small amplitude lateral motion.



a Clean aircraft

b Tanks on

Fig.13 Rolling moment coefficients for aircraft B(β = 0)

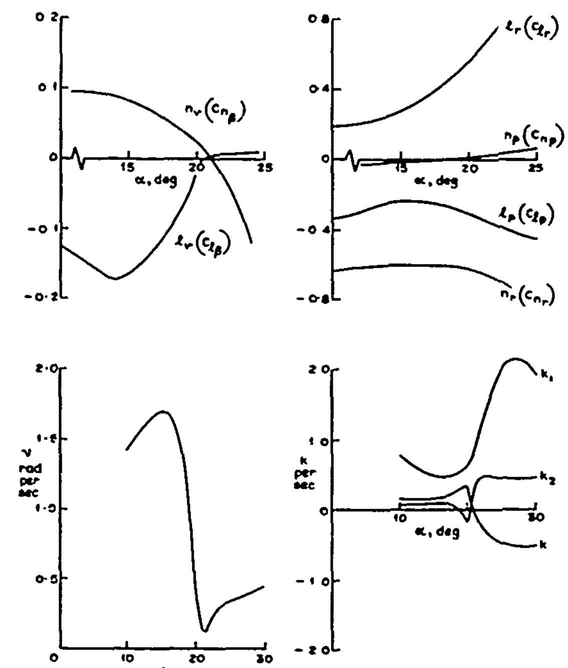


Fig.14 Lateral characteristics of aircraft A

It is reasonable to suppose that certain trends in the aircraft's behaviour would be discernible from a linearized treatment of the lateral motion in view of the fact that non-linearities with respect to sideslip are not so marked as those associated with the angle of attack. However, a complete interpretation of any large angle motion would require a non-linear treatment, particularly if the angle of attack changes during the course of the motion.

4.2.1 Aircraft A

Aircraft A is a twin-jet swept-wing combat aircraft for which early wind-tunnel test data[2], elaborated as necessary by estimates, suggested the set of derivatives shown in Fig.14 for a Mach number of 0.4. Flow separation first occurs at this Mach number for an angle of attack around 12° and from 15° upwards the aerodynamic derivatives change rapidly with the angle-of-attack increase.

On the basis of these aerodynamic data the linearized modes of lateral motion around the steady-turn equilibrium conditions defined by the Mach number of 0.4 and the various g-levels corresponding to different angles of attack all at a height of 25000 ft (7620 m) were determined. As can be seen from

Fig.14 an oscillatory Dutch-roll mode exists throughout the angle-of-attack range. In the neighbourhood of 20° the frequency of the oscillation drops sharply and thereafter remains relatively speaking low. Beyond an angle of attack of about 22° the oscillation is, moreover, unstable. Apart from a very localised instability of the spiral mode near $\alpha = 20°$ the two aperiodic modes are stable.

Because of the large changes in the rolling and yawing moment derivatives with respect to sideslip at the larger angles of attack we should also expect some adverse effect on the roll-response characteristics. It is in fact found that the parameter, which expresses the extent to which the Dutch-roll mode intrudes in the roll-response, namely ω_ϕ^2/ω_d^2 falls to unacceptable (even negative) values for angles of attack greater than 15°.

Later wind-tunnel tests[3], the results of which only came to hand very recently indicate somewhat different trends in the derivatives. The main effect of significance in the present context is a much lower damping-in-roll in the range $15° < \alpha < 23°$. This would make matters worse in this range in the angle of attack.

Flight experience is available and in one recorded incident, when flying at a Mach number of 0.4 the aircraft was put into a banked turn building up to 60° bank, buffet occurred at an angle of attack of about 10°. Just beyond the stall angle of attack of 18° a lightly damped lateral oscillation of small amplitude in sideslip and bank angle is present. This gives way as the angle of attack is increased to a divergent lateral-directional oscillation from which as a result of further gyrations the aircraft entered into spin, from which fortunately the pilot was able to recover. This behaviour is not inconsistent with the indications of the analysis, but it must be stressed that the correlation refers more to the anticipated (and realized) control difficulties one would expect to be associated with the calculated stability and control qualities rather than to a complete explanation of the progression through the various types of motion outlined and described more fully in Re

4.2.2 Aircraft B

During a test programme directed mainly towards a study of the buffet problem using a fighter/trainer aircraft (Aircraft B) handling difficulties were experienced as the normal acceleration is increased and penetration of the buffet regime occurs.

Measured static forces and moments are available from wind-tunnel tests for two conditions of the aircraft, namely, clean and fitted with slipper-type fuel tanks. Due account of the manner in which separation of the flow around the wing affected the static derivatives was taken in the estimates of the other derivatives[4,5,6].

The lateral stability characteristics of the aircraft in the above two conditions were calculated for flight at Mach numbers of 0.6 and 0.85 for linearized perturbation motion about a diving, banked turn. Since in the context of the observed flight behaviour the Dutch-roll mode is the mode of principal interest only the frequency and damping of this oscillatory mode are shown in Figs.15 and 16.

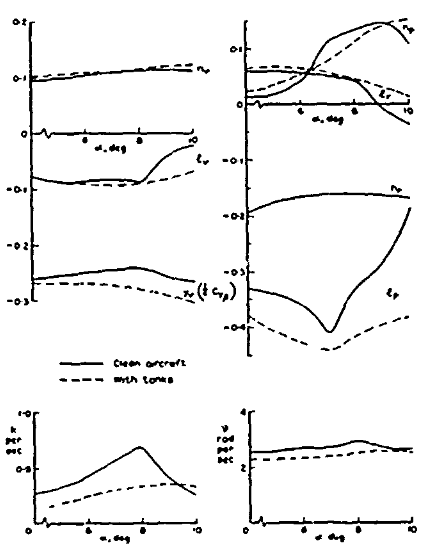


Fig.15 Lateral characteristics of aircraft B (M = 0.6)

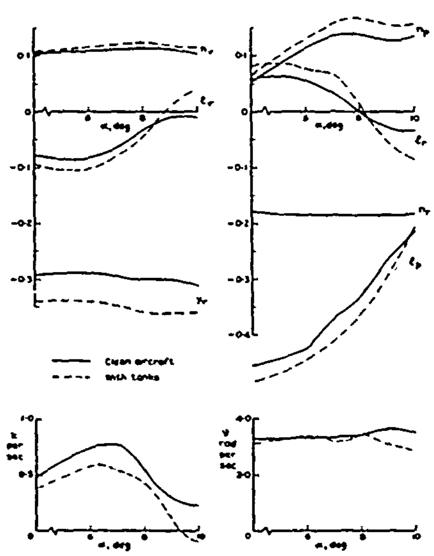


Fig.16 Lateral characteristics of aircraft B (M = 0.85)

Unfortunately tests of a model with control surfaces deflected were not available over the range of the angle of attack of interest. Accordingly it is not possible to assess the aileron-yaw handling characteristics.

To date only qualitative information is available from the flight tests. Pilots' impressions are that it takes the Dutch-roll about 6 to 8 cycles to damp out a disturbance to a very small amplitude in level flight, but as normal acceleration builds up the oscillation becomes more strongly damped with the corresponding decay taking about 3 to 4 cycles. Further increase of normal acceleration is accompanied a very poorly damped, or even unstable, oscillation apparent to the pilot as a 'wing rock' or rolling oscillation. The trends do seem to be reflected in the calculated trends in the damping. However, more tests and analysis are indicated.

4.3 Non-linear lateral motion of HP 115

In Ref.8 it is demonstrated that given the availability of the static forces and moments and oscillatory derivatives over a sufficient range in the angle of attack the non-linear counterpart of the Dutch-roll motion can be predicted. This provides further evidence of the usefulness of calculation of the properties of an aircraft. In the particular case the flight tests and experience preceded the calculations, but it would have been desirable and more reassuring if the order had been reversed.

5 CONCLUDING REMARKS

Not only do the examples just discussed help to illustrate the way that pre-flight calculations can be of benefit in the planning and conduct of flight tests and thus strengthen the arguments of section 2, but they also point to areas where additional work is needed. For instance, it is immediately clear that there is usually a lack of data on the 'dynamic' derivatives. This can hardly be the result of confidence in the methods of estimation particularly for separated flow conditions.

The yawing moment due to aileron and the rolling moment due to rudder, as well as the direct control moments of roll and yaw, respectively, can be subject to large variation during a large angle motion[7]. It is important to take full account of these variations in the moments due to control surfaces in assessing the handling qualities of aircraft and the functioning of an autostabiliser.

A basic purpose of research work in the large angle motions is to formulate meaningful calculations, which refer to a mathematical problem of the utmost simplicity possible without sacrifice of the essential character of the more complex motions that can occur. It is probable that in some cases this can only be achieved by successive simplification of the calculation which reproduces the general flight conditions and results as faithfully as possible. On the other hand the relevance of the linearized treatment in the lateral motion indicates that this is not necessarily always so.

Insofar as success attends the calculation methods so far should it prove possible to tackle in an integrated manner the problems of requirements, design and testing.

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4 A. Goodman and G.H. Adair, "Estimation of damping in roll of wings through the normal flight range of lift coefficient", NACA TN 1924 (1949)

5 J.W. Wiggins, "Wind-tunnel investigation of effect of sweep on rolling derivatives at angles of attack up to 13° and at high subsonic Mach numbers, including a semiempirical method of estimating the rolling derivatives", NACA TN 4185 (1958)

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7 H.H.B.M. Thomas, "On problems of flight over an extended angle-of-attack range", RAE Technical Report 71013 (ARC 32887) (1971)

8 A. Jean Ross, "An experimental and analytical study of non-linear motion experienced on a slender-wing research aircraft", AIAA Paper 72-62 (1972)

OPEN DISCUSSION

D.M.McGregor, Canada: I did not understand your figure of α and $\dot{\alpha}$. Would you please elaborate.

H.H.B.M.Thomas, UK: The figure referred to (Fig.5 of the paper) is an example of the use of the so-called "phase-plane" method of analysing the stability of a non-linear system. The classic reference on the method is "Theory of Oscillations" by Andronow and Chaikin, Princeton University Press, 1949 although it is usually described in most of the text books on non-linear dynamics.

As applied to the equation of motion governing the pitching oscillations of an aircraft for a single-degree-of-freedom, the procedure is as outlined below. The equation of motion for an aircraft having non-linear damping and stiffness characteristics has the following form,

$$\ddot{\alpha} + f(\alpha,\alpha)\dot{\alpha} + g(\alpha,\dot{\alpha})\alpha = \text{constant} = C.$$

This can be rewritten, if we write $q = \dot{\alpha}$, in the form

$$\frac{dq}{d\alpha} = \frac{f(\alpha,\dot{\alpha})q + C - q(\alpha,q)}{q} = \frac{P(\alpha,q)}{Q(\alpha,q)}, \text{ say.}$$

The figure therefore displays the various solutions of this equation, which defines the slope of the solution through any point. For points at which both P and Q are zero the slope is indeterminate and such points are termed singular points. When the equation describes a physical system the singularities are associated with the equilibrium conditions. There are two such points shown in Figure 5a and the one at $\alpha \sim 48°$, $\dot{\alpha} = 0$ is a "saddle-point" or "Col", whilst the one at $\alpha \sim 58°$, $\dot{\alpha} = 0$ is a stable "spiral" point. A curve which enters or leaves a "Col" is called a "separatrix". For all curves, including the separatrix, increasing time corresponds to passage from left to right along the curves in the first quadrant but right to left in the second quadrant.

Of particular interest to us in the context of the present paper is the region between the two left-hand branches of the separatrix. It is further known (see Reference 1 of the paper) that a more relevant definition of the range of α and $\dot{\alpha}$ for which "preventative" recovery is likely to be possible is obtained if the speed for the motion in pitch only is taken as that which corresponds to the "saddle-point" equilibrium. Further information can be found in Reference 1.

FLIGHT TEST EXPERIENCE IN AIRCRAFT PARAMETER IDENTIFICATION

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SUMMARY

An automatic method for determining stability and control derivatives from flight data has been developed. The technique, a modification of the Newton-Raphson method for derivative extraction, has an a priori provision that makes use of initial estimates of the derivatives and that therefore provides a means of checking the validity of the results.

Consideration is also given to future applications of the method.

