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on

Impact of Active Control Technology on Airplane Design

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PREFACE

Active Control Technology is a rapidly emerging technology with the potential of significantly improving performance, operational flexibility and tactics including the changing of air vehicle design concepts, procedures and methods. The most dominant reason for this rapid emergence is the success of fly-by-wire system implementation which is one of the fundamental ingredients for exploitation for these concepts. The majority of technology efforts to date, as evidenced by the papers presented at this Symposium, stress performance improvements with little, if any discussion on design criteria, handling characteristics or control law development for fully exploiting the maneuvering potentials of active control technology. Considerable emphasis centers on dissimilar redundancy as a panacea to protect against the generic failure potentials of similar redundancy techniques without stressing the limitations. Great care must be taken to assure that dissimilar redundancy techniques do not create more problems than they solve. Irrespective of the extensive work that has been performed in all NATO nations on active control technology, there still remains a common need for further control law development and application design criteria before this technology can be fully exploited and effectively applied.

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LA CONCEPTION DES AVIONS VA-T-ELLE ETRE BOULEVERSEE
PAR LES PROGRES DANS LES SYSTEMES DE COMMANDES DE VOL

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RESUME

Cet exposé introductif examine les définitions possibles du terme C C V et de quelques autres notions associées telles que Autostabilisation , Commandes de Vol électriques, etc... Sont simultanément examinées les caractéristiques communes à tous les systèmes dits "C C V" mais aussi les notables différences rencontrées.

Les possibilités actuelles de ces systèmes sont passées en revue en s'interrogeant dans chaque cas, pour les avions de transport et les avions de combat, sur les objectifs de sécurité et les objectifs de performances.

Les perspectives à long terme sont enfin évoquées.

ABSTRACT

This introductory talk examines the possible definitions of the term C C V and some other associated notions such as autostabilization, flight by wire, etc... The characteristics common to all the so called "C C V" systems are examined simultaneously, together with the most noteworthy differences encountered.

The present possibilities of these systems are reviewed, considering, in each case, the safety objectives and performance objectives for transport and combat aircraft.

Finally, long term prospects are evoked.

A peine avais-je accepté de faire cet exposé d'introduction au Symposium organisé conjointement par le FMP et le GCP que j'ai été pris d'un complexe de doute et d'infériorité en réalisant qu'une remarquable conférence synthétique sur cette question avait été faite en Octobre 72 à Florence (Italie) par le Colonel Scolatti et M. Johannes (Réf. (1)).

Je pourrais donc rendre à César ce qui est à César et, me contentant de cette citation, regagner ma place, mais ce serait faire injure aux Organisateurs de cette réunion qui m'ont fait l'honneur de m'y inviter.

Je vous proposerais donc quelques réflexions à caractère philosophique ou sémantique et quelques interrogations dans le but de provoquer, au long de cette semaine, des réflexions contradictoires de nature à éclairer le débat.

1 - LE SORT DES TERMES A SUCCES

Il est beaucoup question depuis plusieurs années des "CCV" (Control Configured Vehicules) et de la technologie correspondante des systèmes de commande ou de controle (active control technology), cette technologie constituant une partie de l'ensemble "CCV". Le concept "CCV" a déjà été fort bien défini notamment à l'AGARD, mais la signification donnée aux mots évolue à la fois dans le temps et dans l'espace, d'un individu à l'autre: ceci est particulièrement vrai pour les mots qui connaissent quelque succès. On peut en trouver un bon exemple dans le mot français "fiabilité" (reliability). Le mot semble avoir été utilisé initialement par les spécialistes de probabilités qui lui ont donné une définition précise résumable en "probabilité d'accomplir une mission donnée". Le mot connaît depuis un grand succès qui s'accompagne d'une imprécision de sens, sans doute à la fois cause et conséquence du succès.

Cette évolution et cette imprécision de sens affectent beaucoup de termes techniques et sans doute ceci fait-il partie de la "nature des choses". En effet, d'une part lorsque les spécialistes veulent se comprendre de façon précise, ils étayent l'emploi de termes généraux par des plans, du papier, des crayons et des gommes: ils ne peuvent prétendre maintenir à un mot à succès un sens précis, s'il l'a jamais eu, le savent et s'en accomodent. (Ce en quoi les spécialistes n'ont pas toujours raison mais ceci est une autre histoire...). (Fig. 1)

D'autre part, dans nombre de discussions rapides entre Directeurs et Spécialistes, ou entre spécialistes de branches différentes, il convient de disposer de termes imagés, mais l'image, le sens, ne sont pas exactement les mêmes pour tous. La déformation apportée par chacun doit rester suffisamment faible pour que la compréhension mutuelle demeure. (Fig. 2)

Par ailleurs, un ingénieur de langue française qui serait assez familiarisé avec les "CCV" a de la peine à reconnaître ce qu'il comprend dans la définition anglaise. S'il essaie une traduction en français, sa peine se transforme en déroute.

Ainsi, l'appellation internationale "CCV" étant soumise au moins à 2 types de perturbations (succès et traduction), n'est-il pas inutile de revenir sur la définition de cet ensemble, ne serait-ce que pour savoir s'il convient qu'elle soit précise et fermement défendue.

2 - CARACTERISTIQUES COMMUNES AU "CCV"

Un ensemble peut être défini par la liste des éléments le composant. En se basant sur les exemples réels ou envisagés aujourd'hui, on pourrait donc proposer la définition suivante :

"CCV" : Aéronefs dans lesquels il est tenu compte, pour leur définition générale, des possibilités suivantes des systèmes de commande :

- stabilisation de l'avion autour de son centre de gravité
- atténuation des effets de la turbulence
- optimisation de la répartition des efforts en manoeuvre
- augmentation des vitesses limites de flottement.

(Voir Fig. 3)

Un ensemble peut aussi être défini par une ou des propriétés communes aux éléments de l'ensemble. Si, à partir de la liste précédente, on recherche de telles propriétés on voit que les "CCV" utilisent des systèmes bouclés, le calcul étant effectué par des systèmes électroniques et l'exécution étant assurée par des équipements hydrauliques actionnant des gouvernes.

Ce sont sans doute là les seuls points communs à tous les "CCV".

Ainsi, si on examine la fiabilité qui commande la structure des systèmes, les exigences diverses suivantes apparaissent :

- atténuation des effets de turbulence envisagée sous l'aspect fatigue : taux de survie : 90 %
- stabilisation autour du centre de gravité, la fiabilité exigée peut être totale ou non selon qu'il existe ou non un domaine de vol sûr en absence de stabilisation artificielle
- flottement : l'existence d'un domaine sûr après perte du système doit être assurée. Il ne doit pas y avoir de perte totale brutale du système, même de courte durée.

Si on examine les performances

a) bande passante

Pour la stabilisation autour du centre de gravité, on souhaite être essentiellement non déphasé pour une fréquence = 1 HZ

La situation est évidemment très différente pour le contrôle artificiel du flottement.

b) performances statiques

On se montrera très exigeant sur le seuil pour assurer une stabilisation autour du centre de gravité, beaucoup moins pour les systèmes d'atténuation rafale.

3 - LA "PREHISTOIRE" DES "CCV"

Pouvons-nous retenir les propriété communes définies plus haut au paragraphe 2 comme caractéristiques des "CCV" ?

Examinons donc certains aspects historiques - ou préhistoriques - des "CCV".

Où commencer ?

- Aux frères WRIGHT qui ont conçu leur avion compte tenu d'une commande (mécanique) "bouclée" sur un pilote, comme le rappelait le Colonel Scolatti à Florence ?

- Au premier avion équipé d'un pilote automatique et de servo-commandes hydrauliques ?

- Aux premiers volets hypersustentateurs qui bouleversèrent l'optimisation structure - aérodynamique ?

- Au premier stabilisateur ou au premier Pilote Automatique "transparent" (Control Wheel Steering) (première modification de la réponse de l'avion par mesure de certaines grandeurs, système bouclé) monté pour rendre acceptable un avion déjà construit ?

- Au premier avion dessiné en tenant compte de l'existence d'un stabilisateur nécessaire au confort ?

On voit que notre énoncé de propriétés communes conduirait à inclure dans les "CCV" tout avion équipé d'un pilote automatique et de servo-commandes hydrauliques, résultat peu satisfaisant par sa généralité - et à constater que beaucoup de gens, depuis longtemps, font des "CCV" sans le savoir.

Revenons donc à ce qu'écrivait le Colonel Scolatti : "CCV" : aéronefs conçus en exploitant les possibilités offertes par les commandes de vol.

Mais alors, tous les avions actuels sont des "CCV" par le dessin de leurs hypersustentateurs. En revanche, les avions auxquels on ajoute un amortissement de modes structuraux par suite de problèmes de fatigue découverts en exploitation ne seraient pas des "CCV" ?

4 - TENTATIVE DE DEFINITIONS DES "CCV"

Mais au fait, quel type de définition souhaiter pour la notion de "CCV" ? Précise et étroite ou large et floue ?

Le flou est mal vu (officiellement) en technique. Précise alors ?

Par exemple, par énumération des fonctions indiquées précédemment ? En fait, "CCV" est une bannière derrière laquelle peuvent se ranger des idées assez hétéroclites, ce qui semble parfait pour le succès d'un sigle, un sujet de symposium, un "thème de recherches". Donc, ne soyons pas trop précis pour la définition de la bannière.

On pourrait désigner par "CCV", les aéronefs dont, dès le niveau de l'étude, un bouclage aérodynamique ----> structure ----> système -----> aérodynamique.... etc. a permis l'optimisation, les systèmes utilisés étant eux-mêmes des systèmes bouclés (asservis) permettant le développement de forces aérodynamiques.

Quant à l' "active control technology", nous dirons que c'est la technologie qui permet la mise en oeuvre des "CCV".

Enfin, en français, l'expression "Intégration des systèmes" englobe assez souvent les "CCV" mais elle me paraît beaucoup plus générale et ne pas constituer un réel équivalent (voir Réf. 2). Son caractère vague et imprécis devrait d'ailleurs lui assurer le succès.

5 - AUTRES DEFINITIONS

On peut opposer cette définition générale et assez floue des "CCV" à la définition plus précise de certains systèmes de commandes de vol. (Des termes français ont été mis en regard des expressions en anglais sans qu'une réelle équivalence soit clairement établie).

1) S A S (Stability Augmentation System) (Autostabilisateur ou Autostab.)

   Système d'autorité limité, en série avec les commandes de vol (mécaniques ou électriques) Fig. 4

2) C A S (Control Augmentation system) (Commandes de vol électriques à "stabilisation")

   Système à grande autorité, par lequel transitent les ordres de pilotage; commande de vol mécanique en parallèle. Fig. 5

3) FLIGHT-BY-WIRE (Pure Electrical Flight Control)
   (Commandes de vol électriques - CDVE)

   Généralement, les auteurs semblent admettre qu'on asservit le mouvement de l'avion (et non pas la position de la gouverne), mais on peut considérer que cette caractéristique ne fait pas partie de la définition. Fig. 6

4) PSEUDO-FLIGHT-BY-WIRE (Pseudo commandes de vol électriques)

   Existence d'une commande mécanique de secours.

   Existence possible d'une stabilisation artificielle. Fig. 7

6 - POSSIBILITES ACTUELLES

6.1 - Remarques liminaires

On voit qu'en pratique aussi bien que par la définition que nous en avons donnée, la technologie du contrôle actif devrait être très diverse dans le détail selon le type de contrôle souhaité, orientée soit essentiellement par des soucis de performances, soit essentiellement par des soucis de sécurité. Les conférences qui suivront permettront peut-être de saisir s'il en est bien ainsi ou si, au contraire, compte tenu des impératifs économiques, une technologie a des avantages tels qu'elle triomphe dans toutes les applications envisagées.

Mais le sujet de ce symposium n'est pas tellement la technologie que les répercussions de celle-ci sur la définition de l'avion, y compris celle des systèmes. Nous devons donc examiner quelles sont les possibilités techniques actuelles et, tout d'abord, en fonction des impératifs de sécurité.

Nous serons amenés à distinguer entre transports civils ou militaires et avions de combat pour lesquels la survie de l'équipage peut être assurée, jusqu'à un certain point, par cabine ou siège éjectable. Dans cet examen des possibilités, nous ferons intervenir des considérations économiques : ainsi, nous considèrerons que la complexité d'un système peut rendre aujourd'hui incertaine sa réalisation en raison de son coût. Un tel jugement ne peut être que subjectif puisque général, alors que les gains économiques apportés par un système évolué dépendent de l'avion d'application. Au moins permet-il de situer les difficultés absolue et relative de certaines réalisations.

Nous ne ferons pas intervenir ici les problèmes de règlementation sur lesquels nous reviendrons plus loin. Nous présenterons l'opinion certainement discutable d'un avionneur uniquement préoccupé de sécurité, performances et prix. Cette opinion est basée sur :

- l'inexistence actuelle d'une flotte d'avions à commandes de vol purement électriques
- la connaissance d'avions à commandes de vol électriques avec secours mécanique
- l'examen des MTBF (temps moyen entre pannes) en exploitation d'équipements actuels (calculateurs numériques par exemple)
- l'analyse des problèmes posés par l'auto-surveillance et la programmation des calculateurs numériques
- les ségrégations géographiques et physiques à imposer aux composants d'un système électrique pour assurer sa survie, au moins partielle, à un impact de projectiles, un début d'incendie, un coup de foudre ou une tension électrique excessive
- un objectif de sécurité chiffré dont nous allons dire deux mots.

6.2 - La Sécurité

Lorsqu'on a affaire à un système électrique complexe, on ne peut se limiter au critère d'absence de panne double catastrophique. En effet, on devra vraisemblablement examiner aussi bien :

- une unique combinaison de pannes particulières de composants simples
- qu'un très grand nombre de combinaisons de pannes de composants complexes (calculateurs par exemple)
- que des combinaisons incluant des pannes cachées depuis de nombreux vols.

Quelles que soient donc les difficultés rencontrées pour estimer les probabilités de panne, cette évaluation semble nécessaire, ainsi que sa comparaison avec un objectif de sécurité exprimé sous forme numérique. Les propositions de jugements que nous portons dans le tableau 1 tiennent compte des objectifs suivants :

Pour un avion de transport, la probabilité de chacun des évènements ci-dessous doit être inférieure ou égale à 10^{-9}/Heure :

- perte totale du système de commandes de vol
- déplacement incontrôlable et catastrophique d'une ou plusieurs gouvernes.

Dans l'évaluation de probabilité, il sera tenu compte des combinaisons de panne d'équipements appartenant aux commandes de vol et aux autres systèmes (génération hydraulique par exemple).

Pour un avion de combat (muni de sièges ou de cabine éjectables)

- la perte totale du système de commandes de vol doit avoir une probabilité $\leq 10^{-7}$/Heure
- un embarquement incontrôlable et rapide des gouvernes doit avoir une probabilité $\leq 10^{-9}$/Heure.

Il est évident qu'une modification de ces objectifs modifierait les jugements portés. (Voir tableau 1)

6.3 - Les Performances

Examinons maintenant les possibilités au point de vue des Performances (Voir tableau 2)

7 - QUELQUES ASPECTS DE LA REGLEMENTATION

Considérons rapidement les problèmes liés à la règlementation. Nous prendrons, à titre d'exemple, les règlements ou projets de règlements civils, ceux-ci poursuivant des objectifs de sécurité généralement plus exigeants, seront plus illustratifs.

Les règlements (FAR 25 - Par. 1309 - TSS Standard 1.1) sont rédigés sous une forme suffisamment générale pour couvrir le cas de commandes de vol purement électriques.

Mais cette partie générale des règlements est habituellement complétée par des recettes et on arrive donc à deux ensembles d'exigences superposées :

- d'une part le respect de certains critères dictés par l'expérience
- d'autre part la recherche par analyse de toutes les configurations dangereuses et l'évaluation, explicite ou non, de leur probabilité.

Sur le plan des principes, on ne peut qu'approuver la sagesse de cette attitude. En pratique, elle constitue, néanmoins, un frein considérable à l'évolution des techniques aéronautiques.

En effet, d'une part la 1ère exigence tend à ne pas reconnaître les progrès technologiques. D'autre part, la 2ème exigence tend à introduire un doute fondamental sur la sécurité des systèmes étudiés : Comment être sûr que toutes les combinaisons de pannes ou d'erreurs de probabilité significative ont été recensées ?

Mentionnons aussi le risque d'un certain formalisme sur la démonstration numérique de la sécurité.

Ainsi, dans le cas des "CCV", on peut craindre une application excessivement sévère de la règlementation.

- en partie, parce qu'avant la certification d'un premier transport entièrement dépendant des "boîtes noires", on aura eu le temps d'analyser, donc de faire apparaître des cas où une démonstration rigoureuse est impossible

- en partie, en raison de la réputation des systèmes électroniques courants. Leur perte n'est jamais catastrophique pour les avions actuels (même en atterrissage par mauvaise visibilité, il est vraisemblable que, pour tous les avions certifiés à ce jour, une remise manuelle des gaz à l'horizon a toutes chances d'éviter une issue catastrophique).

Il semble donc que les constructeurs aient, jusqu'à ce jour, fait plus d'efforts pour augmenter les performances et éviter les ordres erronés, que pour conserver un minimum d'équipements électroniques de pilotage. Il ne sera pas facile de faire admettre que les précautions prises à la suite d'une volonté nouvelle seront suffisantes. Il est donc vraisemblable que, par prudence, la règlementation amènera à une complexité un peu trop grande.

8 - PERSPECTIVES A LONG TERME

Les principales motivations d'une évolution sont :

- généralement économiques

- parfois politiques (avion porte-drapeau, qu'il soit présenté comme un avion expérimental ou un avion de série)

- liées à la recherche d'un accroissement de sécurité ou diminution des charges de travail.

Comment le développement de l'électronique appliquée aux systèmes de pilotage peut-il aider à réaliser des aéronefs optimaux sur le plan économique, objectif qu'on peut interpréter de deux façons : (Fig. 8)

a) A service égal, diminution de prix

b) A prix égal (ou même supérieur), amélioration de la qualité du service (augmentation de vitesse, piste plus courte, absorption de rafales, etc...) entraînant un accroissement du marché.

Considérons tout d'abord les avions de transport subsoniques à décollage conventionnel. L'intérêt économique d'un système de "CCV" est probablement modeste et sans doute discutable : les formes de l'avion sont telles qu'un centrage très arrière ne procure pas d'avantages considérables (sans parler des problèmes de géométrie de train et de basculement au sol). Les problèmes de flottement peuvent être souvent résolus sans pénalité de masse significative. A moins que le "CCV" ne rende possible l'utilisation de formules non conventionnelles telles le canard.

La situation est toute différente pour un avion de transport supersonique, en raison :

- de l'intérêt du vol à centrage arrière
- de la possibilité de réduction de la taille de la dérive
- de la recherche d'amortissement des modes souples
- de l'intérêt éventuel de l'antiflottement
  (comparaison de poids avec une solution classique)

Après les efforts faits pour mettre au point des commandes de vol électriques pour transport supersonique, il sera peut-être possible de les utiliser économiquement sur des subsoniques, d'autant plus que la technologie elle-même aura progressé. On peut alors rêver d'un pilotage automatique et manuel ne nécessitant que 3 calculateurs * numériques et des servo-commandes électrohydrauliques de puissance. Les timoneries, les servo-moteurs, les calculateurs de stabilisation, de mach-trim, de sensation artificielle, de bras de levier variable auraient disparu. Les organes de pilotage miniaturisés s'intègreraient à un aménagement nouveau du poste de pilotage. Mais il s'agirait plus alors d'une refonte des systèmes que d'un véritable "CCV".

Par ailleurs, sans aller jusqu'à l'emploi des commandes électriques vitales, on peut envisager sur des subsoniques classiques :

- de la modulation de portance pour améliorer les performances d'atterrissage, essentiellement automatique
- d'atténuer les effets de rafales sur le confort et la fatigue structurale. Mais apparaissent des problèmes de variation d'assiette ou de débits hydrauliques importants pour mouvoir de "grosses" gouvernes (hypersustentateurs par exemple).

S'agit-il alors de "CCV" ? En d'autres termes, les services structures sauraient-ils tenir compte du système atténuateur de rafales dans leurs dimensionnements initiaux ?

Autre domaine dans lequel des commandes électriques sont justifiées, sinon indispensables : VTOL, convertibles, tant pour pallier la complexité mécanique que pour assurer, avec la redondance nécessaire, la stabilisation artificielle.

Les "CCV" trouvent certainement une application aux avions de combat à hautes performances, mais d'une façon variable suivant les missions. Citons quelques exemples :

- sur avion supersonique, réduction de la taille de la dérive, amélioration de la manoeuvrabilité par centrage arrière
- adaptation des commandes de vol à des conditions très diverses de vitesse et d'altitude aussi bien que de formes (charges extérieures)
- protection entre le flutter dans des cas d'emplois peu fréquents

  .... etc.

Citons deux exemples en rapport avec la <u>motivation sécurité</u> :

- limitation des charges :

  . de manoeuvre par limitation stricte des facteurs de charge demandée aux valeurs acceptables par la cellule

  . de rafales - Aucune étude à ma connaissance n'en montre la faisabilité, mais il semble bien possible d'obtenir une atténuation en fonction d'un détection (nz demandé - nz réel)

* Trois calculateurs : pour pouvoir décoller avec un calculateur en panne et qu'alors un "coup de feu" dans un calculateur restant ne soit pas catastrophique.

- diminution des charges de travail et risques d'erreurs par :

  . intégration des systèmes
  d'où simplification du pilotage des commandes et des alarmes)

  . refonte pilotages automatique - transparent - manuel.

Je ne conclurai pas. Aussi bien ne voulais-je qu'ouvrir la porte aux orateurs qui, dans les jours qui viennent, vont parler du "CCV" de façon beaucoup plus experte que je n'ai pu le faire.

Les quelques points de vue que j'ai exprimés trouveront peut-être un écho ou une contradiction. Je me réjouis à l'avance de l'échange d'idées qui en résultera.

REFERENCES

1 - Colonel C.A. SCOLATTI
"The Evolution of CCV Technology"
2 Octobre 1972 - Florence - ITALIE

2 - J.C. WANNER "Concept CCV et Spécification"
Octobre 73 - Florence - ITALIE

3 - Informations non publiées, aimablement transmises par BAC

TABLEAU 1

SECURITE - POSSIBILITES ACTUELLES

Conséquences des pannes de la partie électrique des commandes de vol	Existe-t-il actuellement une solution sûre et suffisamment économique ?	
PANNE PASSIVE	Avion de combat	Avion de transport (Civil ou militaire)
Ier Cas Il n'y a pas de domaine de vol sûr en cas de panne totale. (Ex. Instabilité rédhibitoire de l'avion naturel).	Oui	Incertain
2ème Cas La panne totale, même de courte durée est catastrophique dans une partie importante du domaine de vol, mais il existe un domaine de vol sûr permettant en particulier l'atterrissage (ex : système antiflottement).	Oui	Incertain
3ème Cas La panne totale n'est catastrophique que pour des phases de vol exceptionnelles ou de courte durée (Ex : atterrissage sans visibilité).	Oui	Oui
4ème Cas La panne totale n'est pas catastrophique sauf combinée à un évènement rare ou exceptionnel.	Oui	Oui
5ème Cas La panne totale n'est jamais catastrophique (Ex : amélioration confort ou tenue à la fatigue par absorbeurs de rafales).	Oui	Oui
EMBARQUEMENT Un embarquement incontrôlé peut être catastrophique dans presque tous les cas précédents (système à grande autorité).	Oui	Oui

TABLEAU 2

PERFORMANCES - POSSIBILITES ACTUELLES

Fonctions assurées par le système	Principales performances requises	Existe-t-il actuellement une solution satisfaisante	REMARQUES
Stabilisation d'un avion instable à courte période. (Ex : marge statique nulle ou négative)	Accéléromètres, gyromètres, servocommandes : Faible hystérisis Faible déphasage jusqu'à 1 Hz Calculateurs : Faible déphasage jusqu'à 1 Hz	 Oui Oui Oui	1) Le "oui" doit être nuancé pour les cas : - de centrages très arrière (très au-delà du point de manoeuvre) - de couplage avec plusieurs modes structuraux de très basse fréquence. 2) Exemples d'application TSR2 réf. (3) "Stabilité de route artificielle à M$\gtrsim$1,7 - Programme SFCS (survivable Flight Control System)
Antiflottement	Détecteurs : Suivant l'emplacement, tenue à l'environnement Servocommandes : Problèmes d'atténuation et de dispersion de phase si la fréquence à amortir est élevée. Calculateurs numériques : Bande passante. Calculateurs analogiques : Bande passante	Selon l'environnement Selon fréquence Selon fréquence, nature des calculateurs et programmes de calcul Oui	1) Problème de MTBF et d'obtention du signal utile 2) Exemple d'application : Programme sur B 52
Atténuation de la turbulence - amortissement de modes souples	Identiques à antiflottement	Identiques à antiflottement	Exemples : CONCORDE (premier mode longitudinal du fuselage. - L 1011 (latéral) - B 52 Programme LAMS load Alleviation and mode stabilisation
- Modification de l'amplitude de la réponse de l'avion rigide.			- En longitudinal , le principal problème semble être celui de l'importance des forces aérodynamiques nécessaires - En latéral, il existe sur avion des systèmes d'amortissement de rafales pour avion de série (B747 - A.300)
Optimisation de la répartition des efforts en manoeuvre.		Oui	

VIE D'UN TERME MATHEMATIQUE
life of a mathematical term

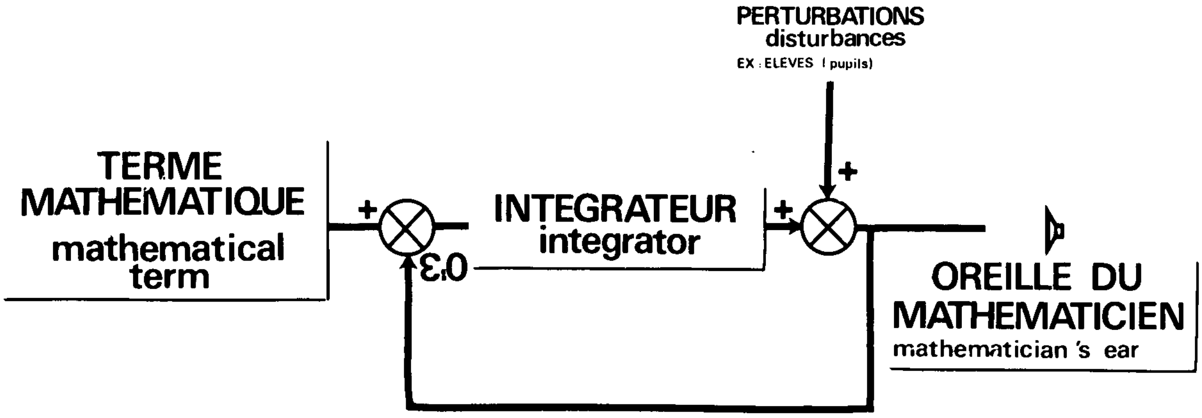

Fig. 1

VIE D'UN TERME TECHNIQUE
life of a technical term

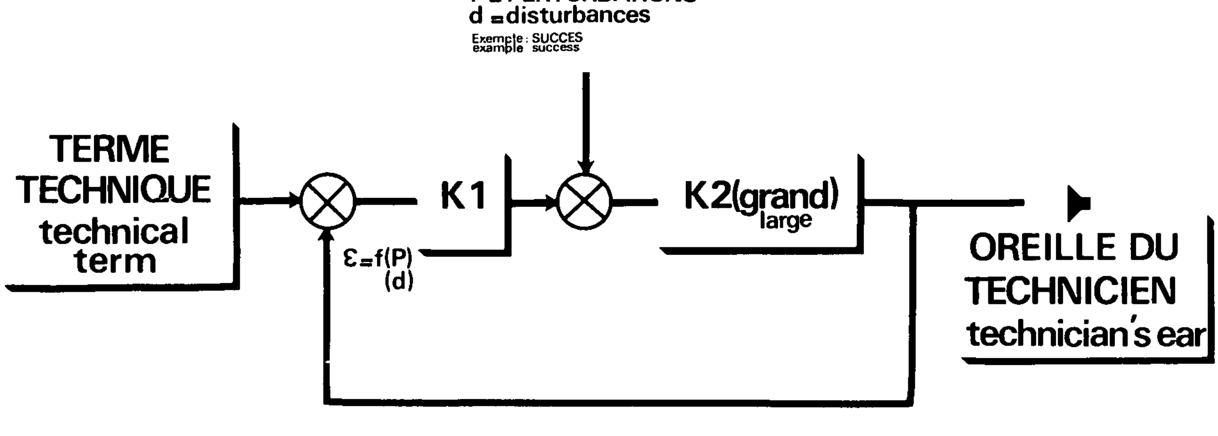

Fig. 2

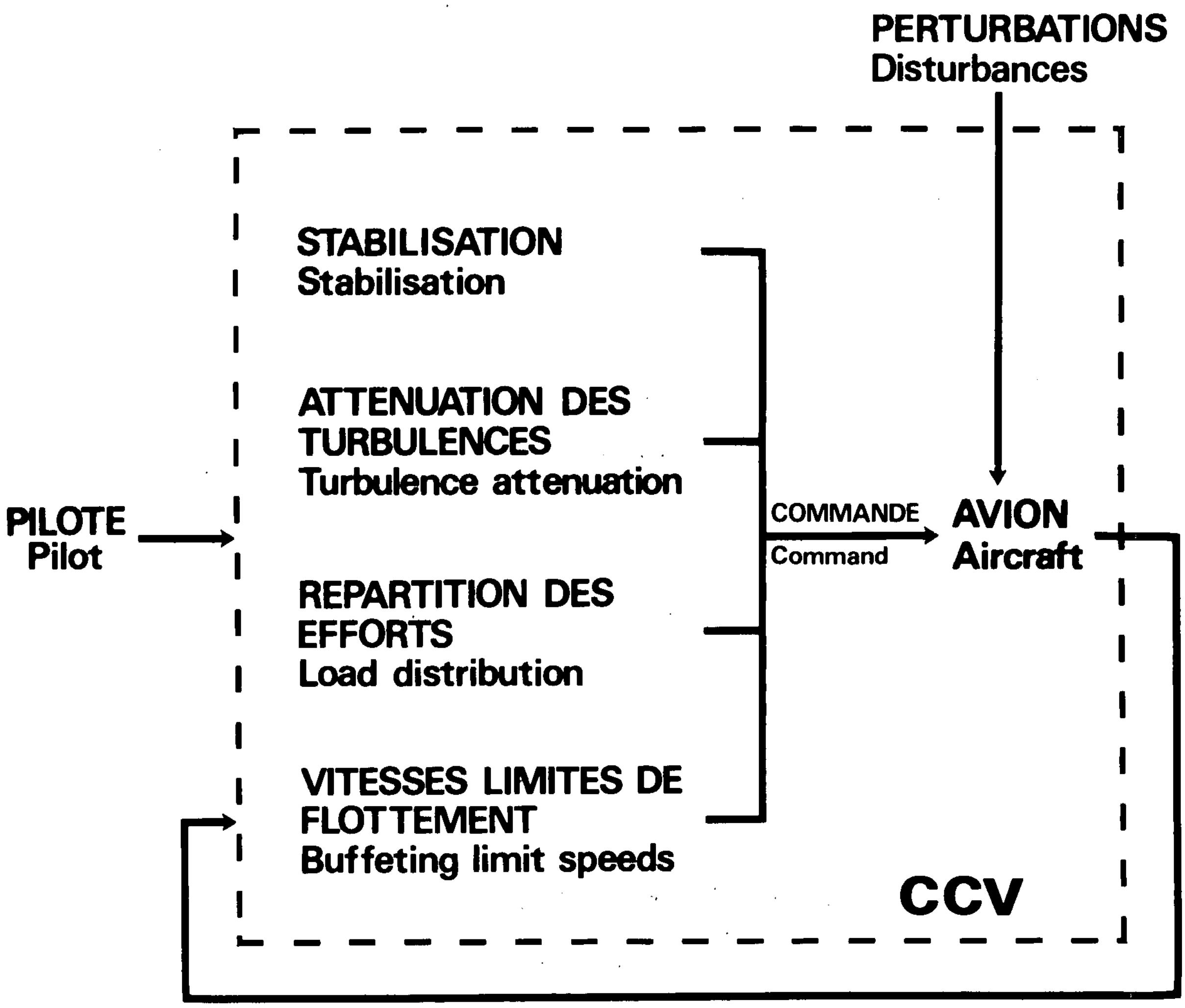
PERTURBATIONS
Disturbances
STABILISATION
Stabilisation
ATTENUATION DES TURBULENCES
Turbulence attenuation
PILOTE
Pilot
COMMANDE
Command
AVION
Aircraft
REPARTITION DES EFFORTS
Load distribution
VITESSES LIMITES DE FLOTTEMENT
Buffeting limit speeds
CCV

Fig. 3

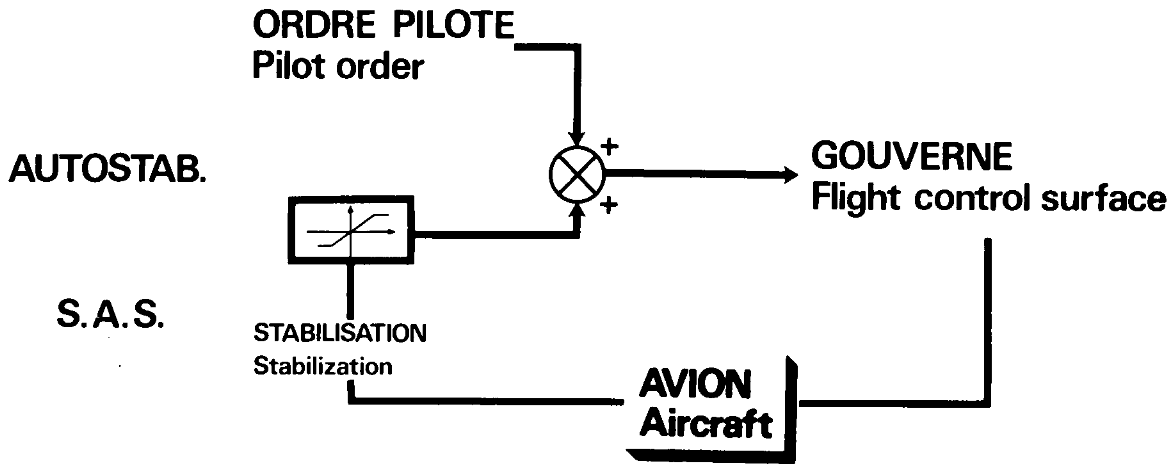

Fig. 4

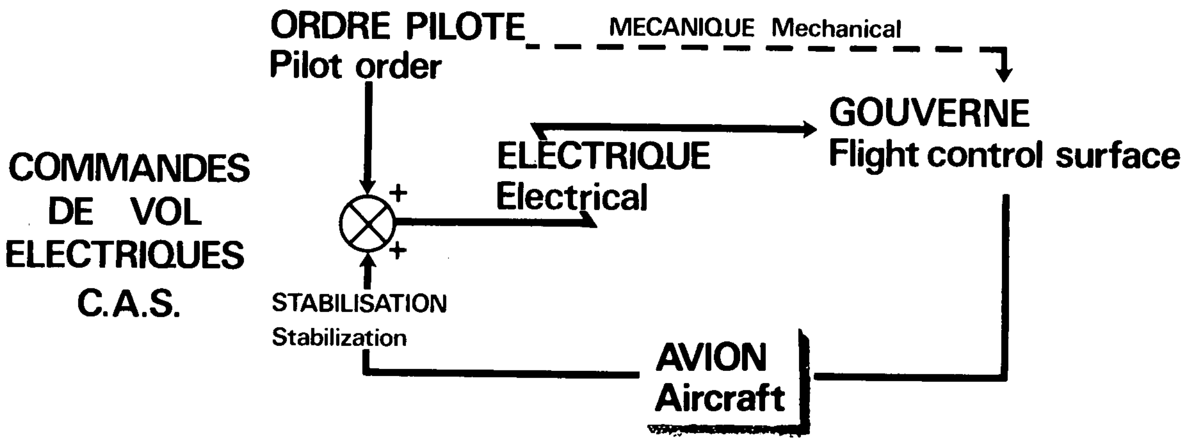

Fig. 5

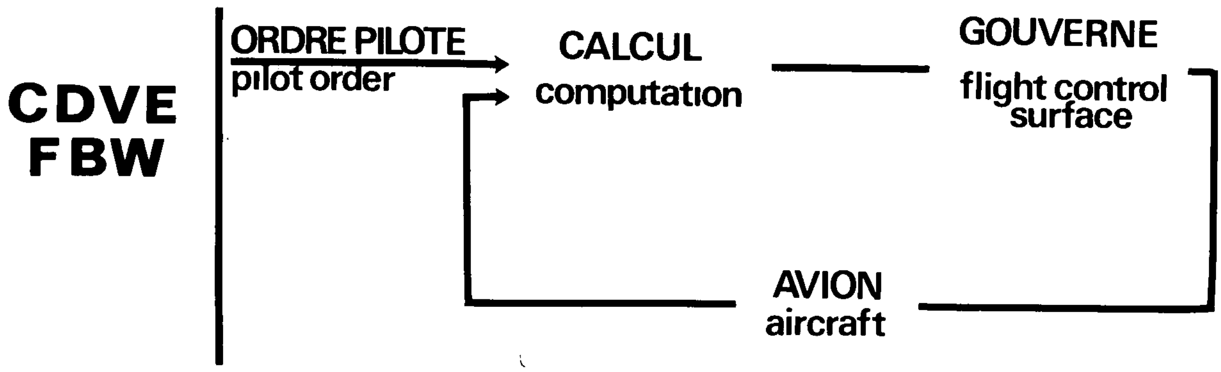

Fig. 6

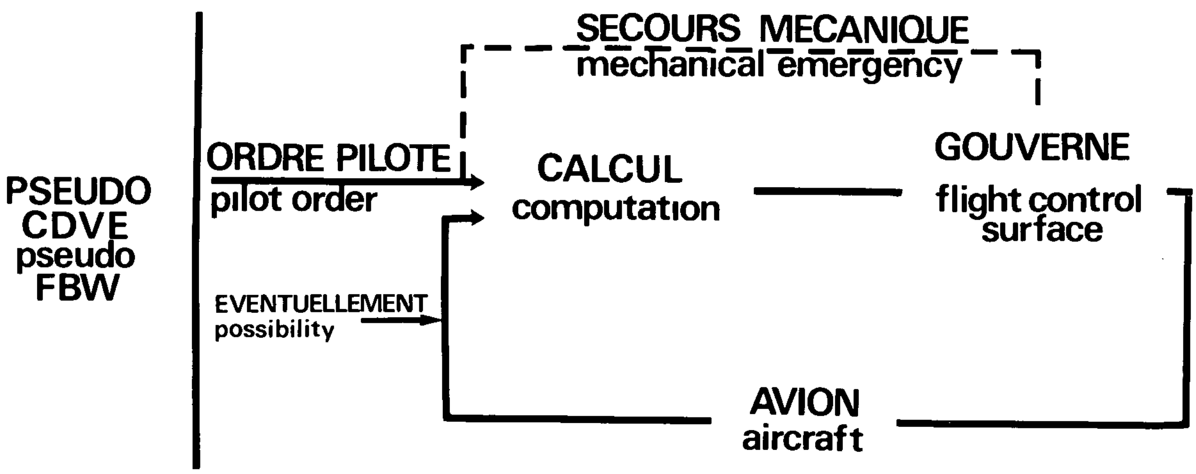

Fig. 7

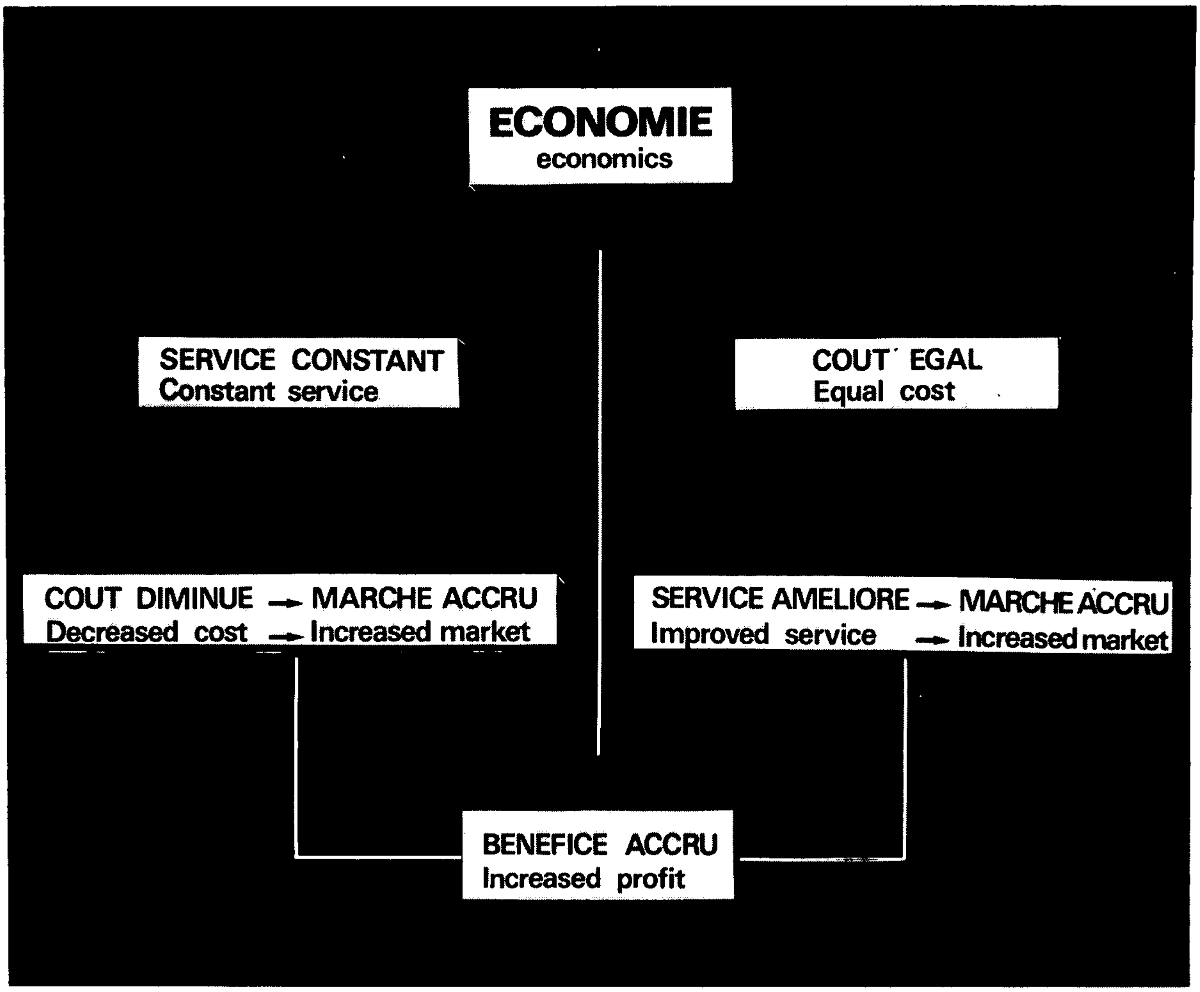
ECONOMIE
economics
SERVICE CONSTANT
Constant service
COUT EGAL
Equal cost
COUT DIMINUE → MARCHE ACCRU
Decreased cost → Increased market
SERVICE AMELIORE → MARCHE ACCRU
Improved service → Increased market
BENEFICE ACCRU
Increased profit

Fig. 8

ACTIVE CONTROL AS AN INTEGRAL TOOL IN ADVANCED AIRCRAFT DESIGN

by

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SUMMARY

The scope of active control in the design and operation of aircraft is broadly reviewed, and the paper covers automatic control, stability and control augmentation, artificial static stability, gust alleviation, stall and spin protection and various methods for reducing airframe loads. It is argued that active control should not be treated as a piece-meal solution to isolated design problems but rather as an integral element in a general advance in aircraft technology. Only in combination with other refinements will the true potential of these powerful techniques be realized. In particular it is shown that many CCV applications require commensurate improvement in the aerodynamic performance of the control surfaces.

1 INTRODUCTION

In a paper[1] recently given to the Royal Aeronautical Society, R.W. Howard was able to speak of "Automatic flight controls in fixed wing aircraft. The first 100 years". The idea of active control of aircraft, it would appear, is by no means new. What is it then that makes us consider the advance towards the so-called control configured vehicle concept as a breakthrough. Are we in fact facing a real watershed or merely an interesting milestone in an orderly evolutionary process.

In the past we have used feedback control almost exclusively to enhance flight characteristics beyond the standard it would be possible to achieve by aerodynamic design alone. Stability augmentation in this form is now essentially an integral ingredient of the modern aircraft. As a consequence the fully operational aircraft is expected to possess handling qualities of a standard which clearly presupposes the assistance of active control. For instance, before the era of the autostabilizer, airworthiness requirements demanded the dutch roll to be damped to about 10% of critical and even this modest standard was hardly met except in a small part of the flight envelope. Today the MIL Spec. asks for a minimum of 19% and this requirement is firmly expected to be satisfied over the full operational flight envelope. Only recourse to active control allows the designer to satisfy these exacting standards. In other handling fields the situation is similar. So we observe that in the area of handling the designer of the modern aircraft relies already entirely on feedback control to achieve satisfactory performance.

However, there is one important reservation. If the CSAS fails we still expect the basic airframe to exhibit a residual degree of stability that allows the pilot at least a safe return to base. What the proponents of at least some of the more adventurous CCV schemes suggest is the abandonment of this principle. If active control can be guaranteed sufficient integrity, then it can be entrusted with the very basic safety of the aircraft and it is this possibility which opens an entirely new range of applications for active control. In particular it allows one to ignore, in the design and stressing of the airframe, certain constraints which in the past have been considered inviolate and to entrust the control system with functions which so far had to be catered for by passive design features, such as large tail surfaces for aerodynamic stability or structural strength and stiffness to withstand all conceivable flight loads and ensure structural stability.

CCV is therefore offered principally as a more efficient alternative to conventional design procedures, not necessarily aiming at superior performance, as is the purpose of, for example, the conventional CSAS; but at the achievement of present standards by more efficient means. When CCV is used in this fashion, its cost effectiveness can be directly assessed by comparing the cost of solution A to that of solution B. Such a clear-cut economic assessment is much less easy if a system is designed to improve some qualitative performance feature, say ride comfort. Then it may be difficult to attach a unique value to the promised improvement and hence to balance it against the cost of implementation.

Only if the savings in structure weight and drag outweigh the penalties associated with installing and operating a new system can that system be considered cost effective and technically sound. In practice some significant gains will be demanded before a designer or customer will commit himself to the inevitable risks associated with a major technical innovation.

This now brings us to a crucial question. Are the benefits promised by the various CCV schemes under discussion really solid enough to warrant serious practical interest? At present, opinions on this point still differ. In the first place, different types and classes of aircraft obviously offer different scope for CCV. Gust load relief will be of little interest to the designer of a combat aircraft stressed to an n_1 of 8 or so. However, he may well be very interested in ride or fatigue life improvement for low level operations. Airworthiness certification is a major obstacle in civil applications. Schemes which would leave the aircraft in an unsafe or even unflyable state when an active system malfunctions will presumably have to be proved in military service before they attract the designer of civil transport.

In the UK as elsewhere, a good deal of effort has been put into feasibility studies and trade-off assessments, most of them unpublished. The broad conclusions from most of these studies appear to agree by suggesting weight savings which would allow the aircraft to be scaled down by something of the order of 9% or thereabouts for the same mission capability (see Refs. 2 and 3, for example). These gains would result from reductions in the size of tail surfaces made possible by active control providing static pitch and yaw stability. Studies on a civil transport design, where the major benefits are likely to come from load alleviation give a comparable gain of 4% or perhaps 6% if several CCV functions are applied simultaneously. A Boeing study[4] arrives at a 13% reduction in take off weight or a 7% reduction in DOC from a study of a near-sonic advanced transport aircraft exploiting artificial static stability, load alleviation and in addition a restriction in the CG range.

Broadly, these and similar studies suggest performance benefits which are certainly not insignificant but perhaps not so demonstrably dramatic that they invite instant commitment. It is heartening to see that General Dynamics have put their money on CCV in the YF-16 design and this may well herald the needed breakthrough in confidence for CCV to get seriously off the ground.

In most of the studies I have mentioned, CCV has been applied restrospectively to a conventional design taken as a datum and the benefit then used to scale the original design down to exploit the drag and weight savings offered. In one case of a combat aircraft study this process had to stop when the airframe was shrunk by 9% (wing area and weight) as further reduction would have left insufficient volume to accommodate fuel. Without this constraint in this case a reduction by as much as 15% would have been possible. Clearly this limitation need not have arisen if the CCV version could have been designed from scratch and the overall configuration optimised appropriately. In fact there was some evidence that a tailless design might have emerged as the optimum solution from this process.

Without CCV, such a configuration would be decidedly unattractive because of the well-known aerodynamic disadvantages of the tailless layout. As we shall discuss later in more detail, artificial longitudinal stabilization can effectively remove much of this handicap from the tailless aircraft and convert it into a powerful competitor to the traditional tailed layout.

What this suggests is that CCV is not to be treated as a cure for some localized design problem but that it must be handled as an integral element in the whole design procedure. Only if this opportunity is fully exploited will we reap the real benefit of these revolutionary techniques. It is this aspect that I wish to make the main topic of this paper as I consider it crucial for both the sound appreciation and the proper direction of effort in this field.

In particular I wish to explore the interaction of advanced control with other aspects of design, operation and certification.

2 A CATALOGUE OF THE POSSIBILITIES

It is perhaps expedient to remind ourselves of the range of functions in which advanced active control can be visualized to be employed. In this list I reiterate all that is well-known but include also one or two possibilities that have not so far been given any publicity. Broadly one can perhaps distinguish two main areas, one in which flight control and response of the rigid airframe is modified and another where the objective is improved structural efficiency.