SYMBOLS

The body system of axes and radian measure are used throughout the paper unless specifically stated otherwise.

a_n, a_y — normal and lateral accelerations of the airplane, respectively, at the center of gravity, g

b — wingspan, m (ft)

C_D — drag coefficient

C_L — lift coefficient

C_l — rolling-moment coefficient, $\frac{\text{Rolling moment}}{\bar{q}Sb}$

$C_{l_{i=p,r}} = \frac{\partial C_l}{\partial \frac{ib}{2V}}$

$C_{l_{i=\beta,\delta_a,\delta_r,\delta_s}} = \frac{\partial C_l}{\partial i}$

C_m — pitching-moment coefficient, $\frac{\text{Pitching moment}}{\bar{q}S\bar{c}}$

$C_{m_{i=\alpha,\delta_e}} = \frac{\partial C_m}{\partial i}$

$C_{m_q} + C_{m_{\dot{\alpha}}} = \frac{\partial C_m}{\partial \frac{q\bar{c}}{2V}} + \frac{\partial C_m}{\partial \frac{\dot{\alpha}\bar{c}}{2V}}$

C_N — normal-force coefficient, $\frac{\text{Normal force}}{\bar{q}S}$

$C_{N_{i=\alpha,\delta_e}} = \frac{\partial C_N}{\partial i}$

$C_{N_q} + C_{N_{\dot{\alpha}}} = \frac{\partial C_N}{\partial \frac{q\bar{c}}{2V}} + \frac{\partial C_N}{\partial \frac{\dot{\alpha}\bar{c}}{2V}}$

C_n — yawing-moment coefficient, $\frac{\text{Yawing moment}}{\bar{q}Sb}$

$C_{n_{i=p,r}} = \frac{\partial C_n}{\partial \frac{ib}{2V}}$

$C_{n_{i=\beta,\delta_a,\delta_r,\delta_s}} = \frac{\partial C_n}{\partial i}$

C_y — side force coefficient

$C_{y_\beta} = \frac{\partial C_y}{\partial \beta}$

$\bar{c}$ — wing mean aerodynamic chord, m (ft)

$\underline{c}$ — vector of unknown derivatives

$\underline{c}_k$ — k^{th} iterated value of the derivative vector

$\underline{c}_o$ — vector of a priori estimates of unknown derivatives

D_1	instrument measurement weighting matrix
D_2	a priori weighting matrix
h_p	pressure altitude, m (ft)
I_X, I_Y, I_Z	moments of inertia about the X-, Y-, and Z-body axes, respectively, kg-m^2 (slug-ft^2)
I_{XZ}	product of inertia referred to X- and Z-body axes, kg-m^2 (slug-ft^2)
$\frac{\|i\|}{\|j\|}$	amplitude ratio of vector quantities i and j
J	cost functional (performance index) or weighted mean-square-fit error
$\underline{\nabla_{\underline{c}} J}, \nabla_{\underline{c}} (\underline{\Delta y})$	vector of gradient of J and Δy, respectively, with respect to $\underline{c}$
$\underline{\underline{\nabla_{\underline{c}}^2 J}}$	vector of the second gradient of J with respect to $\underline{c}$
K	scaler weighting factor (gain) for a priori weighting matrix

$\bar{L}_{i = p, r} = C_{l_i} \frac{\bar{q} S b^2}{2 V I_X}, \frac{1}{sec}$

$\bar{L}_{i = \beta, \delta_a, \delta_r, \delta_s} = C_{l_i} \frac{\bar{q} S b}{I_X}, \frac{1}{sec^2}$

M	Mach number

$\bar{N}_{i = p, r} = C_{n_i} \frac{\bar{q} S b^2}{2 V I_Z}, \frac{1}{sec}$

$\bar{N}_{i = \beta, \delta_a, \delta_r, \delta_s} = C_{n_i} \frac{\bar{q} S b}{I_Z}, \frac{1}{sec^2}$

p, q, r	roll, pitch, and yaw rate, respectively, rad/sec
$\dot{p}, \dot{r}$	roll and yaw acceleration, respectively, rad/sec^2
$\bar{q}$	dynamic pressure, N/m^2 (lb/ft^2)
S	wing area, m^2 (ft^2)
T	total time, sec
t	time, sec
V	velocity, m/sec (ft/sec)
$\underline{\Delta y}$	error vector
α, β	angle of attack and sideslip, respectively
$\dot{\alpha}$	rate of change of angle of attack, rad/sec
Δ	increment
$\delta_a, \delta_e, \delta_r, \delta_s$	aileron, elevator, rudder, and spoiler deflections, respectively
ζ	Dutch-roll damping ratio
$\Phi_{p\beta}$	phase angle of p relative to β
φ	bank angle

Subscript

k	iteration index

Superscript

T	matrix transpose

INTRODUCTION

As new and more extensive flight programs have been undertaken over the last 20 years, the NASA Flight Research Center has continually upgraded the techniques used to determine stability and control parameters. The F-100 program saw the first intensive use of the time-vector technique (ref. 1), which included refinements in its application. With the X-15 program came the first, although rudimentary, use of the analog-matching technique (ref. 2). The lifting-body program, although originally dependent on the analog-matching technique, now utilizes a computerized output-error technique which is a modification of the Newton-Raphson method. The modified Newton-Raphson method has also been used successfully in flight studies of the Convair 990 transport (ref. 3), F-8 and F-111 fighter, and PA-30 general aviation airplane. It is currently being used in all Flight Research Center flight investigations involving stability and control parameter identification.

Automatic methods for determining stability and control derivatives from flight test data, which are being made feasible by the increased capability of more sophisticated computers and data acquisition systems, minimize dependence on the analyst's skill. The techniques established earlier, however (e.g., approximate equation, time vector, and analog matching (see ref. 4)), remain valuable where computer facilities are inadequate or unavailable or for validation of newer methods. The time-vector technique, in particular, continues to be valuable for providing insight into problems encountered in analysis of flight data.

The purpose of this paper is to give a brief review of earlier parameter identification methods, including improvements in the analog-matching technique, and to discuss the formulation and utilization of the modified Newton-Raphson method. The paper includes examples of the application of the method to different situations. Finally, consideration is given to extended applications of the Newton-Raphson method that may be possible in the future.

REVIEW OF EARLIER METHODS

The more successful methods of flight derivative identification have been the time-vector and analog-matching techniques. These methods are well known and only a few observations, concerning their limitations and refinements to them, will be made.

Time-Vector Method

Normally the time-vector method can be applied only to control-fixed time histories of transient-oscillation responses with damping ratios of less than approximately 0.3. The method can solve for only two unknowns in any one equation. The success of the application of the method is highly dependent on the technique and the skill of the analyst. Special considerations must be made to handle maneuvers with the stability-augmentation system on or other forms of dependent control movements, which then also makes knowledge of the control derivatives necessary.

Equation-Error Methods

Equation-error methods are row independent in that each equation is solved independently of the others. The methods minimize the error between the acceleration determined in flight and that predicted by the equation. The equation-error methods do not minimize the errors between flight-determined and predicted variables in the equations (such as angular rates and displacements). As reported in references 5 and 6, the equation-error methods provide inferior solutions compared to those obtained by the analog-matching technique (which is essentially an output-error method).

The row independence of the equation-error methods is one of their weaknesses. Another disadvantage of the methods, as pointed out in reference 7, is that all of the variables in the equation must be measured, and that the accuracy of the instrumentation must be nearly perfect.

Analog-Matching Method

Analog matching has been important because it has successfully analyzed flight data the methods discussed earlier failed to handle. It is not restricted to transient-oscillation maneuvers, as is the time-vector method. Because it is essentially an output-error technique, it minimizes the errors of the various responses iteratively (through the human operator), and thus is an improvement over the equation-error methods. However, when working with maneuvers made with the stability-augmentation system on, the maneuvers are difficult to analyze. Initial applications of this technique were laborious (ref. 2) but have improved with time (ref. 8).

Several refinements have been made to the analog-matching technique at the Flight Research Center by incorporating a hybrid computer. The control inputs programed into the analog are now stored in a digital computer. Before the digital computer was used, the control inputs were stored through a recorder, on magnetic tape. The playback of the tape through the analog caused noise problems that resulted in hashy response outputs on the oscilloscope. The use of the digital computer eliminated this problem.

Flight time histories are now also stored in the digital computer and, through the analog, are displayed on the oscilloscope for direct comparison with the analog response of the mathematical model. This direct display of the flight time histories has replaced the less accurate use of transparent plastic-sheet overlays on the scope, has eliminated potential parallax errors and other distortions, and has also minimized the need for a final match using a strip record from a precision recorder and an overlay to check the fidelity of the scope match.

The nondimensional derivative format of the equations of motion has been replaced by a dimensionalized format, which simplifies analog circuitry. Analysis can be accomplished through the dimensional derivative form independent of the inertias, which are at times not known with precision at the time of analysis. The nondimensional derivatives can be obtained from the dimensional derivatives at any time after the analog match.

The analog technique, as employed, also has an equation-error option which makes it possible to arrive rapidly at an initial estimate of the derivatives. The approximate derivatives thus obtained are used in the normal analog-match procedure to reduce the time involved in the analysis.

With the refinements described above, the time involved in analyzing a lateral-directional flight maneuver, from the receipt of the data to final results, has been reduced from approximately 20 hours to 4 hours or less. The time required for the analog matching itself has been reduced considerably.

Although the analog-matching technique has been refined in many respects, the accuracy of the results obtained with it is still dependent upon the experience and skill of the individual operator, who constitutes the feedback loop to minimize the response errors between flight and computed time histories. It would be advantageous to replace the analog technique with one that eliminates dependence on the operator's skill and yet applies an iterative correction technique.

THE NEWTON-RAPHSON METHOD AND ITS MODIFICATION

In an effort to minimize dependence on the operator's skill, attention was focused on computerized output-error methods that minimize the error between the various recorded flight responses and the corresponding responses of the mathematical model. Representing the error by the error vector

$$\underline{\Delta y} = [\Delta\dot{p}, \Delta\dot{r}, \Delta p, \Delta r, \Delta\beta, \Delta a_y]^T \tag{1}$$

the objective of the output-error methods is to minimize $\underline{\Delta y}$ in some manner using the cost functional

$$J = \int_0^T (\underline{\Delta y})^T D_1 \underline{\Delta y}\, dt \tag{2}$$

where D_1 is a weighting matrix reflecting the relative confidence in the instrument measurements.

Although the equations of motion used are linear, the problem of minimizing Eq (2) is nonlinear in unknown coefficients, so some form of iterative solution is necessary.

Attempts were made to use the standard gradient technique (steepest descent), which is the simplest minimization technique; however, the method was unsuccessful (ref. 6). The minimization of the cost functional was extremely slow; a minimum was never reached.

Of the various other methods available for nonlinear minimization, the Newton-Raphson method was selected and modified to provide successful minimization.

The Modified Newton-Raphson Method

The Newton-Raphson technique uses a two-term Taylor series expansion of the gradient $\nabla_{\underline{c}}J$ of the cost functional to minimize the gradient in successive iterations. On the basis of successive iterations, finding a root of $\nabla_{\underline{c}}J$ for the k^{th} iteration is presented by

$$\left(\nabla_{\underline{c}}J\right)_{k+1} = \left(\nabla_{\underline{c}}J\right)_k + \left(\nabla_{\underline{c}}^2 J\right)_k \Delta\underline{c}_{k+1} \tag{3}$$

With each iteration, $\left(\nabla_{\underline{c}}J\right)_{k+1}$ is considered equal to zero with the result that for any one k^{th} iteration, the change in the derivative vector $\Delta\underline{c}$ for a successive approximation is

$$\Delta\underline{c}_{k+1} = -\left[\left(\nabla_{\underline{c}}^2 J\right)_k\right]^{-1}\left(\nabla_{\underline{c}}J\right)_k \tag{4}$$

which is the Newton-Raphson algorithm.

As pointed out in reference 6, the method attempts to predict where the local minimum point is and step directly to it. If the complete second gradient matrix is used, the computation task is enormous. The computation task is reduced significantly by approximation of the second gradient matrix. With the approximation applied, the first and second gradients are

$$\left(\nabla_{\underline{c}}J\right)_k = 2\left\{\int_0^T (\underline{\Delta y})_k^T D_1 \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k dt\right\}^T \tag{5}$$

$$\left(\nabla_{\underline{c}}^2 J\right)_k \approx 2\int_0^T \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k^T D_1 \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k dt \tag{6}$$

This approximated form of the Newton-Raphson technique has been referred to as the modified Newton-Raphson method.

Figure 1 shows the rapid reduction in the weighted fit error, J, for a typical lateral-directional case when the modified Newton-Raphson method is used. For the example shown, only four iterations were necessary to obtain a solution. This represents a total computation time of about 6 minutes on the IBM 360/50 digital computer.

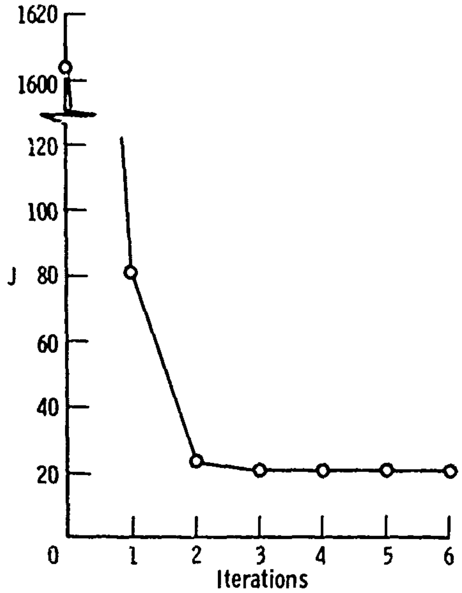


Figure 1. Convergence of fit error for the modified Newton-Raphson method (from ref. 6).

Addition of A Priori Feature

Initial applications of the modified Newton-Raphson method showed that with good data the method had the potential of providing results equal or superior to those obtained from analog matching performed by a highly skilled operator. Further study of the method also showed that versatility could be provided that would allow data with deficiencies or abnormalities to be successfully analyzed. As a result, an a priori provision was incorporated in the method to guide the analysis process so that the best values for the derivatives could be obtained. This a priori provision was made by expanding the cost functional. The expanded cost functional became

$$J = \int_0^T (\underline{\Delta y})^T D_1 \underline{\Delta y}\, dt + (\underline{c} - \underline{c_o})^T D_2 K (\underline{c} - \underline{c_o}) \qquad (7)$$

where D_2 is the weighting matrix for the a priori estimates of the derivative vector $\underline{c_o}$. The term K is a scalar weighting factor for D_2. These two weighting factors are discussed in detail in reference 6.

The first and second gradients of the above cost function, applied to the Newton-Raphson algorithm, were determined to be

$$(\underline{\nabla_c J})_k = 2\left\{\int_0^T (\underline{\Delta y})_k^T D_1 \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k dt\right\}^T + 2D_2K(\underline{c_k} - \underline{c_o}) \qquad (8)$$

$$\left(\nabla_{\underline{c}}^2 J\right)_k \approx 2\int_0^T \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k^T D_1 \left[\nabla_{\underline{c}}(\underline{\Delta y})\right]_k dt + 2D_2K \qquad (9)$$

With the a priori provision included, the modified Newton-Raphson solutions for the flight-determined derivatives may be biased in favor of the best a priori estimates based on wind-tunnel data, previously obtained flight data, or theoretically obtained derivatives. If sufficient information is available in the flight responses to warrant change, a derivative well defined in the responses would converge to a best value. A derivative poorly defined by the responses would tend to remain at the a priori value.

Figure 2 shows the results of the application of the method with a priori to HL-10 lifting-body flight data. The figure compares predicted and flight-determined variation of lateral-directional characteristics with angle of attack at a Mach number of 1.2. The predicted characteristics were used as a priori estimates. Much of the data would have been difficult to analyze by established methods. Some of the problems involved in the analysis of these data are discussed in the latter part of the next section in the discussion of Analysis With Dampers On.

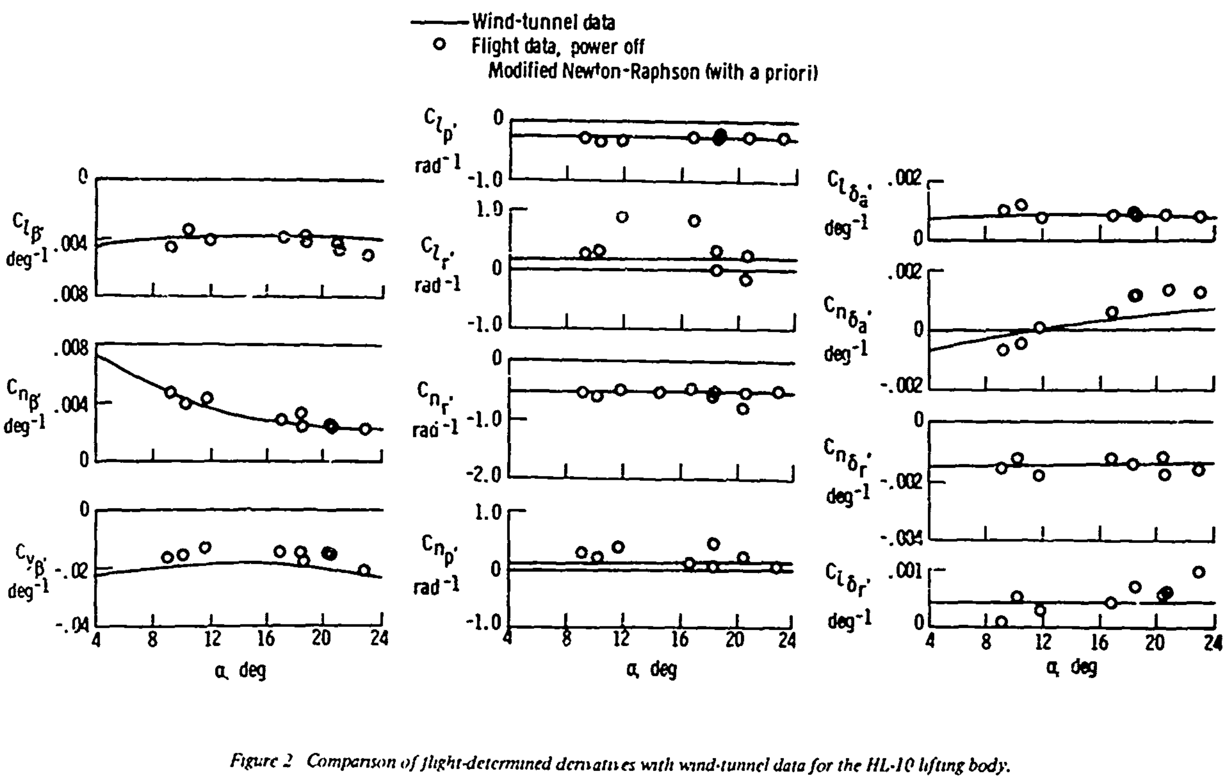


Figure 2. Comparison of flight-determined derivatives with wind-tunnel data for the HL-10 lifting body.

TYPICAL APPLICATIONS OF THE MODIFIED NEWTON-RAPHSON METHOD

Successful application of the modified Newton-Raphson method, as with all other methods, is dependent upon the amount of recorded information and the information contained in the responses to inputs, as well as the quality of the recorded data. Discussions of several situations encountered at the Flight Research Center follow.

Analysis of Multiple Maneuvers

The flight test program of the Convair 990 airplane (ref. 3) showed the lateral-directional mode to be characterized by low Dutch-roll damping (ζ = 0.07) and approximately neutral spiral stability. The angular accelerations were not available, and the sideslip angle, β, was not matched due to inadequate calibration of the vane. A set of three lateral-directional stability and control maneuvers consisting of rudder, aileron, and aileron-plus-spoiler doublets was obtained for each flight condition. The maneuvers were analyzed by the Newton-Raphson method without a priori. As shown in figure 3, significant differences were evident between derivatives obtained from separately matching the rudder doublet

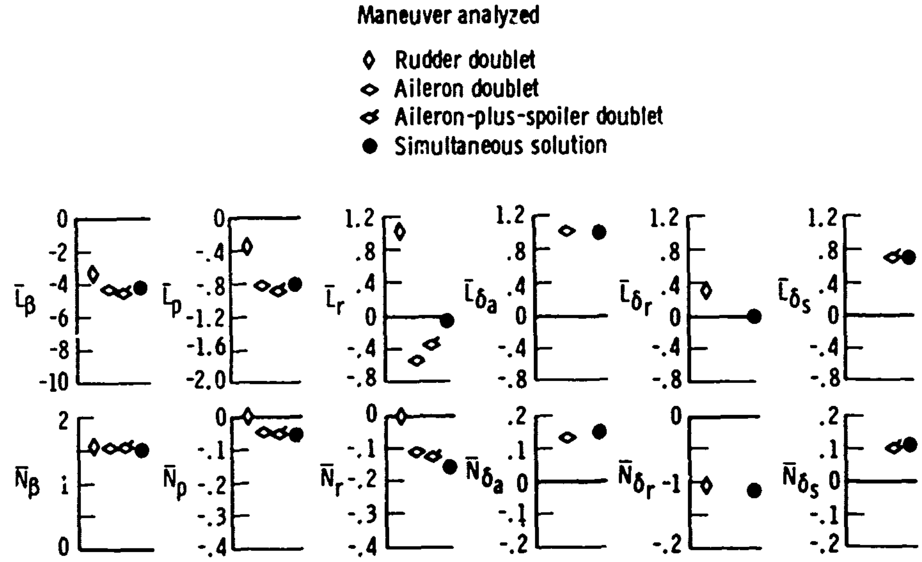


Figure 3. A comparison of derivatives from maneuvers analyzed separately and simultaneously by the modified Newton-Raphson method without a priori. Convair 990, M = 0.80, h_p = 10,670 m (35,000 ft)

and aileron maneuvers. When all three maneuvers were analyzed simultaneously, a set of derivatives was obtained which showed good consistency with derivatives obtained at other flight conditions analyzed in the same manner. Figure 4 shows a typical match for the rudder-doublet maneuver using the derivatives obtained from the simultaneous solution of three maneuvers. Although the flight time history of the sideslip was not used in the analysis, the calculated time history shows reasonably good correlation.

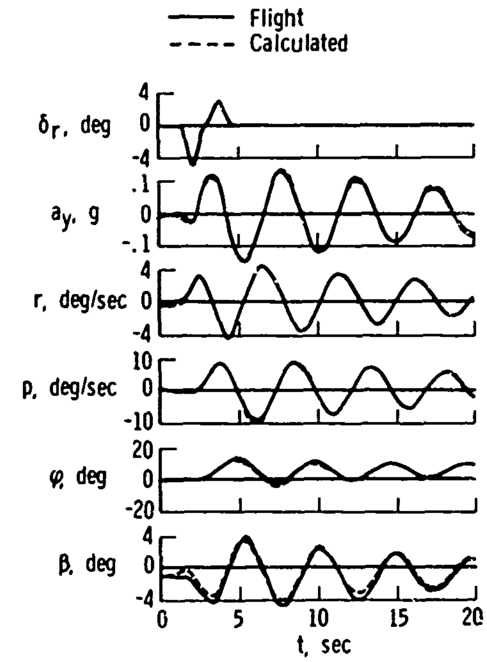


Figure 4. Comparison of Convair 990 flight and calculated time histories of a rudder-doublet maneuver. M = 0.80, h_p = 10,670 m (35,000 ft).

The Convair 990 derivative analysis illustrated the desirability of knowing both yaw and roll control inputs. The various control inputs excite the maneuvers differently and provide additional valuable information. Aileron control inputs provide a better excitation of the roll mode than rudder inputs.

Verification of Flight Derivatives

During recent flight tests of a high aspect ratio sweptwing airplane at transonic speeds, maneuvers were obtained that contained rudder and aileron excitation. These data were analyzed by the Newton-Raphson method with and without a priori and by the time-vector method for comparison with wind-tunnel data (fig. 5). Differences between the non a priori and the a priori results, when there is agreement between the a priori and wind-tunnel results, indicate that insufficient information was contained in the flight maneuver to identify the derivative (see C_{l_r}, for example). However, agreement between non a priori, a priori, and wind-tunnel results (see $C_{l_{\delta_r}}$) indicates that there is sufficient information in the maneuver to identify the derivative, and the agreement can be considered valid. If the non a priori and the a priori results agree but differ from the wind-tunnel data (see $C_{l_{\delta_a}}$), the flight values should be considered valid. The time-vector results, where applicable, tend to confirm the a priori results.

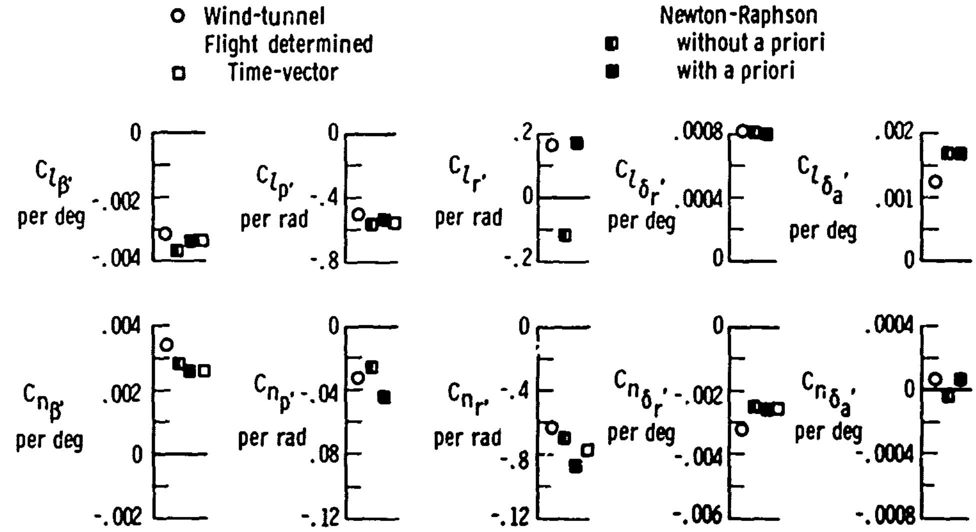


Figure 5 Comparison of predicted and flight-determined lateral-directional derivatives for a high aspect ratio, swept-wing airplane.

The excellent match of the flight and calculated time histories shown in figure 6, using the results of the modified Newton-Raphson method with the a priori provision given in figure 5, indicates that the derivatives represent the airplane for the condition analyzed.

Analysis of Incomplete Data

In the Flight Research Center flight test investigation of a light twin-engine, propeller-driven airplane, responses to rudder-doublet inputs were obtained. After the flight program the records were analyzed to compare flight-determined derivatives with predicted derivatives. As was the case with the Convair 990, the rudder doublet, which excites primarily the Dutch-roll mode, did not provide a complete set of data from which the derivatives could be easily extracted. Preliminary analysis indicated that C_{l_β} was significantly different from its predicted value. As a result, several methods were used to establish the best estimate of the flight value of C_{l_β} from the available data sets.

Initial attempts to solve for C_{l_β} and C_{l_p} by the time-vector technique resulted in the vector diagram illustrated in figure 7(a), which shows the roll rate, p, to be nearly 180° out of phase with the sideslip, β. Since knowledge of precise values of the phase angle $\Phi_{p\beta}$ and the derivative C_{l_r} was crucial to successful determination of C_{l_β} and C_{l_r}, there was no possibility of obtaining C_{l_β} and C_{l_p} with this technique. In view of this situation, attention was focused on determining C_{l_β} and C_{l_r}, using reasonably accurate theoretical values of C_{l_p}, as shown by the diagram in figure 7(b). The time-vector solutions

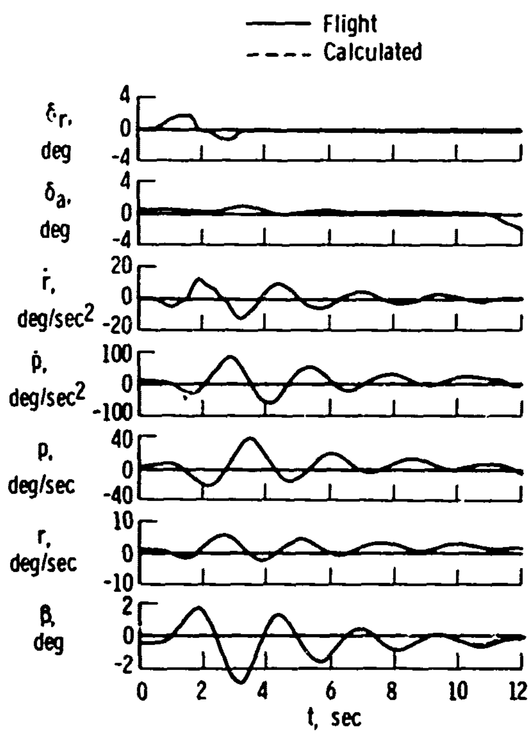
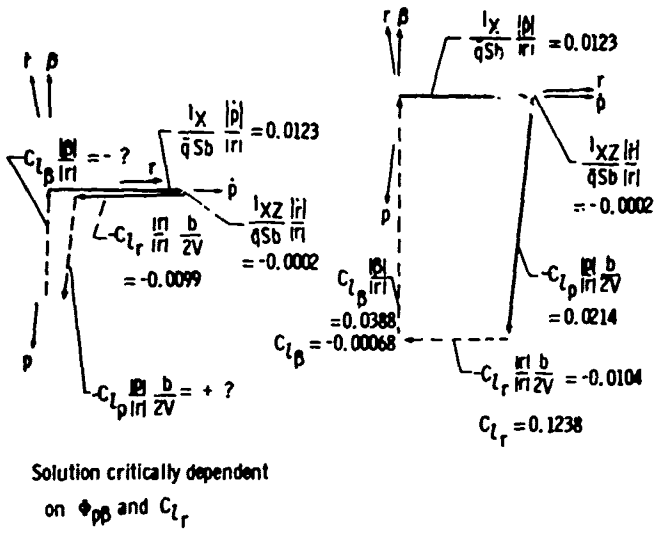


Figure 6 Comparison of flight and calculated time histories of a high aspect ratio, swept-wing airplane.