A FLIGHT CONTROL FUNCTIONS

1 The auto-pilot

2 Stability and control augmentation

3 Artificial static stabilization in pitch and in yaw

4 Gust alleviation for ride comfort

5 Stall control and spin prevention

B STRUCTURAL FUNCTIONS

6 Manoeuvre load alleviation

7 Gust load alleviation

8 Mode control

9 Active flutter control

10 Manoeuvre load limitation

In this paper mainly the flight control functions will be considered, being closer to the author's expertise, but we shall discuss in some detail the concept of manoeuvre load limitation, a less publicised idea and one that affects structural stressing as much as flight control.

It is not possible meaningfully to discuss advanced applications of active feedback control without at least paying some attention to advances in associated design areas. In particular we shall have occasions to consider the demands made by CCV on the aerodynamic performance of the relevant control surfaces; especially we shall discuss

(a) the spoiler as a direct lift control;

(b) high lift capability as a design requirement for the horizontal and vertical tail.

3 ACTIVE FLIGHT CONTROL

This is of course the field where active control - first in the form of the automatic pilot - was first pioneered in aviation. What the more advanced ideas under discussion today suggest may well be seen as the evolutionary development of a well established discipline. We shall first consider the auto-pilot. Although not in itself a novel device we shall use it to demonstrate that in this area too integration with other advanced design features can dramatically enhance its performance.

3.1 The auto-pilot

The modern auto-pilot is capable of a wide variety of control functions. Of these we shall consider only one particular mode, namely automatic landing. The development of the autoland system was inspired originally by the desire to overcome aviation's perhaps most irksome limitation, the inability to operate, i.e. to land, in conditions of poor visibility near the ground. Although fog slows all movement, only the aircraft is totally immobilized.

This is still the primary objective of the autoland system we now see entering operational service. However, once it was shown that in this most critical flight phase the human pilot can be successfully replaced and even bettered by servo machinery, it became obvious that this technique has equal potential in situations where the pilot is not limited by lack of visual reference, but by other impediments. In principle automatic control promises more accurate and consistent performance than can be expected of the unaided human operator. Automatic landings should therefore result in more consistent and more gentle touchdowns. Surprisingly, the present generation of automatic systems does not in fact realize this expectation, manual and automatic landings giving very comparable touchdown statistics in otherwise comparable conditions. At the moment this is not very important because we accept normal manual landing performance as generally acceptable and the autoland system was designed to permit flight operations in fog and not to better pilot's performance in conditions when he could see.

However, the autoland system could readily be developed to improve touchdown performance as such if it were combined with direct lift control, DLC. Fig.1 shows the results of some computer studies made by Lockheed for the L 1011 to demonstrate what could be achieved if DLC were added to the normal autoland system of the aircraft. Illustrated is the improvement in touchdown scatter. Similarly spectacular improvements were predicted for the other touchdown parameters, notably vertical velocity.

In the UK we are just starting to flight test an autoland system on a BAC 1-11 aircraft, which has been modified to make its existing spoilers available for DLC. Not enough flying has been done so far to make firm claims but what we have seen strongly suggests that flight performance will match theoretical predictions which promised virtually halving all significant touchdown parameters. Ref.5 from which Fig.2 has been extracted shows what the computer predictions are for this system. We note in particular how effectively DLC desensitizes the aircraft against turbulence.

On the other hand when used by the pilot in manual control DLC appears to give comparatively disappointing improvements as, e.g., shown in Ref. 6.

We conclude therefore, that two advanced design concepts, automatic control on the one hand and DLC on the other, each on its own are relatively ineffective in bettering the landing performance in conventional manual operation but that in combination they produce dramatic results.

In spite of the generally accepted potential of the DLC - autoland combination there is still little evidence that the aircraft designer is yet ready to exploit this capability. The L 1011 is a good example of a case where the customer apparently could not be persuaded to opt for the better product incorporating DLC. One powerful reason is simply that present touchdown performance is adequate, i.e. that it can be comfortably accepted by the conventional undercarriage and that the margins demanded for the landing run also are capable of accommodating existing touchdown scatter. If this view is correct then improved touchdown performance is merely a luxury, for which no expenditure is justified.

However, I would like to suggest that this is too narrow a view, being based on the acceptance of a whole string of traditional practices, one being the undercarriage design requirements, another landing performance margins and another the adherence to the 3° approach path. DLC-assisted autoland can allow all these design assumptions to be relaxed and it is in this direction that I see the true value of the actively controlled landing.

Of course there is another important hurdle to be cleared. In order to exploit the potential of automatic landing control in the wider context of aircraft design, this mode of operation must be available full time. It will be argued that this pre-supposes the availability of suitable guidance at all airfields and that this is an unrealistic expectation for the foreseeable future. This is true if we discuss automatic approach and landing as an integral package, but not if we merely ask for automatic assistance only in touchdown control. A system with this limited function can be fully self-contained, needing only radio altitude as a guidance signal. What we are in fact advocating is an extension of the function of the oleo in the airborne phase of the flare. If we could equip aircraft with such a system, which will of course also serve as the final stage of a fully automatic landing system and if we can persuade pilots to accept the idea - then we are in a position to reconsider all the design aspects dominated by touchdown control. The undercarriage could be relieved of much of its present design condition, namely the vertical impact, and could be reoptimised as a more accommodating suspension for the ground run with consequent savings in undercarriage weight and reduced ground-induced airframe loads, especially in their fatigue aspects. The latter argument is particularly relevant to transport aircraft, which typically consume 50% of their fatigue life whilst taxying over the airfield surfaces.

Also, steep approaches would clearly become more manageable, possibly even not needing a stepped flare, since there is more than enough lift available to perform a sound flare from substantially steeper approaches than are in use at present. For STOL operations another important benefit is the more consistent placing of touchdown which must help to meet the tight landing-run requirements.

It will be noted that such a scheme would require a fundamental revision of existing airworthiness rules.

I have dwelled on this example at some length as it allowed us to demonstrate that major advances in aircraft design, such as are possible with reliable active control, can have repercussions over a wide field of design, operations and airworthiness and that only an integrated effort in all these areas will allow the full benefits to be extracted.

3.2 Stability and control augmentation

In this area, active control is of course already widely and vigorously exploited. A wide spectrum of possibilities exists from simple rate dampers to electrically signalled manoeuvre demand systems.

So far the only major restriction on the scope of these systems was that they should be grafted only on to a basically stable and controllable airframe, so that the pilot would at least be able to return to base after total system failure. This restriction implies in particular that the basic aircraft stall be statically stable in all axes. Although the provision of static stability by feedback control is essentially a CSAS function, it is treated generally as a special case and discussed under the CCV label. We shall do likewise and restrict the discussion at this point to more conventional CSAS schemes. Artificial static stability will be considered in the next section.

Even with conventional applications we are now beginning to meet configurations where safe controllability of the basic airframe is in serious doubt, i.e. where it fails to meet the bare minimum handling requirements (level 3 in MIL-F 8785 B). This forces one then to seek reliability of the CSAS which equals that usually associated with the more sophisticated CCV schemes or with full-time fly-by-wire. Here the borders between conventional CSAS and CCV become blurred and the development towards CCV standards will happen in the course of natural evolution.

This is not the occasion for a review of conventional technology and I shall therefore only consider,in the CSAS context, areas where more fundamental developments may still be expected.

Stability augmentation originally evolved under the constraint of limited control authority. This offered the most convenient form of protection against malfunction. One of the consequences was that CSAS was and still is today seen mainly as a means of enhancing handling in a relatively narrow 'normal' flight envelope. It does not address itself specifically to flight safety, where one is usually more concerned with excursion towards and beyond the boundaries of the principal flight envelope. This point is worth stressing because it is usually assumed that any improvement in what is generally called handling inherently also enhances safety. In fact an aircraft with exceptionally undemanding handling qualities may present the pilot with a real problem when it is flown beyond the limits of the CSAS authority and suddenly reverts to its marginal basic flying characteristics, even if they are not very poor by absolute standards. I believe that this aspect needs much greater attention, in particular we may need to consider nonlinear control which would be able better to mimic the pilot's natural reaction to dangerous situations, but we shall return to this topic when discussing artificial static stability. There the problem becomes clearly most acute.

Flight safety dominates especially the high incidence regime where a whole catalogue of flight hazards awaits the pilot and when, at present, CSAS gives him little assistance. We shall raise this problem in section 5.

3.3 Artificial static stability

The requirement for aircraft to possess natural aerodynamic static stability in pitch and in yaw has up to this point in time been treated as an inviolate design constraint. If automatic control can be entrusted with these fundamental stabilization functions, the designer then has freedom to configure his project much more for optimum performance. In particular, CCV promises to reduce the demands for vertical and horizontal tail area and the resulting weight and drag savings either lead to improved performance or they allow the size of the aircraft to be scaled down to maintain a given performance.

Many studies have been made on this theme (e.g. Refs. 2 and 3) and most of these have applied this idea to some conventional design as a datum. We have already stressed in the introduction the limitations in such a narrow approach when CCV is treated almost as a retrofit and little use is made of the possible liberation of the designer from traditional constraints.

There is one type of configuration, especially, that CCV may be able to revitalize - the tailless aircraft. We take Concorde as an example. Although an excellent shape for its cruise mission, the tailless slender wing sets severe limits to low speed performance. There are three factors involved. Its low aspect ratio attenuates lift slope although the essential absence of a stall and the presence of leading-edge vortex lift at high incidence make up some of the deficit. Flaps cannot be used as there are no means to trim their pitching moments. Finally, trimming requires the elevons to be deflected up and this implies a further loss in available lift, (see Fig. 3a). However, if one were to configure the aircraft longitudinally in the unstable sense, i.e. place the centre of gravity aft of the aerodynamic centre, the elevons would be deflected down for pitch trim (Fig. 3b). In this position they act as normal high lift devices. Elementary calculations show that for Concorde the consequent gain in trimmed lift would be close to 4% for every one per cent rearward movement of the centre of gravity. With the controls presently installed on the aircraft it appears that in this way the lift available for take off and landing could theoretically be increased by up to 20% - a very formidable improvement. Modest moves in this direction also improve drag, since optimum L/D is associated with a modest positive camber, i.e. with a moderate down-deflection of the trailing edge.

Unfortunately, this potential cannot be exploited on the aircraft now since the implied relocation of weight and undercarriage is not practicable as a retrofit. However, a tailless aircraft designed ab initio as a CCV configuration would be another matter altogether and make this shape a very strong competitor to more conventional designs not only for supersonic transport but for many other roles as well.

Generally the design requirements for pitch and for yaw stability are quite distinct and we shall therefore treat these separately.

3.3.1 Pitch stability

In spite of the fact that the horizontal tailplane is frequently referred to as the stabilizer, its only essential function is to provide pitch control and trim. Static longitudinal stability depends entirely on the relative location of the centre of gravity with respect to the aerodynamic centre and the presence of a tailplane does not alter this fact. Of course once it is installed the tailplane makes a contribution to overall stability, but none that an equivalent forward shift of the centre of gravity could not match. Strictly we should acknowledge, however, the pitch damping contribution of the tail for which there is no such natural equivalent. In this sense, the tail makes indeed a unique contribution to stability, namely dynamic and manoeuvre stability.

The CCV concept of an artificially stabilized aircraft demands that the aircraft be configured so as to minimize requirements for pitch control and trim, so that these can be satisfied with the smallest possible tailplane. The ideal configuration in this sense is a neutrally stable airframe. Theoretically such an aircraft requires no control to trim, the elevator need only have just enough power to overcome inertia and damping in pitch to be able to initiate and terminate manoeuvres. In addition some control must also be provided to allow the CCV system to exercise its stablizing function. In practice this idealized picture is complicated by the need to allow the centre of gravity to vary over a reasonable range and to trim configuration changes. Flaps, airbrakes and disposable external loads are obvious examples. Another design condition is nosewheel-raising for take off. When all these factors are fully accounted for, one may arrive at a situation when the centre of gravity straddles the aerodynamic centre of where the whole centre of gravity range is in the unstable sector, as in the example shown in Fig.4. We note there that in this - perhaps unusual case, trim of the low speed configurations totally dominates tail sizing, so that the optimum CCV layout turns out to be a very unstable aircraft, needing powerful and reliable active control to become flyable.

If an aircraft features a short tail, trim lift becomes an important factor in its own right. In this case the optimum solution may not necessarily favour the smallest possible tailplane but a somewhat larger one trimming an unstably configured aircraft with significantly beneficial trim lift. The tailless aircraft discussed earlier and illustrated in Fig.3 is of course the most extreme example of this condition, when tailplane size as such has completely disappeared as a factor.

When CCV is used to reduce tail-volume requirements then it may be possible that the foreplane becomes an attractive alternative. If its size is reduced, many of the well-known disadvantages of the canard will be scaled down in proportion and it may well then offer the optimum solution.

The aircraft depending on active control for positive static stability poses a substantial safety problem. One well-understood aspect of system integrity which is the province of the system's expert and which I shall leave to them to consider. There is, however, another fact which is perhaps equally important and which has wider implications.

The stabilization system will in practice be limited in its scope by the control power available to it. Once that authority is exhausted the aircraft will find itself effectively in a fatal superstalled condition. Protection against such a possibility must be a major concern. One solution is clearly to provide a generous amount of aerodynamic control but that may not seem very attractive in a scheme which has minimising tailplane size as the prime objective. Close attention will have to be paid in this area to the most effective use of the smallest possible amount of control. I believe that here is a case for non-linear control. Once a situation is reached where only little control power remains the most prudent course of action is to use this in a massive recovery manoeuvre rather than to bring it in gradually as and when the situation further deteriorates. From this crude argument it would seem to follow that a non-linear control law with rapidly increasing gain as the authority limit is approached would offer maximum protection. The stick pusher is of course an example of that form of control, albeit in a very crude version.

3.3.2 Directional stability

As opposed to the role of the horizontal tail the fin has as its main function the stabilization of the unstable fuselage. It alone provides directional stability and there is no natural alternative. In addition it has to satisfy trim and control demands, such as caused by power asymmetry and the need for control of sideslip in crosswinds. Active control can be use to 'amplify' the restoring moment of a fixed fin by suitably deflecting the surface of the rudder attached to it. The sideslip angles an aircraft experiences in flight - such as those produced by gusts - tend to decrease with increasing airspeed. As a result one will generally find that in high speed flight the fin is grossly underutilized as far as its aerodynamic capability is concerned. Typically at M = 2 the sideslip envelope of an aircraft may be within as little as $\pm 1^\circ$. That implies that a fin having 1/10 of the size of the original and actuated to deflect to 10 times the sideslip angle will produce the restoring moment of the original without danger of running out of lift. This miniature fin would therefore be an adequate replacement if driven by an appropriate feedback signal. At low speeds this argument breaks down, because the stalling incidence of the surface would quickly be exceeded. Artificial yaw stability is therefore généerally more appropriate in designs where n_v becomes deficient at high speeds. This was the case with the TSR 2 project and the similarly ill-fated Avro (Canada) Arrow. Both these aircraft were designed to use directional stability augmentation at supersonic speed. This resulted in each case in a substantial saving in the required fin size. BAC estimate that without this solution the TSR 2 aircraft would have had to be scaled up by about 4% to maintain the same performance.

The scope of this technique is severely limited at low speeds when the fin is often required to operate close to its stalling sideslip. 30 knots of crosswind at 100 knots forward speed implies 18° sideslip, a very real design case for take off and landing, especially of RTOL and STOL aircraft. Clearly the fin of such an aircraft can only be reduced significantly with active control if the smaller surface is capable of generating the same yawing moment, i.e. lift as the original fin. As distinct from the pitch case it is not possible to reduce the demands on yawing power from the aerodynamic surface in the directional case. This now brings us to a topic which is equally relevant to pitch and yaw stabilization by active control, namely the role of aerodynamic control efficiency.

3.3.3 The role of aerodynamic control efficiency

In the preceding discussion we had shown that with the use of active control for static stabilization the design requirements for the pitch and yaw controls call for maximum lifting capability. In the orthodox case one is much more interested in tail volume, i.e. in the lift slope of these surfaces. This then implies a changed emphasis in the aerodynamic performance and design of these surfaces. Unless there are improvements in this area the scope of CCV in this field may be severely limited. Indeed one can go further and suggest that if it makes technical sense to employ sophisticated servo-control technology with the object of reducing the weight and size of these so called secondary airframe components then it will also make sense to look at the same time at the aerodynamics efficiency of the traditional tailplane and elevator and fin + rudder combination. Later we shall also cast a similarly critical eye on the principal roll and direct lift control of the modern aircraft, namely the spoiler.

To put some numbers to this proposition let us re-examine the example for pitch control sizing considered earlier (Fig.4). If it were possible to provide a horizontal control surface having twice the $C_{L_{max}}$ available by comparison with the original, then we could satisfy all the design requirements of this aircraft with a very much smaller tailplane as is shown in Fig.5. Whereas originally a tail-volume of 1.06 was needed in the datum configuration, reducing to 0.74 with active control, the latter reduces to 0.37 with the assumed high lift tailplane. More significant perhaps is the fact that even without CCV, the tail could be reduced to a volume of 0.63, smaller than the reduced size possible with CCV alone.

We must emphasise that these results apply only to the particular example chosen. Other design requirements may dominate the picture in other designs and may limit the scope, but there can be little doubt that this technique has formidable potential.

It is perhaps somewhat surprising and disappointing to observe that the whole area of aerodynamic control efficiency has not for a long time been considered a serious subject for aerodynamic research. Once upon a time hinge moments caused some activity but even this is now defunct. Tail surfaces are generally treated as secondary appendages in spite of the evident fact that with the modern combat aircraft they are of very comparable magnitude to the wing. Considering only exposed area, the fin of MRCA has nearly 40% the area of the wing and the horizontal tailplane about 46%. In other words the two together almost equal the wing in size. Surely such items are worthy of the aerodynamicists' very serious attention. This will become particularly beneficial for the control figured aircraft but even without active control there is scope for improvements.

If high lift capability becomes a dominant design requirement for the aerodynamic controls, then we must turn our attention to the high lift devices developed for the wing, as a model. What is needed is a system of articulated surfaces, combining a moving main surface with perhaps several flap type elements. It is not inconceivable that tail surfaces designed to these principles could provide twice the maximum control power when compared with orthodox layouts. There is a clear challenge to the aerodynamicist and airframe designer.

4 GUST ALLEVIATION

The idea of using active control for the attenuation of aircraft response to turbulence attracted attention as soon as powered flying controls became feasible. An attempt was made in the early fifties by the late J. Zbrozek[9,10] to use the ailerons of a Lancaster aircraft for gust alleviation, although the results were rather disappointing. However, with improvements in the understanding of servo control this problem was soon mastered as evidenced by the excellent performance achieved in the LAMS programme[11]. The principal objective of this CCV scheme was to reduce gust-induced fatigue loads and hence to improve the service life of the airframe. Both airframe loading and ride comfort were significantly reduced, both in the longitudinal and the lateral plane. The results are too well-known to need repeating here.

In the case of the B52, gust alleviation was applied as an after-thought, but in the true spirit of CCV one ought, of course, to integrate it in the initial design. The B52 is perhaps a somewhat exceptional configuration, its size and flexibility making it particularly sensitive to gusts. Most modern aircraft, on the other hand, have gust response characteristics that are as a whole considered acceptable, both with regard to airframe loads and ride comfort. In these cases the proper role of active control is perhaps not so much improvement over present standards but the achievement of these standards by more efficient means. Ride comfort considerations normally control the choice of wing loading of the modern strike aircraft to the detriment of other performance parameters, such as manoeuvrability and airfield performance. With active control looking after ride response the designer can ignore this constraint in choosing wing area and optimise it for these other performance aspects. CCV therefore removes a powerful traditional design constraint and it is perhaps in this sense that the true benefit of this technology will best be realized.

Another attractive area is STOL. The simplest way to achieve short-field performance is obviously to reduce wing loading. This would normally lead to an unacceptable ride, unlikely to attract the fare paying public and equally unpopular with pilots. If active control alleviates gust response, such a configuration becomes much more practical and a serious competitor to designs employing sophisticated power assistance to achieve the required lift.

I started by making the broad assumption that current aircraft as a whole have gust response characteristics not in great need of improvement as such and if this is so then gust alleviation will be used to achieve the same standard by more economic means. This assumption may in fact be far from generally valid and it would be interesting to establish to what degree combat efficiency could in fact still be improved by making the aircraft less sensitive to turbulence. The cost-effectiveness of gust alleviation applied for this purpose cannot be properly assessed until we have a realistic exchange rate between its effectiveness for a weapon system and the cost of a CCV solution to this problem. It is worth noting that we still lack handling requirements which specifically set out to control aircraft response to turbulence. This is a serious deficiency in urgent need of attention.

Gust alleviation in the normal acceleration sense demands an efficient direct lift control. Since the old style aileron has now virtually disappeared from the modern aircraft, this function has to be provided by either a spoiler or a suitably modified and activated element of the high lift flap system. For many reasons the spoiler would be more attractive, but in its present form it has many shortcomings. First of all it is only capable of reducing lift. To obtain symmetric control the spoiler has to be operated from a substantial angle as a datum position. This imposes a totally unacceptable drag penalty except at very low speeds when this aspect is less critical. It may be worthwhile to have another look at the spoiler variants not suffering this limitation. The spoiler-deflector combination is one such scheme as a combination of spoiler with a down-going flap another.

Even if we ignore this particular aspect the spoiler appears as a curiously inefficient aerodynamic device, which should not be tolerated even for the roll control function for which it is commonly employed. Fig.6 shows a typical pressure distribution over the upper surface of a swept wing equipped with a spoiler. We note that the spoiler induces two conflicting reactions. By changing effective camber, lift is increased as evidenced by the suction 'bulge' at the rear of the airfoil. This adverse effect is couterbalanced by the true spoiler effect, which slows the upper surface flow and as a consequence reduces general circulation. The fact that the sum of these two effects produces the desired lift decrement appears to reflect good luck more than good design. All the well-known difficulties of obtaining reasonably linear effectiveness stem of course from lack of control over the balance between these two opposing trends.

The spoiler appears to have failed to attract the serious attention of the aerodynamicist. In view of the increasingimportance this control now commands, both as the primary roll control and as a prime candidate for direct lift control, this omission must be remedied. The requirement is very clear, we want a device or devices that control either effective camber or decelerate surface flow to control circulation, but not both at the same time. Here is a challenge to the aerodynamicist overdue even before active control, essential if we want efficient direct lift control.

5 HIGH INCIDENCE STABILIZATION AND SPIN PREVENTION

The traditional CSAS system is designed primarily to improve handling in the principal operating regime of the aircraft. It is not specifically tailored or indeed meant to deal with aircraft behaviour at the extremes of the flight envelope. For most aircraft one is concerned especially with handling up to and at the stall. For combat aircraft, however, control well beyond the regime of flow separation is of real practical interest. Ideally the pilot would like the aircraft to be free from serious control hazards up to the extremes of the incidence range into which the demands of combat may force him. This may well involve angles of attack of 30° and beyond. Safe access to this regime gives the pilot a sometimes decisive advantage, often denied to him because of unmanageable departure characteristics. Once flow separation starts we cannot of course expect the aircraft to maintain smooth and entirely docile flight. Buffeting, wing rock and wing drop, pitch up and nose slicing are the most common irregularities we expect to find in this regime and finally, unless departure characteristics are entirely innocent, the aircraft will spin or indulge in some other form of poststall gyration. The chances of controlling these phenomena by aerodynamic design alone are extremely limited.

There can be little doubt that their suppression by active control would be a very desirable aim, greatly improving combat effectiveness and if spinning could be positively eliminated many accidents would be prevented. What then are the chances of active control in this area? Some limited thought has been given to the idea of active spin prevention[12,13] in the USA, but so far no serious development of appropriate hardware appears to have started. Surely this is an area which ought not to be ignored if we wish fully to assess the potential of active control.

There are in fact two separate areas to which we may have to address ourselves. One is to deal with what I would like to describe as nuisance phenomena, in particular wing rock and deteriorating dutch roll damping. Here conventional CSAS will be able to make a significant contribution, given appropriate gains and sufficient authority. More difficult will be the control of the more serious departure characteristics. It is not possible to generalize since each configuration presents a unique case and may require an individually tailored solution. However, undoubtedly the most common deficiency is loss of effective directional stability[14,15] often associated with adverse sidewash. The solution would seem to be artificial stabilization by active control, i.e. an extension of the technique discussed in section 3.3.2. However, the success of this technique will depend critically on the aerodynamic efficiency of the available directional control. What was said earlier about the need for an articulated vertical surface would seem particularly appropriate here.

Departure prevention is obviously a very wide subject and I can do no more here than advocate some real effort in this much neglected field.

It is worth reflecting that at present we devote a substantial effort to establish spin and spin recovery characteristics of every new design with very dubious results. Even if we can prescribe a recovery technique this will only help if the spin is entered at sufficient altitude as substantial height is lost even in a relatively straightforward recovery. Would it not be more sensible to redirect this effort towards a more positive approach to the problem - that of spin prevention by active control.

6 THE ROLE OF CCV IN LIMITING MANOEUVRE LOADS

Many CCV schemes currently advocated aim at reducing airframe loads or at stabilizing structural modes. Structural design not being strictly within my field I do not feel qualified to comment extensively on these matters. Ref.16 covers the subject in some depth. On the question of manoeuvre load alleviation one aspect, however, may be worth mentioning, as it impinges on aerodynamic design. The principal idea exploited in these proposals is to move lift from the tip regime towards the wing root, thus reducing bending moments. Such a strategy assumes that in fact there is underutilized lifting capacity available near the root and that one can afford to dispense with lift potentially available on the outer portions of the wing. Usually the manoeuvre cases involve operation of the wing close to its buffet boundary, which itself is normally optimised by aerodynamic wing design. This process would not normally permit drastic rearrangements of the spanwise lift distribution without loss of overall capability. In other words there will exist for each case an optimum aerodynamic wing configuration for manoeuvre lift which is likely to differ substantially from that required for load relief. What I suggest is needed, is that the two sides come together so that a true overall optimum be found. The studies that so far have appeared in the literature do not indicate such an approach. In fact in the design of the modern combat aircraft the most difficult task is often the achievement of the required manoeuvre capability. Both instantaneous 'g' without unacceptable buffeting and sustained 'g' are important. It may well turn out that automatic control of spanwise configuration for optimum lift is as useful a concept in this respect as is load minimization for purely structural reasons. Clearly this needs close cooperation between the aerodynamicist, the control engineer and the structural designer.

I would now like to devote the remainder of this paper to another potential role which I can visualize active control of manoeuvre loads to be able to play and which has not to my knowledge so far been much canvassed. I refer to a scheme which I like to define as the manoeuvre limiter. This concept in a sense revives the old idea of the 'g' limiter, i.e. an active system which controls the maximum loads a pilot can apply to the airframe, but in a more ambitious version appropriate to the capability of modern active control technology.

The basic idea is simply that feedback control be used to control aircraft response in such a fashion that full application of the pilot's cockpit control results in a response which takes the aircraft to the limit appropriate to the prevailing flight condition. This limit may be defined by structural strength or by some aerodynamic restriction such as the stall or buffet. The concept is equally applicable to pitch and roll control, but relevance to directional control is less obvious. An essential element of such a control system is a central memory in which are stored the limits of the permissible manoeuvre envelope for each point in the flight envelope. Especially for combat aircraft it may also be necessary to take account of store configurations which frequently demand severe manoeuvre limitations.

The benefits from such a control system are to be found in two entirely separate areas. The airframe designer need stress the structure only for manoeuvres that are operationally useful and not for inadvertent exceedances which I suspect dominate at present virtually all important stressing cases. Moreover if the system can be relied upon to restrict aircraft response tightly to these 'design' manoeuvres the role of reserve factors can also be reconsidered. The overall result ought to be substantial savings in structure weight.

Obviously in designs where gust and manoeuvre loads are of comparable order - as is normally the case with transport aircraft - manoeuvre load alleviation or restriction brings little structural relief unless matched by commensurate reductions in gust loading.

The pilot will benefit similarly as he now commands an aircraft in which he can apply control without inhibition. The full capability is available to him whenever the stick is moved towards the stops.

Clearly the realization of such a scheme demands reappraisal of a whole range of design disciplines, embracing control engineering, structural and aerodynamic design, a fundamental revision of airworthiness concepts and a clear understanding of the actual manoeuvre needs of the user. It represents the total integration and optimisation of handling on the one hand and structural design on the other.

With this challenging proposition I would now like to conclude this paper.

7 CONCLUSIONS

I have attempted to indicate in this paper that the proper application of advanced active control may radically alter many aircraft design and operating traditions. CCV, it is argued, will not realise its full potential if used as a piece-meal solution to isolated design problems. More properly it should be seen as a tool helping in a broad advance of aircraft technology.

Active control makes new and exacting demands on the performance of the aerodynamic controls through which it acts. These assume then a more critical role in the overall design process and it is suggested that the conventional elevator, rudder and spoiler, as we know them today, are perhaps inadequate to this task and badly in need of aerodynamic refinement. In particular, the pitch and yaw controls would benefit from improvements to their maximum lifting capability, as this becomes the design criterion for these surfaces in the actively controlled aircraft. Equally urgent is the need for the development of an efficient direct lift control replacing the present crude spoiler. Improvements here would of course also benefit its performance in the traditional role as a roll control.

Time permitted consideration of only some of the many possible CCV options, but we believe that the central theme of this paper, the plea for integration of advances in all relevant disciplines, applies throughout.

REFERENCES

No	Author	Title, etc.
1	R.W. Howard	Automatic flight controls in fixed wing aircraft. The first 100 years. The Aeronautical Journal, November 1973
2	H. Anders	Parametrische Untersuchung der Laengsbewegung von CCV-Transportern. DLR Mitt. 74-11 (1974)
3	W. Kubbat	Regeltechmische Aspekte cines Flugzeuges Kuenstlicher Stabilitaet (CCV) unter besonderer Beruecksichtigung der Manoeverlaststeuerung. DLR Mitt. 74-11 (1974)
4	L.T. Goodmanson L.B. Gratzer	Recent advances in aerodynamics for transport aircraft. Astronautics and Aeronautics, January 1974
5	F.R. Gill M.J. Corbin	Design and theoretical assessment of experimental glide path and flare systems for a BAC 1-11 aircraft (including direct lift control). RAE Technical Report 74013 (1974)
6	W.J.G. Pinsker	Direct lift control. The Aeronautical Journal, 74, No.718 (1970)
7	B.L. Fister	Wind tunnel tests of a double-slotted rudder. Air Force Inst. of Tech., Wright-Patterson AFB, Ohio School of Engineering. GAN/AE/73-9 (1973)
8	T.B. Huneycutt	Aerodynamic comparison of two double slotted rudders. Air Force Inst. of Tech., Wright-Patterson AFB, Ohio School of Engineering GAN/AE/73A-10 (1973)
9	J.K. Zbrozek K.W. Smith D. White	Preliminary report on a gust alleviator investigation on a Lancaster aircraft. RAE Technical Note Aero 2244 (1953)
10	J.K. Zbrozek	Theoretical analysis of a gust alleviator used on a Lancaster aircraft and comparison with experiment. RAE Technical Report Aero 2645 (1961)
11	Anon	Advanced controls technology. The Boeing Company, Witchita Division. Report D3-8466
12	W.P. Gilbert C.E. Libbey	Investigation of an automatic spin prevention system for fighter airplanes. NASA TN D-6670 (1972)
13	R.T.N. Chen F.D. Newell A.E. Schelhorn	Development and evaluation of an automatic departure prevention system and stall inhibitor for fighter aircraft. AFFDL-TR-73-29 (1973)
14	W.J.G. Pinsker	Directional stability with bank angle constraint as a condition defining a minimum acceptable value for n_v. RAE Technical Report 67127 (1967)
15	J.R. Chambers E.L. Anglin	Analysis of lateral-directional stability characteristics of a twin-jet-fighter airplane at high angles of attack. NASA TN-D-5361 (1969)
16	-	Active control systems for load alleviation, flutter Suppression and ride control. AGARDograph No.175 (1974)

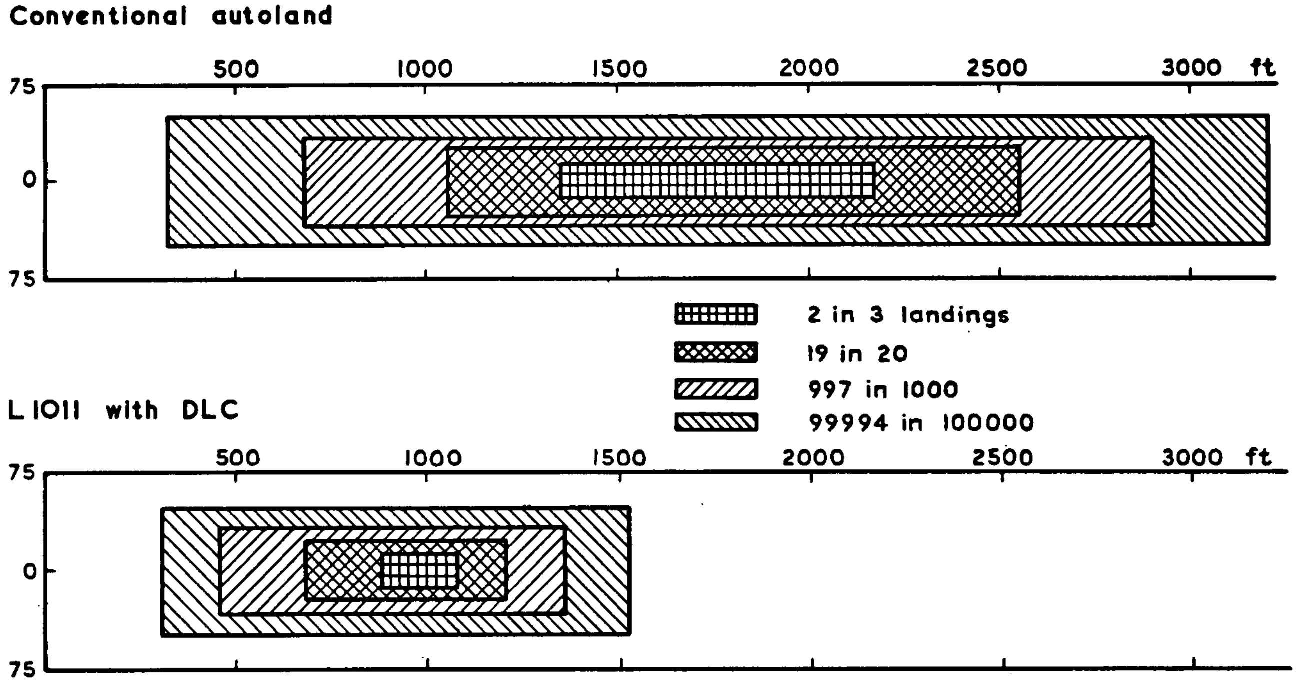

FIG 1 ESTIMATED EFFECT OF DLC ON TOUCHDOWN FOOTPRINT IN AUTOMATIC LANDINGS

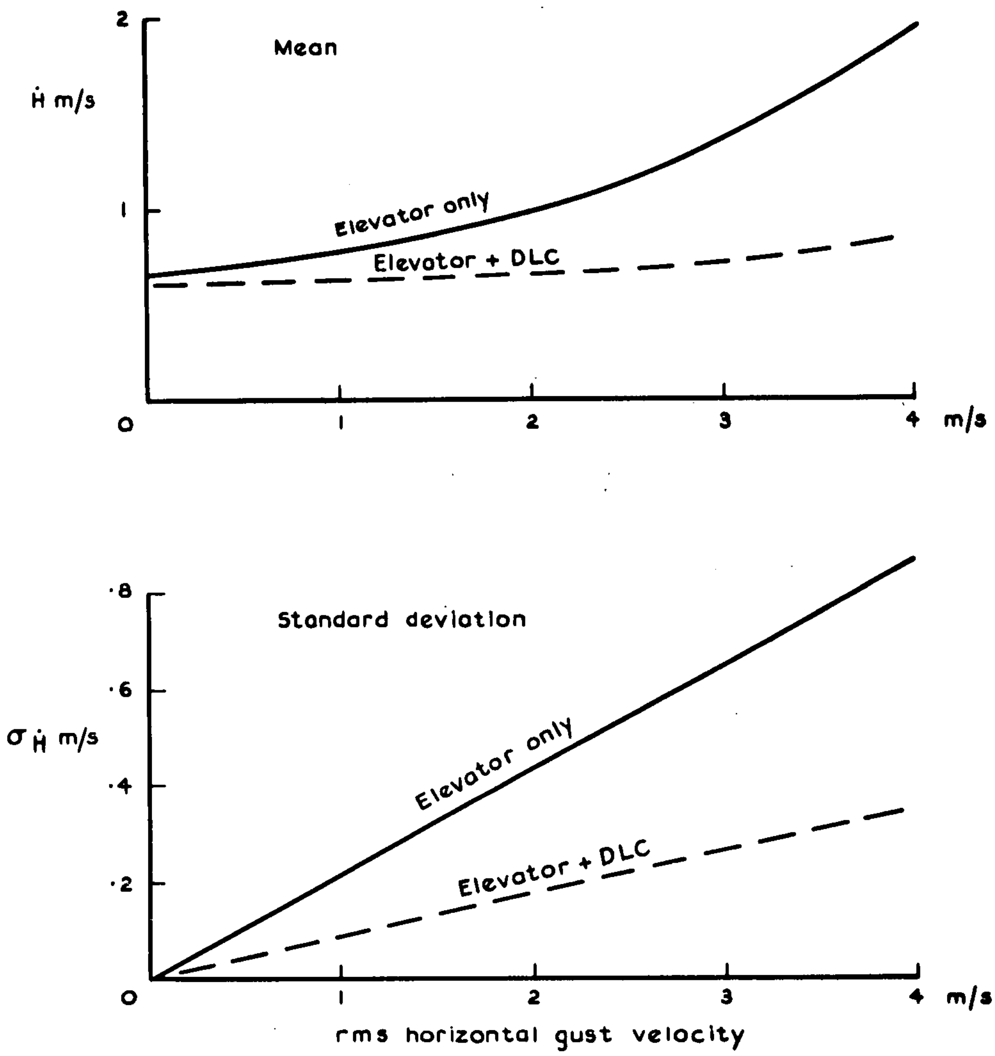

FIG 2 ESTIMATED EFFECT OF DLC ON TOUCHDOWN RATE IN AUTOMATIC LANDINGS OF A BAC 1-11 AIRCRAFT AS A FUNCTION OF TURBULENCE LEVEL (REF 5)

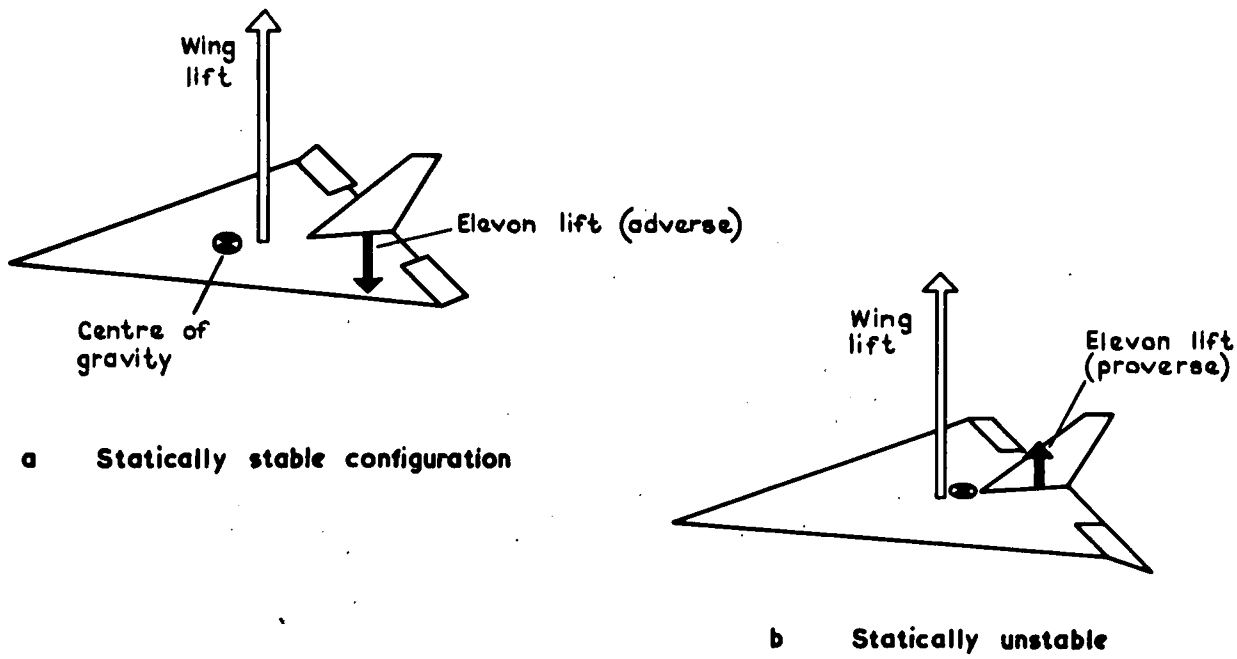

FIG 3a and b TRIMMED LIFT EQUILIBRIUM OF THE STATICALLY STABLE AND UNSTABLE TAILLESS AIRCRAFT

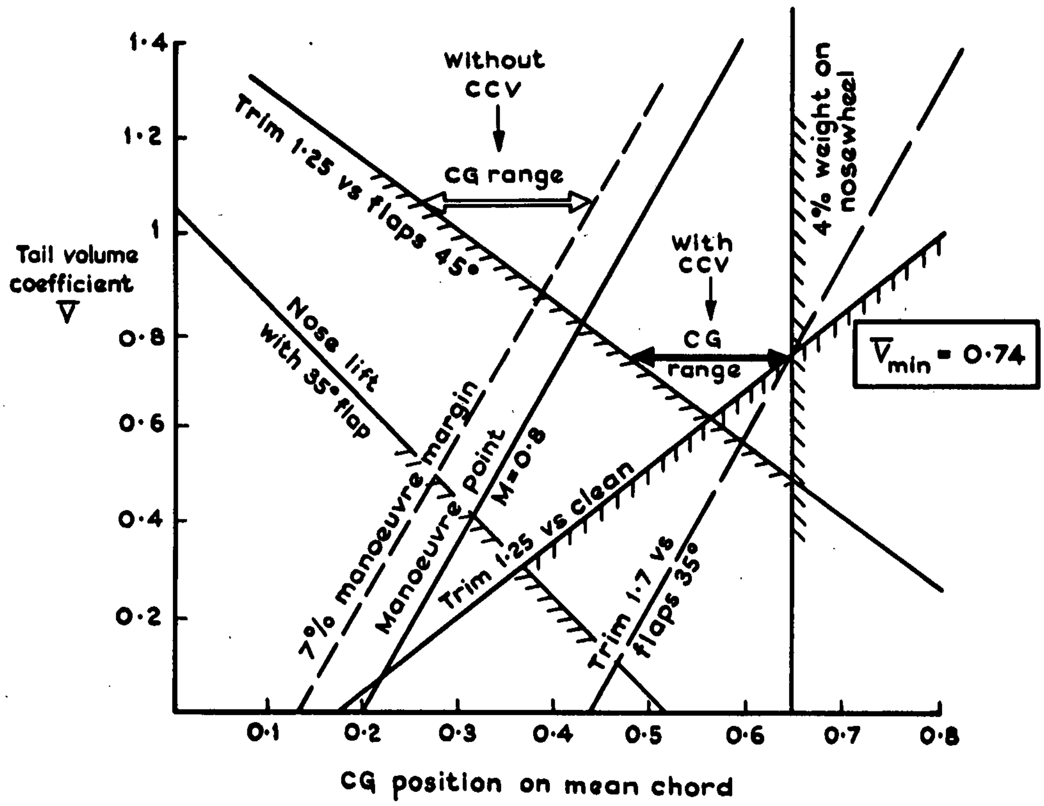

FIG 4 TYPICAL EXAMPLE OF DIAGRAM SUMMARISING TAILPLANE SIZING REQUIREMENTS

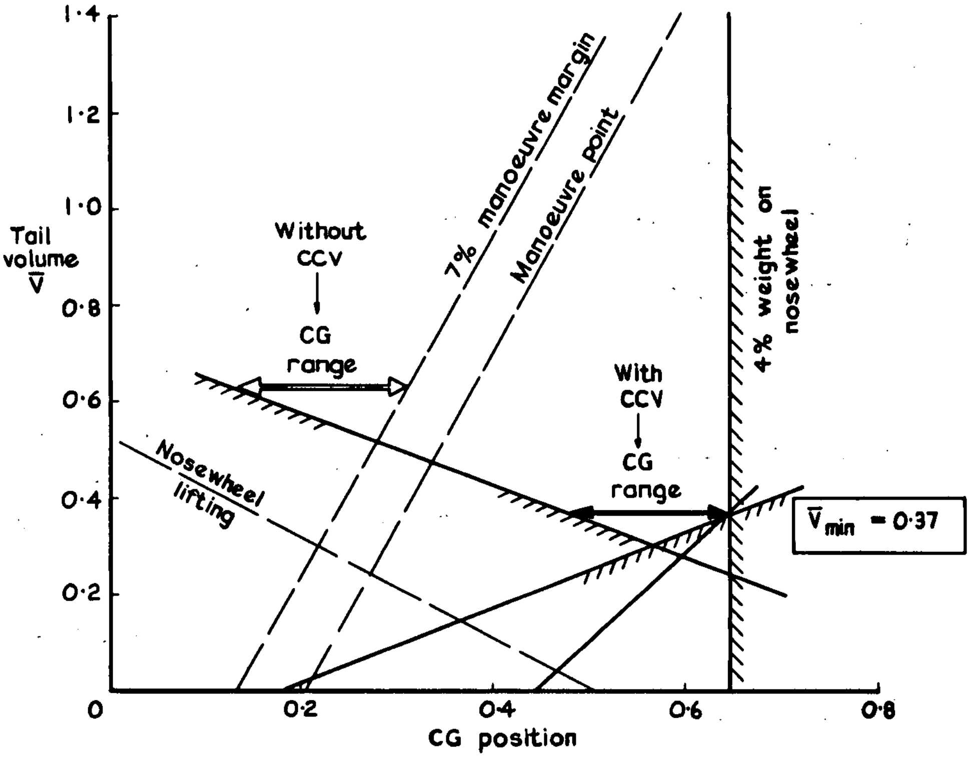

FIG 5 EXAMPLE OF FIG 4 MODIFIED BY ASSUMING TAILPLANE HAVING TWICE $C_{L_{max}}$

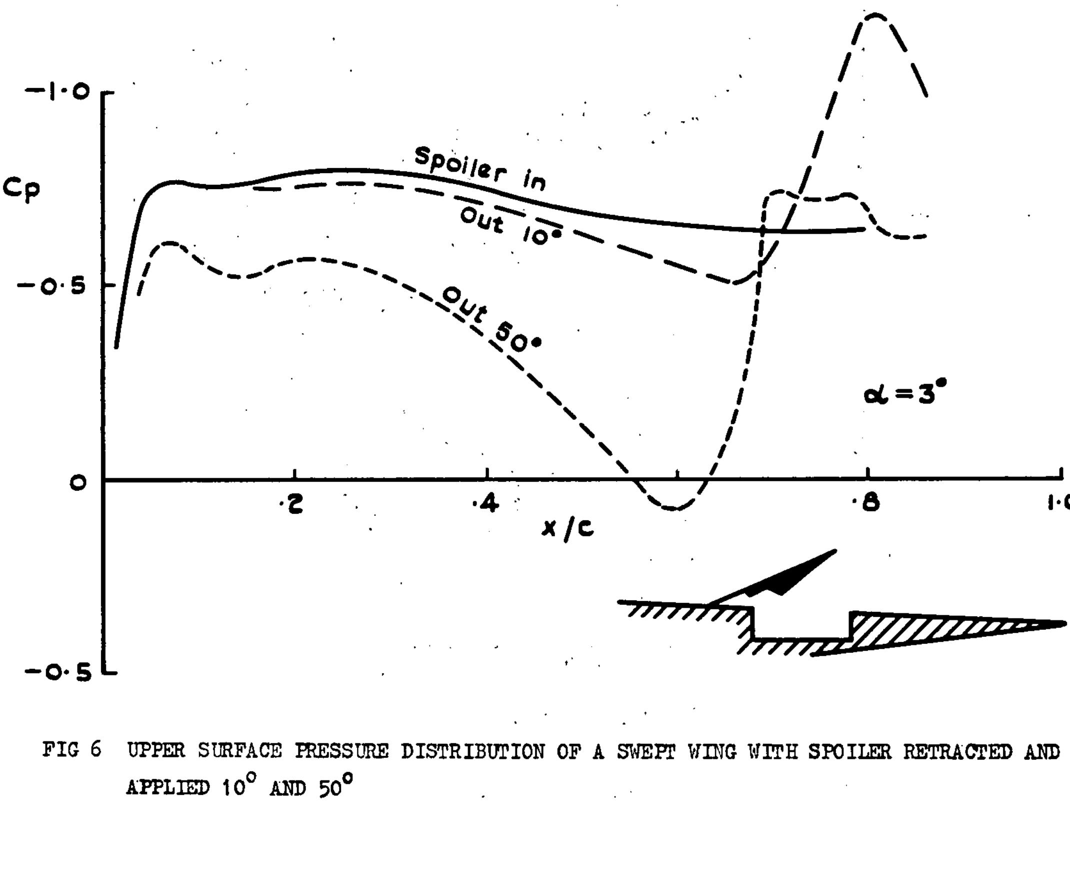

FIG 6 UPPER SURFACE PRESSURE DISTRIBUTION OF A SWEPT WING WITH SPOILER RETRACTED AND APPLIED 10° AND 50°

POTENTIAL BENEFITS TO SHORT-HAUL TRANSPORTS THROUGH USE OF ACTIVE CONTROLS

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SUMMARY

An examination is given of potential applications of active controls for improving the characteristics of transport type aircraft used in short-haul service (< 1,000-kilometer range capability). The types of aircraft to meet future needs (quiet operation, congestion alleviation, fuel conservation, operating economy, and traveler acceptance) are identified as helicopters for shorter stage lengths and fixed wing aircraft of reduced field-length capability for longer stage lengths. Likely uses for active controls for these aircraft are examined regarding payoffs which can be expected and problems and constraints which must be dealt with. Uses showing significant benefits include augmented stability and control, gust-load alleviation, and ride smoothing. Gust-load alleviation is particularly effective for low-wing-loading aircraft employing conventional lift. Ride-smoothing systems are indicated to be the furthest advanced and ready for production commitment for those applications where they can be shown to have payoff.