(a) Attempted solution for C_{l_β} and C_{l_p}. *(b) Solution for C_{l_β} and C_{l_r}.*

Figure 7 Graphical time-vector solution for rolling-moment derivatives of a light, twin-engine, propeller-driven airplane.

for C_{l_β} showed values which were considerably less than reflected by wind-tunnel data or theoretical calculations.

The modified Newton-Raphson method, both without and with the a priori provision, was also applied to this case. Figure 8 shows a comparison of derivatives obtained from a rudder-doublet maneuver as determined by the modified Newton-Raphson method with the time-vector results and with predictions based on wind-tunnel tests and theory. The a priori information was based on the wind-tunnel static stability and control derivatives and theoretical dynamic stability derivatives.

Without a priori, the coefficients obtained with the modified Newton-Raphson solution, using the rudder-doublet maneuver, differ substantially from the solution with a priori and from the time-vector results for C_{l_β}, C_{l_p}, $C_{l_{\delta_r}}$, and C_{n_r}.

The use of a rudder maneuver does not excite the roll mode as effectively as an aileron maneuver, and thus provides poorer conditioning for the determination of C_{l_p} in particular.

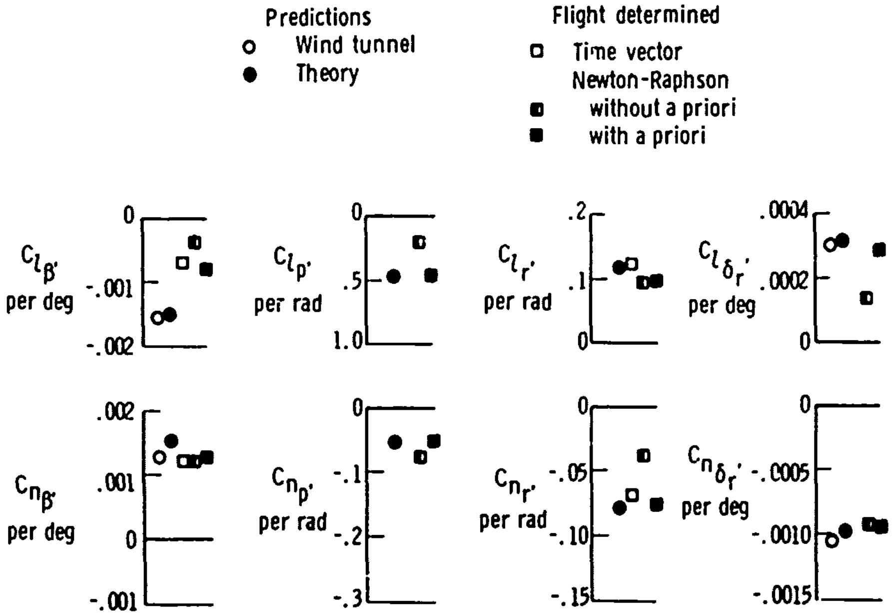


Figure 8. Comparison of predicted and flight-determined lateral-directional derivatives for a light, twin-engine, propeller-driven airplane

With a priori, results for all the derivatives agree well with the time-vector results (where applicable), which also agree well with predictions except for C_{l_β}. The fact that the several analytical techniques used, including sideslip equations, persistently showed the discrepancy between flight and predicted values of C_{l_β}, validates the flight value of C_{l_β}. Figure 9 shows the correlation of computed and flight time histories for this case.

Analysis With Longitudinal Control Movement

At times control movement during a maneuver is unavoidable. Figure 10 shows the recorded input and responses

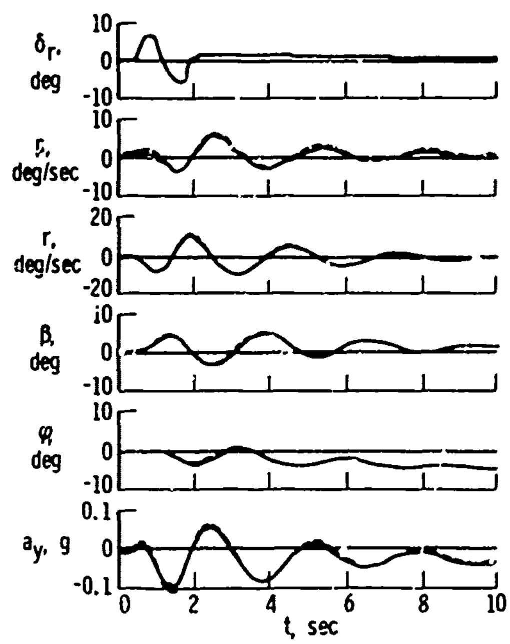


Figure 9. Comparison of flight and calculated time histories of a light, twin-engine, propeller-driven airplane.

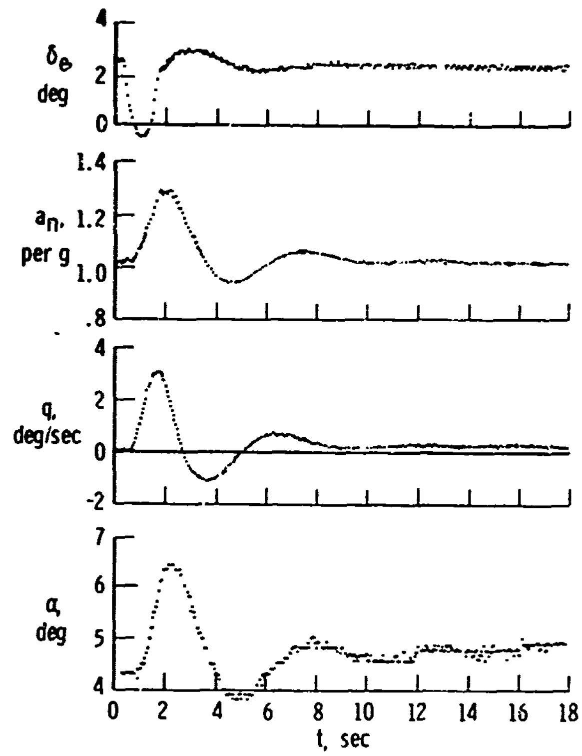


Figure 10. Typical XB-70-1 flight data of pull-up and release maneuver $M = 0.75$, $h_p = 7{,}650$ m (25,100 ft).

of a pullup and release maneuver of the XB-70 airplane at Mach 0.75. Even though the pitch stability augmentation system was off, the elevator continued to move approximately 180° out of phase with the pitch rate, q. The normal acceleration and pitch rate time histories are well defined, whereas the angle of attack and elevator time histories have poor resolutions after the initial part of the maneuver. The maneuver was made during the early stages of the exploratory flight program when a full range of data coverage was provided to the detriment of derivative identification. However, for flight safety, the best estimates for derivatives must be made from the data available.

The maneuver was analyzed by several methods, including the time-vector method, which required the addition of the elevator as a variable. For the Newton-Raphson method with the a priori feature included, the a priori estimates were based on predicted flexible aircraft characteristics provided by the manufacturer. The results are presented in figure 11. The results indicate that careful analysis by any one of the four methods used will give solutions for the major derivatives. For these derivatives, the flexibility effects appear to be properly accounted for by predictions, except perhaps for C_{N_α}. The minor derivatives, $C_{N_q} + C_{N_{\dot\alpha}}$ and $C_{N_{\delta_e}}$, which are generally difficult to assess, are not so clearly identified. However, it does appear that $C_{N_q} + C_{N_{\dot\alpha}}$ is much smaller than predicted. Also, $C_{N_{\delta_e}}$, on the basis of the rough correspondence of the analog-match and a priori results, appears to be predicted fairly well. The analysis by the Newton-Raphson method was straightforward and required no special attention by the analyst.

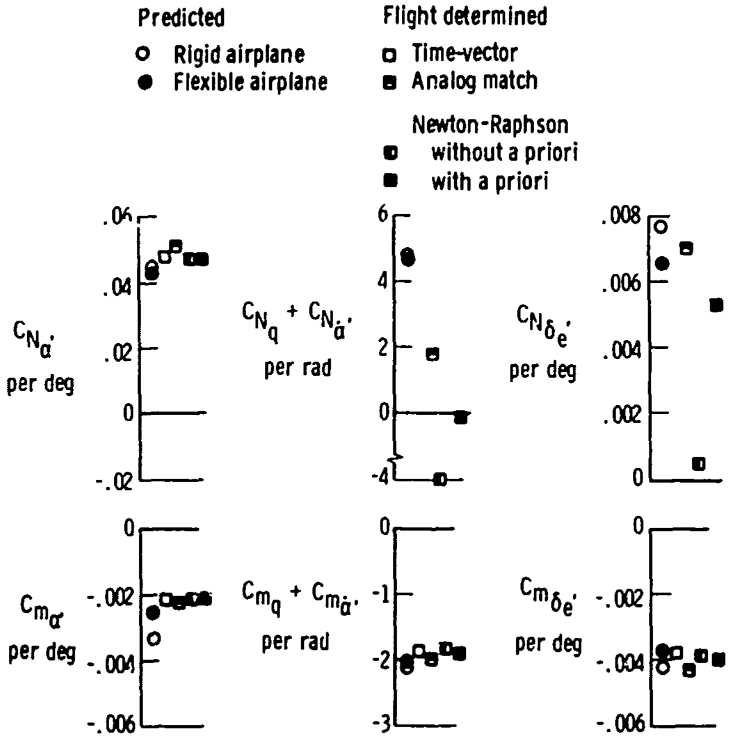


Figure 11. Comparison of predicted and flight-determined XB-70 longitudinal derivatives at Mach 0.75.

Analysis With Dampers On

In the HL-10 lifting-body program, the determination of power-off stability and control data at constant Mach number and angle of attack conditions involved changes in dynamic pressure due to the unavoidable loss in altitude. In a number of instances, the dynamic pressure changed as much as 20 percent. Fortunately, because of the very rigid nature of the aircraft, aeroelastic effects were not a factor and the dynamic pressure change was determined to be acceptable. Each maneuver was planned to generate the maximum amount of response information consistent with flight safety, which in certain flight regimes required roll and yaw dampers to be on.

The lateral-directional maneuver used in the HL-10 program was initiated by a strong aileron-doublet input followed by a short period without pilot control inputs which was then followed by a rudder-doublet input. During the period without pilot inputs, in maneuvers where yaw and roll dampers were on, the damper action caused the aileron to be approximately 180° out of phase with the roll rate and the rudder to be approximately in phase with yaw rate. The dampers made it difficult to analyze the maneuvers with some techniques, but analysis with the Newton-Raphson method with the a priori provision was straightforward. It should be noted that the damper action degrades the conditioning of the responses, so the a priori "option" was more a requirement.

Figure 12 shows a typical Newton-Raphson match with a priori of flight and calculated time histories of a lateral-

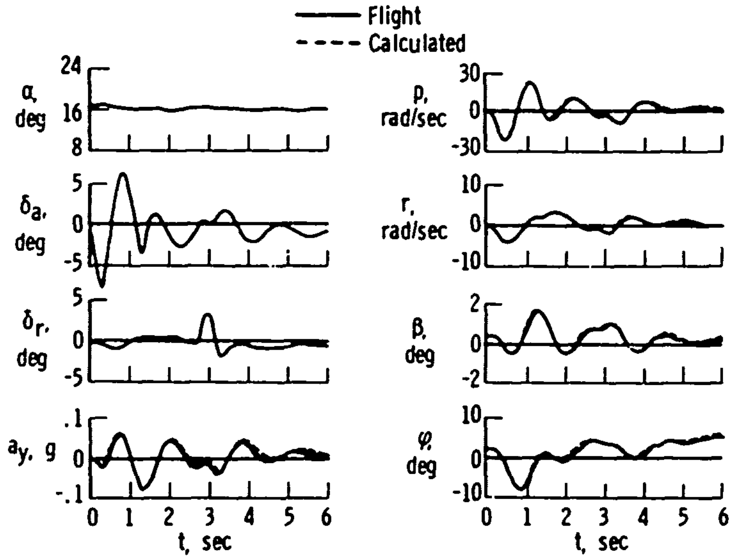


Figure 12 Comparison of flight and calculated (with a priori) time histories of the HL-10 lifting body with stability augmentation system on. Control inputs–aileron and rudder doublets

directional maneuver with dampers on at a Mach number of 1.2 and an angle of attack of 17°. The calculated time histories were obtained using an average value of dynamic pressure. Although the correlations are good, some improvement would probably have been obtained by considering the dynamic pressure as a variable in the analysis of the flight data.

Repeating the maneuver at the same Mach number but different angles of attack made it possible to determine the variation of most of the stability and control derivatives with angle of attack (as previously shown in figure 2) and to ascertain several discrepancies in wind-tunnel-predicted characteristics.

Future Applications of the Modified Newton-Raphson Method

The modified Newton-Raphson method is being used successfully at the Flight Research Center in all the flight programs involving derivative identification, but additional research is needed to improve confidence in the estimates obtained. A procedure (ref. 6) is currently being used that provides an approximation of the standard deviation of the estimates based only upon the information content of a given maneuver. This procedure provides an index of the reliability of the estimates from a given maneuver, but more experience is needed to truly assess the accuracy of the procedure.

Several areas that need investigation to improve the estimates as they are now made are also being studied. One that needs greater clarification is the determination of weighting matrices D_1 and D_2 independent of flight data. The D_1 matrix that is actually desired is the inverse of the error covariance matrix of the measurement instrumentation. The D_2 weighting matrix is the inverse of the error covariance matrix of the a priori values, that is, primarily wind-tunnel values.

Extended applications of the modified Newton-Raphson method are being considered. One of these, real time computation, would permit nearly instantaneous determination of the derivatives either onboard the airplane or on the ground. Vehicle excitation could be either deliberate or unintentional. Real time readouts would be a real advantage in terms of flight safety and timesaving, particularly in extending the flight envelopes of experimental aircraft. A further extension of real time computation would facilitate the design and operation of optimal control systems and the control of aircraft in a gust environment. By extending the method to identify the level of turbulence as well as the stability and control derivatives, it appears feasible, with optimal control techniques, to minimize structural fatigue, improve passenger comfort, or increase the stability of gust-sensitive aircraft like those in the V/STOL class.

Another application presently being developed is the determination of drag polars (C_D versus C_L). This application involves the use of nonlinear equations of motion for matching time-history records from push-pull maneuvers. The results are beginning to show promise.

Application of the method to the determination of derivatives and other characteristics at conditions of stall and spin onset is also being considered. This extremely complex problem involves the coupling of the nonlinear formats of the longitudinal and lateral equations of motion.

CONCLUDING REMARKS

The modified Newton-Raphson method has been successfully used with data on a number of aircraft at the NASA Flight Research Center and has provided generally good estimates of most of the derivatives when all responses were recorded. It has occasionally given poor estimates of derivatives, but good matches with flight time histories, when sufficient response information was not recorded. The reliability of the method—as with any method—is dependent upon the amount of information available in the responses, the type of response data available (accelerations, velocities, and displacements), and the accuracy of the recorded responses.

The use of the a priori provision has substantially increased the versatility of the method and provides a means of testing the reliability of determined derivatives, especially when there are deficiencies in the available information.

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OPEN DISCUSSION

J.L.Wesesky, USA: I wish to add to Mr Wolowicz's statement. The Air Force Flight Test Center has used the hybrid matching technique for derivative extraction, as discussed in Mr Wolowicz's paper, and has used the NASA Newton-Raphson program as a means of spot checking results. This has proven invaluable in the conduct of the X-24A program both in development of the flight control system and in a safe flight envelope expansion exploration. I consider both of these two methods of derivative extraction as more or less routine and they should be used in the flight test of new aircraft to verify predictions, as a safe means of expanding the flight envelope, to develop the vehicle, and to gather historical data for possible future modification and design.

G.C.Howell, UK At RAE, we have experimented with various identification techniques, including hybrid computer model matching and the Newton-Raphson method. With limited information, the Newton-Raphson method with "a priori" modification seems to create a problem as well as solving the convergence of the iteration. Does the method in fact only give a confidence factor rather than improving the absolute results? The choice of weighting factor D_2 seems to be a critical decision.

C.H.Wolowicz, USA: As noted in the paper, the "a priori" provision not only provides a confidence factor but also allows data with deficiencies and abnormalities to be successfully analyzed. This was particularly illustrated in the case of "Analysis of Incomplete Data" and emphasized in the case of "Analysis with Dampers On." In the event that "a priori" estimates of some of the derivatives are not available, the weightings for these parameters in the D_2 matrix are merely set to zero.

The choice of the weighting matrix D_2 is important as is the scalar weighting factor K. In Mr Iliff's paper (Ref.6 of the present paper), Mr Iliff discussed both quantities. In his Figure 11, he discussed K as it related to X-15 data and used a value which doubled the fit error. In other investigations at NASA-FRC, values of K were used which increased the fit error by only 10 to 20 percent.

P.Hamel, Germany: Is it possible to extend your method for determining aeroelastic derivatives like the normal force and pitching moment coupling derivatives due to the first body bending mode for highly flexible airplanes?

C.H.Wolowicz, USA: In general, if an accurate mathematical model is available for the data, nonlinear minimization techniques will still provide good estimates of the derivatives included in the model. The formulation discussed in the paper applies to all linear models although it was only applied to the aerodynamic modes in the present paper and in Reference 6 If the aeroelastic model to be analyzed is linear, then it will work within the existing formulation. If it is nonlinear, the problem becomes more complex and requires analysis by nonlinear methods.

It should be remembered that the reliability of the method – as with any method – is dependent upon the amount of information available in the responses, the type of response data available, and the accuracy of the recorded responses.

J.L.Wesesky, USA: In response to the question of obtaining derivatives of an elastic aircraft – we have had good success with some SR-71 derivatives using the hybrid matching technique such that subsequent simulation using these derivatives matched the airplane, and were appreciably different from the rigid body derivatives previously given to us.

Another comment worth mentioning is that we have found the matching technique successful in extracting derivatives from some semi-out-of-control aircraft motions as well as from particular test maneuvers planned with pulses and doublets.

O.H.Gerlach, Netherlands You state in your Concluding Remarks that the reliability of the method is dependent on the amount of information available in the responses. Could you enlarge on this point, in particular on the way in which the *shape of the manoeuver* should be chosen to obtain the maximum amount of information from the recorded responses.

C.H.Wolowicz, USA: Usually my answer (and Mr Iliff's answer) to this question is to suggest that the questioner refer to some of Professor Gerlach's studies in this area. The inputs should excite each pertinent mode as much as possible. As shown, a rudder input may excite the spiral mode but will normally not provide a good excitation of the roll mode. An aileron input is required for the latter. Since we are trying to get as much information as possible with a minimum of maneuvers, we have been applying rudder and aileron inputs along the lines discussed in

"Analysis with Dampers On" whenever possible. In considering the shaping of the maneuver, a number of factors may have to be considered such as the stability characteristics of the aircraft, the permissible limits of linearity of the derivatives, the pilot's ability to perform the maneuver safety of flight, and other flight considerations.

The "shape of the maneuver" is currently being investigated by the NASA Langley Research Center. Some enlightening results are expected shortly.

H.H.B.M.Thomas, UK: I have always held the view that it was always a good idea to arrange the motion to bring particular derivatives into prominence, e.g., acceleration in roll through zero rate of roll to give the roll control moment derivative. Your remarks on l_p are along the same lines. How far do you go in this direction in planning your flight tests?

Another question I would like to ask is what mathematical model do you use in analysing the non-linear post-stall motion?

C.H.Wolowicz, USA. In considering inputs to maneuvers, the response characteristics of the airplane must naturally be taken into account. A doublet input, properly phased, may be used to bring about a greater excitation without getting into nonlinear control characteristics. On the other hand, the doublet may be used to delay a divergence trend in an unstable aspect of the aircraft which would otherwise necessitate an early termination of the maneuver. It may be that a simple pulse with a slight dwell would be desirable. The most suitable inputs for any one airplane are arrived at during initial flights.

Currently no adequate model has been defined for the nonlinear post-stall dynamics. The formulations being investigated are merely power series expansions of the state variables. No firm results are available at present.

UTILISATION des BOITES NOIRES pour AMELIORER les CARACTERISTIQUES de PILOTAGE DURANT la PHASE de DEVELOPPEMENT d'un AVION.

par

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RESUME

Les moyens modernes d'études des qualités de vol, et principalement les simulateurs de vol, permettent une évaluation des caractéristiques de pilotage relativement tôt dans la phase d'étude d'un avion nouveau.

Par suite, les systèmes de commandes de vol et les divers correcteurs automatiques sont définis, tout au moins dans leur principe, à temps voulu, pour être installés et essayés sur avion au cours de la phase de développement.

Une meilleure connaissance des caractéristiques de l'avion, acquise au cours des essais en vol, ou l'évolution d'exigences réglementaires, conduisent parfois à faire évoluer les systèmes automatiques initialement prévus.

Il peut même se produire que l'on ait à installer de nouveaux correcteurs non prévus dans la définition d'origine, ce qui pose de nombreux problèmes de délais de réalisation, de modification sur avion, et de sécurité.

Ces diverses situations ont été rencontrées au cours du développement de l'avion de transport supersonique CONCORDE. Nous avons retenu, pour les illustrer, trois problèmes qui ont conduit à effectuer sur cet avion des modifications au cours de la phase de développement :

- une tendance au pompage pilote en latéral en vol supersonique, a été éliminée par une modification de la commande des élevons et du stabilisateur de roulis.
- le dérapage consécutif à une panne de moteur en supersonique a été diminué par la réalisation d'un système de contre-automatique.
- les exigences des autorités de certification concernant le contrôle de l'avion après blocage des organes de pilotage, a conduit à l'étude d'un système de pilotage en secours basé sur la détection des efforts pilote.

Dans les deux premiers exemples, il n'a pas été nécessaire d'ajouter de boîte noire, mais seulement de modifier des équipements existants.

INTRODUCTION

Le niveau de connaissances des caractéristiques de qualités de vol et de pilotage que l'on peut obtenir actuellement pour un nouvel avion avant que celui-ci ne vole, est très élevé. Ceci résulte d'abord de progrès accomplis dans le domaine des mesures en soufflerie et des calculs d'aéroélasticité, mais surtout de l'utilisation intensive de simulateurs de vol très perfectionnés. Par suite la définition générale des systèmes de pilotage et d'aides automatiques au pilotage devrait être pratiquement acquise avant les vols du nouvel avion. L'expérimentation en vol ne devrait conduire alors qu'à une optimisation des réglages des divers systèmes sans entraîner de modification profonde. Une telle situation pourrait contribuer à la réduction de la période de développement des nouveaux avions et donc de leur coût. En effet toute modification importante au cours de la phase de développement est très coûteuse en raison, non seulement des nouveaux équipements qu'il faut définir, réaliser et certifier, mais aussi des longs chantiers de modifications qui immobilisent les avions.

La situation que je viens de décrire est sans doute relativement vérifiée pour un avion classique, et ceci contribue certainement à la réduction notable des phases de développement de ces nouveaux types d'avions que l'on constate actuellement. Il n'en n'est pas forcément de même pour les avions de technologie très avancée (avions de transports supersoniques - STOL - VTOL etc...), ceci pour deux raisons principales : d'une part l'évaluation avant vol des caractéristiques de pilotage est plus délicate, et d'autre part les exigences réglementaires, qui souvent évoluent en parallèle avec le nouveau projet, sont mal connues au départ.

Les systèmes de pilotage et d'aide automatique au pilotage de l'avion de transport supersonique CONCORDE, définis avant les premiers vols de l'avion, se sont révélés généralement très satisfaisants au cours des vols d'essais. Leur mise au point avait en effet été très poussée sur simulateur de vol. Un nombre réduit de modifications a dû cependant être apporté à ces systèmes au cours de la phase d'essai en vol, nous en avons retenu trois pour illustrer cet exposé.

COMMANDE DIFFERENTIELLE des ELEVONS

La définition initiale de la commande des 6 élevons de l'avion (pl.1) conduisait à un braquage identique des élevons d'une même voilure, aussi bien en profondeur qu'en gauchissement (voir Pl.2). Cette solution présentait des avantages de simplicité de réalisation des chaînes de commande électrique et mécanique, mais surtout du système de surveillance de fonctionnement de la commande électrique. Il suffisait en effet de comparer entre elles les positions des élevons d'une demi voilure ; un vote à trois permettait de déterminer la commande défaillante (pl.3).

Une meilleure connaissance des caractéristiques d'efficacité en lacet des élevons acquise peu de temps avant les premiers vols, a fait apparaître deux problèmes en vol supersonique liés au couple de lacet important résultant du braquage antisymétrique des élevons internes.

- Un comportement sur vireur de la commande de gauchissement conduisant à des possibilités de pompage pilote mises en évidence par les valeurs élevées du critère $\frac{W\varphi}{W_d}$ (voir pl.4) et confirmées par les essais sur simulateur. Ce dernier a cependant montré que le stabilisateur de lacet installé sur l'avion pour améliorer l'amortissement du roulis hollandais réduisait très notablement cette tendance au pompage.

- Une possibilité d'auto oscillation de l'avion lorsque le stabilisateur de roulis était seul . fonctionnement. Il aurait fallu limiter l'action de ce stabilisateur aux élevons externes et médians, ce qui n'était pas possible compte tenu du système de surveillance des élevons, qui exigeait que les 3 élevons d'une même demi-voilure reçoivent les mêmes ordres de braquage.

L'état de réalisation des avions prototypes étant très avancé, une modification des commandes de vol aurait conduit à un retard notable des premiers vols. De plus il est difficile d'évaluer avec précision sur simulateur une tendance au pompage pilote et l'on pourrait avoir des doutes sur certains coefficients aérodynamiques et aéroélastiques difficiles à évaluer. Aussi a-t'il été décidé de lancer immédiatement les pièces nécessaires pour modifier la commande du vol, mais d'attendre pour appliquer la modification sur avion, d'avoir confirmation en vol.

Les vols en supersonique effectués sur avion 001 ont bien confirmé la nature des problèmes prévus. Le pilotage de l'avion avec tous ces stabilisateurs était satisfaisant, sans stabilisateur la tendance au pompage pilote pouvait être contrôlée par un pilote prévenu, l'oscillation divergente avec stabilisateur de roulis seul se produisait comme prévu, mais pouvait être éliminée sans danger par le pilote, soit en coupant le stabilisateur de roulis, soit en passant les élevons internes en mode de commande mécanique, mode qui élimine les ordres de stabilisation. Cette situation, bien que jugée non acceptable pour un avion en service, a cependant été considérée comme suffisante pour la poursuite des essais, et le prototype 001 vole toujours avec ses commandes d'origine.

Une modification a été étudiée, qui consiste à modifier :

- la cinématique de commande pour réduire le braquage en gauchissement des élevons internes par rapport aux élevons externes et médians (Pl.2)

- le système de surveillance des chaînes électriques de commande des élevons internes (Pl.5)

- le stabilisateur de roulis qui n'agit plus que sur les élevons externes et médians.

Cette modification a été appliquée sur l'avion prototype 002 au cours d'un chantier de plusieurs semaines dans le courant de l'été 1970.

Les caractéristiques de pilotages de l'avion 002 après modification se sont révélées très bonnes et cette nouvelle commande est montrée sur les avions de présérie et de série.

CONTRE AUTOMATIQUE de DIRECTION en CAS de PANNE de MOTEUR

L'étude des pannes de réacteur en haut supersonique avait montré que des dérapages importants pourraient apparaître surtout en cas de panne double. Ces dérapages étaient susceptibles de nuire au bon fonctionnement des entrées d'air des moteurs non affectés par la panne, et par suite d'entraîner leur extinction. Ce sujet est développé dans l'exposé de MM. LEYMAN et SCOTLAND de la BAC.