INTRODUCTION

The development of active-control systems technology broadens the range of aircraft configurations capable of meeting the increasingly stringent demands of today's world. A number of examples of active-controls capability in the areas of aerodynamics and structures were well described in reference 1 for a variety of aircraft. The objective of the present paper is to examine in greater detail the potential application of active controls to a particular class of vehicles: short-haul transports. For clarification, the term "short haul," as used herein, refers to range capability of less than 1,000 kilometers. Attention will first be directed toward outlining technical requirements for short-haul transportation systems and identifying aircraft (together with their characteristics) best suited to meet these requirements. Characteristics which can benefit from active-controls application will be examined to assess potential payoffs. Finally, highlights from a feasibility study of a particular system application (ride smoothing) will be presented to generally assess various factors of system design, implementation, and operation.

SHORT-HAUL AIRCRAFT

The broad objective of short-haul transport operations is to meet mobility needs for transporting passengers over given distances more quickly and conveniently than can competing modes of ground transportation. Situations within this objective vary widely. Distances can be quite short, as exemplified by New York Airways operations (generally between airports) in the New York City area where in 1973 more than 400,000 travelers were transported over stage lengths which averaged only about 25 km. Distances also can be relatively long where a businessman goes to a distant city, conducts business, and returns home in the same day. Great volumes of traffic are generated in this manner between city pairs such as Los Angeles and San Francisco. A third situation concerns less densely populated regions where the terrain or weather lends practicality to air transportation. Examples include traffic between outlying cities in Ontario or between Denver, Colorado, and ski resorts in the mountains nearby. Underlying all the various short-haul situations are common system requirements which can be examined to help identify appropriate types of aircraft.

Vehicle Systems Requirements

The ever increasing complexity of 20th-century life is constantly introducing new factors (such as the energy shortage) which must be considered in the design and operation of advanced systems or system components. These considerations are particularly true for transportation systems where the public is intimately involved. Example requirements for advanced short-haul transport aircraft systems are listed in table 1 for five readily identifiable interests, together with system features which are needed to meet these requirements.

The first requirement concerns the need to maintain low noise levels in the communities surrounding the airports. Demands for an improved quality of life have led to noise control laws and regulations which can be expected to become increasingly more prevalent and strict with the passage of time. Noise certification standards for aircraft have been imposed in several countries and large amounts of money are already being spent to retrofit aircraft to meet the standards. The Civil Aviation Research and Development (CARD) policy study carried out several years ago by the United States Department of

Transportation and the National Aeronautics and Space Administration (NASA) identified noise abatement as the highest priority item demanding attention (ref. 2). Advanced technology features required to minimize noise include quietness of both the aircraft powerplant and of the high-lift system, particularly where the system involves use of high velocity exhaust gases from the powerplant. Also required is the capability for steep climbouts and descents to minimize the area of the noise footprint on the ground.

The second requirement concerns the need for congestion alleviation of the airways system. Air traffic has already reached the saturation point at major travel hubs such as the Washington National Airport. Congestion relief was identified in the CARD study (ref. 2) as having the second highest (after noise abatement) priority for research. Considerable research is underway to improve both equipment and operating techniques for the airways system and the aircraft. Certain features have been identified (see ref. 3) as needed for congestion relief. These include the capability during lift-off for quick short-runway takeoff followed by steep climb. During landing approach, capability is needed for four-dimensional (the three spatial functions plus the time function) control in steep-descent, curved-flightpath operation.

TABLE 1

EXAMPLE REQUIREMENTS FOR ADVANCED SHORT-HAUL SYSTEM

Requirement	Features to Meet Requirement
Low Noise Level in Community	- Quiet powerplant and quiet high-lift system - Steep climbout and steep descent capability
Congestion Minimization of the Airways System	- Quick, short-runway, steep-climb takeoff - Four-dimensional, steep-descent, curved path landing approach
Fuel Saving for Nation	- High lift-to-drag ratio in cruise - Low specific fuel consumption
Economic Operation by Air Carrier	- Efficient high-speed cruise characteristics - Low aircraft weight per seat
Acceptance and Use by Travelers	- Quick, convenient, reliable trip door-to-door - Smooth ride, comfortable seats, ample leg room

A third requirement concerns the need for fuel conservation which, in the present environment of petroleum-related balance of payment deficits, can deeply affect the financial well being of an entire country as well as of individual transport operators. Until recently, the prime concern with aircraft fuel usage has been its impact on the economic factors of direct operating cost and return on investment. Aircraft configurations and operating modes have been selected to optimize these economic factors and fuel utilization is generally not minimized. Ideally, aircraft need to be provided which are not only economically viable, but also fuel conservative as well. Technology features to conserve fuel include high lift-to-drag ratio in cruise, plus low fuel consumption. The powerplants also must be well suited for low-speed, terminal-area operations.

A fourth requirement concerns the need for economical operations to directly benefit air carriers both in their financial well being and in their ability to attract travelers. As pointed out above, fuel conservation practices do not necessarily equate to good economic operations, particularly if fuel economy involves a significant increase in trip time. Cost benefits can be realized, however, from technology which improves efficiency in cruise operations, provided there is no substantial increase in trip time. Additional benefits can be gained from use of features such as advanced structural concepts which may reduce aircraft weight.

A fifth requirement concerns the need for acceptance and use of the vehicle system by travelers. Lack of sufficient sensitivity to the factors which affect one's choice of transportation can lead to near disastrous consequences. A good example in the United States is the extreme decline over the last generation in the public use of railroads for passenger transportation. Features desirable for passenger acceptance in all modes of transportation include the ability for a quick, convenient, and reliable trip over the entire door-to-door journey, plus comfort features such as a smooth ride, good seats, and ample roominess.

The review of requirements presented above has been necessarily brief and admittedly incomplete. Additional requirements, such as safety and maintainability, would expand the list of desirable features. The features presented, however, are considered sufficient to identify the types of advanced technology aircraft appropriate for meeting tomorrow's requirements of short-haul air transportation.

Appropriate Short-Haul Aircraft

The various vehicle systems requirements outlined in the previous section define to some degree the characteristics needed in short-haul transport vehicles. The aircraft should have short field-length capability with good climb and descent characteristics; precise control and good handling qualities; quiet powerplants and high-lift systems; economy in fuel usage without significant increase in trip time for the longer stage lengths; minimal aircraft weight; and, finally, a convenient and comfortable ride.

To the above characteristics must be added the approximate cruise speeds appropriate for various trip distances presented in figure 1. The variation is shown as a shaded band because of individual differences in short-haul market situations. For example, many of the smaller aircraft used in low-density markets will, from economic necessity, probably employ turboprop rather than turbofan propulsive systems and, thus, will operate at lower cruise speeds for a given trip distance than will the larger turbofan aircraft used in high-density markets.

The types of next-generation aircraft appropriate for various trip distances are listed in the shaded area of figure 1. For very short trips, vertical takeoff and landing (VTOL) capability is an obvious requirement. Among the great variety of VTOL vehicles studied or under development, the helicopter is presently considered to be the most advanced for near term transport operations, particularly from fuel utilization and economic considerations (see ref. 4). For very short distances, little trip time is spent in cruise and top speeds can be quite low. In civil passenger carrying operations, helicopter use to date has been limited to very short distances, because of various factors such as inferior ride quality. As technology advances are made and incorporated into hardware, use of helicopter transports can be expected for trip distances up to several hundred kilometers, for cruise speeds up to 200 knots, and for passenger capacities to 100.

Trip distances ranging from 100 to 1,000 kilometers will likely involve use of short takeoff and landing (STOL) or reduced takeoff and landing (RTOL) vehicles. STOL capability, defined in terms of field-length requirements, generally is considered to fall in the range below about 800 meters, while RTOL capability covers the field-length range above that for STOL vehicles and below the approximately 1,200-meter minimum field-length capability generally ascribed to conventional takeoff and landing (CTOL) aircraft. Two types of STOL/RTOL vehicles are discussed in this paper. Conventional-lift type vehicles utilize conventional high-lift devices and achieve reduced field-length capability by use of low wing loading. The de Havilland of Canada DHC-6 Twin Otter is an example of this type aircraft. Powered-lift type vehicles utilize a relatively new principle wherein the high energy airflow from the aircraft powerplant interacts with aerodynamic high-lift devices to produce much higher total lift per unit of wing area than can be produced by conventional-lift configurations. The Bregeut 941 is an early example of powered-lift aircraft. Either propeller or fan-jet powerplants can be used with either type high-lift system.

Likely Applications of Active Controls

Three general classes of short-haul transport vehicles have been identified above as appropriate for next-generation use: helicopter VTOL aircraft; conventional-lift STOL or RTOL aircraft; and powered-lift STOL or RTOL aircraft. For each class of vehicles, active-control applications have been identified in table 2 to improve these characteristics. Examination of these applications is required to determine the extent to which active-control systems can play a significant role in providing improvements.

For helicopter VTOL aircraft, problems generally exist in the areas of the propulsion systems, noise, structural integrity, instrument flight capability, and ride quality. The problem areas of structural integrity, ride quality, and instrument flight capability (as affected by qualities of stability and control) are all influenced by coupling of the dynamic degrees of freedom present in helicopters, and active-controls concepts should find application in decoupling this activity.

For conventional-lift STOL/RTOL aircraft, problems exist in the areas of cruise performance and ride quality caused by the need for low wing loading to achieve reduced field-length capability. With low wing loading, gust inputs cause the aircraft to respond with high dynamic loads in the wing (which results in added structural weight) and considerable accelerations in the passenger compartment. Active controls can reduce both the structural dynamic loads and gust-excited accelerations. An additional application is to provide relaxed static stability for reduction of cruise drag, as described later.

For powered-lift STOL/RTOL aircraft, problems exist in the areas of noise, aerodynamic stability, handling qualities, piloting control, ride quality, and structural loads environment. Problem areas most appropriate for active-controls application include those associated with low speed, terminal-area operations where conventional aerodynamic lift authority is deficient (stability, control, and ride quality).

Some commonality between aircraft exists with regard to likely uses for active controls. A more detailed examination follows in each applicable area of active controls regarding potential payoffs.

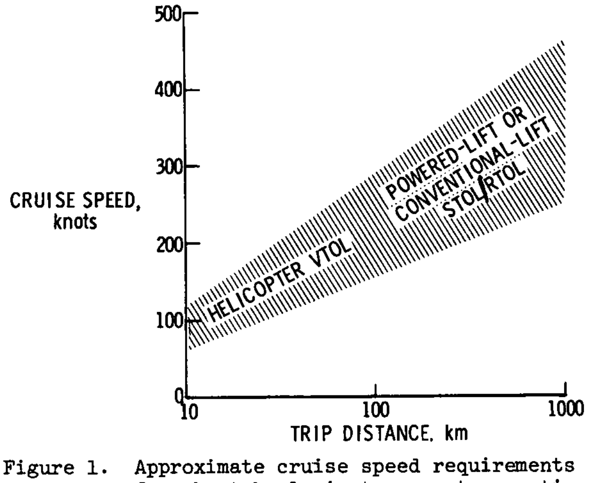

Figure 1. Approximate cruise speed requirements for short-haul air transport operations.

TABLE 2

LIKELY APPLICATIONS OF ACTIVE CONTROLS

Type Aircraft	Active Control Applications
Helicopter VTOL	-Augmented stability or control -Decoupling degrees of freedom -Ride smoothing throughout trip
Conventional Lift STOL/RTOL	-Gust-load alleviation -Ride smoothing throughout trip -Relaxed static stability
Powered Lift STOL/RTOL	-Augmented stability or control -Ride smoothing in terminal area

AUGMENTED STABILITY AND CONTROL

Augmented stability (AS) is a technique for eliminating the requirement for inherent static and dynamic stability by augmenting the stability with an active-control system to a level that provides desirable handling qualities. Augmented stability provides better control response which improves maneuvering performance. Relaxed static stability is another name that has been applied to augmented stability.

Helicopter - VTOL Application

A helicopter operational environment consisting of takeoffs and landings in confined areas without runways involves the use of a steep, circuitous approach path with the aircraft decelerating to zero speed as it descends to landing. Because of the number of variables in the steep, decelerating approach, the level of pilot workload is sufficiently high to directly limit what can be achieved in instrument performance capability. Control and display systems are needed to alleviate many of the tasks for the pilot and to provide capability to perform smooth, accurate approaches.

The piloting task is aggravated by the dynamic complexity of the vehicle, described in reference 4 and illustrated in figure 2, as including: ". . . all elements of the vehicle, with the pilot and controls as a coupled system. All the degrees of freedom are excited continually by periodic inertial forces and by complex aerodynamic loadings. The rotor is an elastic device, with many modes within itself, flexibly attached to the rest of the vehicle with all its masses in a springy relationship. The control system represents yet another elastic path through which many of the dynamic degrees of freedom can couple and interact."

An active-control system can decouple much of this activity and provide stability and control augmentation to a level required for automatic flight (with pilot backup) for foul-weather operations. A pertinent description of rotorcraft automatic landing technology is contained in reference 5.

Conventional-Lift STOL/RTOL Application

Use of conventional-lift configurations with reduced wing loading is a straightforward approach for achieving STOL/RTOL capability. A wealth of experience exists dating back to the low-wing-loading Ford Trimotor, a popular short-field transport aircraft in its day. In today's markets, the low-wing-loading approach tends to have disadvantages in cruise drag, structural weight, and ride quality, all of which, however, are amenable to improvement by active controls. Use of an augmented stability system for reducing cruise drag is the subject of this discussion.

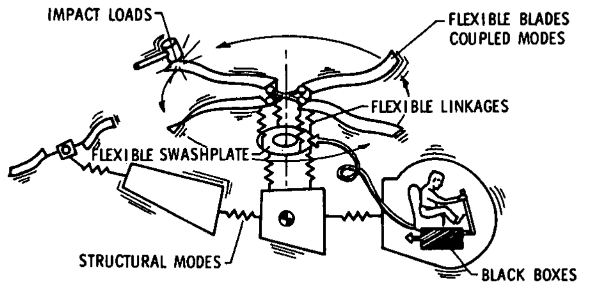

Figure 2. Illustration of the dynamic complexity of helicopters

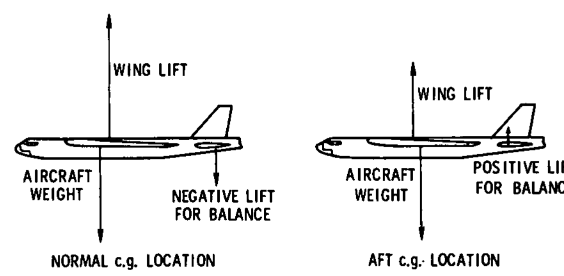

Figure 3. Effects of center-of-gravity (c.g.) location on aircraft lift distribution

Aircraft are carefully designed so that the center of gravity is located properly with respect to the wing and the tail to provide satisfactory stability characteristics. With this conventional arrangement, cruise drag is generally not minimal for reasons illustrated by the diagram on the left-hand portion of figure 3. A normal center-of-gravity location generally requires a negative-lift loading on the tail for balance. This negative lift must be compensated by additional wing lift over and above that required to support the aircraft weight. Lift-induced drag is developed independent of whether the lift is positive or negative. If the center of gravity can be moved aft so that the balancing lift required on the tail is zero or slightly positive, the total required lift is less and the lift-induced drag is decreased. Such an aft movement is possible by use of an active-control system for stability augmentation. The amount of drag saving depends on details of the aircraft configuration and operating conditions.

Although calculations are not directly available for a low-wing-loading configurations, estimates have been made for a higher wing loading aircraft which should be generally applicable. The estimates are for a B-52 aircraft cruising at high altitude, as a representative situation. Reconfiguring the aircraft to shift the center of gravity to the optimal location for cruise drag (about 14 percent of the mean aerodynamic chord aft of the normal location) results in a 2.5 percent saving in fuel.

Cruise drag can also be reduced a modest amount by reducing tail area. To provide sufficient control authority during landing, the size of both the horizontal and vertical tail tend to be significantly larger for STOL aircraft than for CTOL aircraft. With an augmented stability system, size of the empennage could be reduced to that required to provide only trim plus maneuver requirements.

Powered-Lift STOL/RTOL Application

The problems pertaining to stability and control of powered-lift aircraft can be illustrated by figure 4. At the top of the figure is presented the contributions of conventional lift and of powered lift as a function of airspeed. During cruise the entire weight of the aircraft is supported by conventional aerodynamic lift produced by the aerodynamic surfaces moving through the air. In the landing approach as the forward velocity reduces to the degree that the amount of conventional lift generated is not sufficient, powered lift (effected by the engine exhaust flow interacting with the aerodynamic surfaces) is applied and gradually increased until touchdown speed is achieved. More complete details of powered lift and powered-lift configurations can be obtained from references 6 and 7.

Aerodynamic stability and control is normally achieved by use of aerodynamic surfaces which utilize conventional lift to provide effectiveness. With normal size surfaces, stability and control are adequate during cruise, but, as indicated on the lower portion of figure 4, become marginal as the speed is decreased during landing approach. Improvements can be achieved by increasing the size of the surfaces and/or by interconnecting the controls with the propulsion system as was done for the Bregeut 941 (ref. 8). These types of improvements can introduce other complications, however, such as increased gust sensitivity during cruise. An alternate and attractive approach would be use of an active-control system to provide augmented stability and control.

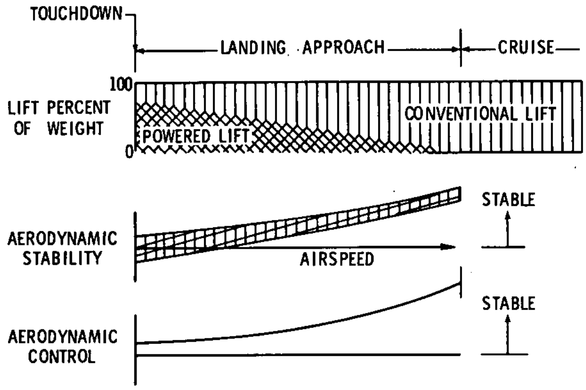

Figure 4. Characteristics during landing approach of powered-lift aircraft equipped with conventional-lift stability and control surfaces

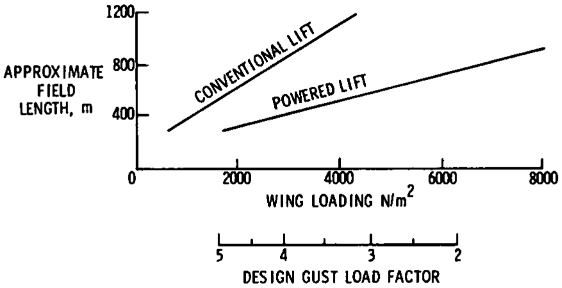

Figure 5. Wing loading and typical design gust load factor characteristics associated with STOL/RTOL aircraft which employ either conventional lift or powered lift

Technology Development Requirements

A very adequate summary, presented in reference 1, of the technology development requirements needed to implement the use of active controls for augmented stability and control is quoted below:

> "Technology improvements in several areas must be accomplished before an AS system will be utilized for anything except limited application. Design criteria for such systems need to be developed, to fully address on a mission basis the questions of redundancy and reliability. The "Hard SAS" criteria of the U.S. SST, i.e., no failures expected in the total lifetime of an SST fleet (ref. 9) is realistic, though perhaps difficult to achieve. To fully utilize the CCV preliminary design philosophy, methods of developing accurate, rapidly iterative aeroelastic mathematical models must be improved in order to assess the effects and payoffs of the AS system. Detailed, accurate weights estimations of the AS system during preliminary design become increasingly significant in that the payoff analyses are critically dependent on them. There is an additional requirement to develop improved methods for predicting non-linear aerodynamic characteristics for elastic vehicles at high angles of attack and yaw, because nonlinearities complicate the design of the control system. The use of aeroelastic wind tunnel models with active control systems is a tool which offers considerable power in CCV airplane design, but one which needs development to a level commensurate with the risks being assumed."

DYNAMIC LOAD ALLEVIATION

Dynamic load alleviation can be applied to a variety of situations affecting the structural integrity of aircraft. Examples include loadings from gusts, aeroelastic responses, and ground taxi over rough surfaces. Active-controls systems have the potential for alleviating such loadings for all types of aircraft. The use of active controls for gust-load alleviation has been the subject of a recent study. The remainder of this section is limited to this particular application.

Gust Sensitivity of Aircraft Configurations

The gust sensitivity of an aircraft is dependent on wing loading. Upper-limit boundaries of both conventional lift and powered lift available for use during landing approach are presented in figure 5 as a function of field length and wing loading. For example, an aircraft designed for a 600-meter field length can have a wing loading no greater than about 2,000 N/m^2 (41 lb/ft^2) if employing a conventional-lift system, and about 5,000 N/m^2 (105 lb/ft^2) if employing a powered-lift system. The design gust-load factor (a multiplier of steady-state loading to account for loading due to gusts) for a given aircraft can be related to aircraft configuration and to operating envelope conditions. A design gust-load factor scale for a typical transport aircraft situation is shown on figure 5 below the wing loading scale. For the same example design field length (600 meter) used above, the gust-load factor is approximately 4.7 for a conventional-lift aircraft and approximately 2.8 for a powered-lift aircraft. The aircraft wing structure, which must be designed to accommodate a variety of imposed loadings (e.g., steady-state aerodynamics, maneuvers, gusts, etc.) probably would not be greatly influenced by a gust-load factor value of 2.8. The higher value of 4.7 would, however, likely result in additional strength requirements which could lead to a significant increase in structural weight.

Gust-Load Alleviation Study

Benefits to be gained in structural weight saving by use of active controls for gust-load alleviation were recently investigated in a design study reported in reference 10. Short-haul aircraft sized for 150 passengers were designed for three STOL/RTOL field-length conditions. The variation in aircraft design gross weight without and with gust-load alleviation (to a value of 2.5 for the gust-load factor) for conventional-lift aircraft is presented in figure 6. Significant weight savings are indicated for gust-load alleviation, particularly at the shortest design field length where weight reduction is 43 percent. For this configuration, gust-load alleviation would be required at all speeds above 180 knots equivalent airspeed. About 4° deflection of an 18-percent-chord full-span trailing edge flap would be required to reduce the gust-load factor from its maximum value (4.7) to the design value of 2.5.

The studies also considered aircraft with powered lift. Because of relatively high wing loading, savings in weight from gust-load alleviation was indicated to be relatively modest, ranging from a few percent for the longest field length to 11 percent for the shortest field length. The results generally indicated that through use of an active-control system, low-wing-loading aircraft with conventional lift can be somewhat lighter, quieter, and more economical than aircraft using the externally blown flap, powered-lift concept. Without gust-load alleviation, the conventional-lift aircraft would be heavier than powered-lift aircraft for field lengths shorter than about 750 meters.

If gust-load alleviation is employed to reduce structural weight, the system must be considered safety-of-flight critical. With partial failure of the system, the aircraft flight envelope would have to be substantially limited. A redundant active-control system is considered mandatory for civil transport utilization. In general, the requirements for technology development to support the use of gust-load alleviation are similar to those presented earlier for augmented stability.

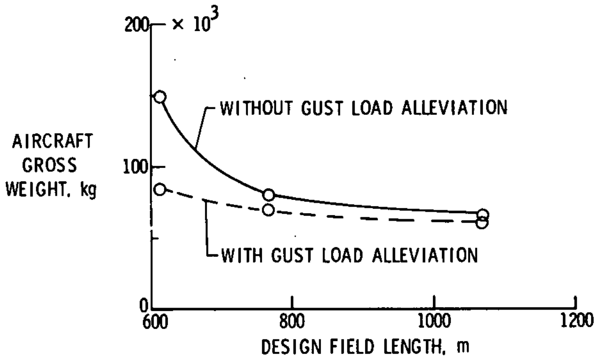

Figure 6. Variation of design gross weight with field length of 150-passenger transport conventional-lift aircraft without and with an active-control gust-load alleviation system

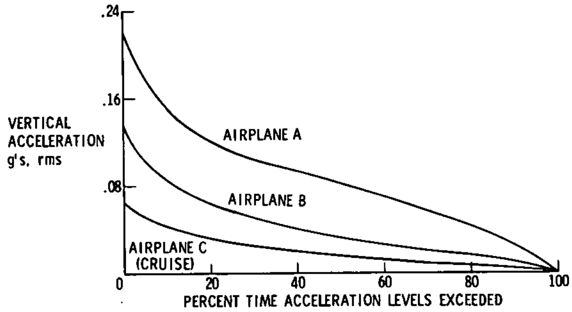

Figure 7. Example differences in aircraft motion characteristics for vertical acceleration

RIDE SMOOTHING

Large differences in ride smoothness can exist for transport aircraft as illustrated in figure 7, where levels of vertical acceleration are presented for three vehicles as a function of percent time that acceleration levels are exceeded. The data shown for airplane A and airplane B are averaged values of measurements obtained in the passenger compartment about every 2 minutes between takeoff and landing during many flights onboard scheduled passenger service in the Eastern Seaboard region of the United States.

For airplane C, data from which averaged values were obtained are more limited, but are representative of cruise flight conditions for present-day large jet transports. Table 3 lists approximate values of several factors believed to influence the levels of vertical response. Acceleration levels for airplane C are favorably minimized by high wing loading, by wing sweep, by low tail volume coefficient, and by high cruise altitude. For the two smaller aircraft which have somewhat similar properties, the vertical acceleration levels for airplane B are significantly lower than for airplane A, probably because of the higher cruise altitude of these studies. The question arises as to how to interpret data such as presented in figure 7 in terms of passenger satisfaction. Before design goals can be established for application of active controls to ride smoothing, information is needed concerning the influence of ride comfort on traveler acceptance and use of vehicles.

Ride Comfort and Traveler Acceptance

Subjective response to motion has been studied in some detail to establish tolerance-limit criteria (e.g., ability to perform a specific task under adverse environmental conditions, exposure-time limit allowable in high-vibration environment, etc.). Such criteria are presented in reference 11. In the area of ride comfort, which involves much lower magnitude motions, meaningful information is limited and criteria are not well established. To fill a need in this area, NASA has underway considerable research (described in ref. 12) concerning ride quality and traveler satisfaction. The research includes studies utilizing field measurements to identify important factors (e.g., motion, vibration, etc.) and to develop approximate criteria. Laboratory and research aircraft experiments under closely controlled conditions are also underway to gain a good understanding of all factors involved and to establish more quantifiable criteria. Much of the field measurement effort has been carried out as part of a traveler acceptance study by the University of Virginia under NASA grant. The study, with some of the findings, is described in reference 13. Information which addresses motion environment and passenger response is used in the following paragraphs to illustrate how evaluation can be made of ride quality. The study also provided the data for figure 7.

TABLE 3

AIRCRAFT PROPERTIES AFFECTING RIDE QUALITY

Airplane		A	B	C
Passengers		20	29	219
Maximum takeoff weight	N (lb)	55,600 (12,500)	104,000 (23,400)	1,490,000 (334,000)
Maximum wing loading	N/m^2 (lb/ft^2)	1,400 (30)	1,920 (40)	5,270 (110)
Wing sweep angle	Degrees	0	0	35
Horizontal tail volume coeff.		0.94	1.02	0.63
Cruise altitude of study	m (ft)	900 (3,000)	1,800 (6,000)	9,000 (30,000)

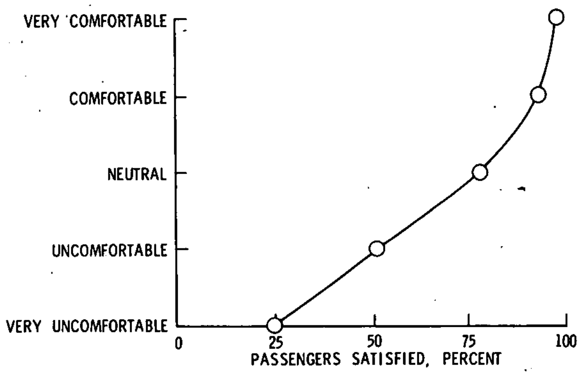

Figure 8. Relationship between comfort and passenger satisfaction

Ride-comfort criteria.- During flights on air carriers and research aircraft, simultaneous recordings were made of reactions of test subjects as well as of aircraft motion environment in all six degrees of freedom. In addition, passengers were surveyed at the end of each trip to obtain their overall assessment of the ride plus an indication of their satisfaction (expressed as willingness to buy another ticket on the same aircraft and to experience the same ride). Even after a trip rated as very uncomfortable, 25 percent of the passengers indicated satisfaction (see fig. 8). As data accumulate from field and laboratory studies on ride environment, subjective reactions, and passenger opinions, correlations are being made and criteria are being developed (see ref. 14 for example). One form of presenting criteria is illustrated in figure 9 where percent of passengers satisfied is shown in terms of specific environmental factors. The two factors used in this figure are lateral and vertical acceleration, both of which affect passenger satisfaction. In addition, a number of other factors have been identified as important, such as rolling motion, terminal-area maneuvers, visual cues, cabin temperature, and seat size.

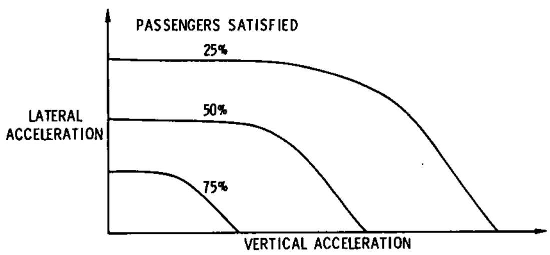

Figure 9. Example form for presenting ride-quality acceptance criteria

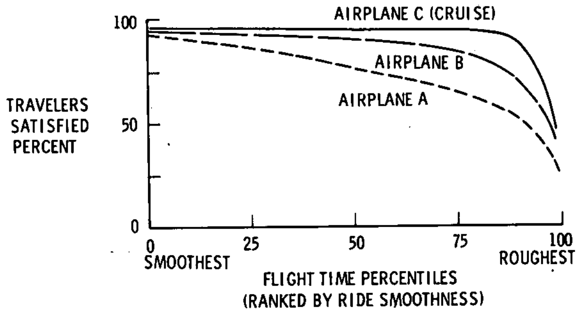

Figure 10. Estimated traveler satisfaction of three aircraft from evaluation of vertical and lateral acceleration characteristics

Aircraft evaluation.- To illustrate the significance of ride criteria to evaluate aircraft, a preliminary estimate of traveler satisfaction has been made for the three aircraft discussed earlier. In this estimate (fig. 10), criteria of the form shown on figure 9 have been applied to measured vertical and lateral acceleration data. Satisfaction is expressed in terms of percent travelers satisfied as a function of flight time percentile ranked by ride smoothness, with the smoothest periods of flight occurring at 0 percentile, and the roughest periods at 100 percentile. The term "traveler" is used rather than "passenger" to point out that about 5 percent of all travelers will not be satisfied in riding an aircraft no matter how smooth the ride may be. For this reason, airplane C, which is considered to have excellent ride characteristics when cruising in smooth air, is satisfactory under the best of conditions to only 95 percent of all travelers. For this aircraft, the ride quality continues to be quite favorable to the 90-percentile time point where about 90 percent of all travelers are satisfied. In contrast, airplane A in its flight situation is satisfactory to only 50 percent of all travelers at the 90-percentile time point and to slightly less than 80 percent of all travelers at the 50-percentile time point.

Other considerations.- The trends shown in figure 10 indicate that, in terms of traveler satisfaction, the relative improvement possible by addition of an active-control system is more modest for airplane C than for either airplane A or B. Decision to incorporate a ride-smoothing system into an aircraft involves a number of other considerations. Typical questions that must be considered are:

What is the ride-environment conditioning of the passengers who will be using the aircraft?

For residents in undeveloped regions, the ride of a DHC-6 could be a big improvement over the ride of an off-road mode of transportation, while for residents of a metropolitan area, seasoned by smooth rides on long-range, heavy aircraft, equally good rides could be expected of smaller short-haul aircraft used by the connecting feeder lines.

Will increase in revenue from additional travelers gained by ride smoothing offset the increased costs of the active-control system?

Carriers serving low-density markets may generate little, if any, additional business by ride smoothing, whereas air carriers serving high-density markets may generate considerable extra revenue by attracting customers from competitors whose aircraft have a poorer ride.

Is there a public responsibility to make the ride acceptable to the greatest possible number of travelers?

Perhaps carriers serving the public should be required to conform to minimum comfort standards as well as to requirements concerning safety or to the amount of service given cities on their route structure.

Answers to the above questions will depend to a significant degree on detailed information on the active-control systems required for ride smoothing.

Ride-Smoothing System Feasibility Study

Concurrent with subjective studies of ride quality, a feasibility study was carried out of an active-control system for the de Havilland DHC-6 aircraft for NASA by the Wichita Division of The Boeing Company, assisted by de Havilland Aircraft of Canada, Limited. The objective was to examine the feasibility of developing and certificating a ride-smoothing-control system for a typical small feeder line aircraft known to have a ride environment not equal to that found on larger, high-wing-loading jet transports. The DHC-6 was selected for study not only because it has a low wing loading and is oftentimes operated extensively in low altitude turbulence, but also because it is the only STOL vehicle presently certificated and extensively used by air carriers in the United States. Its capability to carry out steep-angle climbouts and descents and to perform short-radius, terminal-area maneuvers makes suitable the study of ride-quality situations reasonably typical of those which may be encountered by subsequent advanced STOL/RTOL transports. An example application of this nature is the Canadian STOL Demonstration Program between Ottawa and Montreal, where modified DHC-6 aircraft are being used to obtain passenger acceptance data as well as to study and refine systems operations in advance of introduction of the new and larger DHC-7 STOL transport aircraft now being built for such service.

Description of system.- Quite a bit of information having general application to ride-smoothing systems was obtained from the feasibility study. Highlights of this general information are presented herein; detailed description of the study and findings are presented in reference 15. Investigation of active controls was limited to only vertical and lateral ride smoothing, as preliminary study indicated response to turbulence to be acceptably low for the other degrees of freedom. Aerodynamic surfaces considered in the system are shown in figure 11 and include portions of the existing ailerons, elevators, and rudder as well as all-new spoilers. Consideration of additional surfaces could not be accommodated within the scope of the study. Ride control of each degree of freedom was treated independently. Simplified block diagrams showing feedback loops are presented in figure 12 for the vertical control system and in figure 13 for the lateral control system. Details such as transfer functions are not shown. System effectiveness was determined as reduction of acceleration response to a random turbulence intensity with an exceedance probability of 0.01 which was established as a gust velocity of 2.1 meters per second (rms) for the design flight conditions.

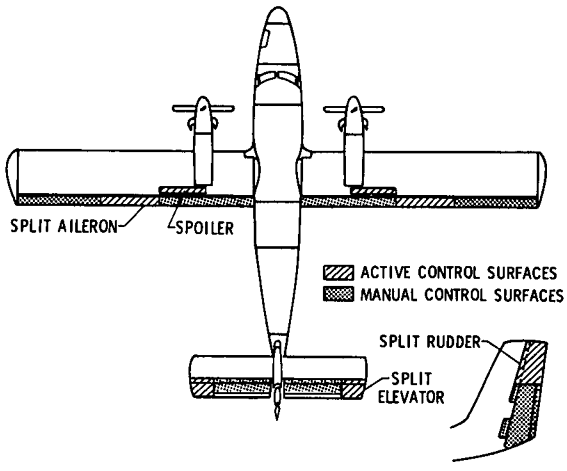

Figure 11. Active-control surfaces studied on DHC-6 aircraft

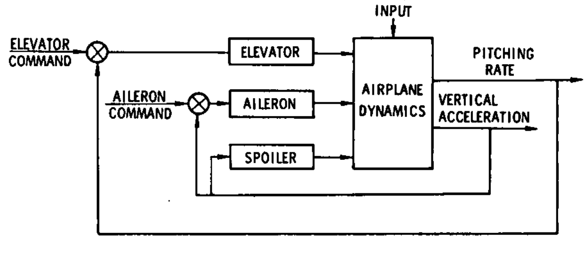

Figure 12. Block diagram of vertical ride-control system from DHC-6 study

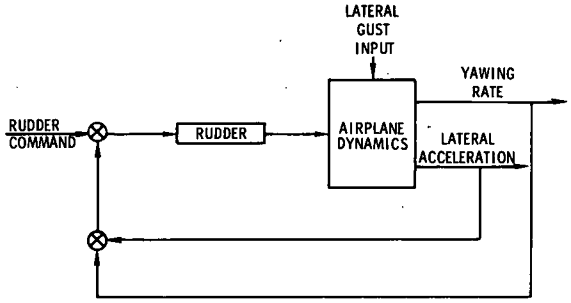

Figure 13. Block diagram of lateral ride-control system from DHC-6 study

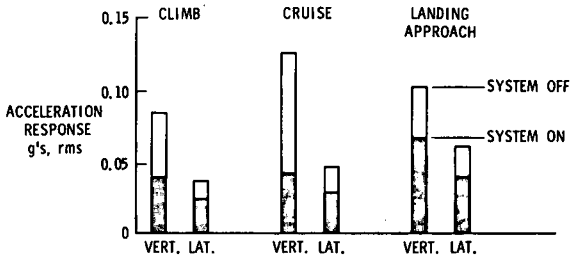

Figure 14. Effectiveness of DHC-6 ride-control system in terms of aft cabin response to 2.1 m/sec gusts

Ride-control effectiveness.- In the area of effectiveness, the most important finding was the requirement for relatively large direct-lift and direct-side-force surfaces located near the airplane center of gravity. As shown by the bar graphs of figure 14, significant reductions in vertical acceleration response were obtained with wing flaps retracted during both climb and cruise conditions. The elevator surfaces contributed only a modest amount to this reduction. For the landing approach condition, new spoilers had to be employed to even achieve the less-than-adequate reductions shown. Design techniques need to be developed for integrating large, direct-lift surfaces for ride smoothing into wing-flap systems. Use of rudder surfaces for reducing lateral response was somewhat effective in the aft section of the passenger cabin, but was ineffective ahead of the cabin midpoint. Efficient (high side-force/drag) direct side-force surface configurations need to be provided at a fore-and-aft location near the airplane center of gravity.

Effects on aircraft properties.- In this area, a ride-smoothing system can be designed which is satisfactory. Considerable attention must be given, however, to various potential problems in order that the system be tailored to minimize adverse effects. In the feasibility study, problems which had to be resolved involved the aircraft low-frequency longitudinal mode, the very-low-frequency phugoid mode, the Dutch-roll mode, and the lateral-directional spiral mode. A detailed control-system synthesis and performance analysis are required to examine various tradeoffs. During the study, problems also had to be resolved in aircraft handling qualities such as one where adding the active-control system caused a loss of effectiveness of the elevator to relatively sharp inputs. In this case, satisfactory short-period handling quality was achieved by introducing a crossfeed signal to the system to initially cancel the ride-control signal which opposed the acceleration, and then to wash out at the same rate as the ride-control signal. Use of groundbased simulators is appropriate to study and help resolve handling problems.

System reliability.- No major problems in reliability are anticipated for the ride-smoothing system. Since use of the system is not critical to the well being of the aircraft, the system can be deactivated if malfunctions occur. The main concern involves transient problems which could arise at the time of any malfunction. The worst problem envisioned would be hard-over deflection of an aerodynamic surface used in the active-control system. If sufficient authority is provided by the aircraft control system to control vehicle motions caused by such a deflection, safety can be maintained. Such authority would be a reasonable requirement for system certification. A fail-soft design control system, such as devised in the feasibility study, can also be incorporated for additional protection. The particular system studied contained dual-signal channels with two stages of monitoring between channels for failure detection. An unfavorable comparison of channel signals would switch off the ride-control signals.

System hardware and maintenance.- Ride-smoothing hardware requirements are not considered to tax the present state of technology. Appropriate sensors, electronic elements, servosubsystems, and actuators are in production. The size and capacities of these components are not necessarily matched to detailed requirements, and modifications of existing designs may be required to obtain appropriately tailored articles. Aerodynamic requirements do require innovation, as discussed earlier, to develop configurations to efficiently produce aerodynamic forces through the center of gravity in both vertical and lateral directions.

Weight and power demands of a ride-smoothing system should not seriously burden the aircraft. Findings of the feasibility study indicated the total additional weight would amount to less than 2 percent of the aircraft gross weight. Additional power requirements of the system would amount to no more than 0.3 percent of the aircraft total engine power. Requirements for larger aircraft would not be expected to exceed these percentage values. Only a small additional volume is needed, but volume requirements in local regions near aerodynamic control surfaces may require special consideration, particularly if an existing aircraft is being retrofitted with a ride-smoothing system.

Specific maintenance information is lacking until a ride-smoothing system is put into service. Considerable experience has been obtained, however, on a closely related active-control fatigue-reduction system, described in reference 16, which was applied to the United States Air Force B-52G and B-52H fleet of 280 aircraft. For this application, system performance and maintenance experience have been excellent and well within guideline limits. Since an active-control ride-smoothing system is essentially a state-of-the-art system competitive with control systems used on modern transport aircraft, maintenance should be similar to that required for current control systems.

Implementation costs and time.- Cost information is lacking because no detailed cost analysis has been carried out. Based on the findings presented above, system development and certification will require considerable effort which will be somewhat independent of aircraft size. Where the system is incorporated into the initial design of an all-new aircraft, the additional costs estimated for the system design through prototype flight tests and certification could range from 2 to 5 percent of the total costs. The additional cost would be expected to be higher if a system were to be designed and retrofitted into an existing vehicle. These higher costs result because of the probability of significant modification, requalification, and retesting of existing systems and structures. Estimated production costs for the system in terms of aircraft production cost could range from about 1 percent for large jumbo transports to as much as 4 or 5 percent for very small transports. Ride smoothing may be included as a feature of a multipurpose active-control system which performs other functions as well, such as gust-load alleviation. Design and checkout of an appropriate multipurpose system would require considerable effort, possibly greater than the sum of efforts required for individual systems.

Little, if any, additional time would be needed if the decision to proceed is made at the beginning of an all-new aircraft project. For retrofit of a ride-smoothing system into an existing aircraft, the total time required is estimated to range between 2 and 3 years.

Technology development requirements.- No significant requirements for technology development have been identified for active-control ride-smoothing systems. Such application is regarded as not being critical to the safety of flight because failure of a properly designed system merely reverts the airplane back to a rougher ride for that period of time when the system is off. Ride-smoothing systems are considered ready for production commitment for those applications where they can be shown to pay off.

REFERENCES

1. Holloway, Richard B.: Introduction of CCV Technology into Airplane Design. Paper presented at the AGARD Symposium on Aircraft Design Integration and Optimization, October 1973.

2. Joint DOT-NASA Civil Aviation Research and Development Policy Study. 1971, DOT TST-10-4, NASA SP-265.

3. Reeder, John P.; Taylor, Robert T.; and Walsh, Thomas M.: New Design and Operating Techniques for Improved Terminal Area Compatibility. Paper presented at SAE Air Transportation Meeting, April 30-May 2, 1974.

4. Tapscott, Robert J.: Rotorcraft Applications and Technology. 1971, NASA SP-292.

5. Kelley, James R.; Niessen, Frank R.; Thibodeaux, Jerry J.; Yenni, Kenneth R.; and Garren, John F., Jr.: Flight Investigation of Manual and Automatic VTOL Decelerating Instrument Approaches and Landings. 1974, NASA TN D-7524.

6. Wick, Bradford H.; and Kuhn, Richard E.: Turbofan STOL Research at NASA. Astronautics and Aeronautics, vol. 9, no. 5, May 1971, pp. 32-50.

7. STOL Technology. Held at Ames Research Center, Moffett Field, California. 1972, NASA SP-320.

8. Marks, M. D.; and Carpenter, D. O.: Low Speed Handling Characteristics of the STOL Aircraft. Paper presented at the 7th Annual Meeting and Technical Display, AIAA, October 1970.

9. Tomlinson, D. O.: Problems and Solutions Related to the Design of a Control Augmentation System for a Longitudinally Unstable Supersonic Transport. 1971, AIAA Paper No. 71-785.

10. Morris, R. L.; Hanke, D. R.; Pasley, L. H.; and Rohling, W. J.: The Influence of Wing Loading on Turbofan Powered STOL Transports With and Without Externally Blown Flaps. 1973, NASA CR-2320.

11. Guide for the Evaluation of Human Exposure to Whole-Body Vibration. 1972, Draft International Standard ISO/DIS 2631.

12. Symposium on Vehicle Ride Quality. Held at Langley Research Center, Hampton, Virginia. 1972, NASA TM X-2620.

13. Kuhlthau, A. R.; and Jacobson, I. D.: Analysis of Passenger Acceptance of Commercial Flights Having Characteristics Similar to STOL. CASI Journal, vol. 19, no. 8, October 1973, pp. 405-409.

14. Stone, Ralph W., Jr.: An Elementary Psychophysical Model to Predict Ride Comfort in the Combined Stress of Multiple Degrees of Freedom. Paper presented at the AGARD Conference on Vibration and Combined Stresses in Advanced Systems. April 1974, Preprint No. 145, pp. B22-1-B22-7.

15. Dodson, R. O.; and Gordon, D. K.: STOL Ride Control Feasibility Study. 1973, NASA CR-2276.

16. Thompson, Glenn O.; and Kass, G. J.: Active Flutter Suppression - An Emerging Technology. AIAA Journal of Aircraft, vol. 9, no. 3, March 1972, pp. 230-235.

TRANSPORT AIRCRAFT WITH RELAXED / NEGATIVE LONGITUDINAL STABILITY - RESULTS OF A DESIGN STUDY +)

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SUMMARY

Application of active longitudinal control on transport aircraft with relaxed / negative longitudinal stability has been studied.

Using two aircraft of different configuration as baseline designs, versions incorporating active longitudinal control were derived. Configuration changes were studied, varying tail size and center of gravity position parametrically. Based upon the requirement for handling qualities equivalent to the baseline designs, optimum control laws were derived. Controllability and stability were checked by simulating various gust cases. Limits for tail size and CG-position were derived. Wing size was changed where required to hold performance unchanged. Structural and fuel weight changes were calculated and the configuration, within the geometrical and controllability limits, giving the highest payload increase, was selected. Sensitivity of payload benefit to performance specification was checked by parametric variations.

It was found that payload benefit depends upon configuration to a high degree. Best payload benefit will be achieved for high wing, T-tail STOL aircraft using large trailing edge flaps. Payload increase may be up to 15 % for such aircraft.

1. Introduction

In the past, active control has been studied mainly for fighter aircraft to give the aircraft the decisive edge in performance and manœuvrability over its enemy. For transport aircraft, economics is of prime importance for commercial and military service. Therefore, active control for transport aircraft must be studied under the aspect of economic improvements.

Transport aircraft have always been designed to exhibit natural longitudinal stability. The horizontal tailplane has been sized so that the aircraft under all sorts of disturbances tends to go back to the trimmed angle of attack. This requires the tailplane to be of sufficient size and the center of gravity (CG) to be forward of the neutral point. In consequence, the tailplane carries less lift than would correspond to its area, tailplane lift even being negative under many flight conditions. Thus more lift must be produced by the wing, with higher induced drag in consequence.

Active longitudinal control (ALC) can remedy this situation. When stability is provided by an automatic control system, natural longitudinal stability can be relaxed or even become negative. In consequence, tailplane size and CG-position can be selected within wider limits, to achieve minimum structural weight and optimum lift distribution.

Figure 1 outlines the beneficial consequences when artificial longitudinal stabilization is applied: The tailplane must be sized for controllability only, its size can be reduced and, therefore, its weight, resulting in a payload benefit. As the CG can be moved aft, the horizontal tailplane provides positive lift, less lift is required on the wing. Therefore, less induced drag will be produced, less fuel will be burnt, thus improving economics. Payload again can be increased. In addition, if the wing is now required to produce less lift in a critical wing sizing flight case (e. g. landing of a STOL aircraft), wing size may be reduced, hence wing weight, and payload increases again.

As this beneficial mechanism might offer substantial economic benefits for transport aircraft, a study was initiated to quantify the benefits to be achieved by active longitudinal control (ALC) on transport aircraft with relaxed / negative longitudinal stability.

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2. Purpose and Methodology of Study

The study was undertaken to analyse in detail the consquences of ALC upon configuration, weights, performance and economics of transport type aircraft. It was felt that a generalized study would produce results faster and at lower cost, but might be misleading as detail problems would be overlooked. It was, therefore, decided to do a preliminary design study, using specific well-known aircraft as a basis from which the ALC version was to be derived. To ensure comparability as far as possible, the following ground rules were established for the study:

- Baseline aircraft are assumed to be perfectly acceptable in every respect, therefore:
- Only changes necessary for ALC introduction must be made.
- All performance characteristics shall be constant.
- Handling characteristics shall be constant.

Take-off weight was held constant to avoid the necessity of "rubberizing" the engines.

Figure 2 shows the flow diagram of the study. Baseline aircraft were selected first. In parametric form, the consequences of varying tailplane size and CG-position upon aircraft configuration were studied. Undercarriage changes were found necessary, and CG-range was to be reconsidered. According to the requirement of unchanged controllability, limits were established for combination of CG-position and tailplane size. For several combinations within this field aerodynamic data were prepared. Optimum control laws were established, and their effectivity was checked by simulation of various gust cases. Performance was checked and the configuration changed where indicated. Weights were analysed, and the configuration giving the highest payload increase was selected and compared to the baseline aircraft. To evaluate the sensitivity of the benefit achieved against basic airplane characteristics, a parametric variation was done.

This paper follows the flow of the study as indicated in figure 2.

3. Baseline Aircraft

Two baseline aircraft were used for the study. They were selected because they were thought to be typical for two important categories of future transport aircraft. In addition, their configuration differs markedly and results were expected to indicate, whether configuration is an important parameter with respect to ALC benefit. Main data for the baseline aircraft can be seen from figures 3 and 4.

3.1 Baseline Aircraft 1: Airbus A 300 B2

The Airbus A 300 B2, as shown in figure 3, is a twin-engine, wide-body medium-haul airliner, developed by the European consortium "Airbus Industrie". It has a low wing, with engines slung beneath it, a conventional high lift system and a low horizontal tail.
Note: Controllability and performance data cited in this paper are based on preliminary design office calculations and, therefore, may differ slightly from performance manual data.

3.2 Baseline Aircraft 2: CON 30

CON 30 is a STOL transport. It is the result of a preliminary design study on high performance STOL transport aircraft. CON 30 has a swept high wing, as can be seen from figure 4. It uses a complex mechanical multi-slotted flap system which, when extended, enlarges the effective wing area even more than today's usual fowler flap systems. Therefore, large nose-down moments cannot be avoided, and must be trimmed by the horizontal tailplane. The empennage is of T-configuration.

As the aircraft is designed for both short field performance (2000 ft) and high cruise speed ($M > 0,8$), wing loading is rather low at 292 kg/m^2 and thrust/weight ratio is very high at 0,6. Payload fraction in consequence is low.

4. ALC Aircraft Configuration

4.1 CG-position, tailplane area, wing position

Application of active longitudinal control gives a wider choice of combinations of CG-position and horizontal tailplane area. Smaller tailplane areas and CG-positions farther back are aimed for. But these aims contradict each other: When the tailplane area is reduced and thus its weight is reduced, too, aircraft CG moves forward. Correction of this shift must be effected in addition to the CG-shift originally intended.

Aircraft CG can be shifted backward relative to the wing by shifting systems and components backward, but there is little freedom to do this, as systems normally are centralized to a certain degree in the aft fuselage. Only by shifting the fuselage in toto, can the desired overall shift be achieved. Speaking practically, the wing must be shifted forward to shift the CG backward relative to it. But, if the wing is shifted forward, its own mass as well as the mass of the engines, if arranged beneath the wing, are shifted at the same time, and aircraft CG tends to move forward. In consequence, the wing must be shifted still further forward.

Because of the combined effects discussed, the wing must typically be shifted nearly twice as much as the CG-shift which is to be effected. This fact can be read from figure 5, where results of a parametric analysis are shown: to achieve CG-shifts of 22 to 24 % backward in relation to the wing, the wing must be shifted forward by 40 to 43 % MAC. In the example shown tailplane area is reduced to 60 % of the original, which may be considered typical.

As can be seen from figure 5, CG-range is actually shifted forward in relation to the fuselage. This would be welcome for an aircraft configuration which tends to be tail-heavy, with too much of the cabin in front of the CG, as in a rear-engines aircraft (but in this case, wing shift forward will be less if engine position is fixed relative to the fuselage, and the overall effect may be the other way round). In a case where the aircraft originally balanced well, as in the case of baseline aircraft A 300 B2, forward shift of the CG-range relative to the fuselage, that is to the cabin, is not welcome at all. Now loading of passengers will tend to make the aircraft tail-heavy. To account for this, a slightly larger CG-range will be necessary as indicated in figure 6, showing the CG-vs-weight diagram both for the baseline configuration A 300 B2 and a typical ALC-version. CG-range for the ALC-version is 21 % MAC against 20 % MAC for the baseline aircraft.

4.2 Undercarriage

To ensure sufficient nose wheel loading in the critical maximum aft CG-condition, and to enable the aircraft to be rotated in the critical maximum forward CG-condition, the main landing gear must be positioned within narrow limits coupled to the CG-range. On the other hand, the main landing gear often is coupled structurally to the wing, e. g. it is attached to the rear spar in a swept low wing aircraft, or there is a main frame carrying the landing gear loads through the fuselage to the wing box in an unswept high wing configuration.

When ALC is applied and CG-range is shifted backward relative to the wing, the main landing gear must also be shifted backward relative to the wing. This can upset the original structural coupling with serious consequences. For example, figure 7 shows the main landing gear of baseline aircraft A 300 B2. As the main strut of the landing gear must fit into the wing rear section, the attachment point cannot be shifted backward indefinitely. Position A indicates the farthest backward position feasible. By inclining the main strut, the wheels can be shifted backward a little bit more. The limiting case is reached if the maximum aft CG-position to be dealt with is about 45 % MAC. If CG-positions still further aft are to be provided for, landing gear configuration must be changed basically. The main gear can no longer be retracted sideways into a fuselage well, but must be retracted backward into a special nacelle, extended aft of the wing trailing edge, as indicated in figure 7. This configuration is not commonly used in aircraft of Western origin and, probably, causes a weight penalty large enough to cancel any benefit to be expected from CG-positions so far aft.

The forward shift of the CG-range relative to the fuselage, as discussed above, has further consequences for the undercarriage geometry. Figure 8 a) indicates the problem: The rotation angle is limited by the fuselage aft end, and as it must not be reduced, the landing gear must be made higher. In this case, the track must be widened to ensure the same tilt-over angle. These configuration changes will, certainly, add to the weight of the undercarriage and can even result in geometrical problems. Also for a military transport a high landing gear is always bad as it makes loading and unloading difficult or puts higher demands upon the landing gear "kneeling" system.

5. Tailplane Sizing

5.1 Stability and Control Requirements

The basic requirement "handling qualities must be unchanged by switch-over to ALC" as stated for this study, must be specified in more detail to be handled in actual analysis. It is assumed that the following requirements are sufficient to ensure the necessary longitudinal handling qualities to the degree of accuracy required:

a) Rotation at take-off must be possible in the extreme forward CG-condition.

b) Pull-up manœuvres must be possible with the same angular acceleration, in cruise, take-off and landing as for the baseline aircraft.

c) Dynamic response in the short period angle of attack motion after a step input must be within the same limits as given for the baseline aircraft within the full flight envelope.

Figure 9 gives the short period motion limit for baseline aircraft A 300 B2 and its ALC derivative.

Data for those requirements must be established specifically for each aircraft type, because it is assumed that handling qualities, though different for different baseline aircraft, are acceptable for the respective tasks.

5.2 Tailplane Size vs CG-Position

The interdependence between tailplane size and CG-position can best be shown in a dimensionless diagram as given in figure 10, which uses the CON 30 STOL transport as an example. The forward limit is caused by the take-off rotation requirement, the aft limit by the nose down manœuvre requirement in the final approach configuration. For the baseline aircraft, the CG-range must be ahead of the aircraft neutral point, so that the aircraft is longitudinally stable, and this fixes the tailplane area for a specified CG-range as indicated. But if active control is applied, stability may be relaxed or even become negative, and tailplane area can be reduced to the minimum required for controllability. The aircraft would be, naturally, unstable in all the corresponding CG-range. Whether this is the overall optimum configuration cannot be said at this point of the study.

Figure 11 gives the tailplane size vs CG-diagram for the baseline aircraft A 300 B2. Here an additional limit is set by the main landing gear geometry as discussed above. So the minimum tailsize is larger than the minimum required for controllability; but it is substantially smaller than the tailplane of the naturally stable baseline aircraft. In this case, CG-range corresponding to practical minimum tailsize comprises relaxed and negative stability regions.

The limits for controllability and, therefore, the minimum tailsize, too, are very sensitive to configuration changes, control surface efficiencies, and manœuvrability requirements. To give an indication, figure 12 shows the take-off rotation limits for varying aircraft fuselage attitudes at rotation speed and controllability limits for two angular accelerations in nose up/nose down manœuvre.

6. Control System

6.1 Control System Layout

An ALC transport aircraft must be at least as safe as a conventional aircraft. The reliability of the electro-hydraulic control system to be used in an ALC aircraft must, therefore, be at least as high as the reliability of the mechanical-hydraulic system which it is superseding. This requirement can be expressed in terms of probability of catastrophic failure or in terms of number of system-failures to be survived.

Taking baseline aircraft A 300 B2 as an example, this aircraft uses three independent hydraulic systems. The first system is powered by two engine-driven pumps, one mounted on each engine. The second system is powered by one engine-driven pump. The third system is powered by a pump on the other engine, but can be powered by a Ram Air Turbine in case of total engine failure. Independently supplied triplex-servo-jacks power the primary flight controls. This system ensures that two failures of any kind can be survived. Catastrophic failure probability for such a system (triple failure) is in the order of magnitude of 5×10^{-10} per flight hour.

In changing over from the conventional to the ALC-version, the standard rigged cable system, which has a very low failure rate, is replaced by an electrical system containing the automatic control unit, electrical data lines, and electro/hydraulic signal converters. This system must match the original mechanical systems in reliability and this can, indeed, be achieved. A quadruplex electrical system contributes only about $1,6 \times 10^{-10}$ failures per hour to the overall failure rate.

Again, at least two failures of any kind are survivable, but a change of procedure is required: The Ram Air Turbine must already be extended when one engine fails, so that un-interrupted hydraulic power is available in case of a second engine failure, whereas a short interruption in case of an aircraft naturally stable can be tolerated.

6.2 Automatic Control System Dynamics

The automatic control system can be designed using optimum control theory. If the requirements to be fulfilled are given in a form like figure 9, computer aided design proves to be a fast and efficient way of finding the optimum control law. Using the elevator alone to control the aircraft, the requirements of figure 9 can be fulfilled with reasonable deflections, but the dynamic response can be improved by applying power setting changes in addition.

The effectiveness of the automatic control system selected must and can be checked by simulating critical flight conditions. As an example, aircraft behaviour when flying through a discrete gust can be seen from figure 13 for the ALC-version of baseline aircraft A 300 B2. Flight condition is final approach at a speed of 128 kts. The CG is 8 % MAC behind neutral point. The gust has the (1-cos)-form indicated in figure 13 a) with the data shown taken from Dryden for clear air turbulence. As figure 13 b) proves the aircraft behaviour is satisfactory in every case, and elevator deflections required are reasonably small.

Further insight into the aircraft dynamics can be gained from power-spectrum analysis. As an example, figure 14 gives power-spectra for g-load and elevator deflection for the ALC-version of the A 300 B2. Gust power spectrum used is based upon Dryden's work. The data shown do not indicate any serious flaw in the automatic control system.

7. Performance and Weights

Payload benefits can be gained from active longitudinal control because of reduced fuel consumption and reduced Operating Weight Empty.

When the CG is shifted back relative to the wing, the tailplane carries more positive lift and the wing has to produce less lift. Overall induced drag is reduced. This can be seen from figure 15, which includes drag reduction because of reduced wetted tail area. Figure 15 shows a pronounced difference between the two baseline configurations: Whereas drag goes down continuously with CG aft-shift for the baseline aircraft CON 30 there is a definite drag minimum for baseline aircraft A 300 B2. This difference stems from the different horizontal tail positions. The horizontal tail of CON 30 atop the vertical fin is almost unaffected by the wing flow field, but the low tail of baseline aircraft A 300 B2 operates in the wing downwash field and because of the downwash angle the lift vectors at the tail have a drag component which tends to cancel the induced wing drag reduction.

With reduction in horizontal tailplane size, not only the weight of the tailplane is reduced. Vertical tailplane weight goes down also in the case of a T-tail, because of the reduced loads transferred from the horizontal tailplane. In addition, because the CG is moved forward relative to the fuselage, see paragraph 4.1, tailplane arm is slightly increased and vertical tailsize can be reduced accordingly. There will be some weight reduction in the fuselage, too, because of reduced tail loads to carry through. On the other hand the undercarriage and its associated structure will be heavier. A weight reduction for the electro-hydraulic control system against a mechanical-hydraulic system could be claimed, but to be fair, no credit should be given because the same technology could be applied to a naturally stable aircraft.

Figure 16 summarizes the weight benefits achieved by reduced fuel consumption and by reduced empty weight for baseline aircraft CON 30, indicating that weight benefit increases steadily with CG aft-shift and tailplane size reduction. In contrast, for baseline A 300 B2 there is an optimum CG-position, aft of which weight benefit falls off again, as is to be expected from the drag characteristics. See figure 17.

Until now the wing has remained unchanged. For both baseline aircraft wing area is determined by field performance requirements. Lift redistribution by means of ALC influences field performance, and may allow/necessitate wing re-sizing for constant field performance.

In case of baseline aircraft CON 30 landing field length is critical. The further the CG is shifted backward, the more the wing is unloaded, the more the stall speed and, in consequence, the approach speed goes down, the shorter will be the landing run. For constant field length, the wing loading can be increased in proportion to CG backward shift. For the minimum tailsize ALC version, which also has the CG-range farthest aft, the wing area can be reduced by 4 %. At a first sight, this seems to be negligible. But as the wing is the heaviest structural component of the aircraft, the weight benefit achieved is appreciable. Figure 18 gives the total weight benefit achieved by ALC for baseline aircraft CON 30. Comparison with figure 15 shows that wing weight reduction can nearly double the weight benefit achieved for this aircraft.

In contrast, for baseline aircraft A 300 B2 take-off field length is critical. With backward shift of CG the drag component of the tail lift - with the tail operating in the wing downwash field - increases rapidly. The reduced acceleration and climb-out angle tend to off-set the reduced lift-off speed, so that little field length reduction is achieved. For large backward CG-shifts, the overall effect is even detrimental. This is indicated in figure 19, which gives the wing-loading for constant field length vs CG-position. So there is nothing to be gained by re-sizing the wing for the ALC version of A 300 B2.

8. Payload, Economics

Figures 20 and 21 give the tailplane size vs CG-limits as figures 10 and 11, but include lines of constant weight benefit derived from figures 17 and 18. These figures imply variation of the wing area. Of course, one definite wing area must be selected for an actual configuration. As the forward CG-limit is critical with respect to field performance, the corresponding wing area must be selected. In accordance, the weight benefit should be read from figures 20 and 21 for the forward CG-limits of the ALC versions. It can be seen, that the ALC version with the minimum tailplane size gives the highest weight benefit for both baseline aircraft. But this weight benefit is only 4, 5 % of the baseline payload of A 300 B2, whereas the weight benefit is about 15 % of the baseline payload of CON 30.

The ALC versions providing maximum weight benefit are considered as optimum and are used for further comparisons.

The weight benefit achieved can be used to increase the payload and / or the range. Figure 22 indicates an increase in range of about 20 % for the ALC version of baseline aircraft CON 30. Range increase for A 300 B2 is much smaller, of course, as indicated in figure 23. Figures 22 and 23 show an increase in payload equal to the weight benefit discussed above. This, of course, holds true only if the fuselage does not have to be stretched to accommodate the increased payload and if Maximum Zero Fuel weight is high enough. In this case, a relative Direct Operating Cost (e. g. ¢ / ton · mile) reduction will be achieved approximately equal to the relative payload increase, equalling the weight benefit. For constant payload density the fuselage must be stretched, and the payload benefit comes down to about 50 % of the original weight benefit. The Direct Operating Cost reduction will also be proportionately smaller. Note: DOC changes because of aircraft price changes are much smaller than DOC changes because of increased payload.

9. Parametric Sensitivity Study

The marked differences in benefit derived from ALC for the two baseline aircraft can, to a certain degree, be explained by the differences in configuration. The differences in performance must be assumed to cause the rest of the difference. To test the sensitivity of ALC benefit against performance parameters, a parametric study was prepared. Baseline aircraft CON 30 was used, and it was scaled (assuming "rubber engines") to fulfil varying performance requirements at constant payload. The benefit achieved by ALC is then indicated by the take-off weight reduction. Figure 24 gives the range of Take-off Weight reductions achieved. Small reductions only are achieved when design range and cruise speed are low, field length is long; large weight reductions are achieved when the aircraft is designed for long range, high cruise speed, and short field performance. Figure 25 indicates the sensitivity to the parameters named.

In short, the higher the performance required, the lower is the payload fraction of the naturally stable aircraft, and the higher is the absolute and relative benefit which can be achieved by application of ALC.

10. Comparison Baseline - ALC version

Figure 26 summarizes the salient differences between the baseline aircraft and the ALC versions. The tables give main data for comparison, and the plan view of baseline aircraft CON 30 illustrates the configurational changes effected by application of ALC.

11. Conclusions

1. Active longitudinal control in transport aircraft can be realized without any loss in handling qualities and safety.

2. Tailplane size can be reduced substantially - down to about 60 % of the original size.

3. Highest payload benefit is achieved with the configuration of smallest horizontal tail area. The aircraft then will be naturally unstable over at least part of the CG-range.

4. The amount of benefit achieved depends to a high degree upon the aircraft configuration. High wing aircraft benefit more than low wing aircraft. Aircraft with large chord extending trailing edge flaps benefit more than aircraft with modest flap systems. Aircraft with T-tail benefit more than aircraft with low tail.

5. The amount of benefit achieved increases with design range and cruise speed, decreases with design field length.

6. For aircraft of suitable configuration and performance specification, e. g. a high performance STOL transport of high wing, T-tail configuration, payload benefit may be up to 15 %, no fuselage stretch assumed. Range can be increased by up to 20 %.

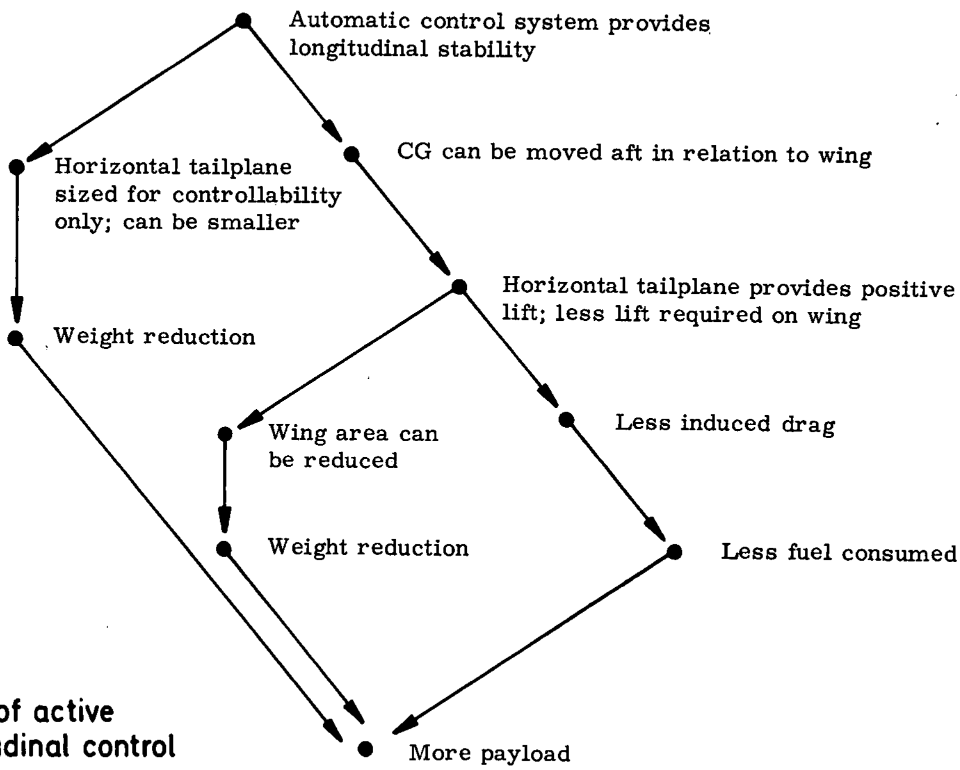


Fig. 1 Logic of active longitudinal control

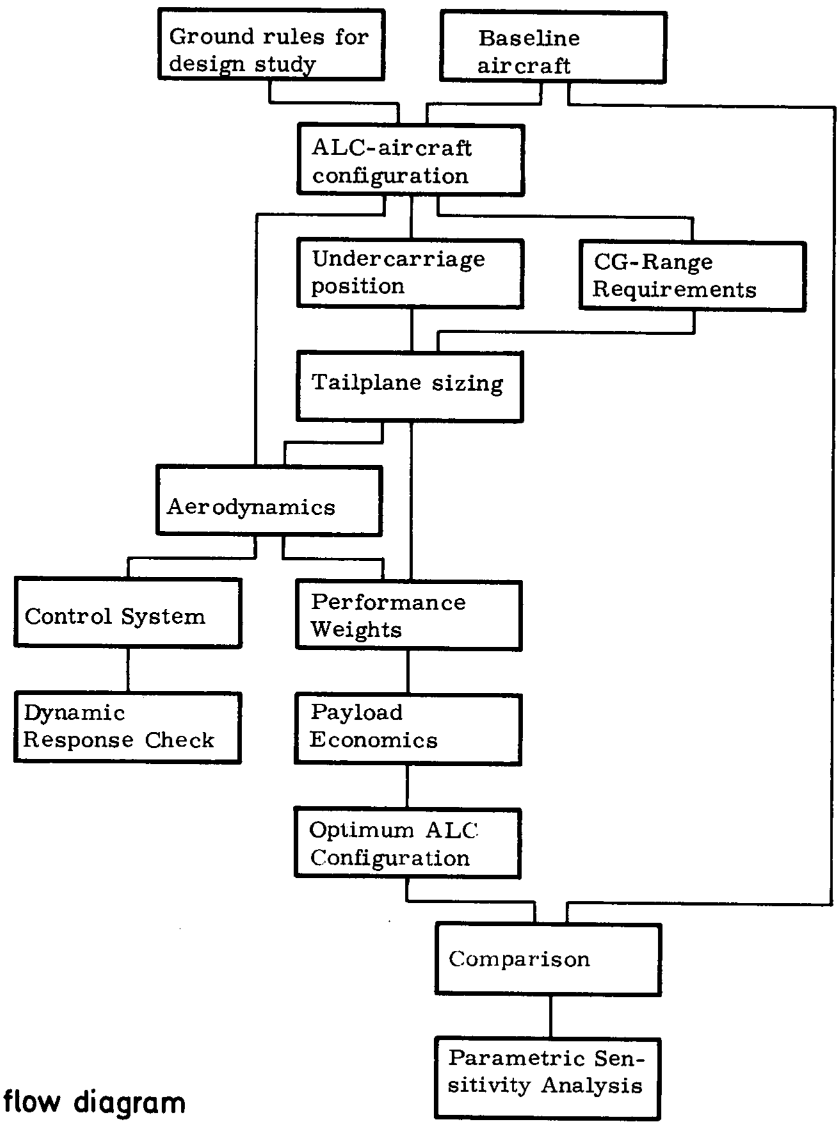
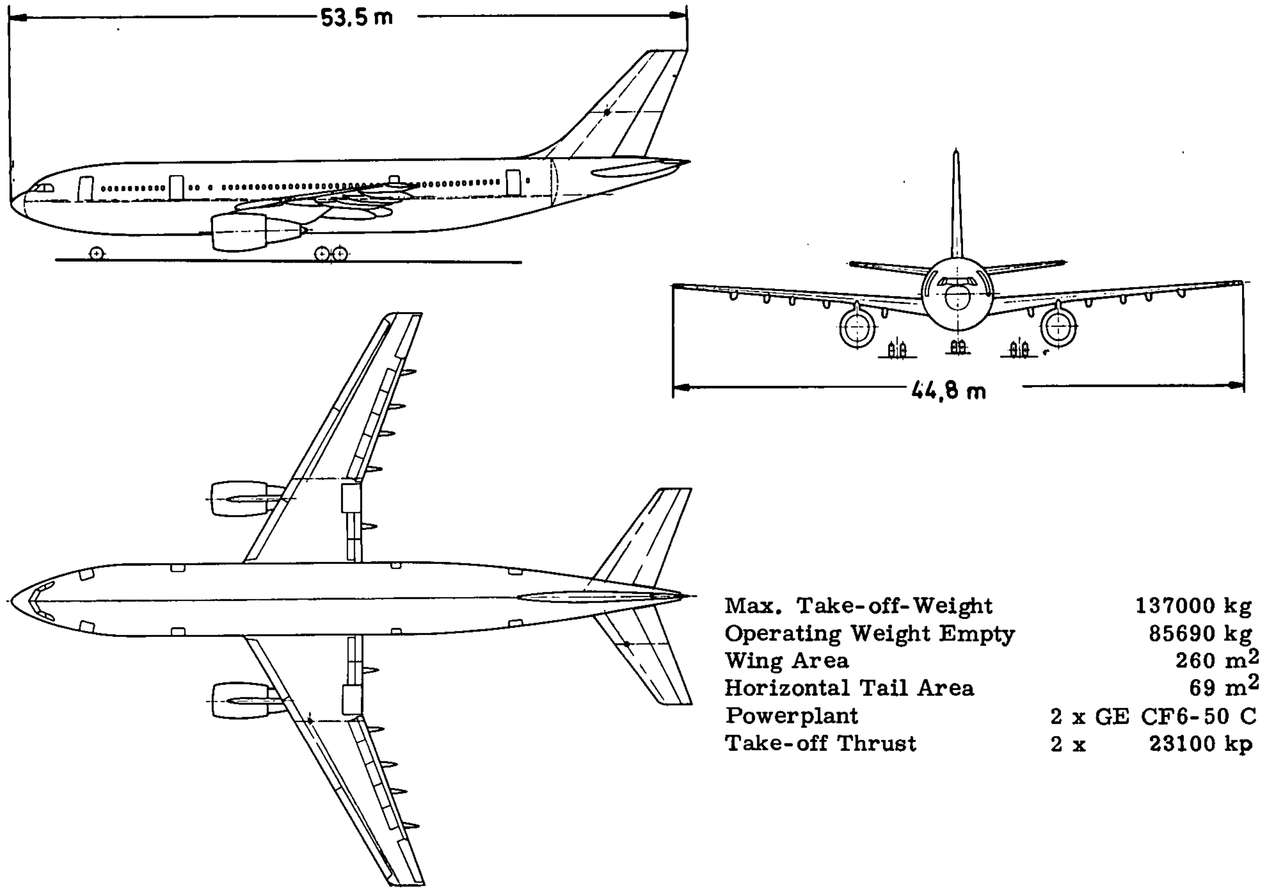
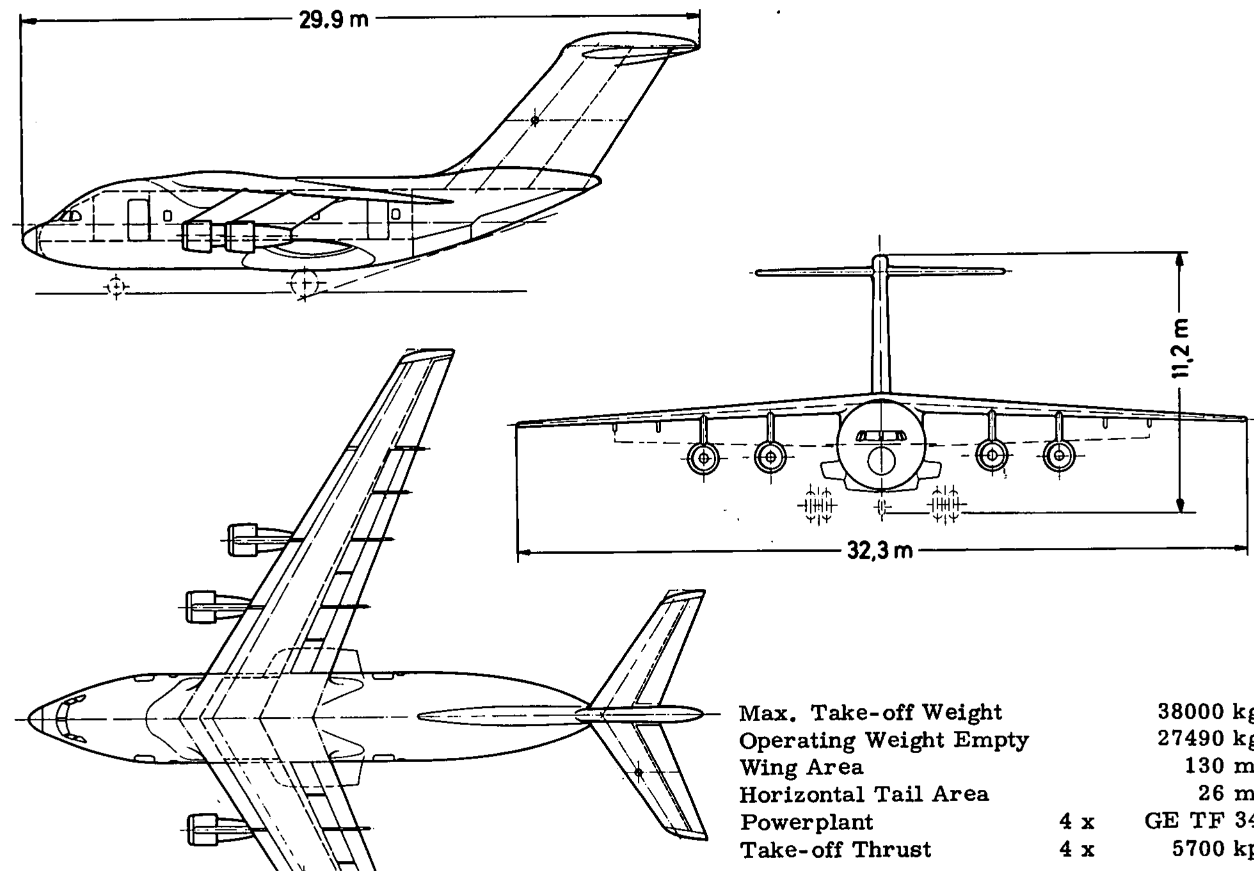


Fig. 2 Study flow diagram



Max. Take-off-Weight		137000 kg
Operating Weight Empty		85690 kg
Wing Area		260 m^2
Horizontal Tail Area		69 m^2
Powerplant	2 x	GE CF6-50 C
Take-off Thrust	2 x	23100 kp

Fig. 3 Baseline aircraft 1: Airbus A 300 B2



Max. Take-off Weight		38000 kg
Operating Weight Empty		27490 kg
Wing Area		130 m^2
Horizontal Tail Area		26 m^2
Powerplant	4 x	GE TF 34
Take-off Thrust	4 x	5700 kp

Fig. 4 Baseline aircraft 2: STOL-Transport CON 30

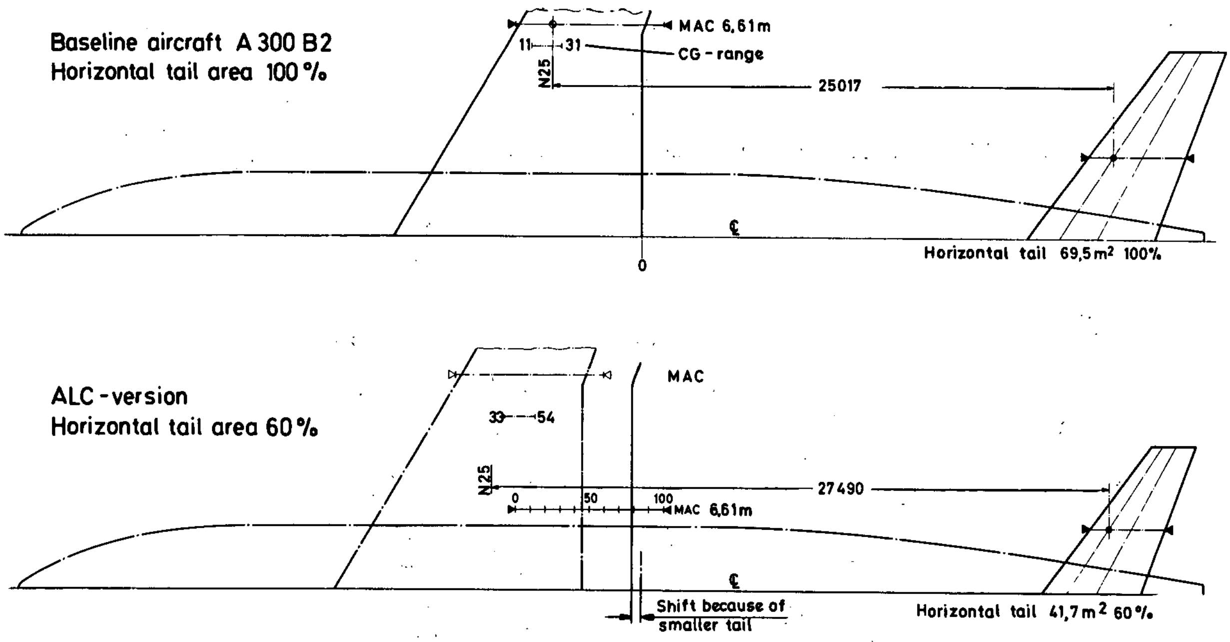


Fig. 5a Wing shift, baseline aircraft A 300 B2

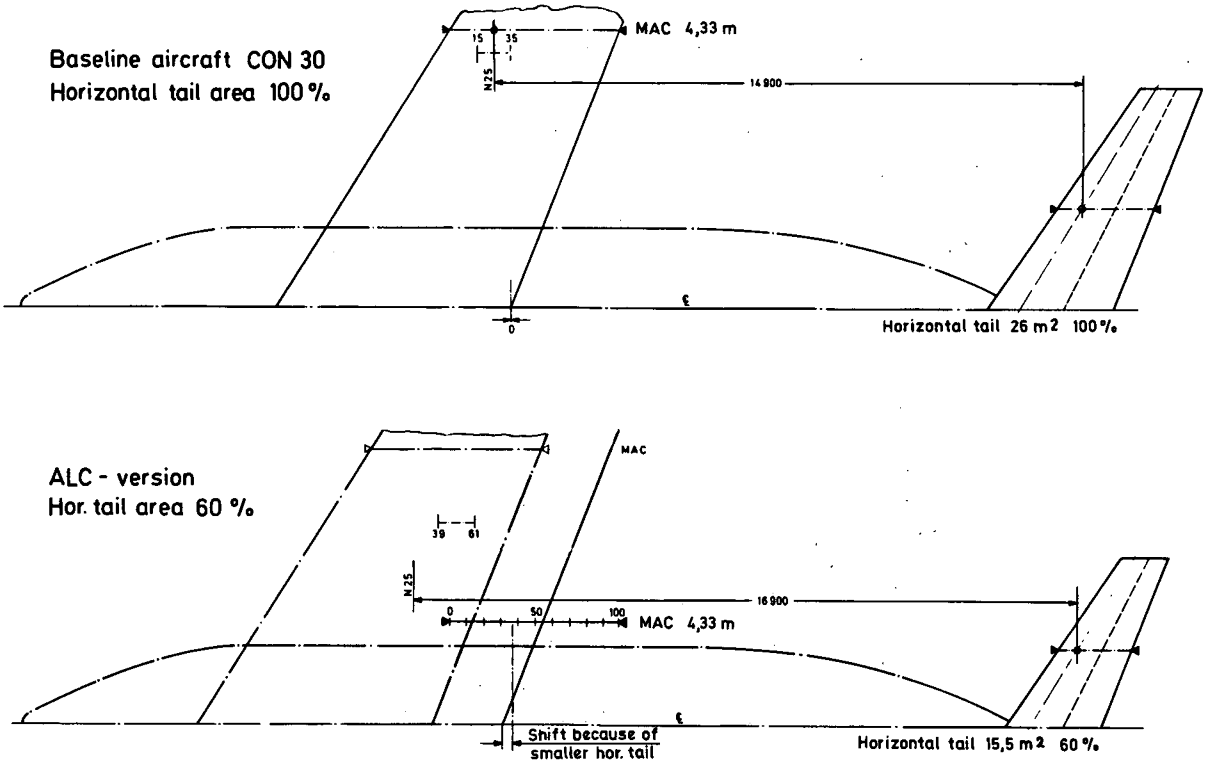


Fig. 5b Wing shift, baseline aircraft CON 30

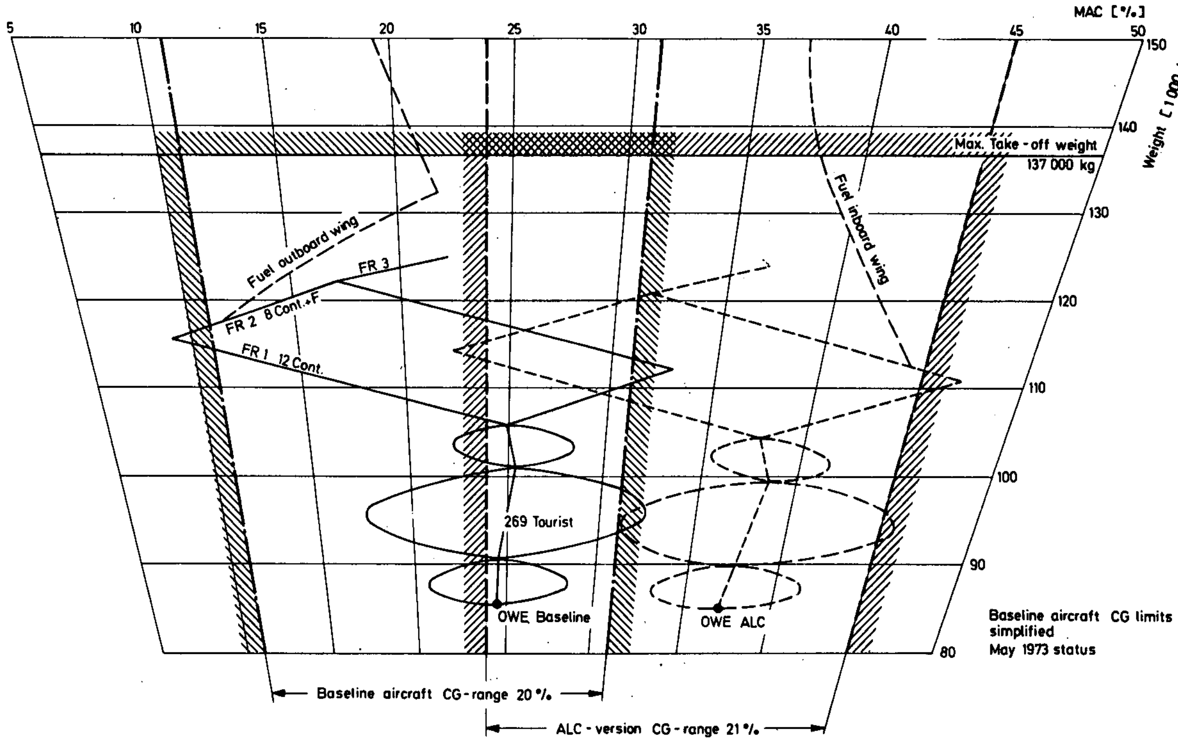


Fig. 6 CG-range comparison, baseline aircraft A 300 B2 vs ALC-version

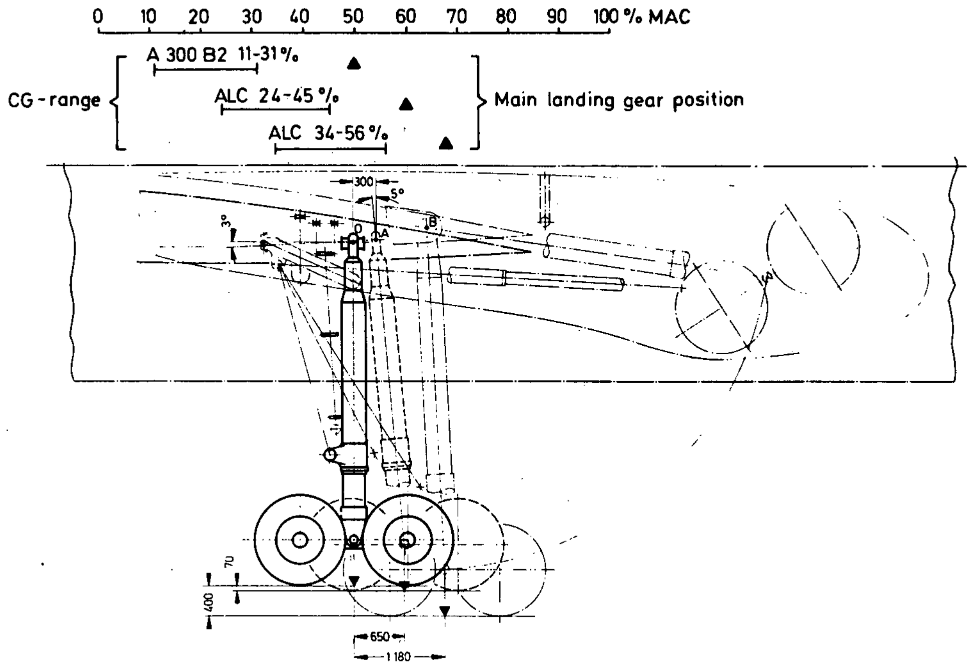
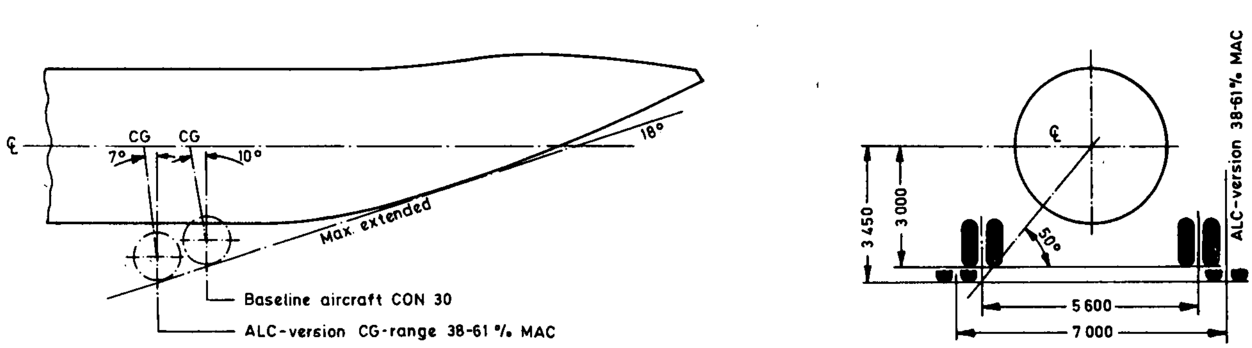


Fig. 7 Main landing gear changes, baseline aircraft A 300 B2



a. Change of landing gear height

b. Change of landing gear track

Fig. 8 Main landing gear changes, baseline aircraft CON 30

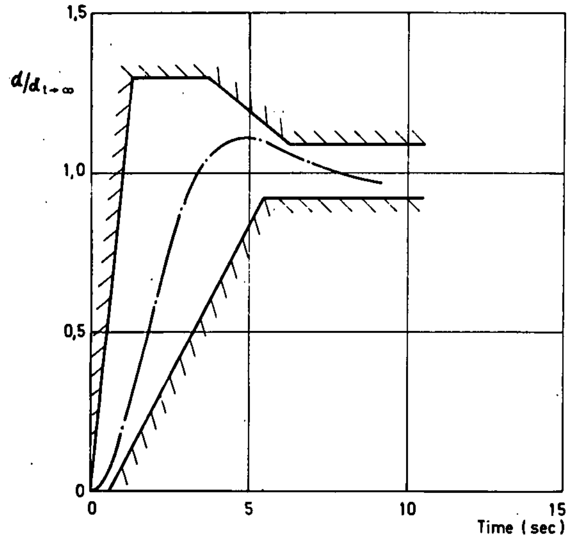


Fig. 9 Angle of attack motion limits
Baseline aircraft A 300 B 2

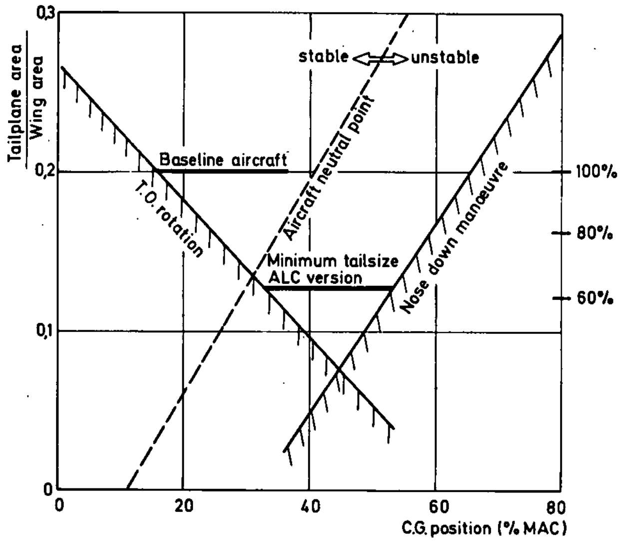


Fig. 10 Tailplane size vs C.G. limits
Baseline aircraft CON 30

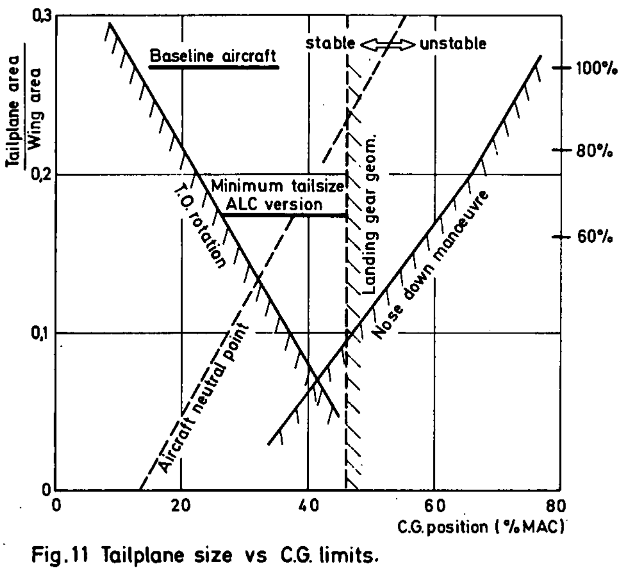


Fig. 11 Tailplane size vs C.G. limits.
Baseline aircraft A 300 B2

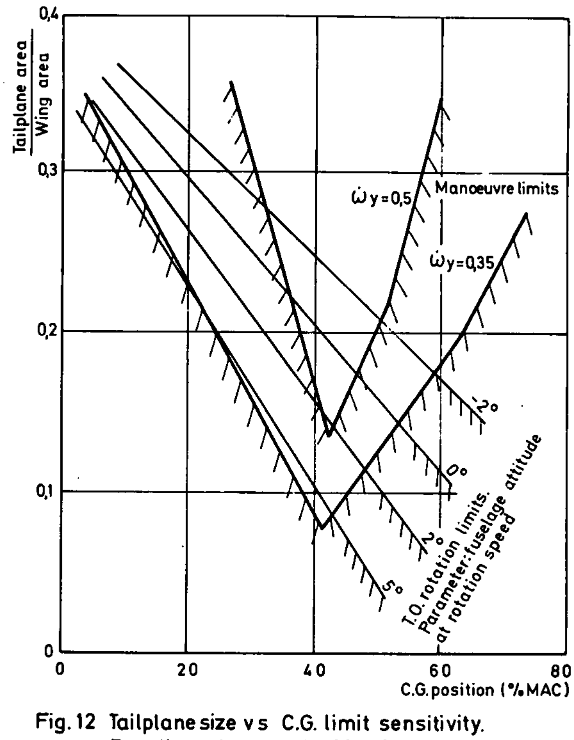


Fig. 12 Tailplane size vs C.G. limit sensitivity.
Baseline aircraft A 300 B 2

dm (m)	dm (MAC)	v_{gust} (m/sec)
49,6	7,5	3,17
82,6	12,5	3,86
115,6	17,5	4,45

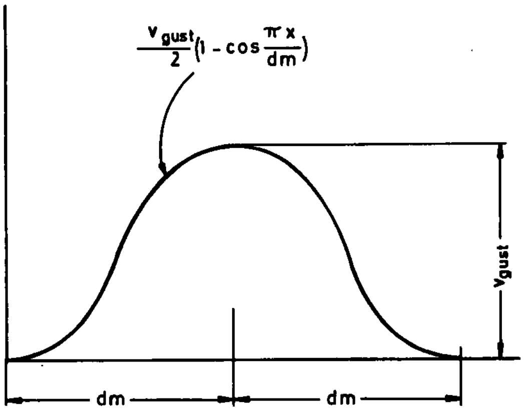


Fig. 13a Discrete gust

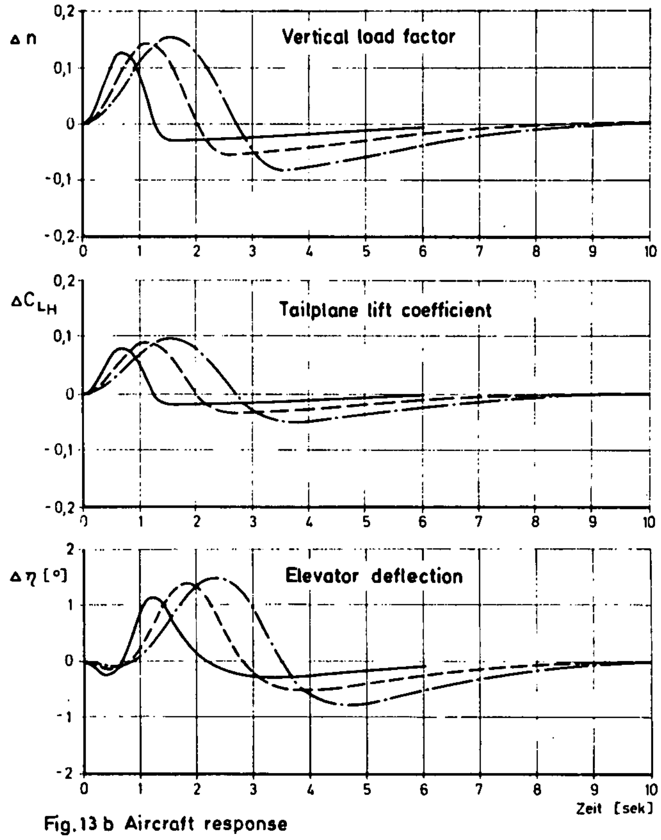


Fig. 13 b Aircraft response

Fig. 13 Dynamic response of ALC version of A300 B2.
Final approach at 128 kts. CG at 8% behind neutral point.
Tailplane area 65% of baseline aircraft

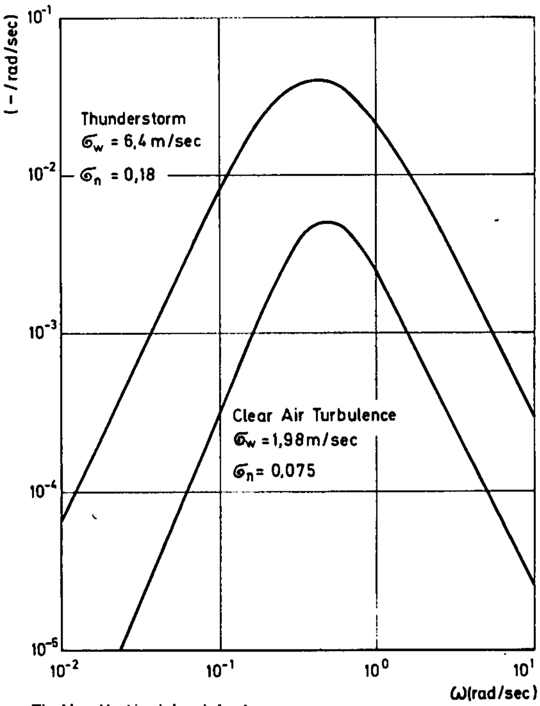


Fig. 14a Vertical load factor

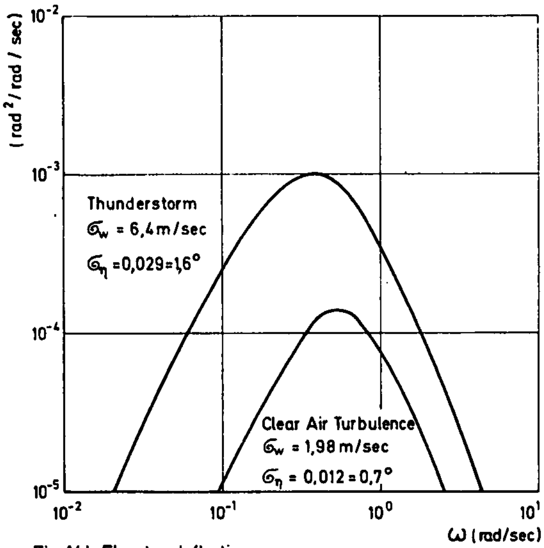


Fig. 14 b Elevator deflection

Fig. 14 Power spectra for ALC-version of A300 B2

Final approach at 128 kts. CG at 8% behind neutral point. Tailplane area 65% of baseline.