Un système très sophistiqué de contre automatique a donc été installé sur les avions prototype. Ce système est basé sur des détections de perte de pression à la sortie du compresseur HP, détection qui déclenche un braquage forfaitaire de la gouverne de direction fonction de la position et du nombre de moteurs en panne (pl.6). Les essais en vol ont montré que son fonctionnement était très satisfaisant, mais que les perturbations, et donc les dérapages en cas de panne moteur, étaient plus faibles que prévu ; seul le cas de panne double de moteur produit un dérapage susceptible d'affecter le fonctionnement des entrées d'air. De plus ce système prototype présente, en raison de sa complexité, les inconvénients suivants :

- son installation est coûteuse en masse (environ 60 Kg) et en prix

- sa fiabilité s'est révélée peu satisfaisante, surtout celle des 6 capteurs de pression installés sur les moteurs. Ce point cependant pourrait probablement être amélioré.

Un autre système a donc été étudié en tenant compte de l'expérience acquise en vol avec le précédent. Le système retenu est très simple ; il consiste à ajouter au stabilisateur de lacet déjà installé sur l'avion un terme d'accélération transversale (voir pl. 7).

On a tout d'abord mesuré en vol les déformées transversales correspondant aux modes structuraux de l'avion, de façon à définir une implantation de l'accéléromètre où le niveau de vibrations structurales soit le plus faible possible. De plus, pour éviter toute agitation inutile de la gouverne de direction, le signal accélérométrique n'agit sur la gouverne que s'il excède une valeur déterminée grâce à un circuit à seuil. L'optimisation des gains des constantes de temps et du seuil a été effectuée par étude théorique et sur simulateur de vol (voir Pl.9). Le stabilisateur de lacet de l'avion a été modifié sans difficulté et l'installation de l'accéléromètre et des câblages correspondants a été effectuée sans retarder les programmes de vol. Les essais en vol effectués sur le prototype 001 ont confirmé les prévisions et montré que ce système permettait effectivement de résoudre le problème posé avec une complexité minimum et sans affecter sensiblement la masse des systèmes de l'avion, en particulier aucune boîte noire supplémentaire n'a dû être installée.

SYSTEME de PILOTAGE de SECOURS

La commande des servocommandes d'élevons (voir Pl.9) s'effectue normalement par une chaîne de commande électrique ; en cas de défaillance de cette dernière, une deuxième chaîne identique à la première prend automatiquement le relais, enfin, en cas de panne supplémentaire, le pilotage s'effectue à travers une commande mécanique. Ces trois voies de commande présentent une partie mécanique commune très réduite au droit des organes de commande et il est possible d'imaginer une possibilité de blocage unique qui les rende toutes inopérantes.

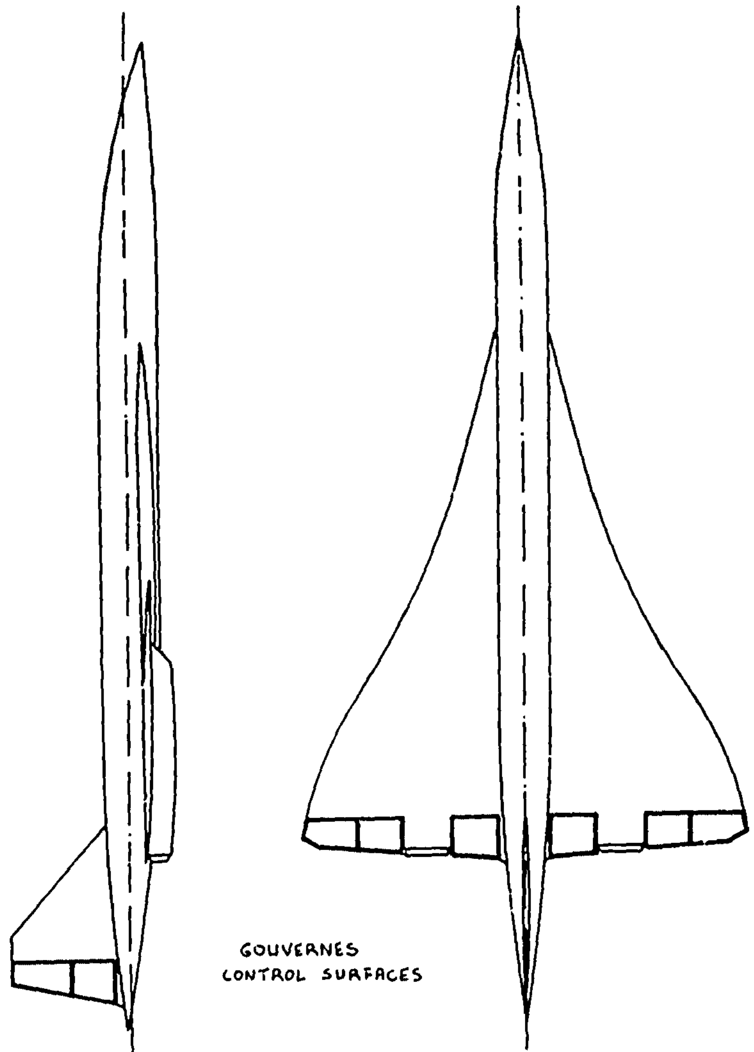
L'objectif des constructeurs était de prendre toutes les précautions de réalisation et de protection dans la zone considérée, de façon à rendre extrèmement improbable l'éventualité d'un tel blocage. Les autorités de certification n'ont pas suivi les constructions dans cette voie et exigent que l'avion soit pilotable en cas d'un blocage unique quelconque de la commande, blocage devant être considéré dans toutes les phases de vol et notamment pendant la rotation au décollage. Cette position a été connue tardivement alors que les avions prototype volaient et que les suivants étaient à un stade de réalisation très avancé. Il aurait donc été très préjudiciable au programme de reprendre l'étude et la réalisation mécanique des commandes de vol pour satisfaire à cette exigence tardive.

Nous avons donc cherché une solution n'affectant pas les éléments mécaniques de commande et ayant le minimum de répercussion sur l'installation de l'avion. La solution retenue consiste à utiliser des détecteurs d'effort placés dans le manche dont les signaux commandent directement les braquages d'élevons (voir planche 10). A ces signaux sont ajoutés ceux délivrés par les détecteurs de hors trim déjà disponibles sur avion et utilisés pour le trim automatique en pilotage automatique, ce qui permet de conserver les possibilités de trim normales de l'avion. L'asservissement électrique des gouvernes de la commande électrique normale est conservé et le système de stabilisateurs de l'avion continue donc de fonctionner normalement. Ce dispositif a été essayé sur simulateur ; on a constaté avec surprise que le pilotage était pratiquement aussi facile et précis commandes bloquées qu'avec le système de commandes de vol normal.

De plus les blocages et déblocages de la commande n'entraînent pas de perturbations importantes de l'avion. Ce système peut être engagé très rapidement par le pilote en cas de blocage en agissant sur un bouton poussoir placé dans le volant.

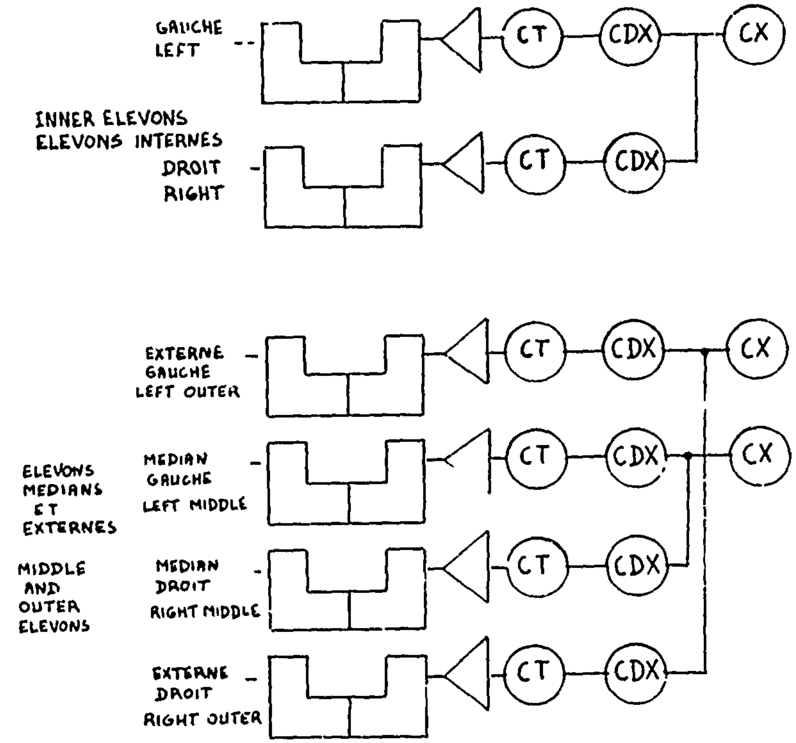
CONCLUSION

Les modifications apportées tardivement au cours de la phase de développement posent toujours de sérieux problèmes. L'étude très complète sur simulateur des caractéristiques de pilotage basées sur un modèle aussi précis que possible doit permettre de les réduire au minimum, ainsi que nous l'avions constaté dans le programme CONCORDE. Certaines modifications se sont cependant avérées nécessaires. L'utilisation de l'électronique et en particulier l'existence de chaînes électriques de commande ont permis de résoudre les problèmes posés sans répercussions préjudiciables au programme de développement de l'avion.



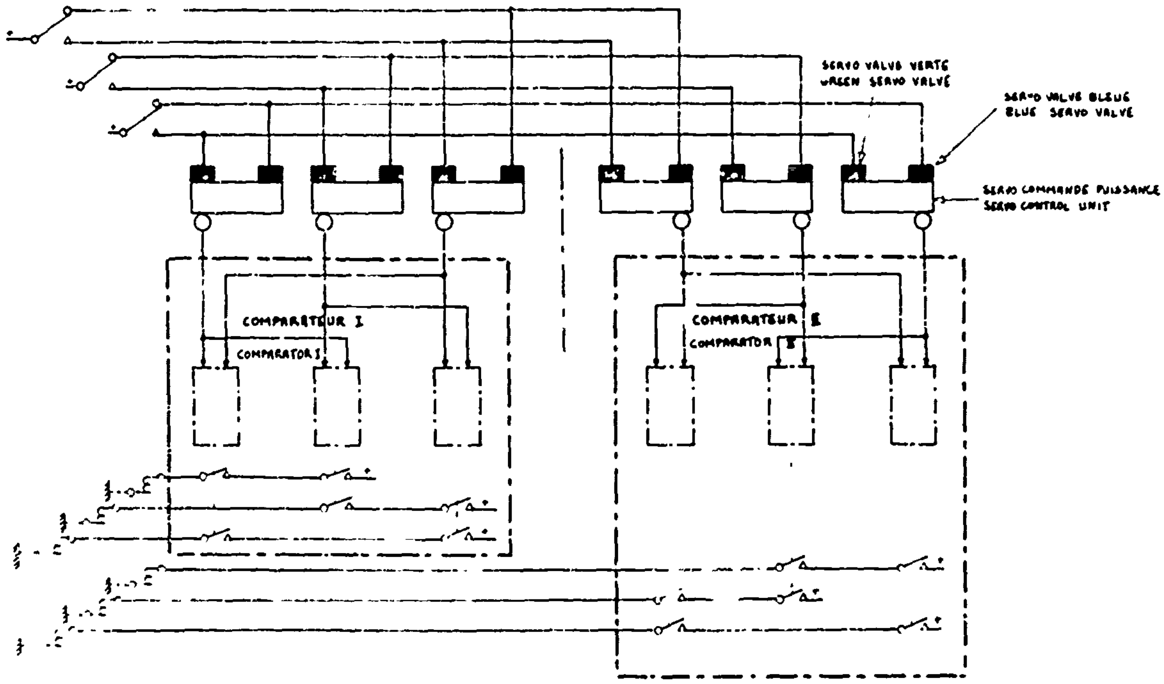
GOUVERNES
CONTROL SURFACES

Pl 1



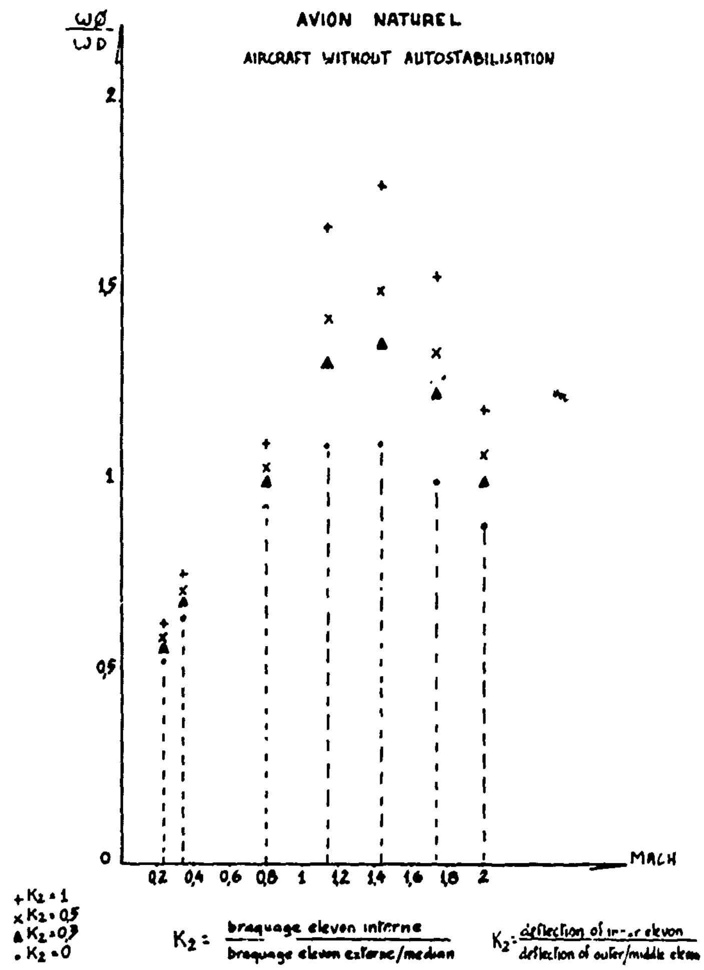
CHAINE DE COMMANDE D' ELEVONS
ELECTRIC CONTROL LINKAGE

Pl 2



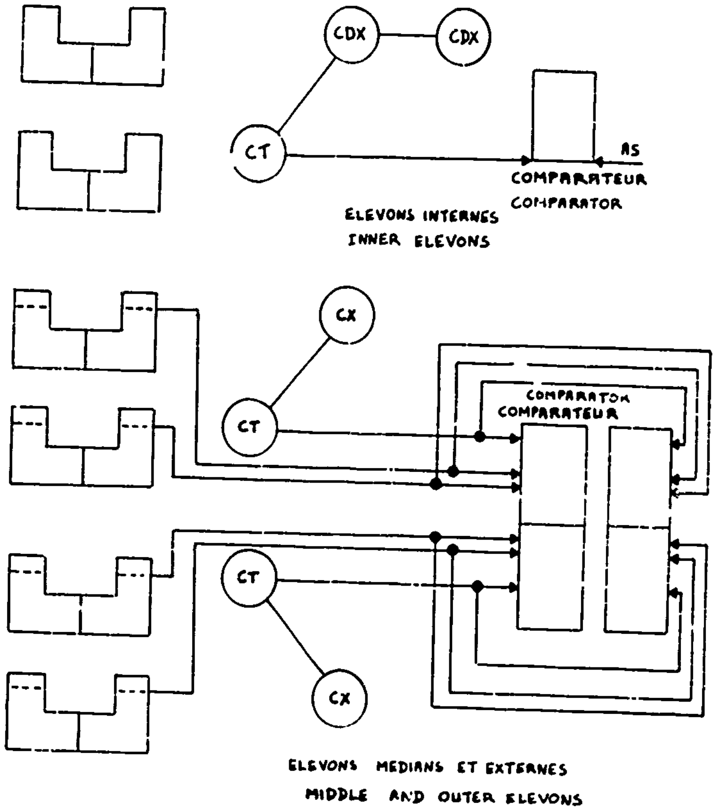
SURVEILLANCE DE LA COMMANDE D'ELEVONS INITIALE
ELEVONS MONITORING SYSTEM (INITIAL DESIGN)