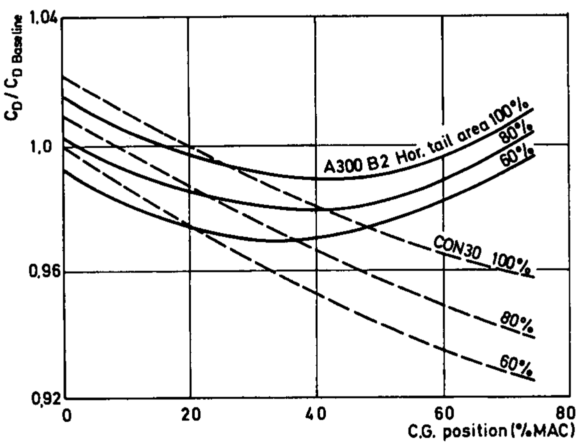


Fig. 15 Drag during cruise
Ma=0,8; H=9000m

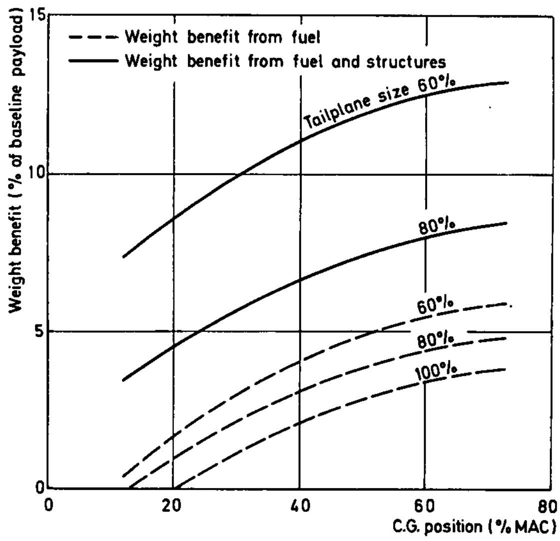


Fig. 16 Weight benefit, wing area unchanged.
Baseline aircraft CON 30
(Cruise at Ma=0,8, H=9000m, stage length 1000 km)

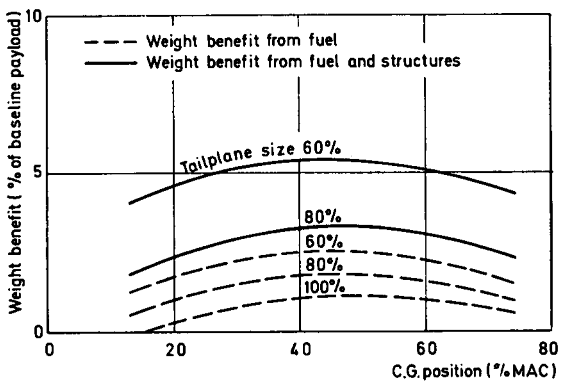


Fig. 17 Weight benefit, wing area unchanged.
Baseline aircraft A 300 B2
(Cruise at Ma=0,8; H=10 000m; stage length 2000km)

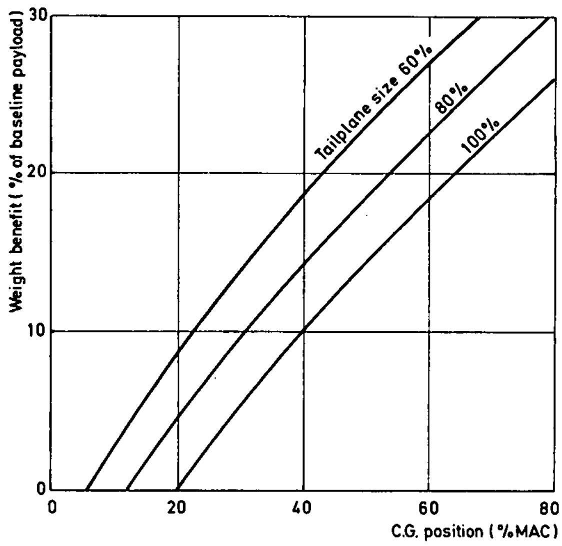


Fig. 18 Total weight benefit.
Baseline aircraft CON 30
(Cruise at Ma=0,8; H=9000m; stage length 1000 km)

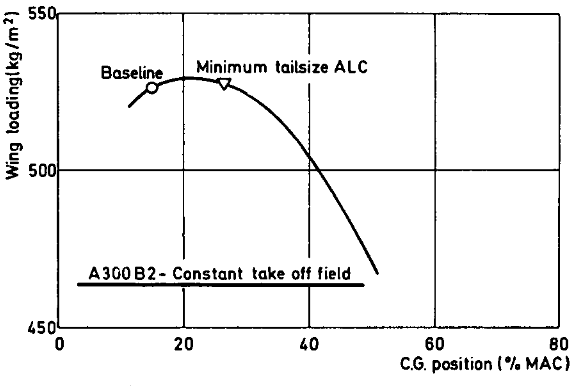


Fig. 19a Wing loading changes for constant field performance
(Points given for forward C.G. range limit)

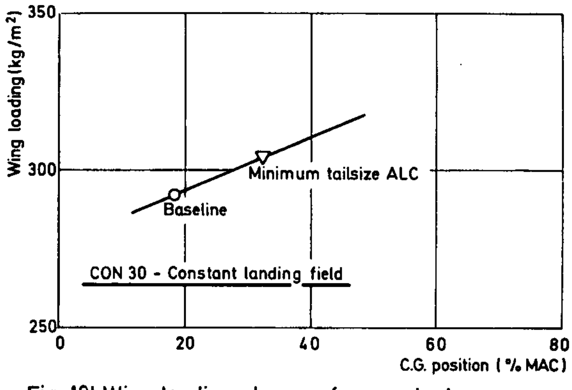


Fig. 19b Wing loading changes for constant field performance
(Points given for forward C.G. range limit)

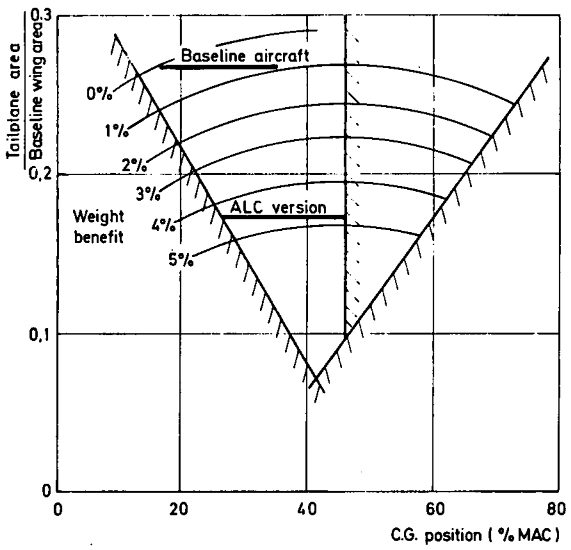


Fig.20 Weight benefit, percent of baseline payload. Baseline aircraft A 300 B2

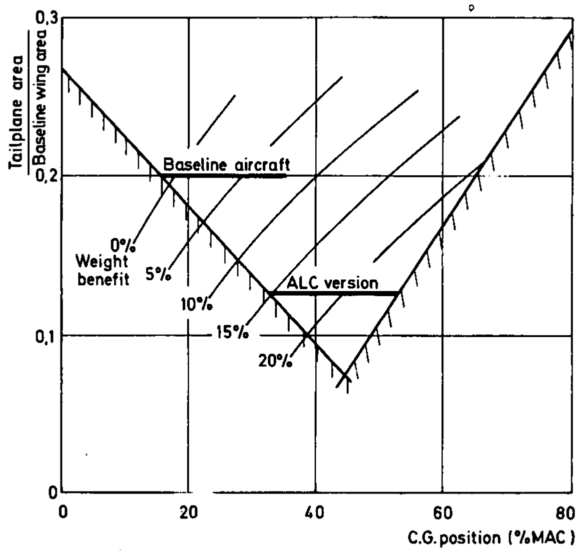


Fig.21 Weight benefit, percent of baseline payload. Baseline aircraft CON 30

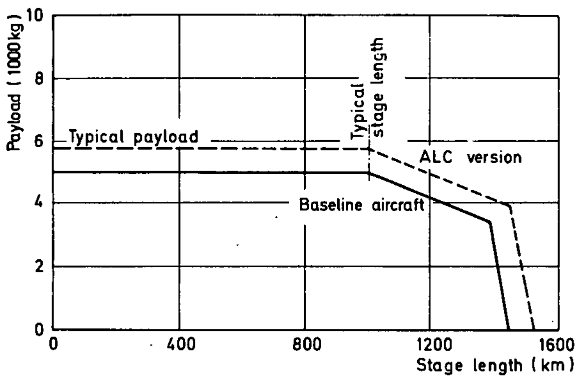


Fig. 22 Payload / range comparison Baseline aircraft CON 30

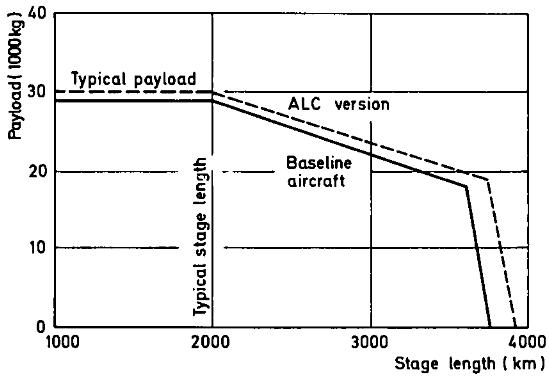


Fig. 23 Payload / range comparison. Baseline aircraft A 300 B 2

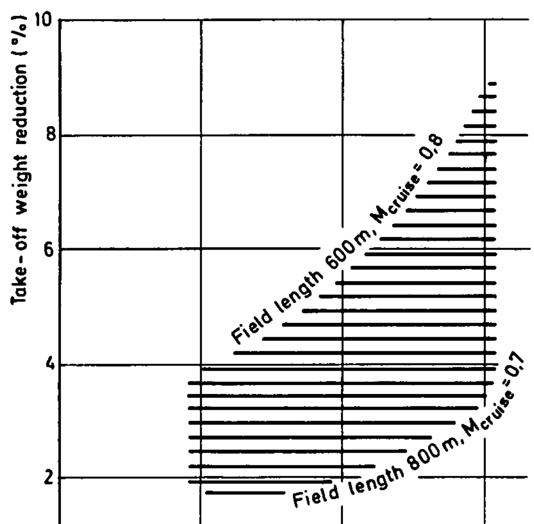


Fig. 24 Take-off weight reductions achieved by ALC Baseline aircraft CON 30

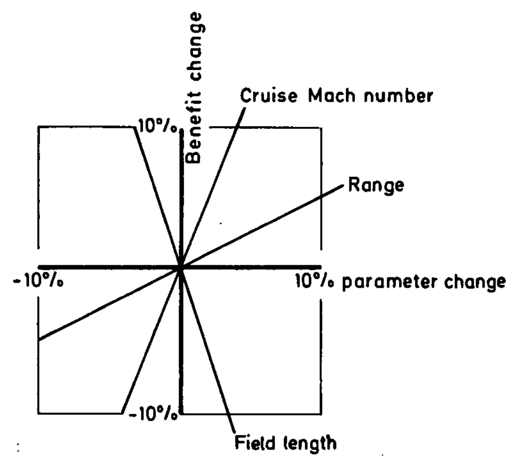
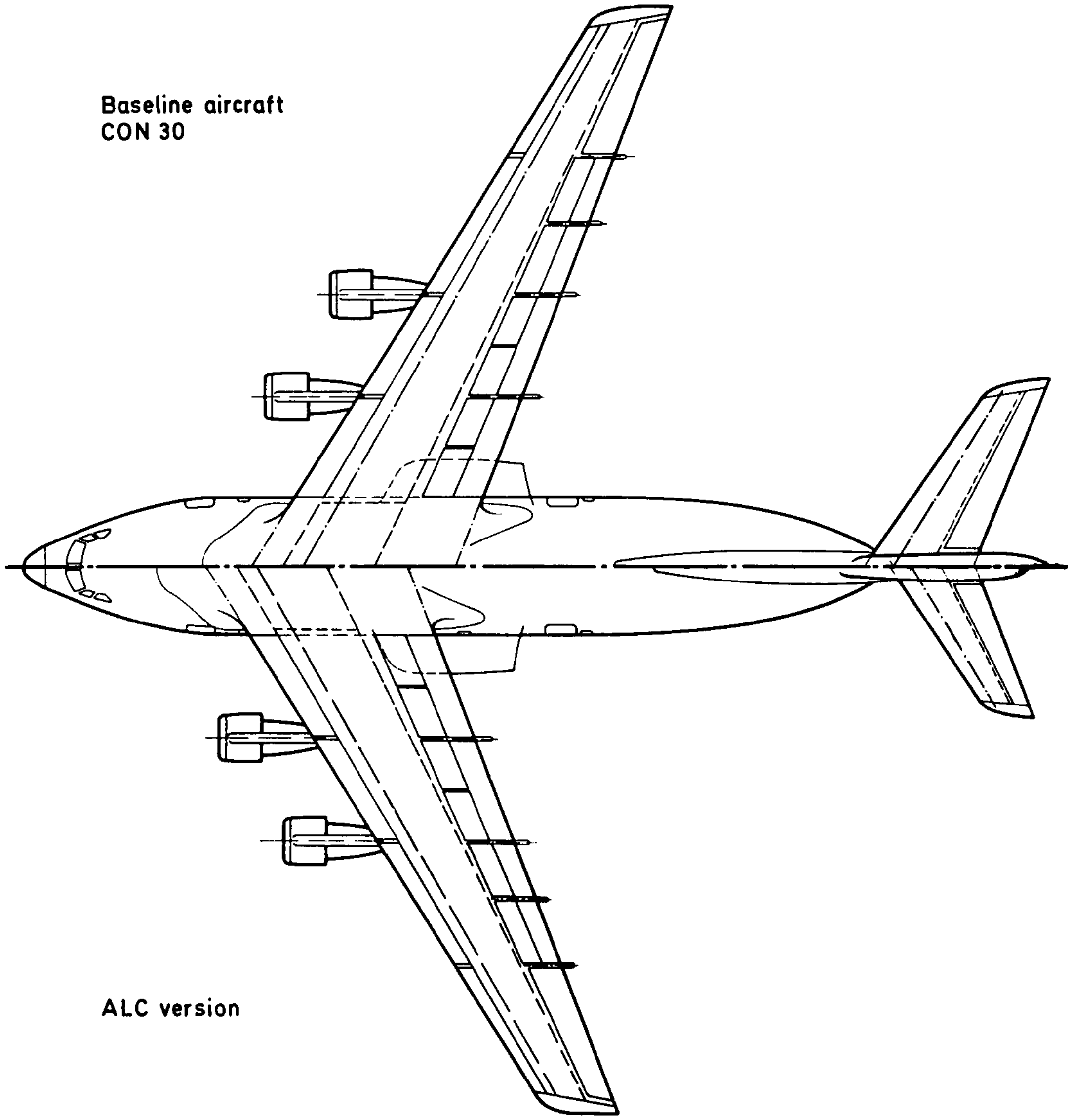


Fig. 25 Sensitivity of ALC benefit to performance parameters Baseline aircraft CON 30



	A 300 B2		CON 30	
	Baseline	ALC version	Baseline	ALC version
Max. Take-off Weight	137,000 kg	137,000 kg	38,000 kg	38,000 kg
Operating Weight Empty	85,690 kg	85,119 kg	27,490 kg	26,764 kg
Wing Area	260 m^2	260 m^2	130 m^2	125,4 m^2
Horizontal Tail Area	69 m^2	45 m^2	26 m^2	15,8 m^2
CG-range % MAC	11-31	24-45	15-35	32-53
Typical Payload	28,850 kg	30,200 kg	5,000 kg	5,730 kg

Fig. 26 Summary comparison baseline aircraft - ALC version

"Impact of active control technology on aircraft design"

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SUMMARY

Use of active control technology on civil transport aircraft is considered, both as regards improvement of a conventional aircraft and as regards development of new configurations to exploit such technology.

Significant gains in weight and operating cost may be made by using artificial stability augmentation and load alleviation on a conventional design, though the precise gains depend on the way in which weight savings are exploited.

Unconventional means are suggested whereby active control technology might best be exploited on short and long range subsonic aircraft, and also on supersonic aircraft. It appears that the largest gains are likely to be made when new techniques are used in combination rather than singly.

Acknowledgements.

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"Impact of active control technology on aircraft design".

1. INTRODUCTION.

Use of active control technology can be made in many ways on many different types of aircraft, but we are concerned here with civil transport aircraft for which cost is a primary criterion. It follows that we consider how the new technology can be employed to produce the biggest cost savings in carrying a given payload over a given range, while meeting given performance requirements. We may approach this question either from a known datum point, say with a given aircraft configuration whose cost per passenger mile is to be improved, or by asking on which type of aircraft the gain using the new technology is likely to be the greatest, and then configuring that aircraft to best advantage. Both approaches are to be considered in this paper. The first is naturally more straightforward and gives what one might describe as the short term answer. The second is necessarily more tentative but provides some interesting speculative thoughts. In all the work described here the incorporation of active control (artificial stabilisation, load alleviation etc.) is assumed, rightly or wrongly, to be capable of eventual achievement. We are concerned mainly with the results which follow.

2. IMPROVEMENT OF GIVEN CONFIGURATION.

2.1. Artificial aircraft stabilisation.

Use of artificial longitudinal stability augmentation allows a reduced or negative level of natural stability to be accepted while retaining satisfactory manoeuvre and trim capability. Achievement of satisfactory reliability for the augmentation system may be assisted by the use of electrical signalling to the controls. The reduced stability levels lead to low trim and manoeuvre requirements as regards tailplane size, and so to reduced aircraft weight. The weight reduction will, in general, 'snowball' from the initial tailplane weight and drag reductions. In addition the reduced stability margins can result in higher values of C_{Lmax} for take-off and landing because trimming losses are reduced, giving further reductions in wing area or engine size for a specified field performance.

The 'reference' configuration used here as a basis for improvement is shown in figure 1. It is a typical medium-range transport aircraft of twin rear-engine configuration. It carries 236 passengers over a range of 2200 n.m. from a 7600 ft. runway, giving a take-off weight of 286,400 lb.

Figure 2 shows the various limitations on tailplane sizing. The design cases for tailplane size involve the ability to raise the nosewheel at take-off on the forward c.g. limit and the maintenance of minimum stability on the aft limit. For the reference configuration the c.g. range is from 0.09 to 0.44 $\bar{c}$ giving a tail volume ratio of 0.93. Two levels of artificial stability were considered, giving variants A and B. In the first case, variant A, the c.g. range was moved back to give a small negative (natural) manoeuvre margin on the aft c.g., with the static nosewheel load reduced to 0.04 x aircraft weight - considered a normal lower limit. The nosewheel load is important because if it is reduced too much steering capability is reduced and even lost at speeds below those for which effective rudder control is available. There may also be a problem in towing with low nosewheel loads. In the second case, variant B, the nosewheel load is further reduced to 0.02 x aircraft weight, as an absolute minimum, leading to a substantially negative manoeuvre margin. In both cases stability is, of course, assumed to be restored artificially to a satisfactory level. Variant A then has a tail volume of 0.83 and variant B has one of 0.73.

For this exercise it is assumed that the undercarriage/wing relationship remains fixed, so that it is nosewheel load which limits the degree of (natural) instability accepted. Other configurations, having for example a fuselage mounted undercarriage, may avoid this limit, and in such cases the level of negative natural stability that is accepted is determined by trim and manoeuvre limits. These limits will arise from the reduction in control power as tail size is reduced with flap moments and aircraft inertias largely unchanged. Clearly it is advantageous to remove where possible non-aerodynamic limits to c.g. travel, but any weight or cost increase due to changing the configuration to achieve this (e.g. by using an unconventional undercarriage) must be included in the overall assessment.

As a result of the reductions in tail volume ratio obtained for variants A and B we find that the A.U.W. of variant A is now 282,000 lb. and that of variant B is 280,400 lb., reductions of 1.5% and 2.1% respectively from the datum value. The d.o.c. is correspondingly reduced by 1.4% and 1.9% for variants A and B.

In achieving these results the aircraft has been assumed to be 'scalable', so that wing, tail and engine size are varied to maintain payload/range and airfield performance capability. As an alternative it is possible to increase the payload, using the saving from reduced tailplane size and improved C_{Lmax}, and so maintain the original A.U.W. with further improved d.o.c.. When this is done for variant B the d.o.c. improvement from datum is some 3%. However, a strict comparison with the reference aircraft scaled to give the higher payload shows that some part of this improvement is due to the increase in capacity (larger capacity aircraft have lower d.o.c. than smaller capacity aircraft, other things being equal), so that the net change due to stability augmentation is close to the value with constant payload.

One may, of course, elect to use the weight saving to give a performance improvement with the original payload and with the engine sized for the original (reference) aircraft weight. This gives typically (for variant B) a reduction of 200 ft. in take-off distance and an increase of 300 ft. in single-engine ceiling. In 'hot and high' conditions a payload deficit of up to 14 passengers may be restored.

For other configurations the limit on c.g. movement may be set by different design cases, so that the gains achieved are likely to be configuration dependent in some degree.

2.2. Manoeuvre load control and gust alleviation.

Using controls already fitted to the 'reference' aircraft (albeit with fast operating jacks) it is thought possible, in principle, to reduce root bending moments on the wing by some 40% in a typical manoeuvre case and by 50% in a typical gust case. Since for the reference aircraft the gust case was just critical this leads to the situation in which gust and manoeuvre loads are about equally critical. The alleviation of gust loads is, however, likely to be the more difficult task in practice, so that this conclusion must be treated with reserve. It should also be mentioned that, as regards gust loads, only 'sharp' gusts (B.C.A.R. equivalent step gusts) were considered since it is known that in principle long period gusts can be dealt with by use of 'black box' and existing controls. Where gust loading is determined by some form of spectral analysis it is thought that to the same standard of assumptions a similar result would be obtained. In the present exercise it is assumed that the control devices (ailerons and spoilers) can be moved sufficiently quickly for the load change due to that application to build up and decay at the required rate. Some allowance has been made for aeroelastic distortion.

Figure 3 shows the change in manoeuvring load distribution spanwise as a result of alleviation. It is clear that while the bending moment on the wing has been greatly reduced the overall load is maintained, so that the loading inboard has been increased. Some care is needed to avoid buffet problems on this account. The normal 2.5 g requirement has been observed - development of a g limiting device could lead to overall load reductions and hence further gains, if its use were to be accepted.

As a result of these bending moment reductions the wing weight can be reduced by up to 20% in the first instance. It must be borne in mind that such reductions in weight may be achieved at the expense of torsional stiffness, so that flutter speed may be affected. This may, however, be dealt with using another form of active control technology - active flutter control.

Following this direct wing weight reduction further weight reductions from wing and engine scaling (with constant payload) may lead to a total A.U.W. reduction of as much as 5%. The d.o.c. compared with the reference aircraft is then reduced by up to 4%. Some allowance in the costing was made for development and additional equipment, as with stability augmentation. As in that case, a larger reduction in d.o.c. may be obtained by increasing payload to maintain the reference aircraft weight, though the same qualifications as in that case apply here.

Examination of fatigue cases suggests that if a much larger reduction in loads took place fatigue would provide the design case on some of those parts previously gust or manoeuvre designed.

3. UNCONVENTIONAL CONFIGURATIONS.

Since the improvements obtained on the reference aircraft are modest one may ask whether a more radical approach involving configuration changes might yield larger gains.

Considering the use of artificial stabilisation it is clear that the primary effect lies in the reduction of tail size. In general this will, of course, include fin size. Bearing in mind that the tail unit of a typical transport aircraft weighs around 3% of A.U.W. it is obvious that while appreciable reductions in weight can be made by reducing tail size, anything other than a large reduction of natural stability will produce only a modest return in cost and weight saving. This thought leads naturally to the proposition that complete removal of the tail surfaces will give the biggest return - i.e. a tailless aircraft is the limiting case. Strictly the limiting case would be a tailless aircraft with a conventional fuselage, but inertia problems with this are serious, so that one is led to consider the all-wing configuration.

In the past various tailless (all-wing) transport aircraft have been proposed (some have even flown) but have suffered from a number of inherent problems :

1. Headroom in the cabin demands a large t/c ratio and/or a large wing chord.

2. Large wing thickness leads to cruise speed limitations.

3. Low wing loading may lead to gust sensitivity.

4. Lack of natural damping in pitch may be a problem.

With the advent of gust alleviating techniques the gust problem can, potentially, be solved. Similarly the lack of pitch damping is easily supplied by use of stability augmentation techniques. It follows that active control technology can assist greatly in the development of a configuration of this type. Further, 'supercritical' wing design makes it possible to design a thick wing section that (at a modest C_L) is capable of achieving a drag rise around $M = 0.8$ in two dimensions - high enough for a normal subsonic cruise. This leads naturally to the use of a rectangular

/box

box of high t/c accommodating passengers over the whole of its span (fig. 4). To give an acceptable aspect ratio and provide fuel capacity, control and some degree of natural stability, swept wing tips may be added. With so thick a wing the engines may be partially buried behind the rear spar, ingesting air from below the wing - some small advantage in s.f.c. may result from this, since boundary layer ingestion can provide an improvement in propulsive efficiency. With artificial stability the c.g. may be moved far enough aft to allow retraction of the undercarriage behind the rear spar, and also to avoid the use of large, up-elevon, trim angles during take-off and landing. Similarly the rear camber characteristic of supercritical wing design could be preserved in cruise.

Such a configuration would have a very clean profile and hence an acceptable L/D despite the low aspect ratio - typically the L/D should be comparable with values for current short/medium range aircraft, the category into which this scheme naturally falls. With such an arrangement the aircraft is simplified to a single component - the wing - and preliminary calculations suggest that substantial weight savings for a given payload might be achievable. Since the advantages of the layout derive from reduced structure weight and efficient 'packaging' of the payload it follows that effort would be needed to ensure both were obtained. The wing structure would need to carry both aerodynamic and pressurisation loads, so that some departure from conventional structure practice would probably be necessary. The need for advanced stability and load alleviation systems would be offset by the lack of complex high lift systems and the use of a basically simple structure.

Considering now how best, in general, to make use of load alleviation, it is at least arguable that since with a conventional layout most of the load is on the wing, and since the heaviest single item on anything other than a short range aircraft is likely to be the fuel weight (fuel cost now begins to contribute substantially to d.o.c.), load alleviation techniques should be used to increase aspect ratio and so increase L/D. This is also consistent with the move towards fuel conservation and will clearly show to best advantage on medium/long range aircraft.

Unfortunately substantial increases in aspect ratio on a swept wing lead to pitch-up. It would be possible to use artificial stability to combat this (perhaps with the aid of stick pushers) but an alternative lies in the use of a straight (or nearly straight) wing of 'supercritical' design, as for the previous scheme so avoiding the pitch-up problem (fig. 5). Using gust and manoeuvre load alleviation techniques, combined with use of carbon fibre composite structure, quite large aspect ratios could be used. As remarked previously, loss of torsional stiffness would present a flutter problem, but with active flutter control this should not present a serious obstacle. An aircraft of this kind should be capable of efficient long-range cruise at moderate subsonic speeds, offering the possibility of cheap mass travel over long ranges.

Both these schemes are for subsonic aircraft and one may ask what can be offered for supersonic travel. Studies done by various project teams lend support to the contention that the most efficient configuration (aerodynamically) for supersonic speeds around M = 2 is probably the so-called arrow-wing layout (fig. 6). This type of configuration is normally associated with the use of variable sweep because of its high mean sweep angle in the cruise and its relatively poor low speed lift capability and handling at this high sweep. Because of the requirement for positive stability in both the low and the high speed configurations one then almost inevitably obtained a large highly swept 'forewing' in the low speed configuration. This is a serious embarrassment as regards pitch-up at low speeds and is structurally inefficient. The result is commonly a high structure weight and a large tailplane. Using artificial stability a negative value of natural stability can be accepted in the cruise or in the low speed configuration, or in both. A convenient option is that for which positive stability is maintained in the cruise but negative stability is accepted for landing and take-off. This allows the 'low speed' wing to be further forward relative to the cruise wing, so reducing the forewing size. However, with realistic values of manoeuvre margin it seems (see fig. 6) that the reduction likely to be achieved is not sufficient to eliminate the forewing, or even to reduce it very substantially. Some assistance in this respect may be obtained by use of fuel transfer, but this technique is available without regard to active control.

An alternative possibility with this configuration consists in using a fixed (cruise) wing of slightly larger size, with no sweep variation, and employing artificial stability to deal with the low-speed handling problems. With this alternative the low speed performance (e.g. approach speed) is worse than with variable sweep, though no worse than with contemporary supersonic transports, but there is a great reduction in mechanical complexity and a corresponding weight saving.

All the unconventional configurations that are described here have one thing in common - they make use of more than one form of advanced technology at a time. Whatever the merits of these particular proposals it does seem likely that this will be necessary in general to give really significant advances when using active control technology.

4. CONCLUSIONS.

Significant improvements in operating costs per passenger mile are possible in principle, by making use of automatic stability and load alleviation techniques. For a conventional layout the improvements are likely to be modest - the actual gain depends on the way in which weight savings are exploited - and for larger gains it seems likely that unconventional configurations employing more than one form of advanced technology may be needed.

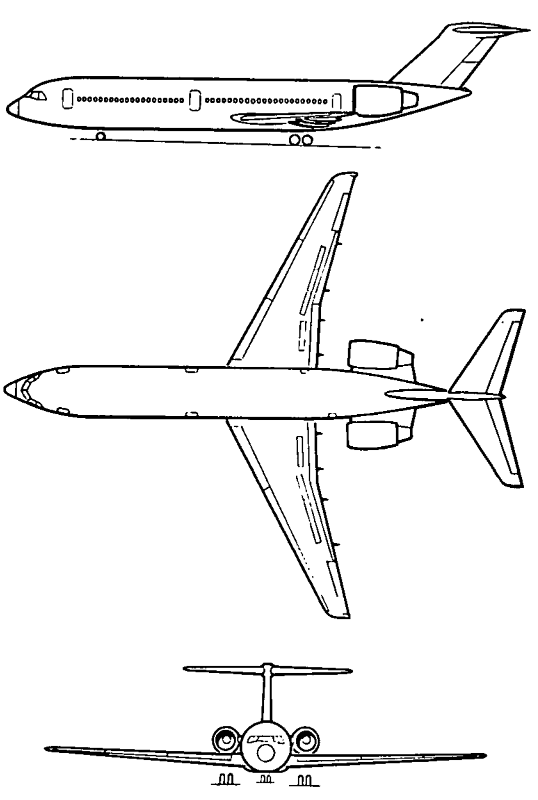

FIG.1, REFERENCE CONFIGURATION

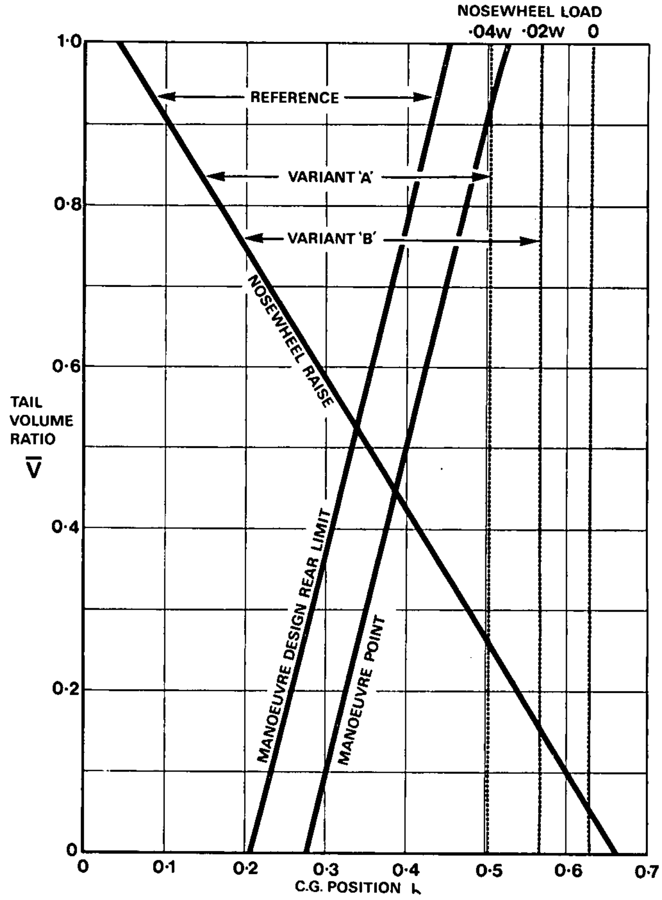

FIG. 2, TAILPLANE SIZING LIMITATIONS

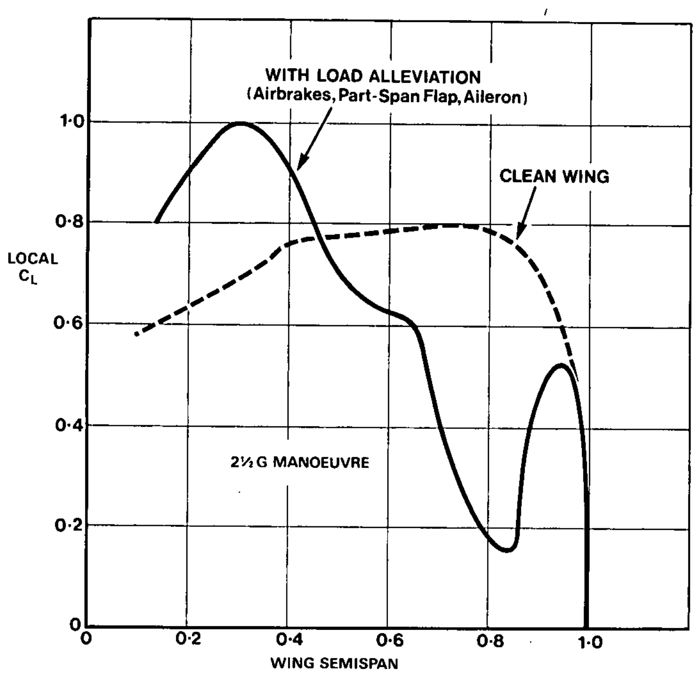

FIG.3, CHANGE IN WING LOAD DISTRIBUTION

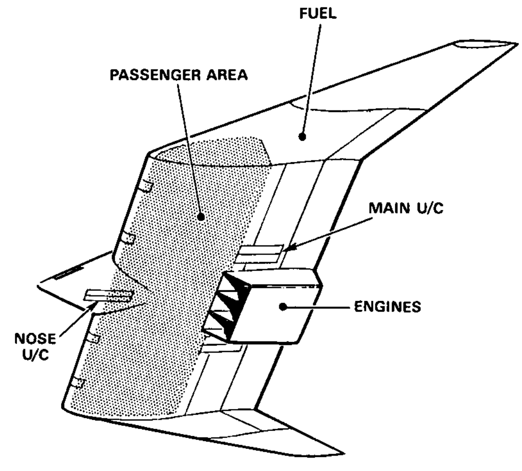

FIG.4 ALL-WING TRANSPORT

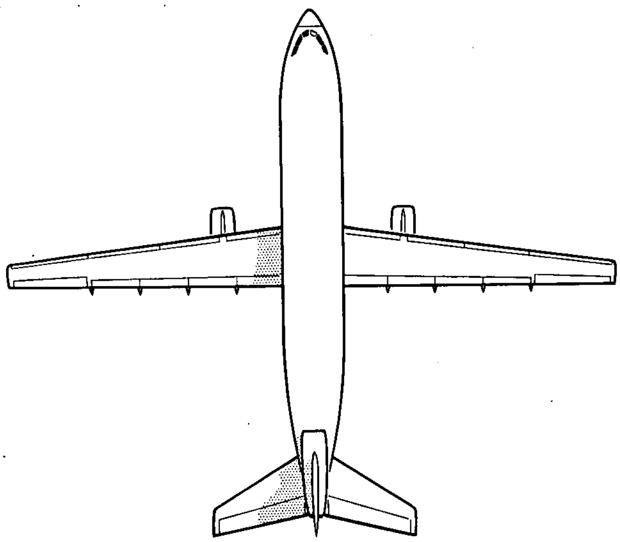

FIG. 5, HIGH ASPECT RATIO TRANSPORT

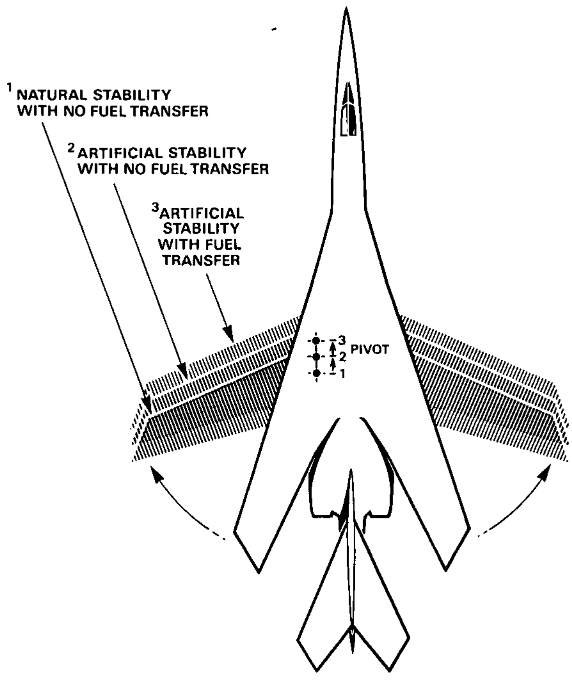

FIG. 6, SUPERSONIC TRANSPORT

HORIZONTAL CANARDS FOR TWO-AXIS CCV FIGHTER CONTROL

BY

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ABSTRACT

This paper describes the potential use of active horizontal canards in the design of fighter aircraft to provide flight path control along both the longitudinal and directional axes. The results are based on wind tunnel tests conducted on two CCV fighter configurations under the Fighter CCV Program of the USAF Flight Dynamics Laboratory. A method for generating direct sideforce using differentially deflected horizontal canards is discussed. The direct lift control capabilities of horizontal canards are also presented. In addition, the use of horizontal canards in implementing the concepts of Relaxed Static Stability and Maneuver Polar Enhancement is described. Finally, the USAF Fighter CCV Program is outlined as it relates to demonstrating the performance improvements achievable through application of advanced control system technology.

1. INTRODUCTION

Recent advances in active control technology have led to new control functions for regulating the flight path dynamics of high performance airplanes. Direct sideforce and direct lift are two control modes for fighter aircraft that have received attention during Control Configured Vehicle (CCV) design studies. Several methods have been investigated for generating direct sideforce and lift. One effective means of generating direct sideforce has been through the deflection of aerodynamic surfaces mounted vertically on the underside of the forward fuselage coupled with rudder deflection. Direct lift control can be obtained using wing mounted flaps deflected in conjunction with the horizontal tail. Although these two procedures can produce adequate levels of direct forces, another control method has recently been identified wherein both the direct sideforce and direct lift capabilities, implementation of Maneuver Polar Enhancement (MPE), and stability augmentation of a Relaxed Static Stability (RSS) configuration, can be provided by a single set of surfaces. The control surfaces that hold this potential for application to CCV fighters are close-coupled horizontal canards.

The attractiveness of horizontal canards as active control surfaces was enhanced with the discovery of an unexpected aerodynamic phenomenon. Wind tunnel tests indicated that differential deflection of close-coupled horizontal canards generates usable levels of sideforce without introducing unacceptable longitudinal or lateral/directional trim requirements. This phenomenon has been documented throughout the speed envelopes of two different fighter configurations in testing conducted under the Fighter CCV Program of the USAF Flight Dynamics Laboratory. The potential of horizontal canard sideforce is that a single set of control surfaces could generate the primary forces needed for flight path control about both the longitudinal and directional axes.

To achieve their maximum usefulness, the horizontal canards will be required to move symmetrically for direct lift control, pitch control, and longitudinal stability augmentation and asymmetrically for direct sideforce control. It also appears desirable to give pilots the option of commanding additional uncoupled maneuver modes which will require automatic control blending between the canards and other primary control surfaces. Realization of the full benefits to be derived from horizontal canards on CCV fighter designs will be dependent on a skillful application of active control technology.

2. DIRECT SIDEFORCE CONTROL

The value to tactical aircraft of direct sideforce control (DSFC) has been investigated for air-to-air combat and for air-to-ground attack. A simulation study by the Boeing Company (Ref. 1) indicates that air-to-ground weapon delivery accuracy can be improved through DSFC. Direct sideforce control enhances the air-to-air capabilities of fighter aircraft through improvements in lateral-directional maneuverability. In addition, DSFC can reduce pilot workload during aerial refueling and in landing approaches under crosswind and gust conditions.

As shown in Figure 1, direct sideforce can give fighter pilots several options for precision flight path control. The lower sketch depicts the capability for lateral translation without rotation, which is the literal meaning of DSFC. Blending DSFC with the standard rotational mode, as in the center sketch, gives variable yaw control for rapid wings-level turning. Finally, independent fuselage azimuth control permits the aircraft to be pointed at the target without altering the velocity vector.

2.1 DSFC Using Differential Horizontal Canards

Under the USAF Flight Dynamics Laboratory Fighter CCV Programs, wind tunnel testing has been conducted on the two fighter configurations shown in Figure 2. Various horizontal and vertical canard planforms at several locations were separately investigated on each of these aircraft. During supersonic testing of the CCV F-4, the horizontal canards were tested at differential deflections. Analysis of the data indicated that differentially deflected horizontal canards can produce a usable level of sideforce without introducing significant rolling or pitching moments. This initial discovery encouraged additional testing to further investigate and document the phenomenon. Follow-on wind tunnel testing conducted for the Air Force Fighter CCV Programs by The McDonnell Douglas Corporation, The General Dynamics Corporation, and by the National Aeronautics and Space Administration (NASA) has consistently verified the differential horizontal canard effect and its potential for application to CCV fighter design.

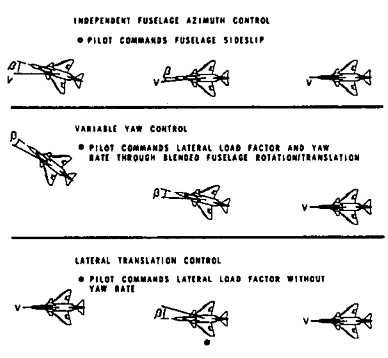

FIG. 1: DIRECT SIDEFORCE OPTIONS

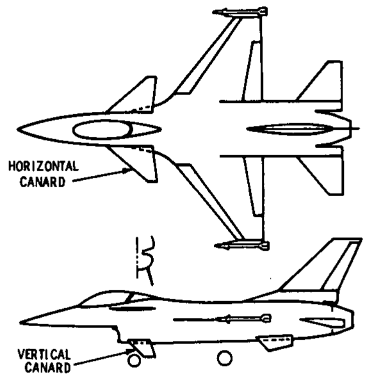

FIG. 2a: CCV YF-16 CONFIGURATION

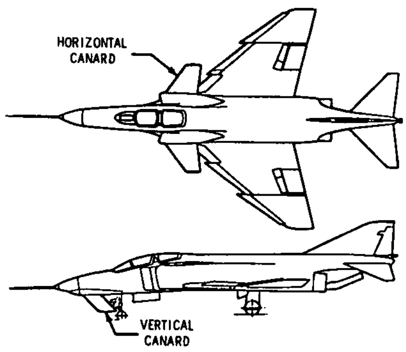

FIG 2b: CCV F-4 CONFIGURATION

The sideforce from differentially deflected horizontal canards occurs as a result of unbalanced pressure distributions between the right and left sides of the aircraft. Wind tunnel pressure testing will be required to accurately define these distributions. Force testing has indicated that the fuselage, wing, and vertical tail all contribute to the phenomenon. The net sideforce is in the direction of the side having the leading-edge-down canard, and is accompanied by a yawing moment which must be trimmed out by the rudder to achieve "direct" sideforce. The center-of-pressure of the sideforce remains forward of the aircraft center-of-gravity. Thus, the sideforce produced by the rudder trim is in the same direction as the sideforce generated by the canards.

A plot of the level of direct sideforce control attainable on the CCV YF-16 configuration is given in Figure 3. The lower curve represents the untrimmed force contribution of the differentially deflected horizontal canards. Trimming the yawing moments with the rudder results in a sideforce capability more than double that of the untrimmed level, as shown by the upper curve. The sideforce coefficients shown in Figure 3 are based on low speed wind tunnel data. The accelerations on the right of this plot correspond to the dynamic pressure seen at Mach 0.9, 10,000 feet. It should also be noted that the 40-degree differential deflection is the maximum envisioned for the canards. Approximately 1g of direct sideforce is available at low trim angles of attack, with more than 1.5 g's available at maneuvering α's. The direct side accelerations achievable under these conditions have not been flight demonstrated to date, and from a pilot standpoint, may be approaching maximum usable levels.

The directional trim requirements for these high levels of DSFC are shown in Figure 4. As mentioned above, the large rudder trim deflections are responsible for about half of the direct sideforce. Drag levels were found to increase significantly, especially at large differential deflection angles. However, these drag increases seem to be comparable to that encountered with other means of DSF generation.

Aircraft directional stability was also found to be affected by the horizontal canard installation as shown by CCV YF-16 test results (Ref. 2). Figure 5 shows the value of Cn_β at high angles of attack to be a function of the horizontal canard deflection. These levels of Cn_β generally reflect stability degradations from the baseline YF-16 configuration. This emphasizes the need for a thorough description of the directional stability characteristics for use in the design of stability augmentation systems of canarded CCV fighters.

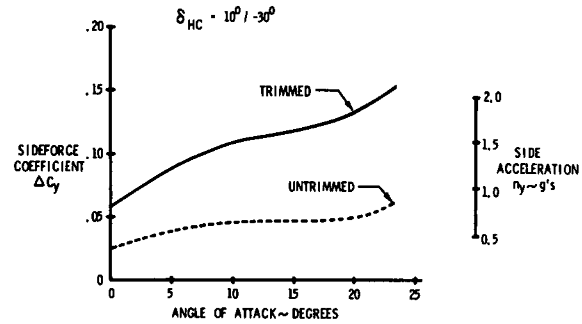

FIG 3: SIDEFORCE FROM DIFFERENTIALLY DEFLECTED HORIZONTAL CANARDS (CCV YF-16)

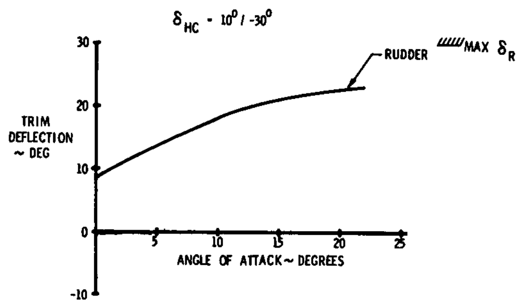

FIG 4: TRIM REQUIREMENTS FOR HORIZONTAL CANARD SIDEFORCE (CCV YF-16)

The DSFC from differential horizontal canards appears to be well behaved with angle of attack and canard deflection. Reasonably smooth aerodynamic characteristics are highly desirable for control system implementation. As shown in Figure 6, the level of DSFC produced by the horizontal canards increases smoothly as a function of differential deflection on the canards. Figure 7 shows the effects of Mach number on the sideforce capabilities for a nominal 20-degree differential canard deflection. Although this is less than the maximum canard deflection, high levels of DSFC are available to the CCV YF-16 over a wide Mach range, from landing speeds up through high transonic combat speeds. The sideforce coefficients of Figure 7 translate into accelerations of approximately 0.1g at landing conditions and 0.5g at combat conditions of Mach 0.9, 10,000 feet.

Adding dihedral to close-coupled horizontal canards was found to increase the direct sideforce produced through differential deflections. The sideforce is generated by taking advantage of both the differential canard pressure effect and the resolved lateral components of the normal force on the surfaces. Figure 8 shows the increase in sideforce that can be realized with 15° of canard dihedral on the CCV YF-16. An increase of approximately 40% is seen at 5° angle-of-attack. Canard dihedral and canard placement must be carefully accomplished to avoid degradation of inlet performance or pilot visibility.

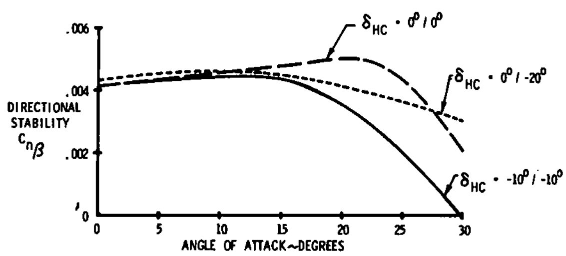

FIG 5: EFFECT OF CANARD DEFLECTION ON DIRECTIONAL STABILITY (CCV YF-16)

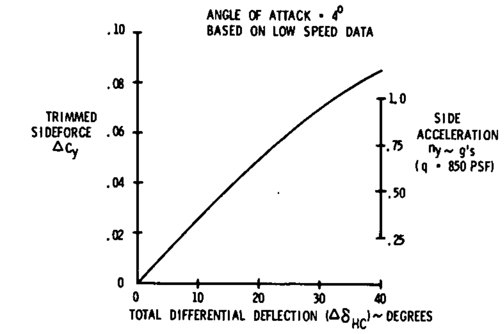

FIG 6: VARIATION OF SIDEFORCE WITH HORIZONTAL CANARD DEFLECTION (CCV YF-16)

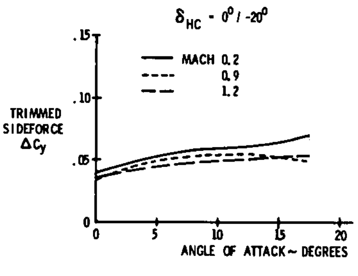

FIG 7: EFFECT OF MACH NUMBER ON HORIZONTAL CANARD SIDEFORCE (CCV YF-16)

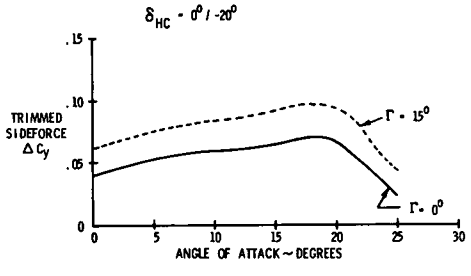

FIG 8: EFFECT OF DIHEDRAL ON HORIZONTAL CANARD SIDEFORCE (CCV YF-16)

The horizontal canard effects reported above for the CCV YF-16 configuration were also found on the CCV F-4. The direct sideforce capabilities of the CCV F-4 are shown in Figure 9. Approximately 0.5 g of direct side acceleration is available at combat conditions. These levels of DSFC are somewhat lower than those of the CCV YF-16 because the F-4 canards were relatively smaller. As on the CCV YF-16, small trim changes were found on the CCV F-4 and the directional stability levels were reduced.

2.2 DSFC Using Vertical Canards

Another surface for implementing the DSFC mode is the vertical canard mounted on the underside of the forward fuselage. Several vertical canard planforms and locations were investigated on both the CCV F-4 and CCV YF-16 configurations. Twin inlet-mounted vertical canards were found to give high levels of sideforce on the CCV YF-16. The DSF capability of a twin vertical canard configuration is shown in Figure 10. The effectiveness of the vertical canards falls off rapidly with angle of attack. Although DSFC may be obtained from vertical canards, they do not possess the two-axis control capability of close-coupled horizontal canards.

3. DIRECT LIFT CONTROL

Direct lift control (DLC) is the longitudinal axis counterpart to direct sideforce control. Together, DLC and DSFC can provide improved capability for flight path control particularly valuable during air-to-air combat and air-to-ground weapon delivery. As summarized in Figure 11, direct lift control offers the pilot several options for flight path control. The lower figure shows that vertical translation without pitch rotation can be obtained through command of normal load factor without pitch rate. This mode of DLC may lead to improvement in tracking capability and evasive tactics. The center figure points out that DLC can be used to provide quickened aircraft response leading to more stable tracking through the blending of direct lift and pitch rate. Finally, independent control of fuselage attitude can be made possible without altering the flight path. This mode, often referred to as independent fuselage aiming (IFA), holds promise for the air-to-air tracking task since it can assist in converting an imprecise tracking condition into a firing solution. Figure 12 shows that the CCV F-4 fuselage may be rotated through approximately four degrees of angle of attack without altering the flight path while maintaining a constant load factor.

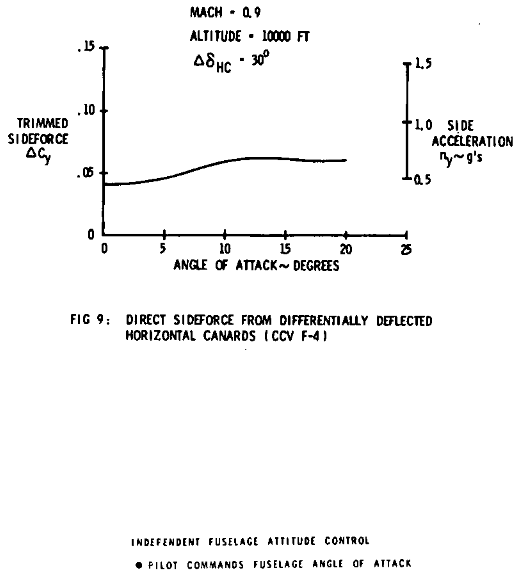

FIG 9: DIRECT SIDEFORCE FROM DIFFERENTIALLY DEFLECTED HORIZONTAL CANARDS (CCV F-4)

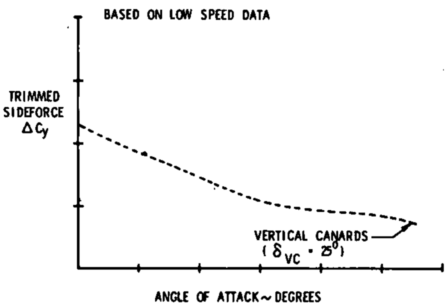

FIG 10: VERTICAL CANARD SIDEFORCES (CCV YF-16)

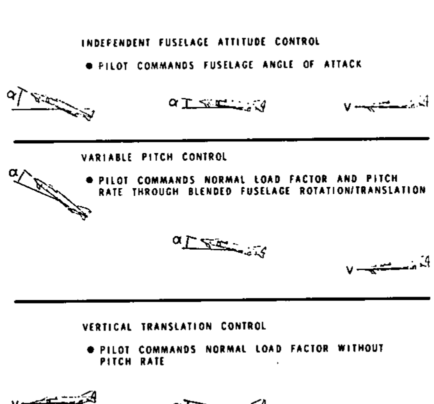

FIG. 11: DIRECT LIFT OPTIONS

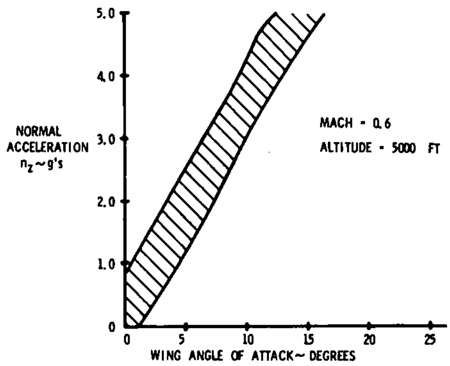

FIG 12: FUSELAGE ATTITUDE CONTROL (CCV F-4)

The incremental lift necessary for DLC can be generated using various combinations of control surfaces such as all-movable horizontal canards, flaperons and spoilers, and an aft horizontal tail. Recent fighter CCV studies at the Air Force Flight Dynamics Laboratory have considered each of these control surface combinations for producing DLC and concluded that significant levels of direct lift can be obtained with each combination. However, combinations employing close-coupled horizontal canards were found to produce the highest levels of direct lift while providing the most flexibility for overall CCV design considerations.

3.1 DLC Using Symmetric Horizontal Canards

All-movable close-coupled horizontal canards can be employed on CCV fighter designs to generate high levels of direct lift control. It should be noted, however, that the direct lift capability of horizontal canards does not arise from their effectiveness as lift producers. Wind tunnel tests have shown that symmetrically deflected close-coupled horizontal canards are relatively ineffective in producing net lift because the lift generated by the canards is offset by the decrease in wing lift resulting from the downwash induced on the wing by the deflected canards. However, the canards can generate a considerable nose-up pitching moment which may be trimmed by a trailing edge down horizontal tail deflection. This positive tail deflection for trim contributes a relatively large amount of incremental lift which, when added to the untrimmed canard increment, leads to a significant level of DLC. As with direct sideforce, the trimming of the canard moment produces the favorable result.

Figure 13 presents a plot of the CCV YF-16 transonic DLC capability using symmetrical horizontal canard deflection. At low angles of attack, the direct lift results almost entirely from trimming with the horizontal tail, and more than 2 g's of incremental normal load factor can be generated at Mach 0.9 and 10,000 feet. Figure 13 also shows the horizontal tail for trim as a function of angle of attack. It is seen that the tail deflection to trim the horizontal canard moments is well below the 25-degree tail deflection limit for the CCV YF-16.

Similar DLC results were obtained from CCV F-4 wind tunnel tests. However, the levels of direct lift were considerably lower than those of the CCV YF-16. This is due to the smaller canard-to-wing area ratio of the CCV F-4 (7.5% as compared to 12% for the CCV YF-16) and the lower combat wing loading of the CCV YF-16. The effectiveness of horizontal canards for DLC application is enhanced by the increased aerodynamic efficiency of the basic YF-16 airframe.

3.2 DLC Using Symmetric Horizontal Canards and Flaperons

The horizontal canard/aft tail configurations discussed in this paper can produce sizeable direct lift increments consistent with the aerodynamic design of the baseline aircraft from which they are derived. Other methods for producing direct lift for CCV fighters have also been investigated. Use of wing trailing edge flaps and spoilers have been considered during fighter CCV wind tunnel tests. The wing trailing edge devices (maneuvering flaps or flaperons) are attractive surfaces for direct lift because they are reasonably uncomplicated from the control system standpoint and act directly on the wing to produce the lift increments. The lift increments generated by trailing edge flaps are reduced since the nose down pitching moments they generate must be trimmed by a down load on the tail. Nevertheless, reasonable levels of DLC are possible with spoilers and flaperons as shown in Figure 14 for the CCV F-4.

Each of the individual methods discussed above for producing direct lift on CCV fighters can be used to generate usable levels of incremental lift or load factor at combat flight conditions. For the configurations studied, maximum DLC capability can be obtained, however, through the blending of the horizontal canard, flaperon and aft tail control surfaces. The combination of these control surfaces can generate the highest level of incremental lift without losses due to trimming. As mentioned above, the flaperons generate a nose down pitching moment and the horizontal canards generate a nose up pitching moment. When trimmed, the simultaneous use of flaperons, horizontal canards and aft tail can result in positive lift on each surface. Within their deflection limitations each surface reinforces the lift generated by the others.

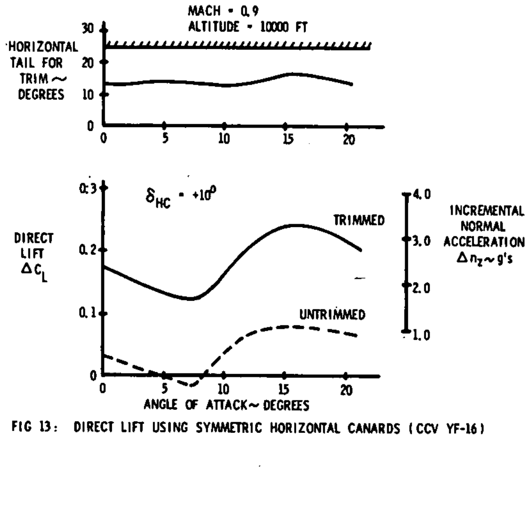

FIG 13: DIRECT LIFT USING SYMMETRIC HORIZONTAL CANARDS (CCV YF-16)

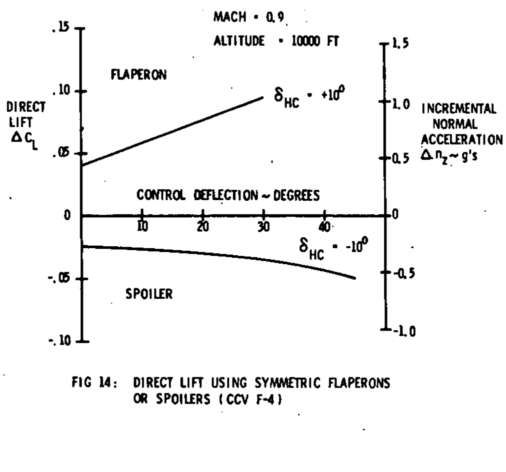

FIG 14: DIRECT LIFT USING SYMMETRIC FLAPERONS OR SPOILERS (CCV F-4)

As shown in Figure 15, substantial improvements in transonic DLC for the CCV YF-16 are possible with the blending of horizontal canards and flaperons. At low angles of attack, over 4.0 incremental g's are available for flight path control.

4. RELAXED STATIC STABILITY AND MANEUVER POLAR ENHANCEMENT

Two additional concepts for improving the maneuver characteristics of fighter aircraft are Relaxed Static Stability (RSS) and Maneuver Polar Enhancement (MPE). A relaxed static stability configuration is one which has a reduced level of bare airframe longitudinal static stability below that of more conventionally designed vehicles. The vehicle relies upon a stability augmentation system (SAS) to provide the stabilizing function. The reduced stability level may be negative in some portions of the flight envelope. Relaxed static stability improves maneuvering performance by reducing trimmed drag and by increasing lift capability. Maneuver Polar Enhancement also depends on lower drag and higher lift for performance benefits. The concept of MPE refers to high angle of attack optimization of the aircraft lift and drag polars through use of fixed or variable geometry aerodynamic surfaces. Active horizontal canards can be used in conjunction with small horizontal tails to implement both of these concepts on CCV fighter designs.

4.1 Relaxed Static Stability with Horizontal Canards

Until recently, it has been standard design practice for fighter aircraft to have inherent aerodynamic longitudinal stability. Trimming a stable configuration requires a download on a conventionally located aft horizontal tail over most of the flight envelope. An unstable configuration, on the other hand, would require an upload on the tail which increases the total lift and may improve the trimmed drag polar. This benefit is the rationale behind the Relaxed Static Stability concept.

CCV studies at the Flight Dynamics Laboratory have indicated that it may be desirable to have both horizontal canards and a small conventionally located horizontal tail on RSS designs. A combination of fore and aft surfaces would allow trim and maneuvering deflections to be optimized to provide the best lift to drag (L/D) characteristics. To take advantage of a trim upload, the tail would probably be the primary trimming surface in subsonic flight, where the aircraft is statically unstable. In supersonic flight, even RSS configurations are normally quite stable because of the large aft shift in aerodynamic center. Supersonically, optimum L/D might be obtained by using the canard as the primary trimming surface to again take advantage of an upload. In all speed regimes, the active canard can contribute to stability augmentation and improvements in longitudinal control power.

The added control power available from the combination of canards and tail would be particularly useful in supersonic flight. The maximum normal load factors achievable on many fighters are constrained supersonically by control power limitations. Reduced supersonic stability levels coupled with increased control power from the canard/tail combination will result in improved supersonic maneuvering capabilities.

The increases in maximum usable load factor gained on the CCV F-4 configuration by means of horizontal canards and relaxed static stability are shown in Figure 16. For this plot, the canarded configuration is 7% less stable subsonically and 3% less stable supersonically than the baseline configuration. As seen in Figure 16, both the subsonic lift limited and the supersonic control limited load factors are substantially increased.

4.2 Maneuver Polar Enhancement with Horizontal Canards

Accepted methods of improving lift and drag characteristics at maneuvering angles of attack include wing leading edge slats and large forebody strakes. Both of these aerodynamic devices use vortex control for obtaining high angle of attack lift to drag improvements. The vortex generated by the strake improves wing flow characteristics while the slats delay separation of the wing leading edge vortex. Close-coupled horizontal canards were reported in Reference 3 to give favorable vortex interference effects and both the CCV F-4 and CCV YF-16 programs verified the usefulness of horizontal canards for Maneuver Polar Enhancement.

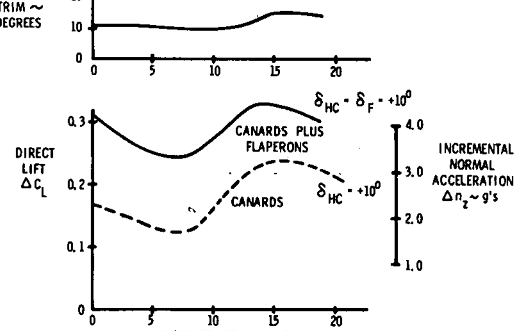

FIG 15: DIRECT LIFT USING SYMMETRIC HORIZONTAL CANARDS AND FLAPERONS (CCV YF-16)

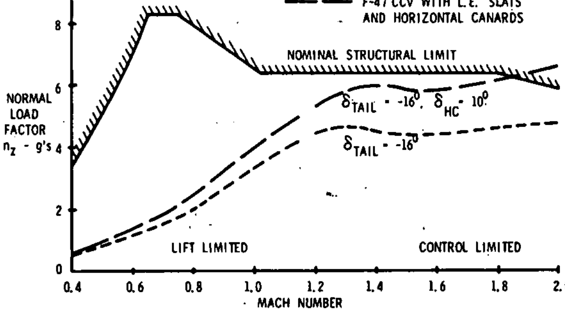

FIG 16: CCV F-4 MAXIMUM USABLE NORMAL LOAD FACTOR

The lift and drag polars for an F-4 aircraft having various amounts of MPE are shown in Figure 17. Deployment of the leading edge slat results in meaningful improvements in the polars at high angles of attack. The addition of horizontal canards to the slatted configuration produced significant further benefits. These improvements are gained at the expense of small profile drag penalties at low angles of attack. The maneuvering drag reduction shown in Figure 17 for the canard/slat configuration translates into an energy maneuverability (Ps) increase of approixmately 300 fps at 3.6 g's, Mach 0.9 and 30,000 feet.

The baseline YF-16 configuration uses leading edge flaps and a blended wing-body strake for Maneuver Polar Enhancement. For the CCV wind tunnel investigations the strake was removed to permit installation of the horizontal canards. The canards were found to be more effective than the strake in providing MPE, except at low angles of attack where there was a small profile drag increase. At a combat Mach number of 0.9, and a 15-degree maneuvering angle of attack, the canarded CCV YF-16 had a 4% higher L/D than the baseline aircraft with wing-body strakes.

The performance benefits to be gained from the employment of both Maneuver Polar Enhancement and Relaxed Static Stability will hopefully be additive. Each potential application of close-coupled horizontal canards has been documented separately from the others. Further study is necessary to determine the compatibility of MPE and RSS. Implementing RSS with horizontal canards will require an active canard, and it has not been demonstrated that an active surface can be fully effective in implementing Maneuver Polar Enhancement. Realizing all the possible benefits from horizontal canards in a CCV fighter design should prove challenging to both aerodynamicists and to control system designers.

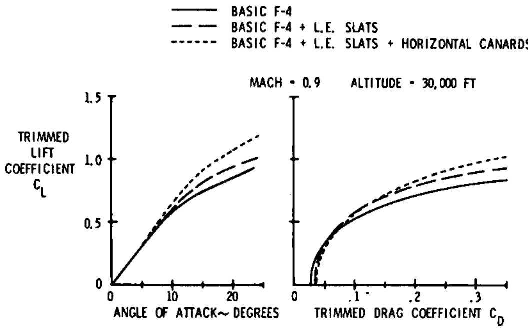

FIG 17: CCV F-4 MANEUVER POLAR ENHANCEMENT

5. USAF FIGHTER CCV PROGRAMS

The Control Configured Vehicle Advanced Development Program Office of the United States Air Force Flight Dynamics Laboratory is conducting test programs to flight validate predicted performance benefits achievable through application of advanced control system design technology. The CCV fighter program is dedicated to the development of advanced control system technology to improve fighter aircraft performance and to stimulate user/builder acceptance of CCV concepts. The scope of the program involves all aspects of design including preliminary analytical studies, wind tunnel testing, flight simulation and flight demonstration. Initial analyses and wind tunnel testing (Ref. 4) were conducted using the Survivable Flight Control System (SFCS) F-4 (Ref. 5). The YF-16 Lightweight Fighter Prototype aircraft was selected to replace the SFCS F-4 as the test aircraft to take advantage of the advanced aerodynamics and control system technology incorporated in the YF-16. The YF-16 was designed to fly at negative static margins and has a quadruply redundant advanced fly-by-wire flight control system.

The CCV YF-16 Program is contracted to The General Dynamics Corporation. After completion of its use in the LWF Program, the baseline YF-16 will be modified to the CCV YF-16 configuration and flight tested following extensive analytical studies, wind tunnel testing, and flight simulations. During the preliminary analysis and wind tunnel test phase, various aerodynamic surfaces, such as close-coupled horizontal canards, twin vertical canards, and flaperons, were investigated to determine their effectiveness for accomplishing the CCV control functions. The CCV YF-16 will be modified with twin vertical canards mounted just aft of the engine inlet and canted out at 30 degrees to provide DSFC. The existing trailing edge flaperons will be mechanized to operate symmetrically for DLC.

As mentioned previously, the baseline YF-16 is designed to operate with negative static margins. The CCV YF-16 will flight test further reductions in static margin to demonstrate maneuvering stability control and to determine performance sensitivity to increasing negative stability. Precision flight path control through direct lift and direct sideforce control will also be flight demonstrated using the trailing edge flaps (DLC) and the vertical canards (DSFC). Blending of the various control modes for maneuver enhancement will be investigated. The starting date of the flight test program is contingent on the outcome of the Air Combat Fighter source selection.

6. CONCLUSIONS

All-movable close-coupled horizontal canards may be effective aerodynamic surfaces for two-axis CCV fighter control. Direct sideforce and direct lift control, relaxed static stability control, and Maneuver Polar Enhancement can be incorporated into fighter designs using horizontal canards. Wind tunnel tests on the CCV YF-16 and CCV F-4 have indicated the differential horizontal canard sideforce potential. Levels of direct sideforce in excess of 1g are possible on the CCV YF-16 at combat conditions without adverse longitudinal or lateral/directional trim requirements. Canard dihedral can increase the sideforce capability of the canards by taking advantage of the resolved components of the normal force on the surfaces. Differential horizontal canards retain their sideforce effectivenes over a large angle of attack range.

Significant direct lift increments are available through symmetric use of close-coupled horizontal canards deflected in conjunction with an aft horizontal tail. When blended with wing flaperons, the DLC capability is enhanced. On the CCV YF-16, incremental normal load factors in excess of 4g's are obtainable at combat flight conditions. Symmetric operation of horizontal canards can also provide additional longitudinal control power for maneuvering and for stability augmentation of relaxed static stability configurations. Improvements in maneuver lift-to-drag characteristics as a result of favorable canard/wing aerodynamic interference are also possible.

The use of horizontal canards introduces additional considerations and complexities for the aircraft designer. In order to realize their maximum benefits, the canards must be carefully sized and positioned. Pilot visibility must be taken into account. Horizontal canards can be expected to reduce directional stability. These considerations, together with the difficulty of blending the various control modes and of insuring concept compatibility, should provide a formidable task for airframe and control system designers. Nevertheless, the projected performance advantages described in this paper should justify the increased design complexity. Therefore, use of horizontal canards should not be overlooked as a viable means of effectively implementing active control technology in CCV fighter designs.

REFERENCES

1. Carlson, E.F., The Boeing Company, An Investigation of the Potential Benefits of Direct Sideforce Control from a Mission Standpoint, 1973, Report No. D180-17508-1.

2. Powell, C.B., General Dynamics Corporation, Analysis of Low Speed Directional Stability Data - 1/9 Scale Fighter Control Configured Vehicle, 1974, Report No. FZM-620-005.

3. Behrbohm, H., Saab Aktiebolag, Basic Low Speed Aerodynamics of the Short-Coupled Canard Configuration of Small Aspect Ratio, 1965, Saab TN60.

4. Strahota, R.A., US Air Force Flight Dynamics Laboratory, Program Summary for Control Configured Vehicle Concepts Applied to Fighter Aircraft, 1974, AFFDL-TR-74-51.

5. Hooker, D.S., et. al., McDonnell Douglas Corporation, Survivable Flight Control System Final Report, 1973, AFFDL-TR-73-105.

Active Control Technology
A Military Aircraft Designer's Viewpoint

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SUMMARY

This paper considers the most likely gains to be obtained by the application of Active Control Technology to small combat aircraft. There are seen to be considerable attractions, although in the Author's view the most significant benefits may turn out to be orientated towards the improved control and design freedom offered by ACT rather than towards revolutionary shapes or greatly increased efficiency or reduced weight. This, however, is not to deny some of the gains attributed to the more publicised aspects of ACT but in some instances they are viewed with some scepticism.

In the design of the A.C.S. itself, it is considered essential that a mechanical back up is avoided in order to produce a more flexible, efficient and safe system, and to this end a suitably progressive system design philosophy must be developed.

Despite some doubts as to the more ambitious claims for ACT, its ultimate adoption is expected for all but the simplest of aircraft, though considerable effort will be required to make this achievement possible.

1. Introduction

This paper is expressly intended to provoke thought and discussion as to how much real advantage might be gained by the exploitation of active control technology (A.C.T.). To do this realistically one has to take note of the practicalities of the situation in which aircraft design and development now finds itself. Something of a plateau has been reached. No dazzling break through in any direction seems probable. Every new proposition has to be subjected to the severest operational and financial scrutiny. And this situation is not likely to change for a considerable time.

It might well seem, from the thoughts set down, that the probable reality is presented as being on a much lower and less enterprising plane than that of the propositions put forward by the ardent advocates of these principles. Nothing, however, could be further from the purpose of this paper than the pouring of cold water over ideas which could certainly lead to significant advances in aircraft design. The questions before us all are - (See Fig. 1)

(a) What are the ultimate goals ?
(b) By what steps should we progress towards the ultimate ?
(c) What are the major hurdles to be cleared ?
(d) What are the practical constraints ?

Expressed in the simplest terms these goals must be more useful aircraft, more easily produced and with a greater certainty of achieving their predicted performance.

The term "more useful" embraces in some measure each of the following:-

(i) Better performance
(ii) Greater versatility
(iii) Greater reliability
(iv) Greater economy.

The term "more easily produced" implies that the aircraft will be

(i) Easier to design
(ii) Easier to manufacture
(iii) Easier to develop for service
(iv) Relatively less costly

A greater certainty of achieving its expected performance would mean that an aircraft would be easier to conceive and easier to "sell" to prospective customers. This goal should not be under-rated. But to achieve it, the certainty that the active control system itself will attain its predicted performance must be at least as great as that for the present generation of non-active systems.

2. The Current Situation

It is usual for system requirements to be the logical outcome of desired aircraft performance characteristics. In this case, however, the aircraft designer has before him a proposition from the system designer to the effect that an active control system would significantly improve aircraft performance and possibly revolutionise aircraft design. Furthermore, states the control system designer, these advanced systems are perfectly logical developments of existing systems.

Though unusual, this is not a unique situation. There is some analogy with the proposition of turbo jet propulsion which was predicted to, and did in fact, revolutionise aircraft performance and led to significant changes in aircraft configuration.

to Improve	or	to Ease
PERFORMANCE		DESIGN
VERSATILITY		PRODUCTION
		DEVELOPMENT

but

to Maintain (or Improve)

RELIABILITY

INITIAL COST

SERVICE COST

FIG.1. ACTIVE CONTROL TECHNOLOGY OBJECTIVES

The analogy remains however somewhat superficial. The aero engine designer could indeed say with some justification that his technological advance made an opportunity which the aircraft designer took up and exploited. In reply, the aircraft designer could say, with equal justification, that the aerodynamics of the high subsonic aircraft, the potential of sweepback and the principles of supersonic flight were already appreciated and that he was waiting for a suitable propulsive system.

This is a circular argument and it is necessary to step back to see how these two technological advances came to be combined and exploited.

The first factor was the occurrence of World War II. There was a clear requirement for improved fighter aircraft performance, and all concerned could agree on the simple goals of increased speed and altitude. Moreover, these potential increases were great - of the order 50% in each case.

The second factor was the post-war defence policy of the nuclear deterrent based upon the unarmed bomber flying too high for the ground defences and too fast for enemy intercepters.

The third factor was the driving force within the air transport industry which, having noted the advances made in the military field, sought to create a revolution in civil transport which at one and the same time would create a bigger market whilst introducing a more profitable aircraft.

Note that the first factor was based to some extent on tenets which never had to be put to the test in World War II, that the second was based on a logical extension of experience with the fast unarmed bomber in that same war, and that the third was to a large extent an act of faith (though whether the "revolutionaries" could be induced to admit this is another matter). Be that as it may, however, the act of faith was based on the fact that some practical demonstration of all the technology involved had taken place.

In the case of the Active Controls proposition we have quite a different set of circumstances. There is neither the compulsion of war nor that of a particular solution for "peace-keeping". Despite a good deal of eloquent pleading, there are not, in the writer's opinion, revolutionary advances to be made in aircraft design and performance.

But there is a revolutionary philosophical step to be taken in that total reliance on an all-electronic control system has to become acceptable. For this to happen, some reasonable confidence in its reliability under all foreseeable circumstances has to be created. The way in which this might be done must exercise our minds as much as any of the technical considerations.

3. Present Claims (See Fig. 2)

3.1 General Remarks

Much has been written and perhaps even more has been said about the advantages which would follow from the wholehearted adoption of A.C.T. The emphasis appears to be on the following:-

(a) The improvement of aerodynamic efficiency by "Relaxed Static Stability" (Proponents of this really mean "Reduced CG Margin" - a quite different thing !)

(b) Improvement in lateral stability and control.

(c) The extension of aircraft controllability by the use of direct lift and side force control - a facility not hitherto practicable.

(d) Modification of wing spanwise load distribution during manoeuvres or due to gusts.

(e) The reduction of structural response in turbulence.

(f) Flutter suppression.

One of the crosses which the aircraft designer has to bear is the ever-increasing task of weighing claim and counter-claim from the specialists. It is to be hoped that no-one will take it amiss if one designer presents some entirely personal views on the present claims for ACT, displays a certain scepticism here and there, interspersed with downright disbelief and finally introduces some further thoughts and suggestions whereby in his particular view ACT can be made to do a useful and probably necessary job.

This is perhaps the point at which to remark upon the term "Control Configured Vehicles" for the aircraft which will appear as a result of A.C.T. In point of fact, the aircraft which we have now are largely configured for controllability. What we are aiming at are "Performance Configured Vehicles" freed from constraints of natural stability and controllability. There is no wish to introduce this new term, however, and the Author has a fervent wish that the other term will die quickly. We have a proposition which demands clear thinking and the coining of terms such as these rarely helps and usually hinders.

3.2 Aerodynamic Efficiency

Two arguments have been put forward based on "Relaxed Static Stability". The first is based on the positioning of the CG relative to the Aerodynamic Centre of the wing/body so as to minimise the total drag, whilst the second puts forward the reduction of tailplane size, and hence weight and drag.

The first argument has to be examined carefully, looking at the whole range of flight

For the Jet Engine

PROPOSED	REALISED
50% MORE SPEED	YES
50% MORE HEIGHT	YES
EQUAL EFFICIENCY	YES
GREATER SIMPLICITY	NO
EASIER MAINTENANCE	?
-	GREATER ECONOMY
-	GREATER RELIABILITY
-	LONGER LIFE

For Active Controls

PROPOSED	REALISABLE
GREATER EFFICIENCY	MAYBE - 10% MAX. IN SPECIAL CASES
BETTER STABILITY	YES
BETTER SPEED MANOEUVRE	MAYBE
LESS STRUCTURE WEIGHT	DOUBTFUL
RADICALLY NEW SHAPES	DOUBTFUL
BETTER AIMING	YES
-	NICER CONTROL
-	DESIGN EASEMENT
-	EASIER FLIGHT DEVELOPMENT

FIG.2. HOPES - v - REALISATIONS

conditions for which the aircraft has to be designed. This introduces the opportunity to deal with another and yet more dangerous misnomer, i.e. "Off-Design Conditions". There are no such conditions. If an aircraft is likely to encounter such conditions they have to be designed for. If we take however two of the simplest possible cases, for instance the high subsonic high altitude unarmed bomber, and the supersonic bomber as proposed in the mid-1950s we have two single-mission types, each intended to spend almost the whole of its time in one flight condition. Some observations can be made based on the Author's personal experience.

In the first case (a vehicle with a cruise speed approaching 0.9M and a gross weight of 150/200,000 lb) it was possible by adopting a T-tail layout, to find a tailplane size and tail volume which made it possible to raise the nosewheel, and to have adequate stability margins with mid-CG coincident with the wing/body Aerodynamic Centre. The tailplane normally carried a small down-load to balance a small resultant nose-down C_{M_O}. This C_{M_O} was the resultant of three contributions, two of which (wing washout and wing/body setting) were substantial, roughly equal, and opposite. The third component came from wing camber. It is necessary to go into this little bit of detail to show how, in a real case of design, tail load is more than a matter of CG/Aerodynamic Centre offset (a few generalised graphs can be misleading). In the event, and to some extent by accident, that small down-load at the tail proved to be just about the optimum for cruise efficiency. As to stability and trim, it was possible to design for a CG range of $\pm$ .04 $\bar{c}$ which meant that at aft CG the problem of control sensitivity in terms of stick travel per 'g' or per knot of trimmed speed seemed tractable. (But more of this later). (See Fig.3)

The second case was by no means so tractable. Here we were aiming at a vehicle with a cruising speed of 2.5M and a gross weight of 200/250,000 lb. Tailed, tailless and canard designs were exhaustively examined. Always, the chief stumbling block was CG position and the ability to raise the nosewheel, with cruising efficiency a close second. Had it been acceptable to proceed with a canard design in which subsonic stability was entirely artificial a gain of some 10% in cruise efficiency could have been achieved.

These, however, were two very simple cases. If the subsonic aircraft had had to do a significant part of its mission at high speed low level, a compromise CG position would have been necessary. Such compromise was later found possible with a smaller aircraft of about 50,000 lb gross weight. A similar compromise would have been necessary if the supersonic aircraft had had to spend significant time at subsonic speed but the Author has no first-hand experience.

With the sub/supersonic aircraft the situation is nothing like so clear, but some experiences in the design of small to medium sized aircraft (20,000 to 60,000 lb gross weight) can be stated. Much depends on the C_L Max target. For aircraft with C_L Max up to 2.0 there is a steady gain in cruise efficiency as the CG is moved aft to the point where the CG margin subsonic is about zero - this with the tailplane sized by nosewheel raising. Should this aircraft however be expected to spend a significant time (3 mins say) at supersonic speed and elevated 'g' the best CG position could be at least 5% $\bar{c}$ further aft. The difference between these two cases is important. In the first, with the CG margin zero, there would be enough natural dynamic stability for unassisted control in emergency. In the second case there would probably be not. (See Fig. 4)

If we now increase the C_L Max to 3.0 - quite a practicable figure with blown flaps, we find that the tailplane size needed to raise the nosewheel with a reasonable CG range enables one to have the CG midway between the sub and supersonic positions of the wing/body Aerodynamic Centre and to effect a very good compromise in drag at subsonic and supersonic speeds over a good range of 'g' whilst retaining inherent stability. (See Fig. 5)

It has been necessary to dwell on this question of longitudinal stability to show hew in practical design cases the situation is complex and not capable of assessment from simple pictures.

3.3 Improved Lateral Stability

This is one of the areas in which significant advantages can be gained. There are three reasons for this. Firstly, weathercock stability is not a feature which it is easy to obtain naturally throughout the speed range. One faces the problem of ever-increasing instability of the wing/body combination with Mach No. accompanied by (at supersonic speeds) an ever decreasing effectiveness of the vertical tail. To add to the difficulty there have been many instances in which tests at wind tunnel scale have produced results very different from those discovered at full scale. In no instance known to the author would it have been necessary to have more than a few degrees of travel either way on a suitably activated rudder to have produced all the stability desired.

The second factor which bedevils the attempts of the designer to achieve a good compromise in aircraft configuration is the interaction between the aerodynamic effects of rolling, yawing and sideslipping. Rolling moment due to sideslip is a difficult enough phenomenon to cater for and creates enough difficulty in compromising between long-term (Spiral) or short-term (Dutch Roll) effects without having such effects as yawing moment due to rate of roll to contend with. This latter effect is due to the effects of the sidewash at a central fin resulting from the anti-symmetric loading on the wings producing a significant de-stabilising effect on the short-term motion. Often the only way to avoid this is to adopt a twin-finned layout with the spanwise positioning of the fins constrained to null or nearly null sidewash points and to accept the inferior efficiency and extra weight of the vertical surfaces thereby incurred.

Thirdly there is the age-old difficulty of getting enough natural damping from any acceptably sized vertical tail surfaces. In earlier days the vertical tail surface required to balance an outer engine failure or to give adequate yawing moment at low speed on the runway was enough in itself to provide acceptable damping in yaw at the modest speeds and heights of the day.

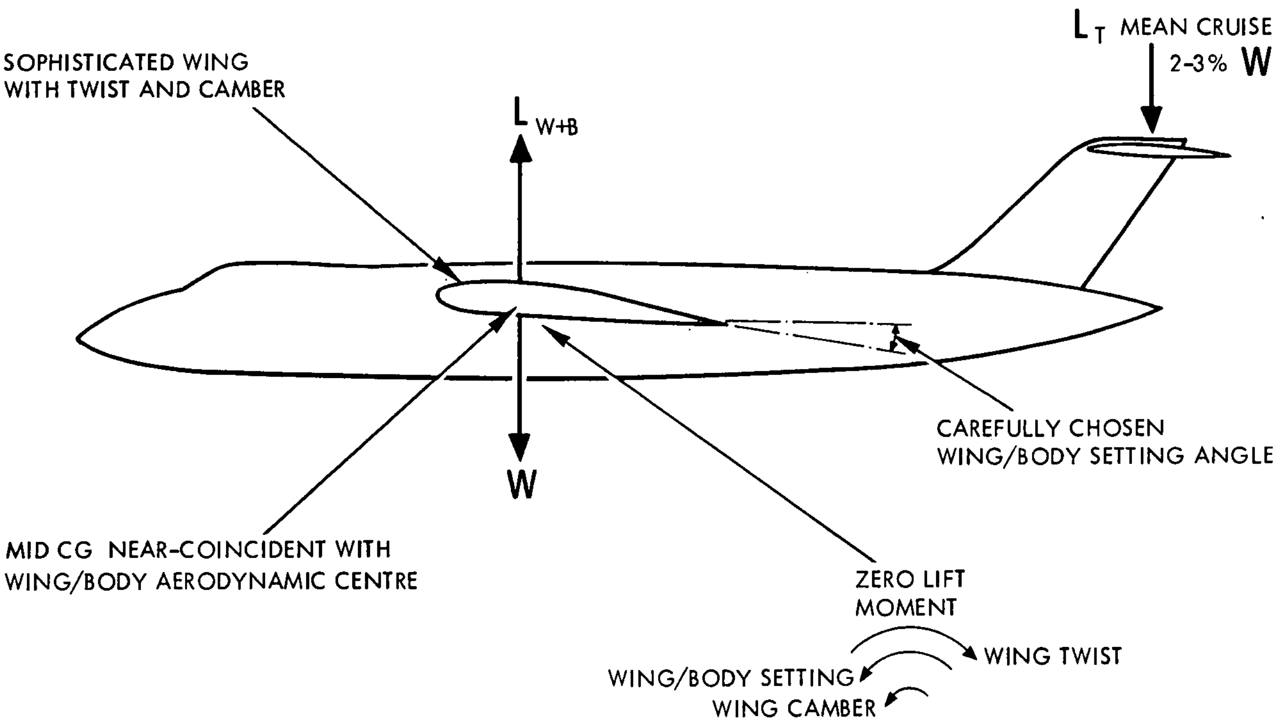


FIG.3. COMPLEX TRIM CONSIDERATIONS – HIGH ALTITUDE SUBSONIC BOMBER

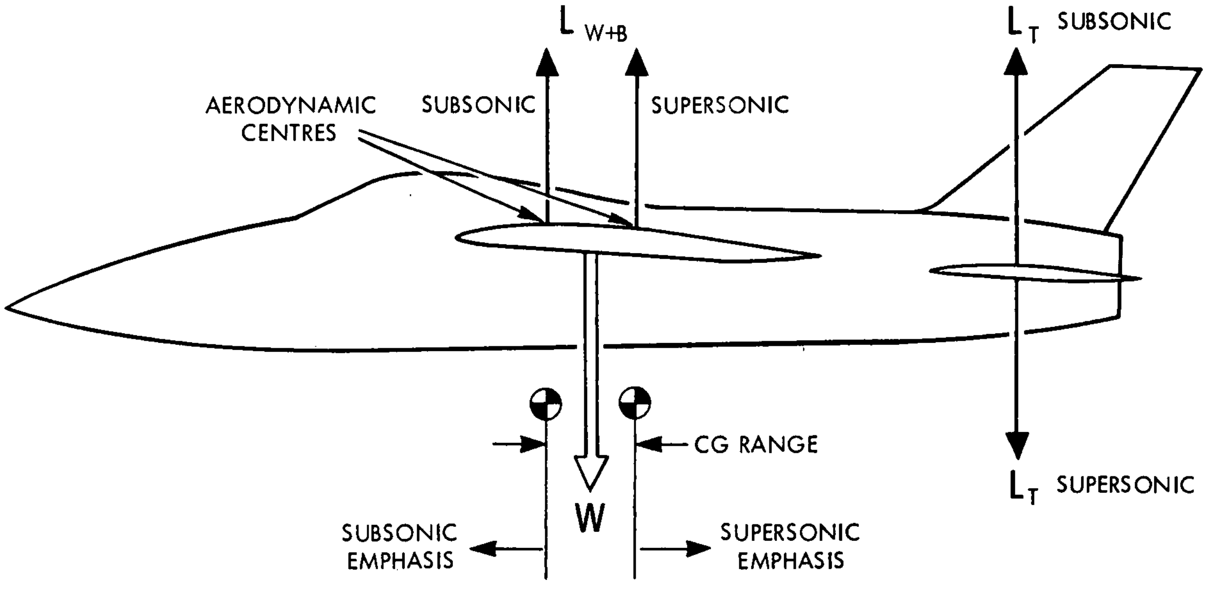
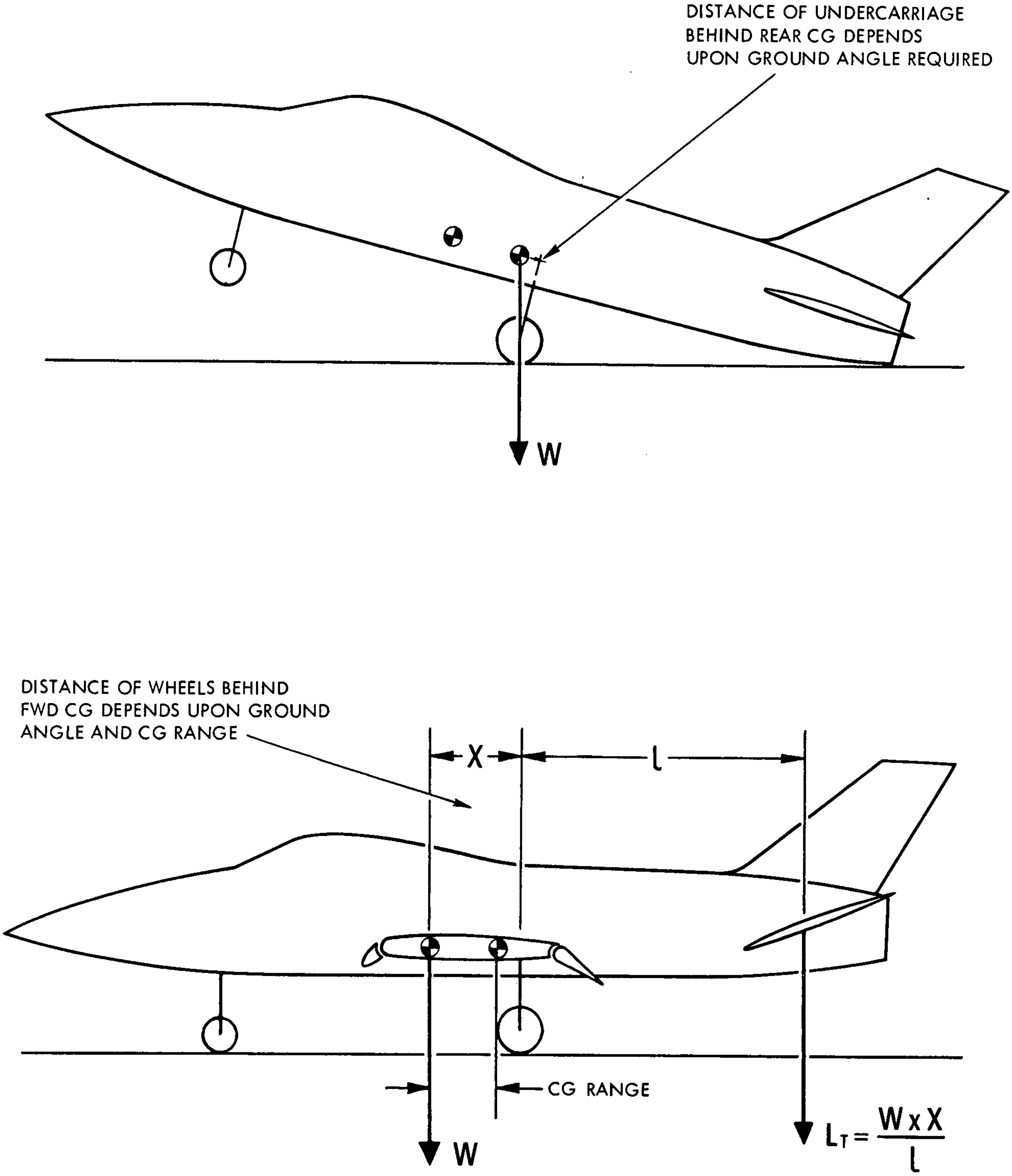


FIG.4. TRIM CONSIDERATIONS – TYPICAL SUB/SUPERSONIC COMBAT AIRCRAFT



TO RAISE NOSEWHEEL

$L_T = \frac{1}{2}\rho V^2_{\text{ROTATION}} \times S_T \times C_{LT}$

THEREFORE

TAILPLANE AREA S_T REQUIRED

VARIES DIRECTLY WITH WING C_L MAX
VARIES INVERSELY WITH TAILPLANE C_L MAX

FIG.5. TAILPLANE SIZING – GENERAL PURPOSE COMBAT AIRCRAFT

Nowadays and for the future we have the problem of providing a suitable combination of damping and frequency over a wide band of operating conditions.

3.4 Direct Lift and Sideforce Control

3.4(a) Direct Lift Control

Whilst the case for this on a large aircraft rests largely on the elimination of the lag in pitch, a similar argument in the respect of the typically sized military combat aircraft does not seem tenable. It could well be however that this facility would make a valuable contribution to the precision of touching down at a given point. The airstrip length needed is the ground roll plus the touch-down error allowance which can amount to hundreds of feet. For the STOL combat aircraft with C_L Max 2.5/3.0 the elimination of touch-down error is as significant as a substantial increase in maximum lift coefficient.

It is to this end, possibly by coupling the aircraft control system into an automatic or pilot-aided landing system, that the development of a direct lift facility for the small/medium combat aircraft should be directed.

3.4(b) Direct Side Force Control

None of the propositions seen to date by the Author seem convincing. In the combat aircraft field they have mostly been directed towards lining up the aircraft onto an aiming point. As the use of the area of the aircraft in side elevation or "keel surface" is such an inefficient means of generating aerodynamic force, it is certainly suited only for the production of small correcting forces. Calculations done for an existing strike/interdicter aircraft in the 50,000 lb class show that, with the hypothetical addition of a "vertical canard" the lateral acceleration would be limited to about ½g by fin or "canard" strength. In certain circumstances, rudder hinge moments could impose an even greater limitation.

Quite apart from this, one questions the soundness of the philosophy. Is it really better to try to translate the whole aircraft sideways for 100 ft or so than to make the relatively trivial angular change in the aircraft velocity vector ? It would seem that this proposed "side-stepping" facility would be made irrelevant by a control system which enabled the pilot to make small angular changes in any given direction, freed from unwanted changes due to aerodynamic or inertia coupling. This is a point which will be dealt with later.

3.5 Span Load Modification

In considering this feature we have to bear in mind two opposing factors, lift dependent drag and wing root bending moment. Any system which relieves wing root bending moment by shedding load from the outer wing incurs a drag penalty. Such a system may suit a large transport aircraft but it can have no relevance to a combat aircraft which is required to sustain high 'g'.

One might indeed turn the proposition around in the context of the combat aircraft and consider span load maintenance instead. Here we come to the manoeuvre flap. Much has been said about the benefits due to the deflection of a leading edge flap at high 'g'. Little has been said about the evils of the deflected flap at low 'g' or the evils of the transient pitching moments during flap deflection at roughly constant incidence at any appreciable Mach No. above about 0.7. The first group of evils is characterised by large drag penalties - up to 30% of the total aircraft drag at moderate C_L. The evils of the second group are much harder to predict, incurring changes of trim large enough to pose stressing as well as piloting problems. By the nature of things they are impossible to calculate, and results at wind tunnel scale are liable to be extremely misleading, being directly affected by boundary layer thickness and shock-wave patterns.

In the author's view the success of any manoeuvre flap system (leading or trailing edge) will depend upon a control system which can match flap deflection precisely to wing incidence, C_L and Mach No.9. In other words this could be a perfect application for an Active Control System.