Pl 3



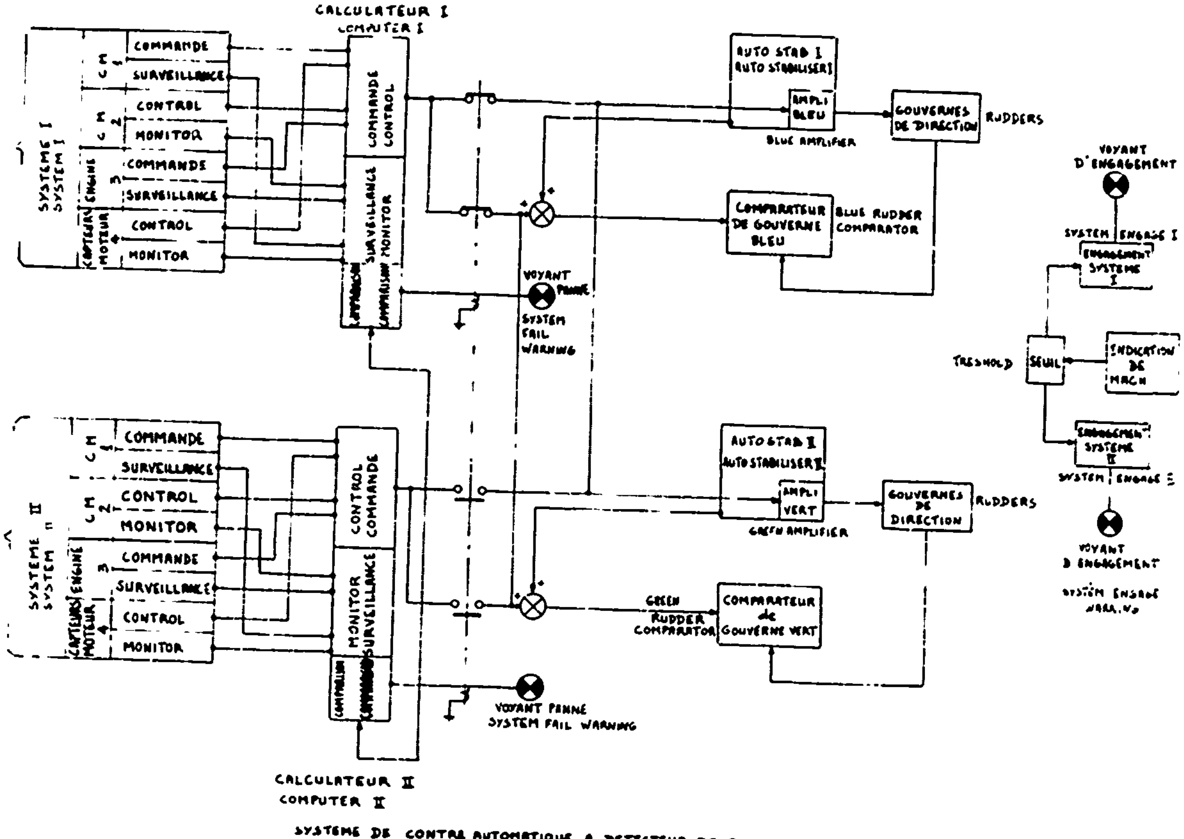
EVOLUTION DU RAPPORT $\frac{\omega\phi}{\omega D}$ EN FONCTION DE K_2
$\omega\phi/\omega D$ AS A FUNCTION OF K_2

Pl 4



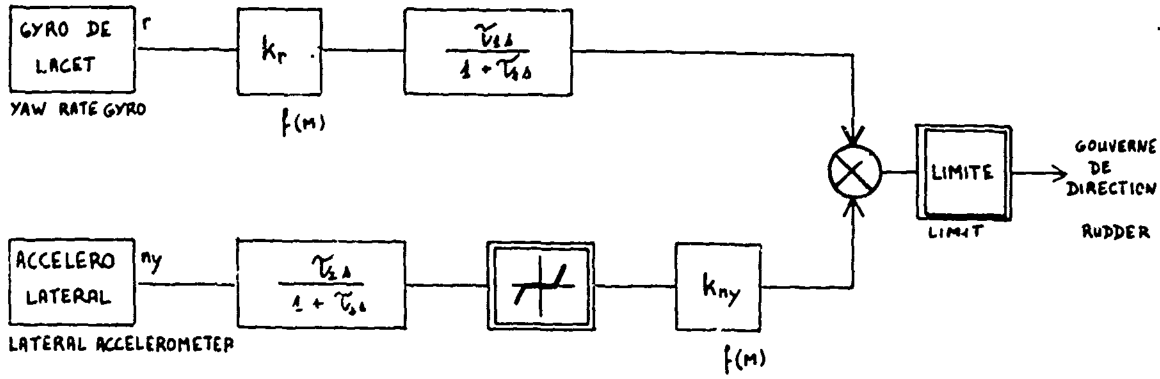
SURVEILLANCE DE LA COMMANDE D'ELEVONS MODIFIEE
ELEVONS MONITORING SYSTEM (MODIFIED SOLUTION)

Pl 5



SYSTEME DE CONTRE AUTOMATIQUE A DETECTEUR DE PRESSION
AUTO RUDDER WITH ENGINE PRESSURE DETECTION

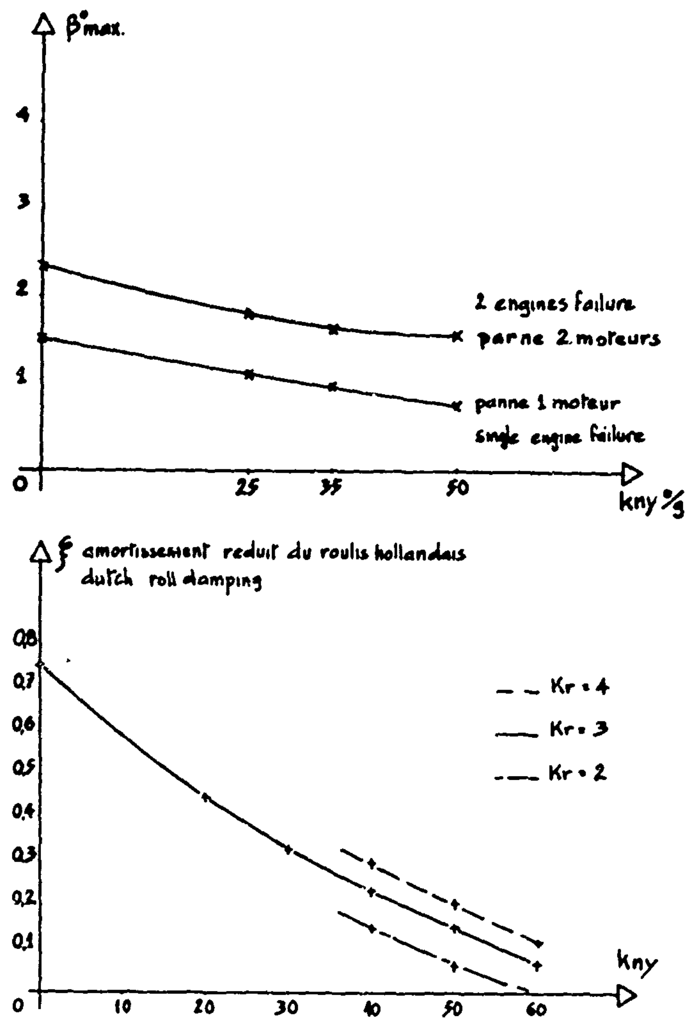
Pl 6



SYSTEME DE CONTRE AUTOMATIQUE A DETECTEUR ACCELEROMETRIQUE INTEGRE

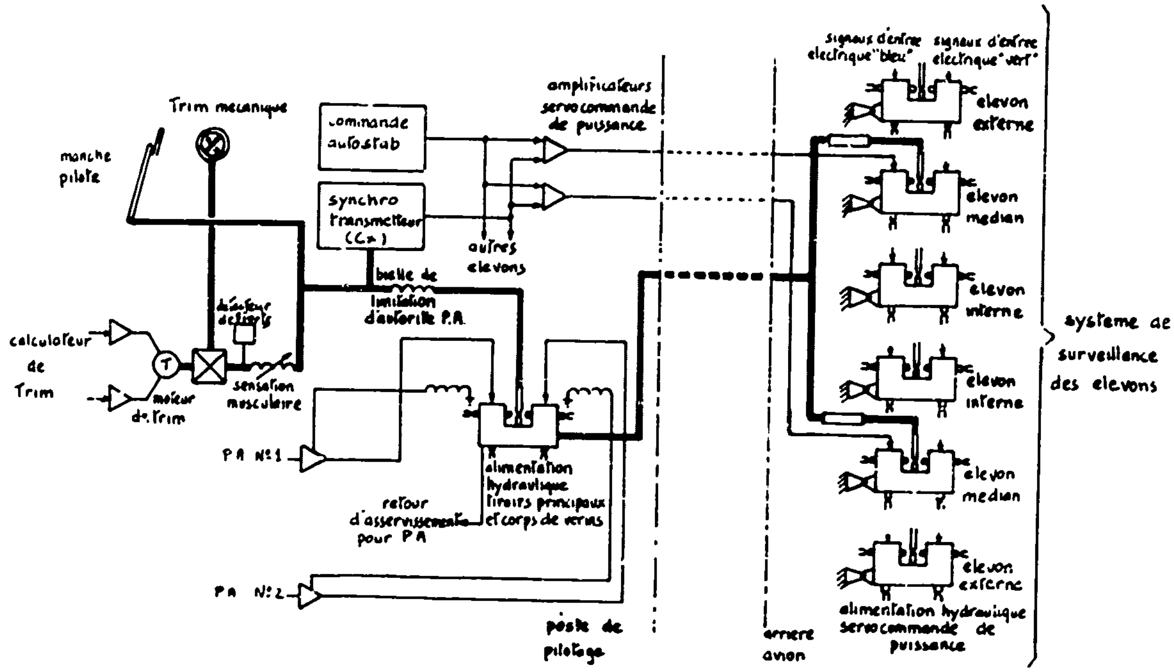
AUTO RUDDER SYSTEM WITH ACCELEROMETER DETECTION