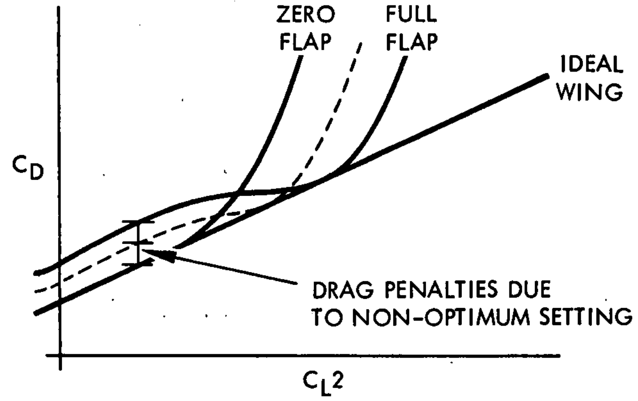


FIG.6. THE NEED FOR AN ACTIVE MANOEUVRE FLAP CONTROL

3.6 Reduction of Structural Response

Under this heading one could consider the reduction of structural response in certain critical modes to turbulence or control inputs along with the reduction of wing root bending stresses due to turbulence with the object of improving fatigue life or perhaps ride comfort.

The first proposition could only be applied to large flexible aircraft, and the Author has no direct experience beyond that for the high altitude bomber of the 1950s with gross weight up to 200,000 lb and Aspect Ratio about 5.0. These aircraft had significantly greater longitudinal flexibility than typical airliners of today and considerably less wing flexibility. Potential problems were indeed foreseen but none actually arose.

It could well be said to be an ill-advised approach altogether. To this designer at any rate the problems of developing a system purely for control will, for a generation, be enough. To conceive a system which at the same time would work its will on the structure itself seems to be asking too much. With small stiff aircraft there would in any case be the problems of providing rapid enough response in the actuators. With large or small aircraft one would have to tread a very wary path past the flutter pitfalls.

It could be however that with an existing aircraft for which the relevant characteristics could be established as distinct from a new design, that ACT techniques could be used to deal with specific defined problems and this is how ACT might eventually become to be accepted.

Turning to the wing root bending moment situation for small combat aircraft, one is faced with a totally different design situation. In the early days, low-level turbulence was thought to be the predominant factor in airframe life but later experience has shown that high 'g' manoeuvring is the killer. If one has to go for minimising lift dependent drag one has to live with the corresponding wing root stresses.

3.7 Summary

One can summarise this review of present claims for the advantages of ACT as follows:-

Relaxed Static Stability and Enhanced Efficiency

In some special circumstances this holds good. It does not for a high subsonic aircraft with a modest $C_{L\,Max}$ - say 1.5/2.0 for which the tailplane areas needed either raise the nosewheel or to cope with the changes in trim and stability for a CG range of about $0.08\bar{c}$ are about equal. It clearly does for an aircraft which does almost the whole of its mission at supersonic speed. It may well do for a combat aircraft with a modest $C_{L\,Max}$ and which is required to expend a good deal of fuel at supersonic speeds either in pursuit or in combat manoeuvres; in this case an increase in top speed or in excess power for manoeuvre may be as important as fuel consumption. It does not hold for a sub/supersonic combat aircraft with a high $C_{L\,Max}$ - say 2.5/3.0 for which the tailplane size is set by the nosewheel raising requirement.

Improved Lateral Stability

This is an important area in which ACT can play a considerable part.

Direct Lift and Sideforce Control

Direct lift control might well make a significant contribution to precision landing utilising mobile landing aids. This could lead to important advances in the deployment of airforces.

Direct sideforce control introduces complications in aircraft layout and systems which seem out of proportion to the small potential gains.

Span Load Modification

Whilst this principle could be used to relieve wing root bending stresses during manoeuvres on a large aircraft, provided that the attendant drag increase can be accepted, no such possibility exists for the combat aircraft. The situation is rather the other way round in that the span loading has to be maintained and the stresses accepted in the interest of minimising drag; this is one of the principles of the manoeuvre flap and ACT may well be the key to its successful development.

Reduction of Structural Response

This proposition is unlikely to be exploited at the design stage. There could be opportunity to apply ACT for special purposes on large existing aircraft with known characteristics.

4. Some Alternative Design Objectives

4.1 Precision of Control

It has been notoriously difficult to obtain suitable stick gearings to match the ever-widening speed range of aircraft and the progressively more demanding requirements for precise control under difficult flight conditions. Individual histories are rarely written up and those that are tend to remain buried in the archives of the firm concerned. Two instances can be quoted in each of which a mechanical solution was only barely possible.

The first of these concerned an aircraft required to cruise for long periods at 0.85/0.9M at altitude and with a maximum design speed of about 400 KTS EAS. This aircraft was designed on the principles mentioned in 3.2 and, despite its small tail volume of about 0.27 a considerable tailplane sensitivity problem at 400 KTS EAS was expected. The form of the stick/tailplane gearing is shown in Fig. 7.

Design tailplane angle was zero in the altitude cruise condition, and the low geared section of the gearing curve was biassed towards positive tailplane angle to cover the high EAS condition. The control surface was fully powered and stick force was reduced to zero at the appropriate stick position by means of the trimmer.

Test flying on the first prototype showed that the tailplane angle to trim at altitude was 1.0 to 1.5 degrees more negative than anticipated. Because of the bias of the gearing curve towards positive angles, the slope at tailplane angle -1.5 degrees was about 30% steeper than intended and this led to considerable difficulties in maintaining a steady trimmed speed at altitude. The input circuit was re-rigged (sacrificing some tailplane travel) to bring the stick central and the difficulty disappeared. Later test flying fortunately showed that there was no need to strive for the "lost" tailplane travel.

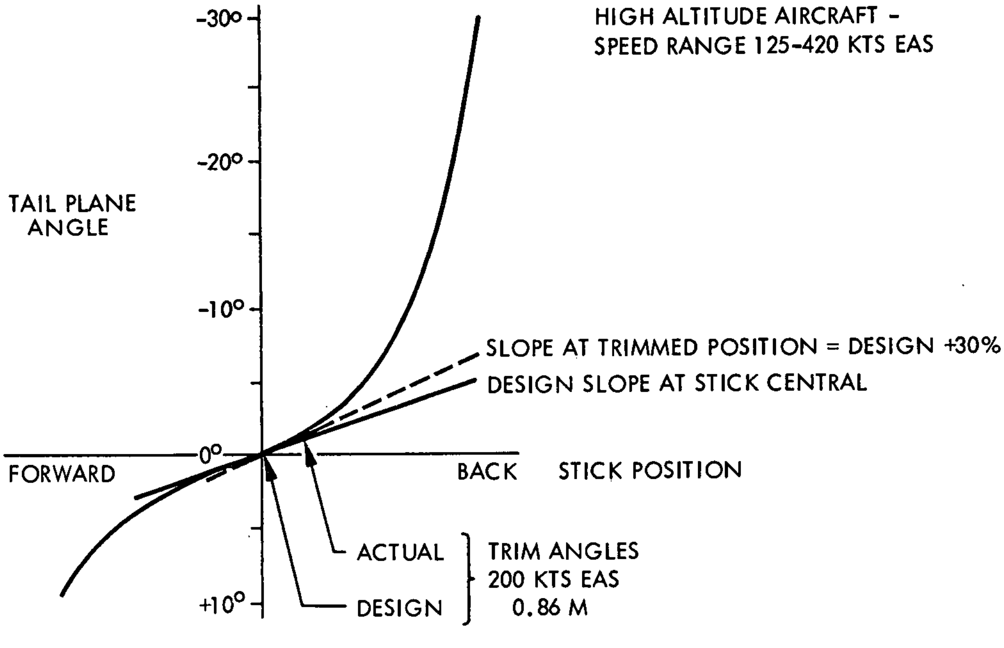


FIG.7. AN ACTUAL GEARING PROBLEM

The next instance concerned a medium-sized low-level attack aircraft of 40,000/50,000 lb gross weight. The speed range had now widened from 125/400 KTS EAS to 125/660 KTS EAS and the dynamic pressure range had thereby almost doubled. Fortunately several additional factors were favourable. For example the fuselage structure was much stiffer, the input circuits were shorter and advances in the design of control surface actuators and servo valves had led to a notable improvement in the precision of that link in the chain. The design of a stick/tailplane gearing curve was also eased by the adoption of a variable datum input system whereby the trimmed stick position was always central.

Despite these improvements and despite the fact that the tail volume was no larger than in the previous instance, the combination of the greater speed range and the more arduous requirements for precise control in low-level flight made it nearly impossible to solve the problem mechanically. In the event it was necessary to produce a series of about six cams before arriving at a design which gave a suitable gearing variation without undue frictional variations and with adequate total tailplane travel. In the end it was the effect of the pitch autostabiliser which turned a marginal system into a completely acceptable one. By opposing the initial response due to tailplane angle, the autostabiliser motion effectively reduced the tailplane gearing for those small but vitally important displacements about the mean position as shown in Fig. 8.

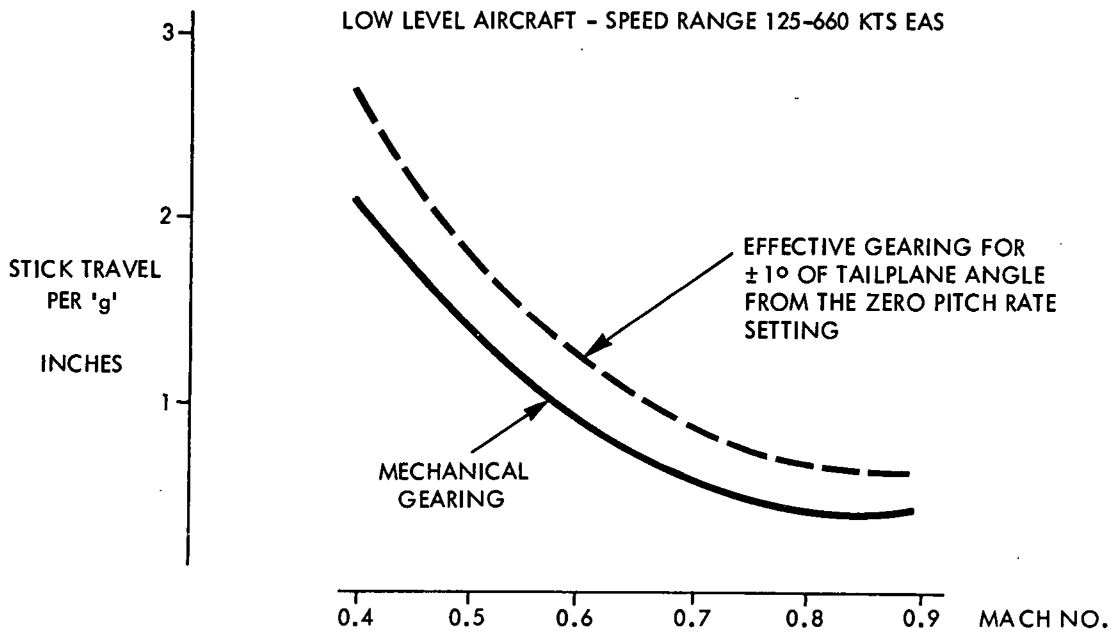


FIG.8. EFFECT OF AUTOSTABILISER ON STICK GEARING

In addition to these fundamental problems of gearing ratios, a number of purely mechanical problems arise. With the aircraft of the first example input circuit friction and the relative motion between the rigid rods and the long flexible fuselage itself led to imprecision of control. With the aircraft of the second example, many frictional problems could be attributed to the deformation of brackets, trunnions, frames etc. Stick centring spring rates, break-out forces and feel gradients were equally productive of mechanical problems. In total, in addition to the work entailed in producing the various gearing cams, several designs of self-centring spring box had to be manufactured and tried out, together with three designs of artificial feel unit. About 10,000 hrs of laboratory investigation into the causes of input circuit friction or harshness and ways of preventing these things from being inadvertently built in on the shop floor were also entailed.

Here again, as with the subject of static stability and trim, it has been necessary to go into some detail with real cases. The mechanical problems are seen to be great. There is every probability that, as the requirements become more severe, or the deficiencies of present control systems become more recognised, it will become impossible to provide mechanical solutions to the input circuit problem.

4.2 Control Harmonisation and Blending

The difficulty of organising suitable mechanical gearings for control about the individual axes has already been made clear. Harmonisation of the pitch and roll controls is a further complication. If the designer could have a greater command over the gearing of each by the use of an electrical link then his chances of getting a well harmonised set of controls would be so muchthe greater.

The aerodynamic cross-coupling effects have already been mentioned. Inertia cross-coupling can be no less embarrassing; in some ways it can be positively inhibiting. For instance there are often good reasons for adopting a high tailed layout which gives overall advantages in aerodynamic efficiency and structure weight and keeps the tail clear of the wake of wing-mounted stores. The other side of the coin is the downward inclination of the longitudinal inertia axis and its deleterious effects on lateral stability and weapon-aiming.

Mechanical interconnections to deal with the aerodynamic cross-couplings seem barely practicable. Active systems alone can deal with the inertia effects.

It would seem that the use of a full-time active system could make an important break-through in this particular area of aircraft control whilst also giving the designer some important new options as to aircraft layout.

4.3 Other Design Freedoms

There are several inhibitions on aircraft design which could be removed if the need for inherent stability were to be removed. For instance -

(a) Much better use could be made of internal volume for full tankage. Every gallon of fuel carried internally in a small to medium-sized combat aircraft is worth nearly two gallons carried externally.

(b) Guns and ammunition tanks could be sited with greater regard for aerodynamic considerations, intake interference, ammunition capacity, servicing and re-arming.

(c) The design of the outer wing pylon on swept wing combat aircraft could be considerably improved. There is often a serious conflict here between the placing of the store with relation to the CG of the aircraft, the engineering of an efficient pylon structure incorporating an ejector release unit and the achievement of good aerodynamics. This problem is illustrated in simplified fashion in Fig.9.

(d) Typical short-range missiles, the stock-in-trade of the high agility fighter, might be carried with integral launcher tubes in the nose of the aircraft giving a notable increase in the all-important thrust/drag margin.

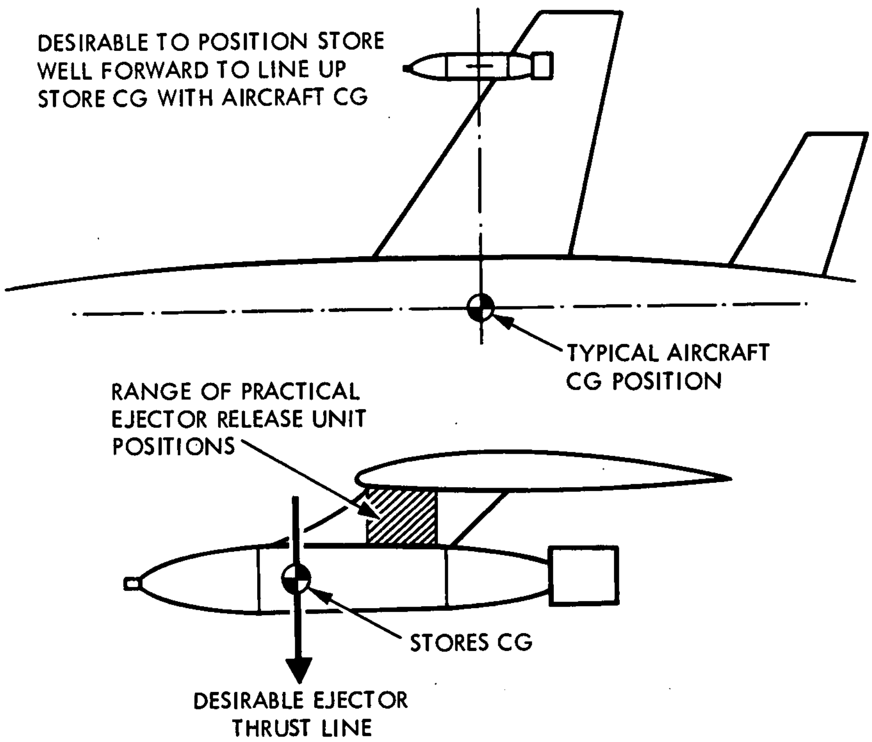


FIG.9. THE OUTER WING PYLON DESIGN PROBLEM
(EXAGGERATED FOR CLARITY)

5. Some Technical Considerations

5.1 Aerodynamic Considerations

When planning this paper it had been the intention to make a critical review of the accuracy and consistency with which the important technical facts could be established. For instance, how practicable is it to provide multiplex sensing of incidence, sideslip, static and pitot pressure with sufficient self-consistency to avoid misleading "error" signals between multiplex lanes ? Or again, can the aerodynamic and dynamic characteristics of a design be estimated from theory and/or experiment with sufficient accuracy to enable a control system rig to be set up at an early stage in the design ?

It proved difficult to establish these facts. There are clearly some fundamental difficulties in establishing rotary derivatives for the aircraft as a whole. There is clearly work to be done in extending our capability to establish the basic force and moment characteristics to high angles of incidence and sideslip before we can properly exploit the advantage of ACT.

5.2 Structural Considerations

In some ways the uncertainties of forming a basis for structural design will be eased by ACT, particularly for highly manoeuvrable combat aircraft. It could well be that the suppression of unwanted excursions in local sideslip and incidence at the tail will contribute to a reduction in actual loads, or in the "factors of ignorance" which have to be applied.

On the other hand, the adoption of a radically different form of control, whereby large control angles are liable to be applied initially followed by large angles in the opposite sense to check the motion will introduce loading conditions quite different from those which we have traditionally designed for or experienced. This needs serious consideration.

5.3 The Installation of Sensors

Gyro and accelerometer installations give no particular grounds for concern, not even for quadruplex installations. Multiplex A.D.D. and pitot or static pressure sensor installations are quite another matter. There are notorious difficulties in getting even a single satisfactory static point on a fuselage. There is the fundamental problem of static pressure sensing in the transonic speed region. There are problems associated with high incidence and sideslip angles which are particular to specific aircraft configurations.

These considerations lead towards the idea of a system which has a very simple, gyro/accelerometer based "core" onto which are grafted such additional air data based facilities as are deemed necessary. In the event of malfunction in these additional facilities the system would progressively revert towards its central "core".

6. Some Targets to bear in mind

6.1 Weight

No hard evidence of the comparative weights of production Active Control and Mechanical Systems is available. This should cause no surprise. Several useful speculative papers have been produced and some data are available from prototype or development systems. Studying the evidence available today and making some hopeful allowances for weight reduction with system refinement one arrives at Fig.10. It seems that for the typical two-seat fighter or interdicter aircraft of 40,000/50,000 lb gross weight there will be little difference between the two types of system. It seems quite likely that the large aircraft would stand to gain by about 10% and that the small single seat aircraft would inevitably have to put up with a weight increase. It is clear, however, from the figures that the weight differences need not be decisive factors. Taking the small aircraft in particular, should it be found that the advantages in control system performance were great enough to justify taking the big step in the direction of ACT a weight difference of 20 or 30 lb would hardly sway the argument.

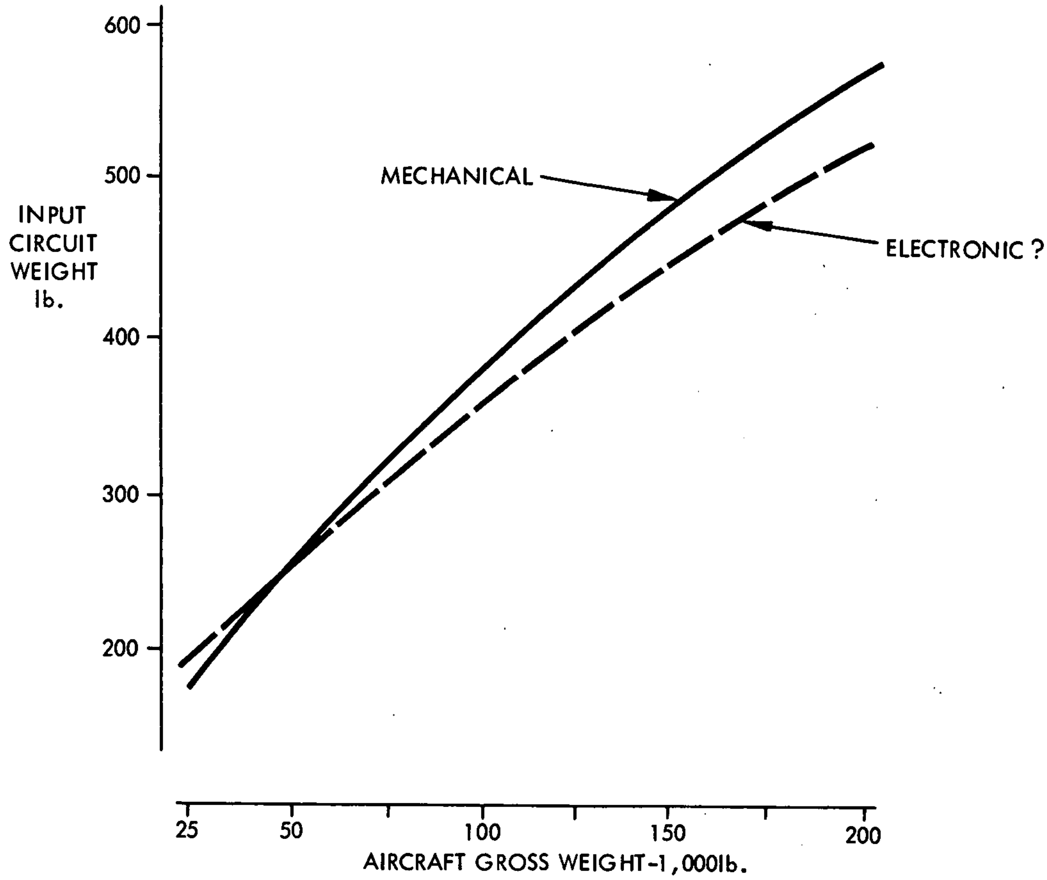


FIG.10. INPUT CIRCUIT WEIGHT COMPARISON

6.2 Reliability

Present speculation in terms of failure probability per sortie or per flying hour seems not at all productive and scarcely even relevant. There seems to be no possibility of amassing enough flight time to establish figures. The best we can aim for in military aircraft is two-failure survivability. How the military experience will be translated eventually into the civil design philosophy no-one can foresee at this time.

6.3 Cost

In the one example available to the Author, i.e. the medium-sized combat aircraft of para.4.1 for which high-speed low-level flying qualities were of extreme importance, the initial design cost of the control input system and its installational details was about 5% of the design effort needed to get the first prototype into the air. Development and refinements took up a further effort of roughly equal magnitude. Some of that development need not be incurred on a new design if previous experience could be drawn upon. As a target therefore the Active Control System designers should attempt to keep design costs to (say) 5% or 7½% of initial prototype design costs or be prepared to put a very strong case for any figure substantially higher.

A production cost target could be derived from the corresponding mechanical input system weights using man hours per lb of assembled aircraft weight as a suitable measure for assembly cost. A "complexity factor" for high quality flying control assembly as compared with average assembly could be between 2.0 and 2.5. Thus the cost of ACS input systems for aircraft in the 25,000/50,000 lb weight bracket must be measured against that of 150/200 lb of high quality mechanical parts and their assembly. It seems unlikely on current evidence that ACS costs could be brought down to such figures.

One has the impression, however, from informal discussions that there could be a significant exchange rate between initial design cost, system production cost and overall aircraft development cost. If the characteristics of an active control system can be changed as easily as is sometimes claimed, there could indeed be the possibility of lower aircraft development costs outweighing higher control system unit costs. It must be remembered, however, that the cost of spares and servicing tends to be a direct function of initial cost.

Nothing like enough has been made available, or even said, on this subject of costs. It is of the utmost importance to the aircraft designer and he would like to be able to make at least an initial appraisal.

7. Conclusions

Despite some doubts as to the more ambitious claims for Active Control Technology, the Author sees its ultimate general adoption for all but the simplest of aircraft.

Its adoption might well be considered the final stage in the process of providing the pilot with adequate control. In the very early days it was possible to have controlled flight by such simple means as warping the wings by direct mechanical linkage. Increase of speed led to configurations constrained to some extent by the need for inherent stability or controllability and flap type control surfaces, the design of which was completely dominated by the needs to keep maximum hinge moment under control and yet to trim with zero hinge moment. This necessitated much flight development.

The introduction of the fully powered control freed the aircraft designer from hinge moment tyranny, reduced drag,and improved control characteristics for about the same weight as before. By and large, the weight of the powered actuators was offset by the elimination of mass balance. In apparently modest but nonetheless significant ways the fully powered control gave the designer greater freedom with the aircraft configuration.

The introduction of active controls could take this process a stage further, freeing the designer from mechanical tyranny and freeing him from the constraints associated with control surface position geared irrevocably to stick position. It would also give significant extra freedoms with regard to aircraft configuration and mass disposition.

A stage to avoid at all costs is the combination of electronic control and mechanical back-up. The unfortunate experiences with the introduction of powered controls should have been an object lesson to us all. In the author's experience nearly every nasty incident in the transition to fully-powered controls was connected with the manual reversion system. Let us not drop any further into that trap than we have done already.

How should we proceed to exploit the potential of ACT ? The author detects here and there the feeling that the basic technology is all available and all that is needed is simply to apply it to the next generation of aircraft. Nothing could be further from the truth. Much basic knowledge has still to be gained; much basic philosophy needs to be developed.

It was the intention for instance, when planning this paper to devote some considerable time to reviewing the accuracy with which some of the more important aerodynamic characteristics could be estimated and the degree of consistency with which such important things as incidence, sideslip and static pressure could be sensed from multiplex sources. It did not seem possible to get this information. Perhaps more energy should have been put into the search. But, if ACT

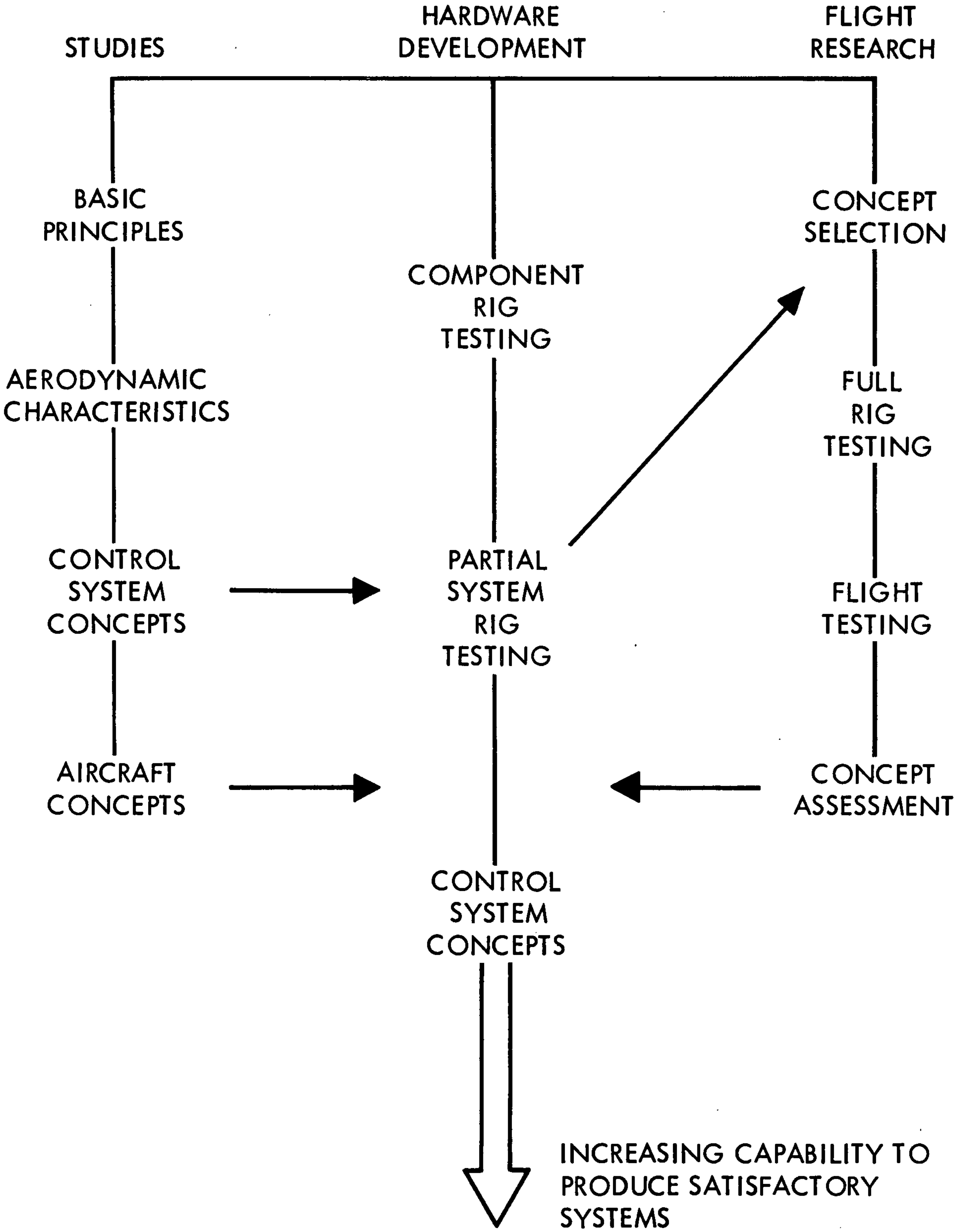


FIG.11. THE PATTERN FOR A.C.T. DEVELOPMENT

is to be exploited properly, the information and its import must become widely available and easily obtainable.

Much good work has been done. Of this there is no doubt. The decisive step achieved by the YF.16 team is something for which we should all be very grateful. But "one swallow does not make a summer" - all aircraft are different with different duties and different environments.

We must therefore proceed with a three-pronged attack on this front.

1. There must be continuing study into the application of the technology. This is predominantly a job for the aircraft designer but he cannot do it in isolation.

2. There must be continuing development of suitable hardware. This is predominantly a job for the equipment manufacturer, but he cannot do it in isolation either because it has to be <u>suitable</u> equipment.

3. Systems must be conceived and <u>flight-tested</u>. Only the latter imposes that collective discipline, across the board, which is essential to success. Only the latter imposes that collective discipline, across the board, which is essential to success. Only the latter puts these systems into their true environment.

An outline of this scheme for continued progress appears on Figure 11.

Acknowledgment

The Author is indebted to Hawker Siddeley Aviation Ltd. for permission to present this paper. The views put forward are, however, entirely his own.

HANDLING QUALITY CRITERIA DEVELOPMENT FOR TRANSPORT AIRCRAFT WITH FLY-BY-WIRE PRIMARY FLIGHT CONTROL SYSTEMS

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SUMMARY

The introduction of fly-by-wire primary flight control systems in future transport aircraft, in some cases including direct-lift-control, makes it highly desirable to initiate further studies into handling quality criteria for future guidance in system design. The handling quality criteria for such aircraft must be based on parameters which describe the combination of the aircraft and its closed-loop flight control system.

Approach flight simulation and compensatory tracking, performed on a three degrees of freedom flight simulator as applied to a conceptual jet transport developed around the relaxed static stability concept, is described. The stiffness of the pitch attitude system and the effectiveness of the direct-lift-control-system were varied.

The following topics are discussed:
- required direct-lift-control-effectiveness for an aircraft with a very low value of the normal acceleration sensitivity,
- required bandwidth of the pitch attitude control system for an aircraft with a value of the normal acceleration sensitivity typical for the present-day jumbo aircraft,
- pilot opinion on the absence of a stable stick force/(deflection) versus airspeed gradient for pitch-stabilized aircraft,
- results of compensatory tracking experiments,
- evaluation of the applicability of the C^*-criterion for the configurations tested.

1 INTRODUCTION

Advanced flight control techniques will be employed in the new generation of high subsonic cruise transports (Ref. 1) as well as the next generation of very large transport aircraft (Ref. 2). As a guide to flight control system design for these aircraft, new handling quality criteria are needed.

This paper describes a part of a research program (Ref. 3) aimed at the formulation of longitudinal flying qualities criteria for CTOL transport aircraft equipped with electrical primary flight control systems including direct-lift-control (DLC) when aircraft with a low value of the normal acceleration sensitivity are considered.

Two developments in the design of future transport aircraft are responsible for renewed research into handling qualities for approach and landing. Firstly, the emergence of Control Configured Vehicle (CCV) technology, leading to aircraft requiring full time artificial stability, may be noted. Secondly, increased wing loading, leading to low values of normal acceleration sensitivity, requires DLC to maintain adequate flight path control capability. A few remarks concerning these two developments are in order here.

Development of CCV-technology directed towards optimizing an aircraft configuration for a given mission requirement, including controls and control systems capabilities as primary design variables, is gaining momentum in the last couple of years. In reference 1 a comprehensive survey of the status of this technology in 1973 is presented. The most promising concept for improving the efficiency of flight is that of "relaxed static stability". Sizable improvements in cruise efficiency and reduced airplane size for a given mission of transport aircraft are indicated. A significant factor is that the introduction of this concept is possible only when fly-by-wire (FBW) flight control systems are accepted. Of all concepts considered in the context of CCV-technology, "relaxed static stability" is the most dominant as far as piloted flight control is concerned. The NASA-Advanced Technology Transport (ATT) systems study program, described in reference 4, is one among those aimed at improvements in economics and reduction of terminal area noise. One NASA contract study, reference 5, emphasizes that high subsonic cruise commercial transports with acceptable noise characteristics can only be developed without economic penalty provided advances are made in structures, aerodynamics, propulsion and flight controls.

The overall conclusion resulting from studies made to date indicates that relaxation of inherent static stability produces an approximate 10 % reduction in take-off gross weights and around 5 % improvement in cruise L/D. The aft center of gravity limit should be placed in the vicinity of the manoeuvre point, in order to obtain this result.

Increasing interest in DLC application for large CTOL transport aircraft is based on the need for improved flight path control for more complex manoeuvres (noise abatement approach, etc.) and on the trend towards lower values of normal acceleration sensitivity primarily due to increasing wing loading. DLC application, as related to the latter point, is considered here. Wing flap components, spoilers singly or in combination can be used very effectively as lift modulating devices.

Considering the input-side of the DLC-system we may observe that there are many reasons for avoiding an increase in the number of separate pilot's controls. A form of mechanization, where the DLC system is integrated with the pitch control system, should be used. High-pass filtering of signals commanding the lift modulating devices should be applied to block low frequency DLC-surface deflections. In this manner a quickening effect of flight path response to stick deflections is obtained, while low frequency path control is obtained through aircraft attitude control only. References 6, 7 and 8 are examples of research into DLC application for large transport airplanes.

The following Chapter explains why the contemporary criteria are not applicable to the design of advanced flight control systems. Chapter 3 describes an exploratory flight simulation experiment while Chapter 4 presents some results obtained so far.

2 FLY-BY-WIRE FLIGHT CONTROL SYSTEMS AND FLYING QUALITIES CRITERIA

In an attempt to correlate the handling characteristics of aircraft equipped with fly-by-wire flight control systems with the descriptors used in contemporary criteria many area's of incompatibility are encountered.
To indicate effectively some of these inconsistencies, at least for certain types of advanced flight control systems, the following should be borne in mind:

- The adoption of the relaxed static stability concept (with c.g. ranging between the neutral and manoeuvre point) requires full-time artificial stabilization. This can only be obtained by applying FBW technology.
- A pitch-rate-command/attitude-hold system can be considered a prime candidate as flight control system configuration for landing approach conditions. An important parameter characterizing the aircraft/flight control system combination is the frequency of the dominant pitch attitude mode, indicated here by ω_θ.
  (A limited qualitative flight evaluation of such a system used in the landing approach is reported in reference 9).
- Side-stick controllers will probably be used in these electrical flight control systems.
- Direct-lift-control is one of the few available means for providing adequate flight path control for aircraft with a very low value of normal acceleration sensitivity; this handling quality parameter will be indicated here by the symbol n_α.

An inspection of the contemporary criteria in the form of requirements/specifications was made. This survey indicated that the (US) Military Specification "Flying Qualities of Piloted Airplanes", reference 10, could best be used as a reference due to its comprehensive background documentation, reference 11.

The two most fundamental requirements for longitudinal control in the landing approach may be mentioned as follows:

1. "Longitudinal static stability" (par. 3.2.1.1).

   No aperiodic airspeed divergence allowed. This requirement will be considered satisfied if the variations of elevator control force/(deflection) with airspeed are smooth; gradient stable.

2. "Short-period response" (par. 3.2.2.1).

   Short-period undamped natural frequency $\omega_{n_{sp}}$ and the normal acceleration sensitivity n_α shall be within the limits shown in figure 1. (In relation to this paper special attention is asked for the absolute limits on $\omega_{n_{sp}}$ and n_α shown in this figure).

In relation to the first requirement it is remarked that no stable stick force/(deflection) versus airspeed gradient exists for aircraft with a pitch-rate-command/attitude-hold flight control system. With respect to the second requirement, it is observed that FBW infers that some of the parameters describing the bare airplane characteristics, such as $\omega_{n_{sp}}$, are no longer applicable as handling quality indicators.

In this discussion the Level 1 boundary of reference 10 is considered to be equal to the distinction-line between the positive "yes"/negative "no" answer to the question "Is it satisfactory without improvement" of the Cooper-Harper Handling Qualities Rating Scale (Pilot Rating distinction between 3 and 4). In this paper this distinction will be called the satisfactory/unsatisfactory boundary.

The absolute lower limits for $\omega_{n_{sp}}$ and n_α (Fig. 1) as prescribed in reference 10 are quite arbitrarily selected due to a lack of experimental data in this area, as is indicated in the background document (Ref. 11). The format of this criterion seems maintainable for pitch attitude stabilized aircraft when $\omega_{n_{sp}}$ is replaced by ω_θ, the frequency of the dominant pitch attitude mode. An important question is what should be the minimum value of ω_θ of the aircraft/flight control system combination.

As stated before it is expected, that future aircraft will show a further (substantial) reduction in the value of n_α as compared to contemporary aircraft, see figure 1, thereby approaching the existing n_α boundary. DLC is needed for aircraft with values of n_α below that boundary. Thus an important question is how this minimum value is affected by the amount of DLC augmentation.

It is concluded that the prime interest of an exploratory investigation in the problem area here indicated, should be directed in the first place towards obtaining answers to the following:

1. what is the pilot opinion on aircraft control without stable stick force/(deflection) versus speed gradient?
2. what is the boundary value for the minimum pitch attitude dominant mode frequency (ω_θ), for "satisfactory" n_α ?
3. what is the boundary value for the minimum value of the normal acceleration sensitivity, n_α, for "satisfactory" ω_θ for control without DLC and what is the trade-off between required DLC-effectiveness and n_α for n_α-values below the boundary just mentioned?

For answering the third question, an investigation of a matrix of n_α and DLC-effectiveness values is required. When combined with the variations needed for satisfying question 2 this leads to a prohibitive number of conditions to be evaluated. Therefore it was decided to apply two procedures in a complementary manner to find answers to these questions. This method applies flight simulation and pilot-vehicle analysis to the coverage of the range of parameters to be checked.

3 FLIGHT SIMULATION AND PILOT-AIRCRAFT ANALYSIS

Flight simulation

To obtain experimental data as described in Chapter 2, a flight simulator experiment was designed with a neutrally stable hypothetical transport aircraft equipped with a pitch-rate-command/attitude-hold flight control system. The stiffness of the pitch attitude system (ω_θ) was an experimental variable. Two values of n_α were selected, the highest being around the value for contemporary Jumbo aircraft, the other being half this value. DLC of various effectiveness was investigated for the lowest n_α condition. The transport aircraft simulated was of the Boeing 707/DC-8 category as far as size, weight (150.000 lbs) and approach speed is concerned. Lateral-directional stability and control characteristics selected were typical for this category. Table 1 presents some information on the configurations tested while in figure 2 the parameter ranges chosen for the flight simulator experiment are indicated together with the conventional longitudinal manoeuvring criterion of reference 10.

Although it is expected that autothrottle will certainly be engaged in regular operation of the transports under consideration here it was decided to require, for the time being, that a handling qualities rating of the aircraft in the category "satisfactory" should be obtainable without autothrottle.

Pilot-aircraft system analysis

Rules to derive the mathematical expressions describing the pilot's behaviour in certain single-loop control situations (e.g. compensatory tracking) are fairly well known. Reference 12 gives a survey of the status at the time.

Flight path control with flight control systems under consideration here is a multiloop control situation and there is less knowledge about the rules to derive the mathematical expressions in this case.

Verification of the rules just mentioned for the experimental circumstances as existing during flight simulation was therefore undertaken for single-loop (attitude) as well as multiloop (attitude/ altitude) control situations. In these experiments the same configurations as in the flight simulation were used, in order to develop an understanding of the relation between pilot opinion of a configuration in simulated flight and dynamic pilot-aircraft performance measured in a compensatory tracking task. Results of an earlier program aimed at the in-flight measurement of human pilot describing function and remnant for pitch attitude control using the same side-stick controller is described in reference 13.

4 EXPERIMENTS

The flight simulator of the Department of Aeronautical Engineering of the Delft University of Technology (Ref. 14) was used for flight simulation as well as for compensatory tracking. The equipment includes a hybrid computer installation and a cockpit equipped with a visual display mounted on a three degrees of freedom motion system. Figure 3a gives an impression of the cockpit/motion system combination. The instrument panel and the projected runway image is visualized in figure 3b. The pilot's controllers consisted of the NLR Side-Stick-Controller (NLR-SSC), a (left-hand) throttle and conventional rudder pedals. Figure 4 presents a photograph of the two-axis dual spring gradient side-stick controller.

The flight control system that was mechanized for the simulation is shown in figure 5. The Electronic-Down-Spring (EDS) shown in the figure was programmed to provide the pilot with an artificial stick-force/(deflection) versus speed gradient at speeds occuring during the flare and touch-down (1 degree stick deflection per knot airspeed only below V = 132 KTS; Reference speed on glide slope was 135 KTS).

The simulation included wind and turbulence effects as well as ground effect. The flight task was the execution of a standard ILS (3° slope) approach under CAT I conditions. The simulated cloud-break occured at 300 ft altitude. During the approach the pilot had to fly an offset manoeuvre in the vertical plane: a glide slope offset equal to 3/4 of full scale deflection on the glide slope deviation indicator was introduced for part of the trajectory in order to permit a more thorough analysis, in particular as far as the DLC-system is concerned. Figure 6 shows an approach path flown as example.

Pilot opinion was obtained through ratings on the Cooper-Harper scale and through commentary. This commentary was given after completion of an "evaluation" run according to questions stated in a pilot comment card, as reproduced in figure 7. The comment was recorded on tape integrally.

Eleven parameters were recorded on a digital tape for further processing. Eight-channel trace recordings of the most important data were made continuously. Directly following the termination of each run, "on-line" calculated approach and landing performance data were printed on a teletype which facilitated close monitoring of the experiments. Four professional pilots participated in the simulation program and they performed 109 successful approaches.

A total of 82 compensatory tracking runs of 200 sec each were executed for 13 different configuration/ pilot combinations of which 64 were used in the analysis. Besides the pilots performing the approaches, two additional pilots participated in the tracking experiments. Somewhat less than half of the compensatory tracking runs were in an "altitude-tracking" mode using the projected visual display. Because of the necessity to have a stationary measurement situation during 200 sec the aircraft was artifically held at a constant range (1000 m) to the runway threshold. The nominal height at this range was 53 m. A VASIS (visual approach slope indicator system) deviation signal was used here as the "error" to be minimized. The effective bandwidth of the forcing function disturbance used, was 0.4 rad/sec. The other runs were in a "pitch-attitude-tracking" mode using a flight director pitch bar as "error"-indicator. The effective forcing function bandwidth here was 1.0 rad/sec. Data of the tracking experiments were FM-recorded on magnetic tape for frequency analysis.

5 RESULTS

Although the data analysis of the flight simulator results is not yet completed and the system analysis to find answers to the third question formulated in Chapter 2 has just been started, some interesting results obtained so far will be presented.

Flight Simulation

Because the boundary to be determined here, is the distinction satisfactory/unsatisfactory, the degree of difficulty of the configurations to be evaluated by the pilot was aimed to fall in the Cooper Harper rating range 1-5. This, combined with the generally observed fact that pilots will compensate for deficiencies in handling characteristics of aircraft rated up to 6, indicates that pilot commentary should be weighted much heavier than performance data in the analysis of the results. For that reason the results presented are those derived from commentary only.

The consistency of the Cooper-Harper ratings, being part of the commentary, was not completely satisfying. The fact that the participating pilots were never before exposed to this rating technique is probably the main reason here. These ratings will not be discussed here.

Detailed analysis of pilot commentary showed good agreement between pilots. The most important observations to be made after a thorough screening of the commentary follow below:

Speed control

Critical reactions were given on this subject for the configurations with a low value of n_α Within these critical comments a systematic improvement in acceptability for an increase of the DLC-effectiveness (A-1 to B-4) is noted however.

Electronic-Down-Spring (EDS)

Surprisingly enough the EDS-feature was not commented upon by the pilots. A mean stick deflection at touch-down of 12 degrees (maximum travel is 16 degrees) existed for a mean airspeed at touch-down of 120 KTS. Just how much influence the EDS has had on the landing performance cannot be checked. For the time being it is nevertheless hypothesized that the EDS feature is a positive asset for pitch stabilized aircraft.

DLC-effectiveness and ω_θ boundary values

The conclusion from pilot commentary for the low n_α configurations, with variable DLC-effectiveness (A-1 to B-4) is that the satisfactory/unsatisfactory boundary should be placed somewhere between B-3 and B-2. This means a required DLC-effectiveness of $\left|\frac{a_z}{\ddot{\theta}}\right|_{\omega \to \infty} > 10$ ft rad^{-1} at the low n_α value of 64 ft sec^{-2} rad^{-1}.

For the configurations with the higher value of n_α and differing in ω_θ value (A-10 up to A-14) it is deduced from the commentary that the satisfactory/unsatisfactory boundary should be placed very close to the A-13 configuration. This means a required value of the pitch-stiffness of $\omega_\theta > 0.5$ rad/sec.

Rate of descent at touch-down

Distinct from the observations based on pilot commentary, it is considered interesting to present results of just one performance indicator measured in the program. The reason for this is that the ability to make consistent landings in a flight simulator with outside-view displays is often considered doubtful. Figure 8 shows mean values for rate of descent at touch-down for all runs made with four configurations (variation of DLC-effectiveness). The standard-deviation value for each of the four configurations was around 0.5 m/sec.

Compensatory Tracking

The compensatory tracking experiments were primarily performed for future pilot-aircraft system analysis. However, some interesting observations can be made as to the correlation between the results of these experiments and the results of the flight simulation.

The pilot describing function data show only small run-to-run variation in amplitude as well as in phase, in spite of the fact that no extensive learning period was used. Intersubject variability was also small in nearly all cases.

Pilot Modeling

With the measurement method used, pilot modeling could only be done explicitly for single-loop tracking situations. The pilot model used to represent the results for single-loop tracking was taken to be:

$$Y_p = K_p \frac{T_L j\omega + 1}{T_I j\omega + 1} e^{-j\omega\tau_e}$$

Figure 9 presents, as an example, the measured describing function for configuration B-4.
The following results were obtained:

$0.29 < \tau_e$ (sec) < 0.37	for all configurations tested (A-1, B-4 and B-7)
$0.75 < T_L$ (sec) < 1.01	for configuration A-1 and B-4
$1.86 < T_L$ (sec) < 2.67	for configuration B-7
$0.30 < T_I$ (sec) < 0.57	for all configurations tested (A-1, B-4 and B-7)

As is indicated above pilot compensation in the form of lead (T_L) for the B-7 configuration was substantial.

For the multiloop (altitude) tracking data, a simple representation of pilot behaviour in the form of only a gain in the outer loop could be obtained here with the so called "series-model" loop structure when measured pilot model data for attitude control, are used for innerloop compensation in a multiloop analysis. This is an important observation related to the loop structure to be used in pilot-aircraft system analysis. (This form of analysis will be performed with the results of the tracking experiments to answer the question on how the minimum value of n_α is affected by the amount of DLC augmentation, third question formulated in Chapter 2).

Pilot-aircraft performance

Pilot-aircraft performance expressed in crossover frequency ω_c, phase margin φ_m for the forcing function used (bandwidth ω_i) was:

pitch-attitude (single-loop)

$$\left.\begin{array}{l} \omega_i = 1 \text{ rad/sec} \\ 1.4 < \omega_c/\omega_i \text{ (rad/sec)} < 1.8 \\ 35 < \varphi_m \text{ (degrees)} < 55 \end{array}\right\} \text{for A-1, B-4 and B-7}$$

altitude (multiloop)

$$\left.\begin{array}{l} \omega_i = 0.4 \text{ rad/sec} \\ 1.5 < \omega_c/\omega_i \text{ (rad/sec)} < 1.8 \\ 15 < \varphi_m \text{ (degrees)} < 30 \end{array}\right\} \text{for B-2, B-3 and B-4}$$

The range indicated for the ω_c/ω_i and φ_m values obtained for the different configurations tested in pitch respectively altitude tracking is fairly small.

Remnant

Pilot behaviour expressed in remnant, which is defined as that portion of the pilot's response not accounted for by his describing function, will be considered.

For pitch attitude tracking no significant difference in the human pilot linearity for the three configurations tested is indicated. Figure 9 gives an example of a fit with a model for normalized (by the variance of the error signal) injected error power spectral density. The model used is

$$\frac{\varphi_{rr_e}}{e^2} = \frac{0.035}{|0.3\,j\omega + 1|^2}$$

A fairly close fit was obtained for all cases.

In the case of altitude tracking no distinction could be made between input injected noise in the inner and the outer loop. A comparison of the closed loop pilot output remnant power spectral density (φ_{ee_r}), normalized by the variance of the pilot output signal, however shows differences between three configurations clearly, figure 10. Increasing the amount of DLC-augmentation (from B-2 to B-4) results in decreasing levels of normalized remnant.

C^*-Response

An exercise was performed to find out the usefulness of the C^*-criterion as described in references 15 and 16 in relation to the results obtained. The C^*-criterion is a time history criterion (with a Bode-diagram counterpart) based on the weighted sum of the two aircraft state variables, normal acceleration at the pilot's station and pitch rate, in response to stick inputs. The expression for C^* can be written as

$$C^*/F_s = n_{z_{cg}}/F_s + (V_{co}/32.2)\,\dot{\theta}/F_s + (l_p/32.2)\,\ddot{\theta}/F_s$$

(l_p is the distance between the pilot and center of gravity)

The weighting factor of $\dot{\theta}/F_s$ has the dimension of speed and is called the "crossover velocity" V_{co}; it is chosen to be 400 ft/sec by the originators of the criterion.