Pl 7

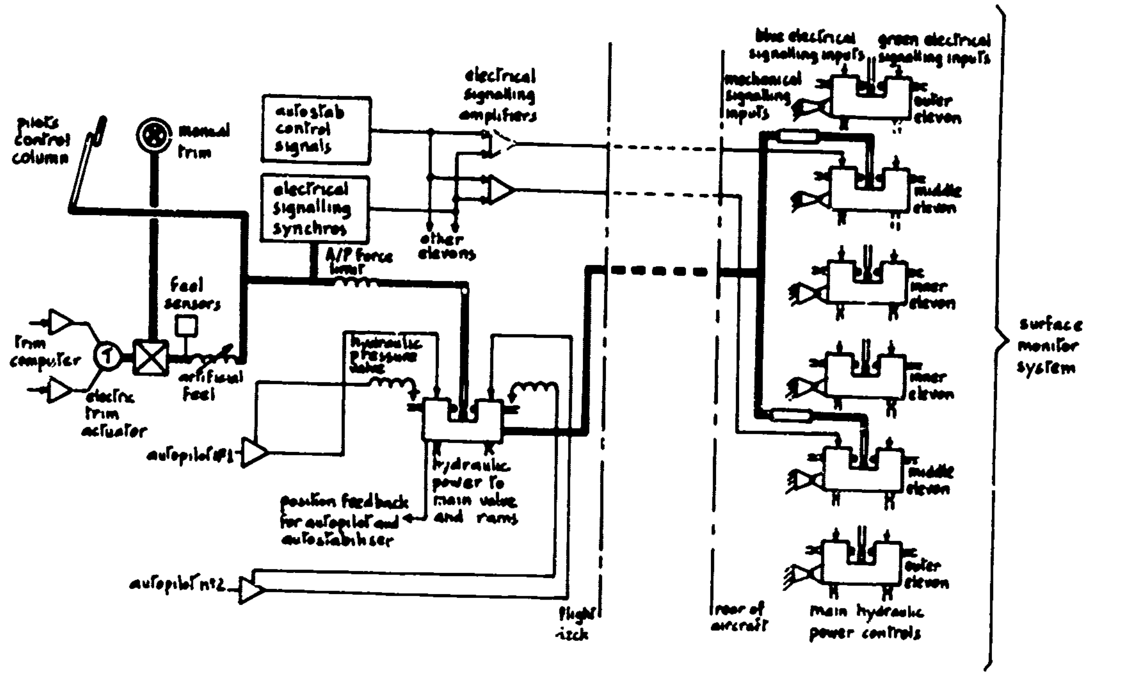


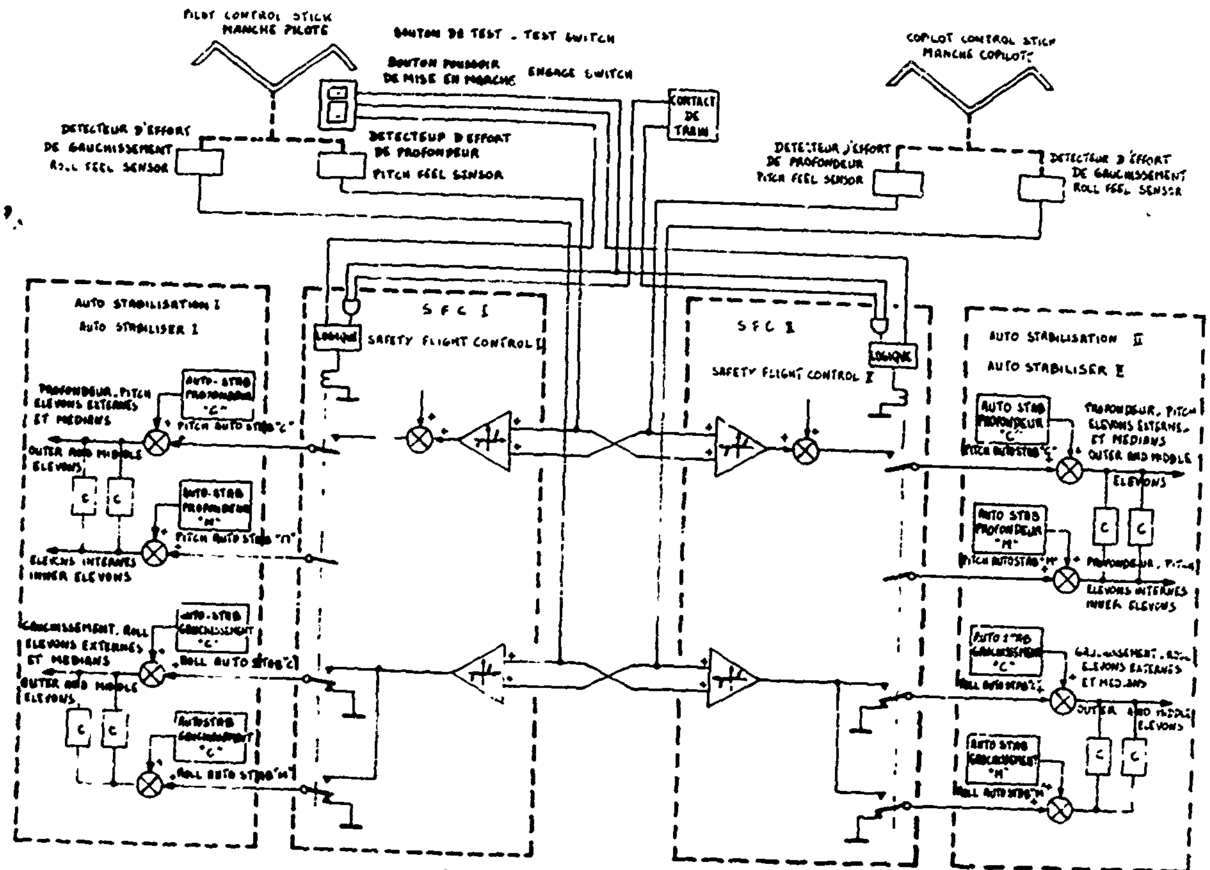
OPTIMISATION DU CONTRE AUTOMATIQUE A DETECTION ACCELEROMETRIQUE
OPTIMIZATION OF AUTO RUDDER SYSTEM WITH ACCELEROMETER DETECTION
PL 8

SCHEMA DE PRINCIPE DES COMMANDES DE VOL CONCORDE



SCHEME SHOWING THE PRINCIPLES OF THE CONCORDE FLYING CONTROL SYSTEM





PILOTAGE DE SECOURS - SAFETY FLIGHT CONTROL

Fl 10

OPEN DISCUSSION

R.L.Schoenman, USA: Are the force sensors which are used to accommodate a jammed control column or control wheel turned off during normal operation?

R.Deque, France: Answered, that these sensors are normally switched off and it is only in case of a jamming in the flight controls that by pushing on a button the pilot selects this mode of control. This mode of control is not operative during normal flight.

M.Hacklinger, Germany: The necessity for an additional stick force sensor back-up system shows again that we are paying a considerable penalty by insisting on mechanical control. The large aeroelastic effects on a slender design will complicate these problems and therefore I would like to ask Monsieur Deque whether at some stage of the Concorde control system design the possibility of later replacing all the mechanical gadgetry by a sufficiently redundant electrical system had been considered.

R.Deque, France: Answered, that it is true, that the mechanical control system gives probably the most design and testing work. But at the time they developed the Concorde, they did not see that it was possible to develop an aircraft control system without a mechanical back-up. So all of the flight control system of the Concorde is based on the existence of this mechanical back-up. If they had to do a system without mechanical back-up it will be another system for which one needs a higher level of redundancy. Such systems will be developed for the next aircraft generation.

PANEL DISCUSSION

THEME: THE IMPORTANCE OF STABILITY AND CONTROL

Chairman: L.P.Greene, Department of Transportation, Washington, DC, USA

S.B.Anderson, NASA Ames Research Center, Moffett Field, California, USA
K.H.Doetsch, DFVLR, Institut für Flugführung, Braunschweig, Germany
D.Lean, Royal Aircraft Establishment, Bedford, Beds, UK
P.Lecomte, Aerospatiale, Toulouse, France
C.B.Westbrook, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, USA

Introduction by L.P.Greene: The purpose of this Panel Discussion is to look back at the material that we have listened to, to reflect on it and to use that reflection as a lead to consider the new things that we should initiate through the Flight Mechanics Panel to and on behalf of the NATO Military Committee. This particular technique of having a Round Table or Panel Discussion at the end of a Symposium was first tried at Ottawa last year and it appeared to be successful. We are hoping that this one will be equally successful.

The "importance of stability and control" is perhaps a paradox to begin with. That is, if you have a substantial level of stability you probably have difficulty getting any control. If you have no stability you probably have lots of control. This point is being emphasized by those who advocate the fly-by-wire approach and at the same time is made equally strong by those who believe in inherent aerodynamic stability characteristics.

I have invited the Panel Discussion members to make a few remarks, to be as concise as possible and as provocative as possible.

C.B.Westbrook: I would like to address a broader point regarding the theme of this Symposium which is, if you recall, "good stability and control characteristics are a primary objective for each of the phase of development." I think that statement is wrong. The people who buy and use airplanes would almost call these characteristics a necessary evil. They cost them money, cost them drag and cost them weight. Their primary motivations are how to make money or to perform a mission. We stability and control people need to reorient our thinking when we say good stability and control is our primary objective. We need to turn the statement around, step aside and look at ourselves. What is really wanted? It is not good, but adequate and necessary stability and control to do a job. If we could do this in some better way, some of the frustrations I have had in working on airplanes, and seeing them not done the way I would like, would go away. If we could put the requirements in a form that meant something to the operator that is, in terms of cost, performance, or whatever, we would probably get better stability and control in most cases than we currently demand by more arbitrary methods. Possibly combat simulation for military aircraft is one way of achieving a meeting of minds between stability and control people and people who use the aircraft. We need to look at ourselves the way people who really use the aircraft look at us.

D.Lean: What has been achieved between this week's Symposium and the one held six years ago in Cambridge? I think really not very much! Where has been the big breakthrough that we really ought to be expecting after all these years of struggling with stability and control problems? We are still worried about the same stability and control problems in the basic aircraft and have the same sort of tools to attempt to solve them. I would like to ask the black-box experts, the advocates of manoeuver demand controls and control configured vehicles (CCV), when are we going to be offered some major clear-cut well-established advantages for these techniques? I have read three survey papers on the possible advantages of the control configured concept and these have attempted to present the advantages in terms of weight saving. Each of the three papers came to the same answer: about nine percent saving in take-off weight. Is this the best the black-box experts can offer? Ought they not perhaps be thinking in terms of the saving in time and cost in the development of a new aircraft? Many of the presented papers have stressed the long, painful and expensive process of developing an aircraft from the prototype stage to the production stage. Couldn't someone come up and say that we can forget about derivative estimation and extensive simulation, leaving it all to the black box people? All you need is sufficient control power to trim the aircraft and a bit more. Is this not the sort of thing which, during all these years, we have looked forward to? Can we expect to have a Symposium on this topic in five, fifteen or fifty years? I would like to hear some reactions from the electronics and avionics people.

K.H.Doetsch: We have made some progress in the last six years. We have now achieved more confidence in the black boxes. I remind you that we have now quite a bit of flying experience on the Concorde, that this aircraft is going into large scale production with the help of fly-by-wire and black boxes, and the ground work is going on. The difficulty that remains is that we have an awkward interface between the aerodynamicist and the aircraft designer and the black box people. Both sides should know about the problems. The black box manufacturers should also think in terms how to simplify the whole arrangement instead of making it more complex.

I have picked up from this Symposium two points which I want to stress here. The first one is what came up after Pinsker's lecture. We seem to have lost the art of derivative estimation since the fifties. The second point I want to make is that one should properly prepare flight tests and select the right test procedure to get what you want without much disturbance. You can save an awful lot of effort afterwards if you think seriously before you start the flight tests. Don't leave it to the mathematicians to find the answers from your flight test data. Normally this can be only the second best way to approach the system; that is, in a statistical manner. If the pilot selects the proper test procedure beforehand, a lot of mathematical evaluation can be skipped in the end.

S.B.Anderson: I want to pick up the subject of the need for stability augmentation systems and get a little more specific in regard to their application to VTOL and STOL aircraft. I think we have recognized from the flight test results obtained to date that certainly VTOL aircraft are not going to get good stability and control characteristics from their inherent aerodynamics and we might as well face up to the fact that STOL aircraft are also suffering along the same lines. The point I would like to make is that we must just start out and say that VTOL and STOL aircraft will need stability augmentation systems (SAS), they will need an attitude-hold rate-command type of SAS based on the experiences of flying these vehicles. They are all disturbed to a large degree by turbulence and a large pereccentage of the control power required to fly these vehicles is accounted for by the turbulence upset aspect. In addition, we must recognize that these systems have to be full-authority. We are not going to get by with the partial authority systems used in the past on conventional aircraft. With this in mind, I then want to make the point that on our new STOL powered-lift aircraft we might as well put in an attitude-hold SAS system in all axes, and further we certainly are going to have to tailor these systems to the operational requirements for these aircraft. From all the flight test results that I have observed, even with the most advanced systems like the DO-31, VJ-101 and VAK-191, some of the more modern flight control systems have not been adequate over the entire operational range or even at low speeds for flight path control, flare and turn entry. Therefore, my point is that we must recognize that we have to incorporate such systems, go ahead and concentrate on their refinement, and report back, I don't know in how many years, on our results.

P.Lecomte: My comments are on the controversy of using black boxes or the difference between the plain, simple and inexpensive aircraft and the complicated flying black box. I think the worm was in the fruit when somebody invented the spring-tab. The worm expanded very strongly when, instead of a black box, a grey irreversible jack appeared without manual reversion. Between them there is no clear-cut situation but good and bad design.

The question, which is of course provocative, is my wondering if there still is the need for handling qualities specialists or whether the companies might lay off all of them. The reason behind this is that in the very beginning one of the tasks of the flight mechanics pioneers was to help solve the four-degree-of-freedom equations, and there was a need to develop some simple criteria which could be handled by the paper and pencil method. Now, if you have a good computer you can compute everything, even wrong data giving wrong results. Computation is no longer a problem. We have realized that a significant part of the aerodynamic data is not right or accurate enough. This might be due to basic shortcomings of the wind tunnels, of the mountings or the measuring techniques. The example mentioned by Mr Leyman in his paper was a good one, and Mr Pinsker gave others in his paper. For some reason there is something wrong with the process. The other half of the wrong data is coming from the structures people who have difficulties in getting a proper assessment of the aeroelastic behaviour of complex airplanes. So, mixing wrong data you can get wrong answers.

I don't think that we have to lay off the handling qualities specialists, but their task is now more a task of synthesis work, being highly critical of the data and results, and trying to understand what happens and why it happens that way.

Summary by L.P.Greene: Drawing on my memory as an active and practising aerodynamicist, I can remember that the aerodynamics designer assumes everything but the responsibility. What is bothering me a little bit here is that they have now learned to assume the responsibility and delegate it to the avionics division.

I think that at this Symposium we did hear some very pointed presentations that evidenced the importance of making very serious trades between the fundamentals that could be practically accomplished and the refinements that could be done to assist the airplane. When the dependability, numbers, experience and confidence has been increased then we can think very seriously about a truly control configured vehicle. But I really resist using the excuse that it saves weight as a justification for such a CCV-program. If it makes the vehicle's performance function better and if it is doing it more economically then it makes sense.

It does not seem to me today that weight per pound costs any more than it did a few years ago in terms of the efficiency of the airplane. Unnecessary or poorly supported requirements for avionics packages that do everything and probably some functions more than are required is not an even trade on dollars per pound! They cost a lot more per pound than the airplane does.

The message, I think, to the Flight Mechanics Panel is that we have to increase our attention to those portions of the flight spectra that are poorly defined aerodynamically for the avionics system applications and, in fact, to encourage our friends of the black boxes to do with us the integration of these functions.