Figure 11 and 12 show normalized C^*-response curves for several configurations tested. It is shown that while A-14 is rejected as is in accordance with results of the flight simulation, the configuration A-1 and B-2, both unsatisfactory in flight path response, are not rejected by the C^*-criterion. This indicates that the criterion is not suited for aircraft with low values of n_α.

Replotting the C^*-response for a new weighting factor of 100 ft/sec, which means more weight for normal acceleration, results in the Bode-diagrams presented in figure 13a and 13b. A first observation now is that a horizontal high frequency asymptote is reached in the vincinity of a frequency of 10 rad/sec for all configurations. Also it is shown that for configurations with decreasing handling quality the distance between the low frequency peak and the high frequency asymptote value, is increasing. This trend can be observed for both parameter ranges investigated (DLC-effectiveness, Fig. 13a and ω_θ, Fig. 13b).

Modification of the original C^*-criterion boundaries for the crossover velocity V_{co} of 100 ft/sec is in progress and therefore no firm conclusions with respect to the usefulness of this modified C^*-criterion can be made.

6 CONCLUDING REMARKS

Some summarizing remarks will be made in conclusion.

The following two boundary values for obtaining "satisfactory" pilot ratings have been obtained:

- a minimum amount of DLC-augmentation of $\left|\frac{a_z}{\ddot{\theta}}\right|_{\omega \to \infty} = 10$ ftrad^{-1} is needed for an aircraft with the low value of $n_\alpha = 64$ $\text{ftsec}^{-2}\text{rad}^{-1}$.
- a minimum value for pitch stiffness of $\omega_\theta = 0.5$ rad/sec is required for aircraft with adequate values of n_α (no DLC-augmentation).

It was observed that airspeed control on the ILS-glide path needed constant attention due to the absence of a positive stickforce/(deflection) versus airspeed gradient as is inherent to the pitch-rate-command/attitude-hold flight control system used.

Side stick control was readily accepted by the four pilots performing approaches in the flight simulation program.

It is shown that valuable data can be obtained when compensatory tracking experiments are performed in the course of a flight simulation program. Results of the measurements show that the "series-model" loop structure leads to a straightforward interpretation of the combination of data measured in multiloop and single-loop control tasks.

Normalized closed loop pilot output remnant power spectral density is shown to be a usable indicator for the degree of difficulty of an altitude-tracking task performed for several aircraft configurations in a situation where the measured crossover frequency and phase margin hardly varies for the same range of configurations.

The C^*-criterion in the form proposed by Malcom and Tobie is not suitable as a requirement format for aircraft configurations depending on DLC-augmentation for "satisfactory" pilot ratings.

The results obtained so far, stem from exploratory research and should be confirmed by results of more piloted flight simulation experiments.

7 REFERENCES

1	Holloway, R.B.	Introduction of CVV technology into airplane design. AGARD Conference Proceedings No. 147 on "Aircraft design integration and optimization"	1973
2	Mooij, H.A. Boer, W.P. de	An exploratory study of flying qualities of very large subsonic transport aircraft in landing approach. ICAS Paper No. 72-07.	1972
3	Mooij, H.A.	Handling quality criteria development for transport aircraft with fly-by-wire primary flight control systems. NLR TR 74141 U (to be published)	1974
4	Hood, R.V.	Active Controls changing the rules of structural design. Astronautics and Aeronautics, August	1972
5	Williams, B.	Advanced Technology Transport Configuration AIAA Paper 72-756	1972
6	Pinsker, W.J.G.	The Control Characteristics of Aircraft Employing Direct-Lift Control Aeronautical Research Council, R. and M. 3629	1970
7	Barnes, A.G. Houghton, D.E.A. Colclough, C.	A Simulator Study of Direct Lift Control. Aeronautical Research Council, C.P. 1199	1972
8	Tomlinson, B.N.	Direct Lift Control in a Large Transport Aircraft-A simulator study of proportional DLC. RAE TR 72154	1972
9	Mooij, H.A.	Flight experience with an experimental electrical pitch-rate-command/attitude-hold flight control system. AGARD Conference Proceedings No. 137 on "Advances in Control Systems"	1973
10	Anon	Military Specification-Flying Qualities of Piloted Airplanes, MIL-F-8785 (ASG)	1969
11	Chalk, C.R. Neal, T.P. Harris, T.M. Pritchard, F.E. Woodcock, R.J.	Background information and user guide for MIL-F-8785B (ASG), "Military Specification - Flying Qualities of Piloted Airplanes", AFFDL-TR-69-72	1969
12	McRuer, D.T. Krendel, E.S.	Mathematical Models of Human Pilot Behaviour AGARDograph No. 188	1974

13 Mooij, H.A. — In-flight measured human pilot describing function and remnant for pitch attitude control. Proceedings of the ninth annual conference on manual control, M.I.T., Cambridge, U.S.A. — 1973

14 Baarspul, M.; Dooren, J.P. van — The hybrid simulation of aircraft motions in a piloted moving base flight simulator. Delft University of Technology, Report VTH-178 — 1973

15 Malcom, L.G.; Tobie, H.N. — New short period handling quality criteria for fighter aircraft. Boeing Document D6-17841-T/N — 1965

16 Tobie, H.N.; Elliott, E.M.; Malcom, L.G. — A new longitudinal handling qualities criterion. 18th NAECON, Dayton — 1966

17 Flora, C.C.; Kriechbaum, G.K.L.; Willich, W. — A flight investigation of systems developed for reducing pilot workload and improving tracking accuracy during noise-abatement landing approaches. NASA CR-1427 — 1969

Configuration	n_α	ω_θ	$\left\|a_z/\ddot{\theta}\right\|_{\omega\to\infty}$	DLC
	$ft\,sec^{-2}\,rad^{-1}$	$rad\,sec^{-1}$	$ft\,rad^{-1}$	
A-1	64	1.33	-8.4	No
B-2	↓	↓	+3.8	Yes
B-3	↓	↓	+15.7	↓
B-4	↓	↓	+38.9	↓
B-5	↓	1.07	+34.0	↓
B-6	↓	0.72	+20.6	↓
B-7	↓	0.32/0.46 1)	+ 9.8	↓
A-10	128	1.25	- 8.4	No
A-11	↓	1.04	↓	↓
A-12	↓	0.77	↓	↓
A-13	↓	0.53	↓	↓
A-14	↓	0.32	↓	↓

1) two complex poles in close proximity

$\left|\ \right|_{\omega\to\infty}$ is high frequency modal ratio

a_z is normal acceleration at the center of gravity

$\ddot{\theta}$ is pitch angular acceleration

(Main data source for development of the mathematical model for the hypothetical transport aircraft was reference 17).

Table 1: Characteristics of the configurations simulated

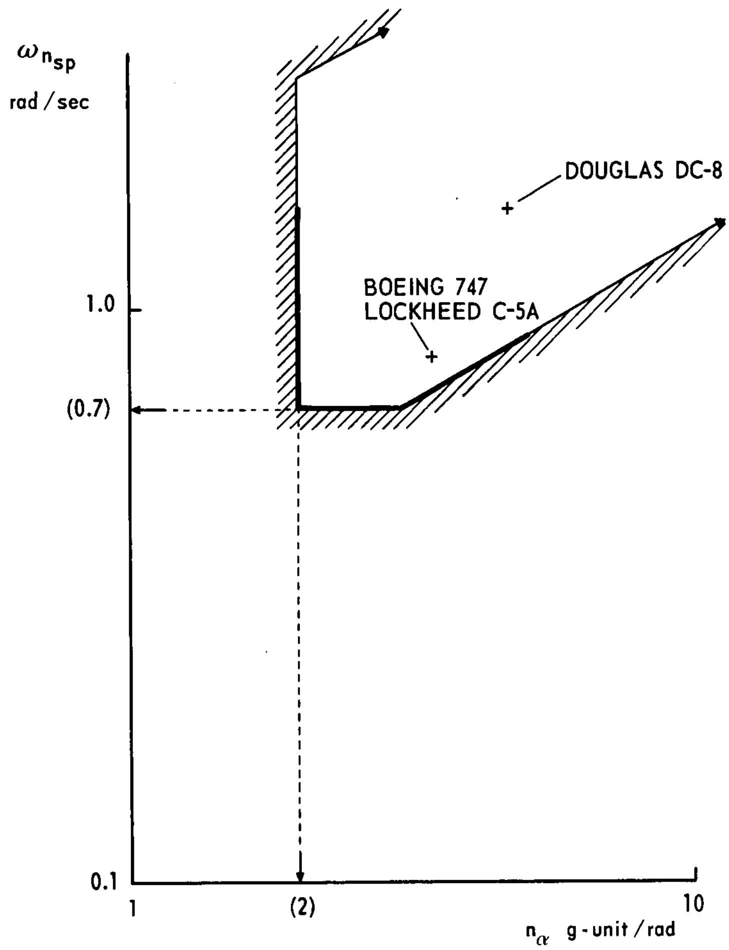

FIG. 1 LONGITUDINAL MANOEUVRING CRITERION FOR AIRCRAFT WITH CONVENTIONAL FLIGHT CONTROL SYSTEMS

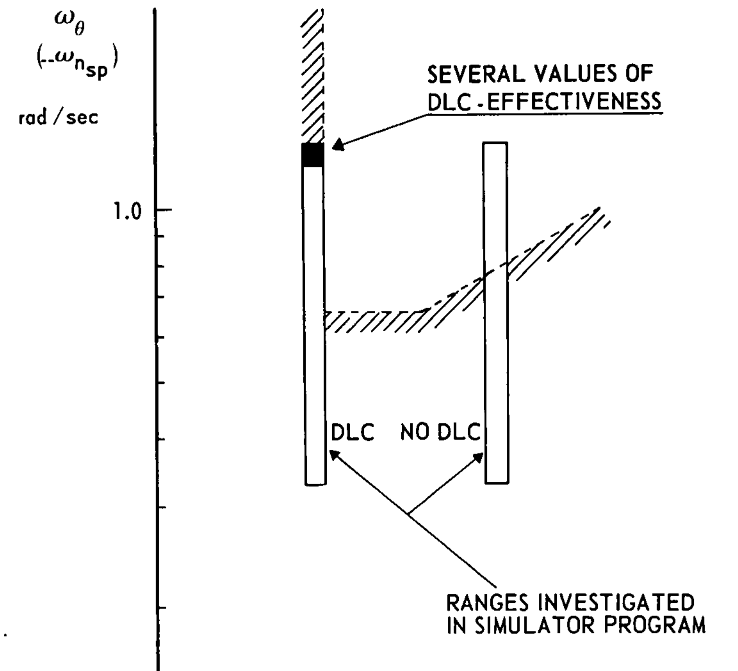

FIG. 2 PARAMETER RANGES INVESTIGATED IN THE FLIGHT SIMULATOR PROGRAM, AND BOUNDARIES OF THE CONVENTIONAL LONGITUDINAL MANOEUVRING CRITERION

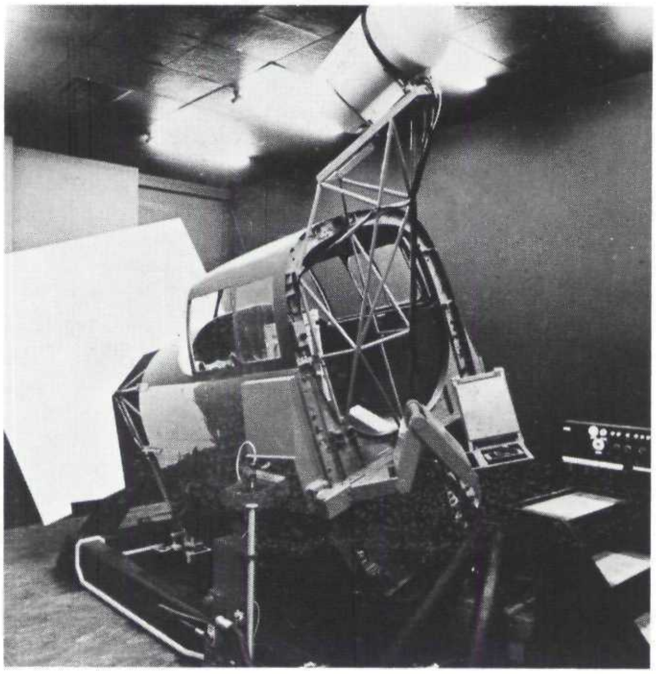

FIG. 3a. MOVING-BASE SIMULATOR

FIG. 3b. INSTRUMENT PANEL / PROJECTED RUNWAY IMAGE

FIG. 4 NLR SIDE-STICK CONTROLLER

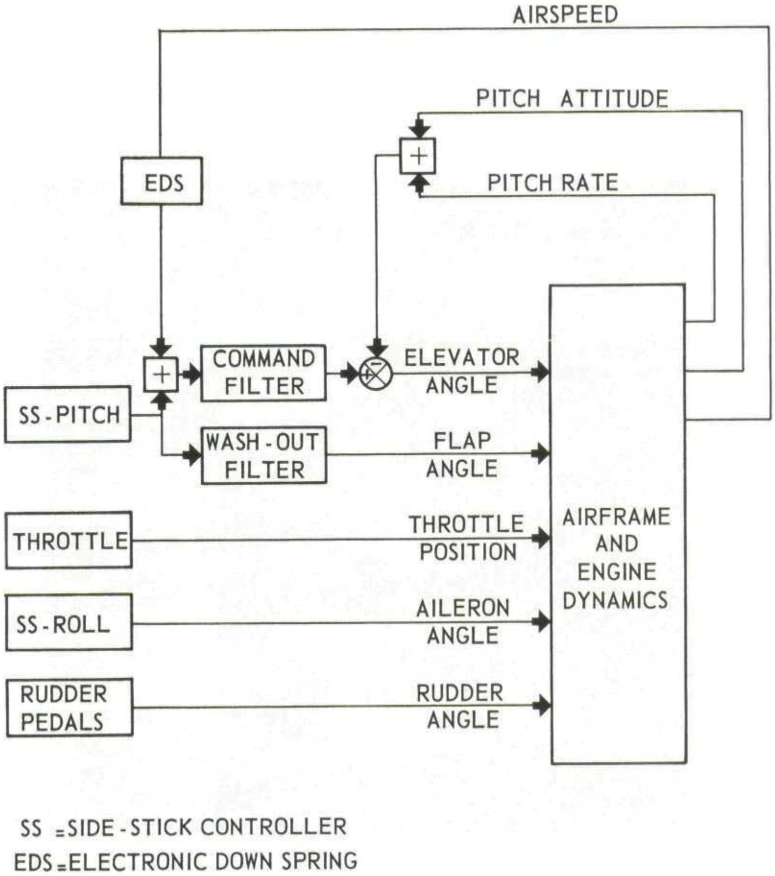

FIG. 5 FLIGHT CONTROL SYSTEM USED DURING INVESTIGATIONS ON THE MOVING BASE FLIGHT SIMULATOR

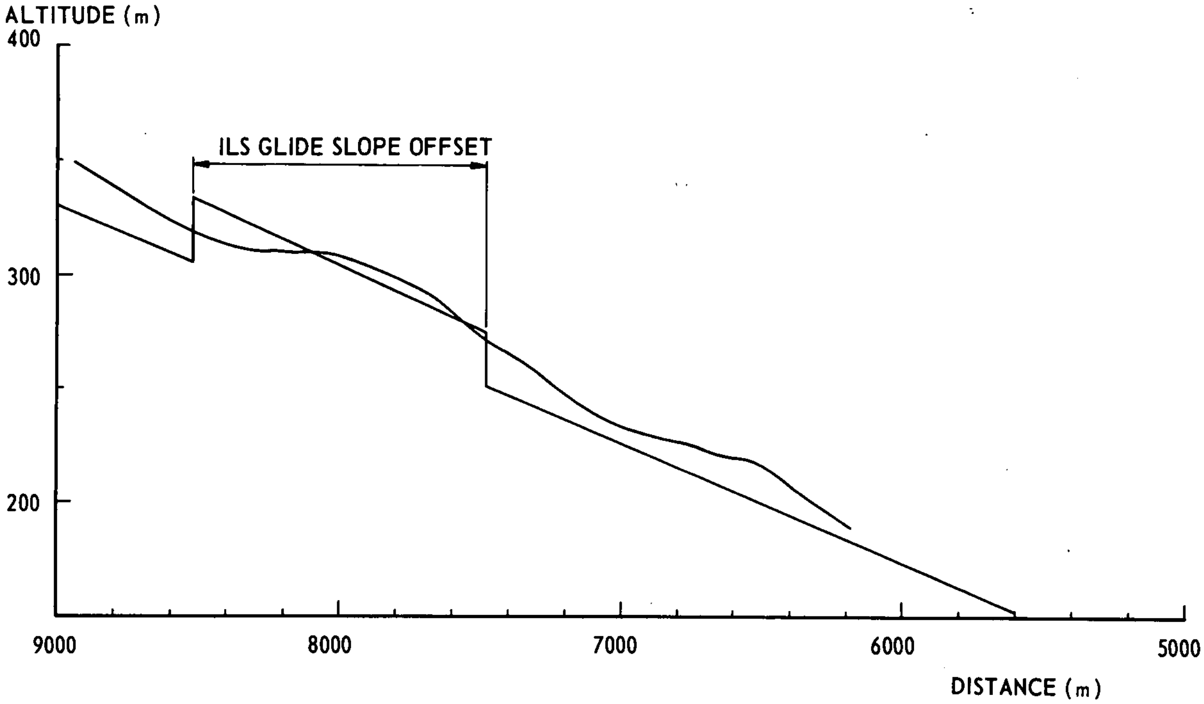

FIG. 6. EXAMPLE OF FLIGHT PATH FLOWN WITH OFFSET MANOEUVRE

1. STABILITY: DOES AIRPLANE STAY AT GIVEN PITCH ATTITUDE AND AIRSPEED
2. ABILITY TO HOLD ALTITUDE
3. RESPONSE TO THROTTLE
4. LOCALIZER ACQUISITION, WORKLOAD
5. GLIDE SLOPE ACQUISITION, WORKLOAD
6. CONTROL ON THE GLIDE SLOPE (G.S.)
   - 6a. PRECISION OF G.S. TRACKING
   - 6b. AIRSPEED CONTROL
   - 6c. ABILITY TO CHANGE ALTITUDE AND LOAD FACTOR (OFFSET - MANOEUVRE)
   - 6d. EFFECT OF LOCALIZER TASK ON GLIDE SLOPE CONTROL
7. FLARE: SINK RATE CONTROL, AIRCRAFT ATTITUDE
8. STICK FORCE AND DISPLACEMENT CHARACTERISTICS
9. RESPONSE TO TURBULENCE
10. CONFIGURATION RATING (COOPER - HARPER)
11. WHAT WAS THE MOST OBJECTIONABLE FEATURE OF THE CONFIGURATION ?
12. WHERE THERE ANY MALFUNCTIONS DURING THE EVALUATION ?

FIG. 7 PILOT COMMENT CARD

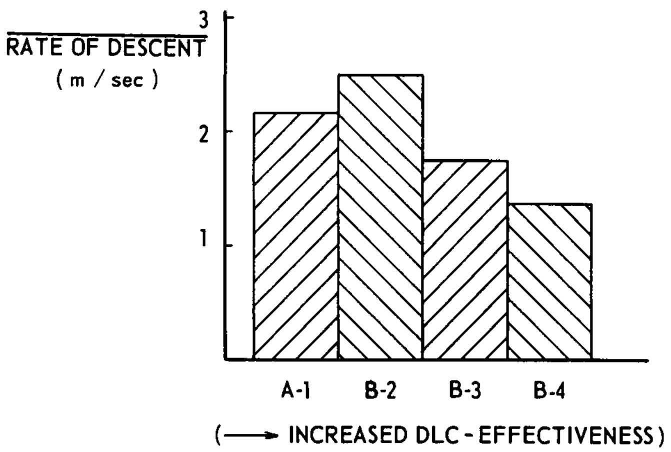

FIG. 8. RATE OF DESCENT AT TOUCHDOWN (DLC - VARIATION)

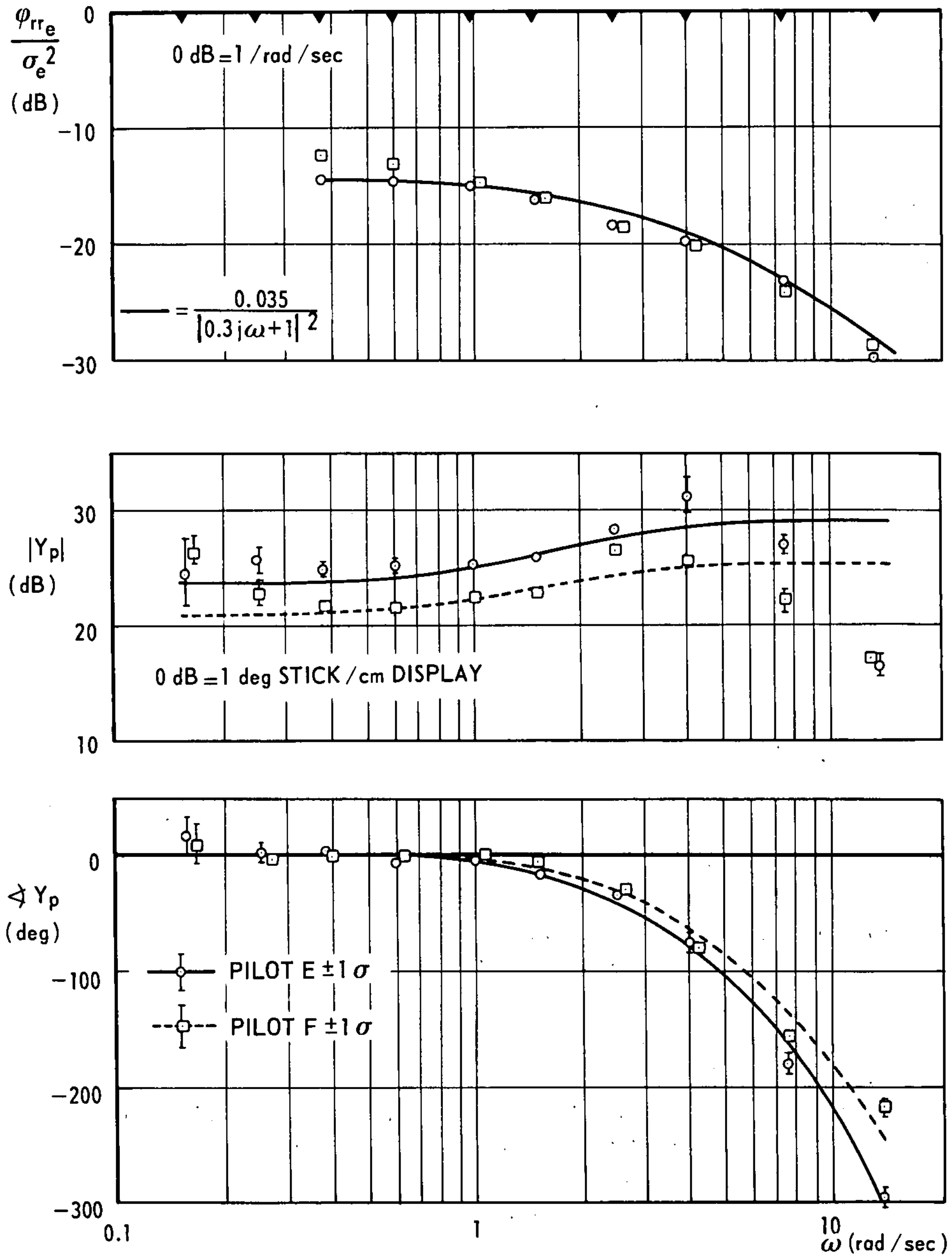

FIG. 9. EXAMPLE OF PILOT MODELING; PITCH ATTITUDE CONTROL

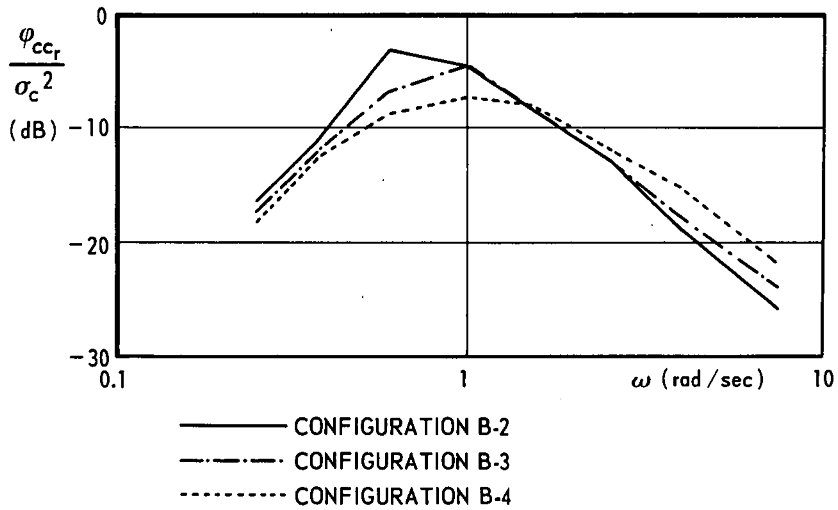

FIG. 10. NORMALIZED CLOSED LOOP PILOT OUTPUT REMNANT POWER SPECTRAL DENSITY FOR ONE PILOT

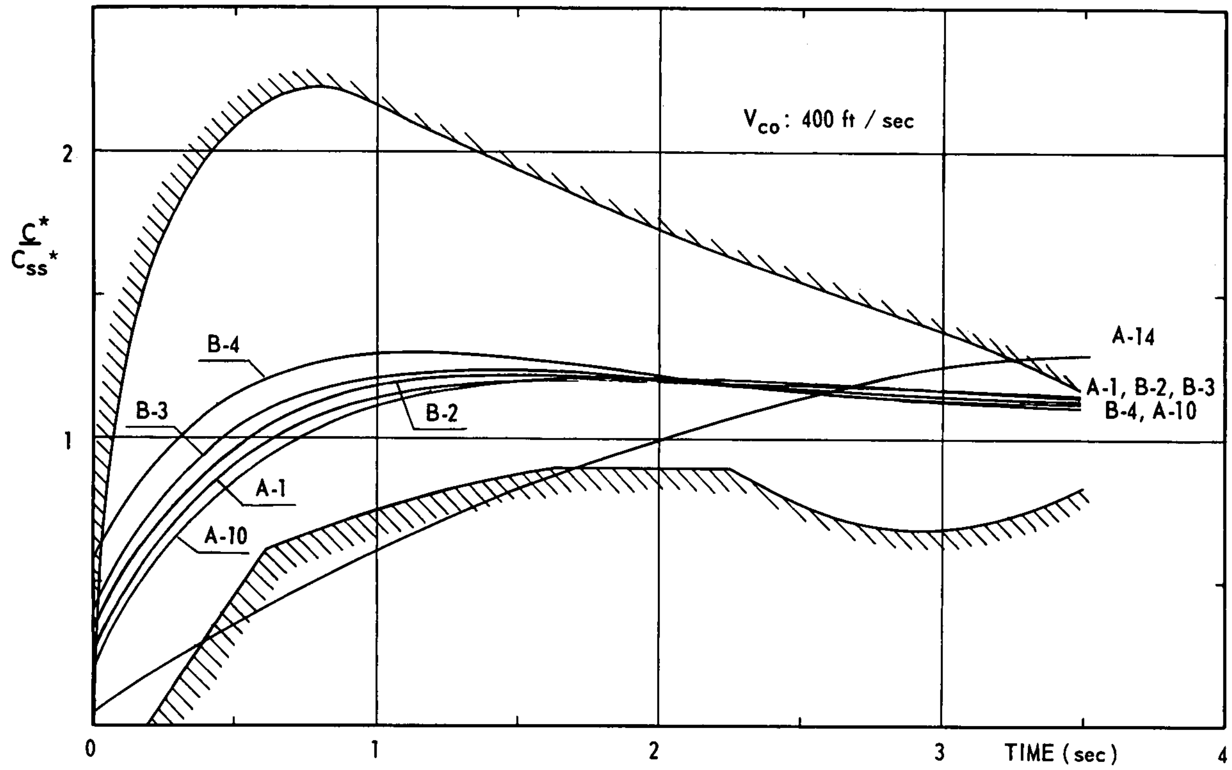

FIG. 11. C*- TRANSIENT RESPONSE FOR SEVERAL CONFIGURATIONS AND THE CRITERION OF REFERENCE 15

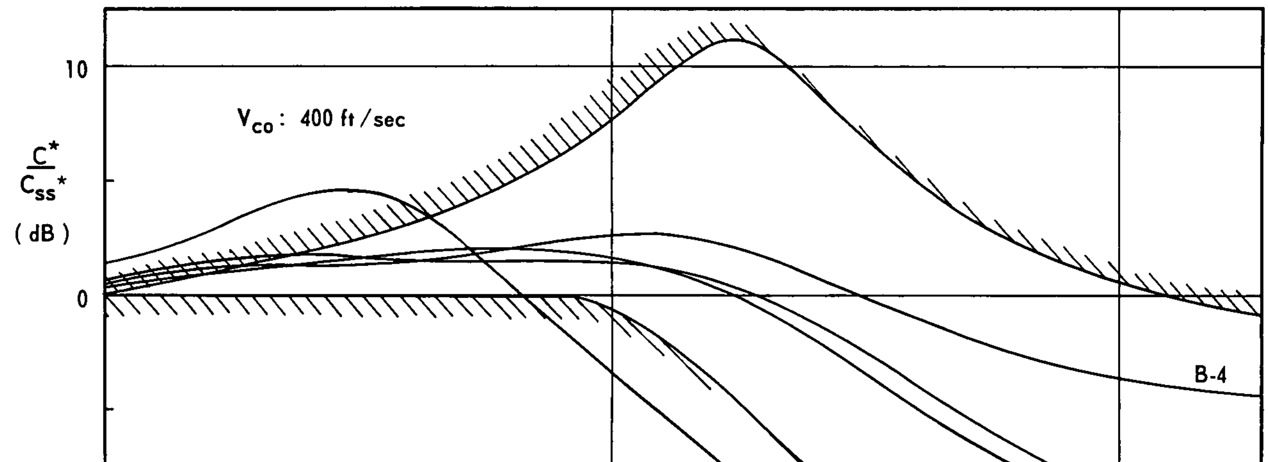

FIG. 12. C*- BODE DIAGRAM FOR SEVERAL CONFIGURATIONS AND THE CRITERION OF REFERENCE 15

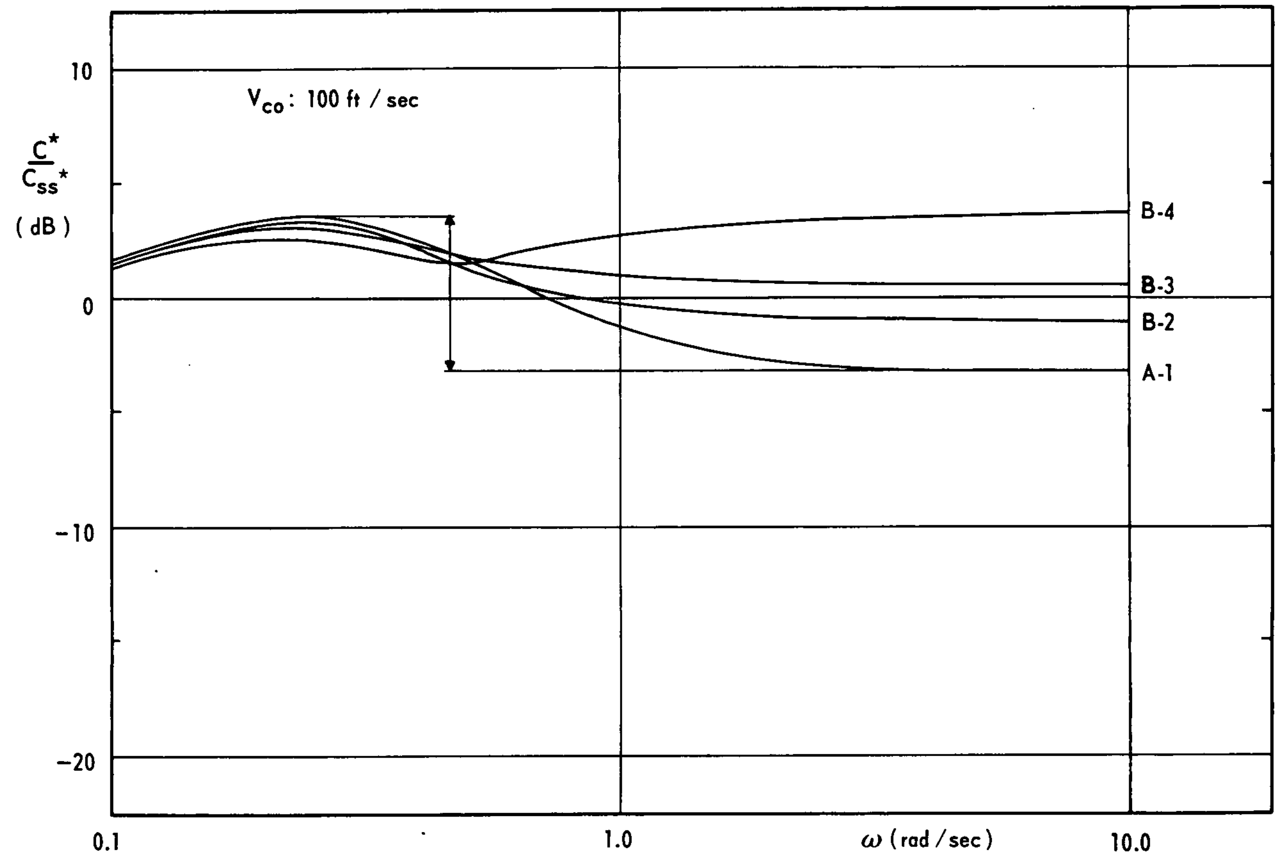

FIG. 13a. C*- BODE DIAGRAM FOR CONFIGURATIONS A-1, B-2, B-3 AND B-4 (DLC VARIATION). CROSSOVER VELOCITY V_{co} = 100 ft / sec

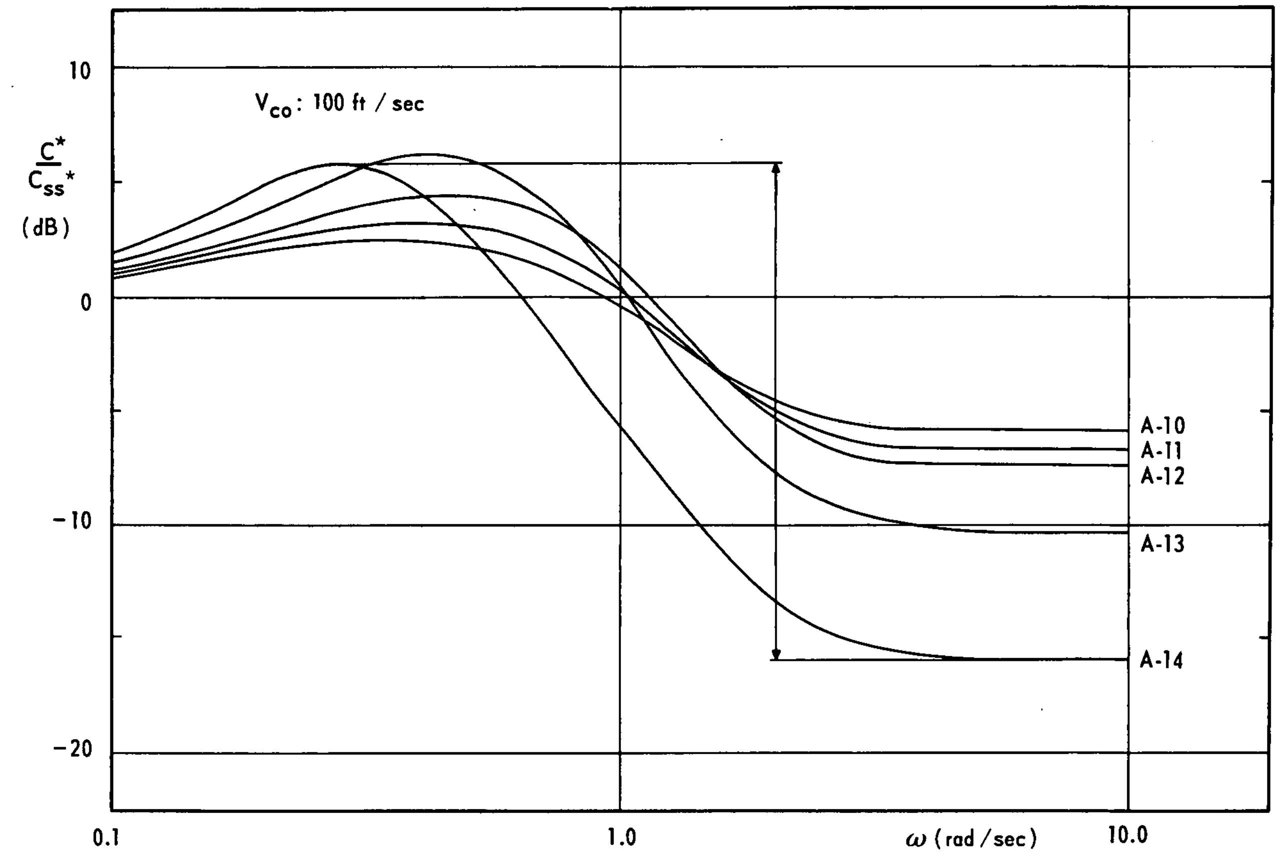

FIG. 13b. C*- BODE DIAGRAM FOR CONFIGURATIONS A-10 UP TO A-14 (ω_θ VARIATION). CROSSOVER VELOCITY V_{co} = 100 ft / sec

CONTROL OF AN ELASTIC AIRCRAFT USING OPTIMAL CONTROL LAWS

by

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ABSTRACT

In this paper the design of a multivariable control system for gust alleviation will be demonstrated. The use of computers for control design, summarized under the name "computer aided design" will be described. The gust control system for gust alleviation will be integrated into an overall flight guidance control system. Two control designs, using optimal control laws, will be achieved, one with complete and the second with incomplete state measurement. In the model description the elastic behaviour of the wing is included as well as the non-steady aerodynamic lift generation and the dynamic behaviour of the actuators. For a STOL-transport aircraft the efficiency of gust alleviation will be shown in a flight through turbulent air. The increase of wing lifetime and the corresponding decrease in structure weight by use of a gust alleviation system will be calculated.

1. INTRODUCTION

In the past few years a lot of CCV-concepts have been tested, which served all the main objectives to improve the flight characteristics or performance of aircraft by use of controllers. One of these CCV-concepts is the manoeuver load control and gust alleviation. The purpose of this CCV-measure is to reduce the mechanical stress of aircraft wings in the case of manoeuver or under the influence of atmospheric turbulence. The principle of the manoeuver load control is to shift the aerodynamic lift distribution over the wing closer to the fuselage by suitable deflection of the outboard ailerons or other manoeuver load control surfaces resulting in a reduction of the bending moment at the wing root.

By use of the manoeuver load control or gust alleviation the performance in flight guidance is affected and must therefore be integrated into the overall flight guidance control system. This new control system is a complex multivariable control system regarding the large number of state and control variables. Single input - single output control must probably be eliminated as a solution of the problem, because of the severe coupling between the state variables. Only a multivariable control system, which takes into account all couplings, will give a good control performance.

In modern control theory a lot of methods for the design of multivariable control system are developed. All these methods can be realized by use of computers. The control design described within this paper is designated as CAD (Computer Aided Design). The main purpose of this paper is to show that by use of modern control design procedures the engineer will have a good tool to find within a relatively short time the solution of any complex control problems.

2. DESIGN OF MULTIVARIABLE CONTROL SYSTEMS IN THE TIME DOMAIN

2.1 CAD - Computer aided design

By the use of computers for solving technical problems a lot of new methods for multivariable control design have been developed. These are design methods in the time domain as well as in the frequency domain. All these methods use a linearized description of the system. Each of these methods optimizes in a certain way a given control problem either by the minimization of a mathematically formulated function or by definition of specific characteristics. There is only one problem: The control system designer must define certain parameters of which the effect on the overall control behaviour can not be exactly predicted without testing.
The optimization process must be devided in three steps.

optimization by finding the best design method	(a)
optimization by finding the best parameters for a given method	(b)
optimization of a postulated mathematical function belonging to a certain method with fixed parameters.	(c)

Step (c) can be solved automatically by the computer. Step (a) and (b) generally can not be solved by the computer and must therefore be done by the engineer, to find the optimal way of control. Computer aided design now means, that those steps, which cannot be optimized automatically, will be supported by the computer in that sense to build up a communication between man and computer, which comprises both the capability of man to find optimal decisions and the capability of the computer to find very rapid solutions of a given mathematical problem. By use of a graphical interactive screen, which is connected to a computer, this man-computer communication can be established. Fig. 1 shows the whole configuration.

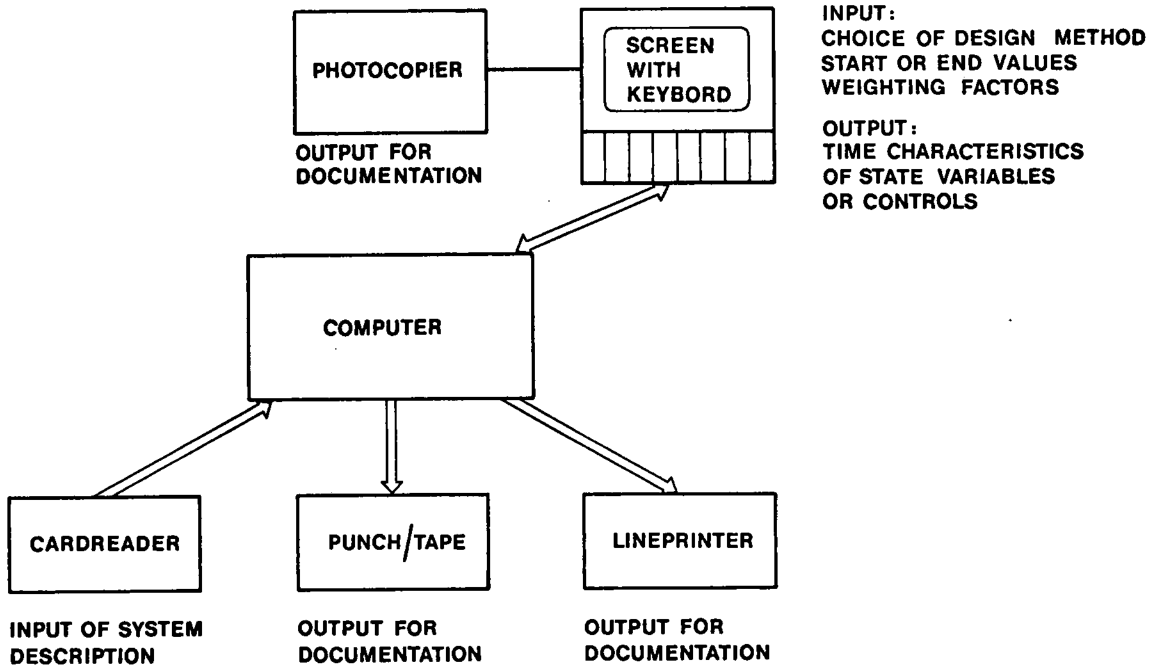


Fig. 1: "Computer aided design" General Configuration

2.2 Optimal control with complete state feedback

Today an often used method for multivariable control design is that of optimal control with complete state feedback. This method prescribes the minimization of a quadratic performance criterion with exponential time weighting.

The system is described by

$$\dot{x} = F\,x + G\,u \tag{1}$$

To find the optimal control the quadratic performance criterion

$$I = \int_{t_0}^{\infty} (\,x^T Q\,x + u^T H\,u\,)\,e^{2\alpha t}\,dt \qquad \alpha \geq 0 \tag{2}$$

is minimized.
This leads to the well known optimal control law

$$u = K_{opt}\,x \qquad\qquad K_{opt} = -\,H^{-1}\,G^T\,P \tag{3}$$

where P is the steady state solution of the Matrix Riccati equation.

$$-\,P = P\,F + F^T\,P - P\,G\,H^{-1}\,G^T\,P + Q \tag{4}$$

The choice of the time weighting factor α influences the position of the eigenvalues of the closed loop (the real part of all eigenvalues must be less (or at least) $-\alpha$.)
The most difficult task of the design procedure is the choice of the weighting matrices, especially for complex systems. With computer aided design this choice is made easier by use of an interactive screen. This screen serves as input and output device. The screen input parameters (Q, H, α) can be changed via typewriter and the effect of this change can be seen immediately after the computer has found the optimal control law. After a few seconds the design engineer will see on the screen the time histories of the state variables x and the controls u. So the engineer will get very quickly a feeling for the behaviour of the system and can find the "optimal" solution rather rapidly. For documantation the screen picture can be copied.

The following picture shows that screen.

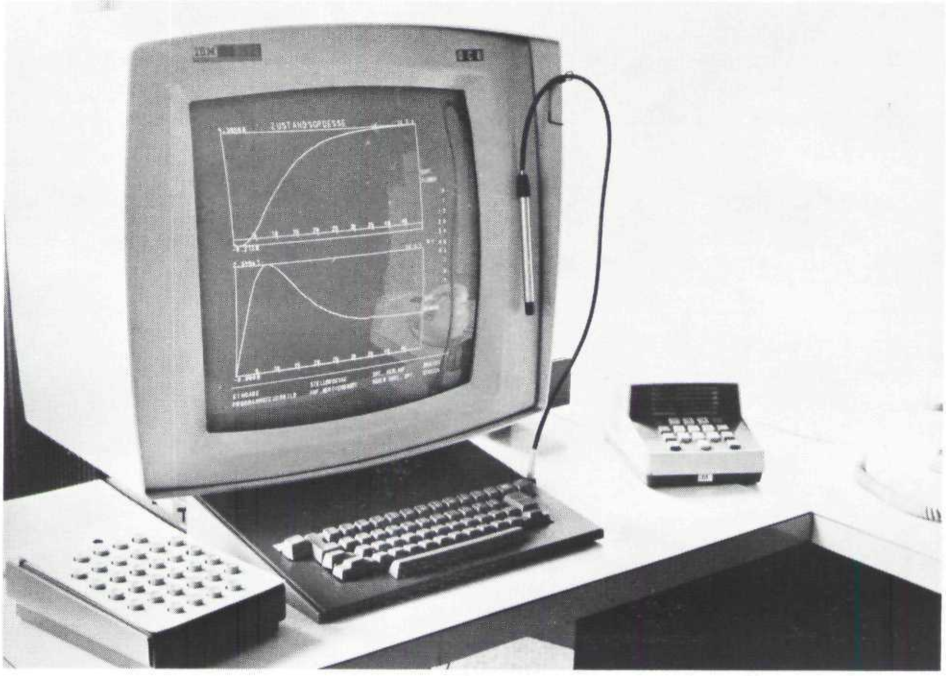


Fig. 2: Graphical screen together with keyboard

2.3 Optimal control with incomplete state feedback

The disadvantage of optimal control with complete state feedback is, that the whole state vector must be measured. To reduce the number of sensors, observers are used in modern control techniques. These observers are dynamic systems, which give an estimate for the unknown state variable by knowledge of the measured state variables and of the controls. In the following part a new design procedure will be shown, which is called "optimal control with incomplete state feedback."

In principle this design is a special case of a design method which is called in the literature a "reduced observer".

Plant $\quad \dot{x} = F\,x + G\,u$ (5)

Output $\quad y = C\,x$ (6)

F	(n, n)	plant matrix	x	n - state vector
G	(n, m)	input matrix	u	m - control vector
C	(l, n)	output matrix	y	l - output vector

Matrix F may have rank n. Following from equation (5) and (6) it must be

$$\int x\,dt = F^{-1}\,x - F^{-1}\,G \int u\,dt$$

or

$$\int y\,dt = C\,F^{-1}\,x - C\,F^{-1}\,G \int u\,dt \tag{7}$$

$$\int y\,dt^2 = C\,F^{-2}\,x - C\,F^{-2}\,G \int u\,dt - C\,F^{-1}\,G \iint u\,dt^2$$

written in matrices

$$\bar{y} = \begin{vmatrix} y \\ y\,dt \\ y\,dt^2 \end{vmatrix} = \underbrace{\begin{vmatrix} C \\ C\,F^{-1} \\ C\,F^{-2} \end{vmatrix}}_{T_o} x + \underbrace{\begin{vmatrix} 0 \\ -\,C\,F^{-1}\,G \\ -\,C\,F^{-2}\,G \end{vmatrix}}_{T_1} \int u\,dt + \underbrace{\begin{vmatrix} 0 \\ 0 \\ -\,C\,F^{-1}\,G \end{vmatrix}}_{T_2} \iint u\,dt^2 + \ldots. \tag{8}$$

$$\bar{y} = T_o\,x + T_1 \int u\,dt + T_2 \iint u\,dt^2 + \ldots.$$

The matrix T_o must be filled up with rows of $C\,F^{-1}$ or $C\,F^{-2}$ until T_o has the rank n. Rows, which do not increase the rank of the matrix T_o, must be scratched.

If the system is completely observable by vector y there must exist a matrix T_o of rank n.

If the matrix T_o is found one can calculate x from

$$x = T_o^{-1}\left[\bar{y} - T_1 \int u\, dt - T_2 \int\int u\, dt^2 - \ldots\right] \tag{9}$$

With the control of incomplete state feedback the same time history of x and u shall be created and therefore the new control law must be

$$u = K_{opt}\, x = K_{opt}\, T_o^{-1}\left[\bar{y} - T_1 \int u\, dt - T_2 \int\int u\, dt^2 - \ldots\right] \tag{10}$$

Fig. 3 shows in form of a block diagram the new control law. It is easy to derive the new matrices K_1, K_2, ... from the above equation.

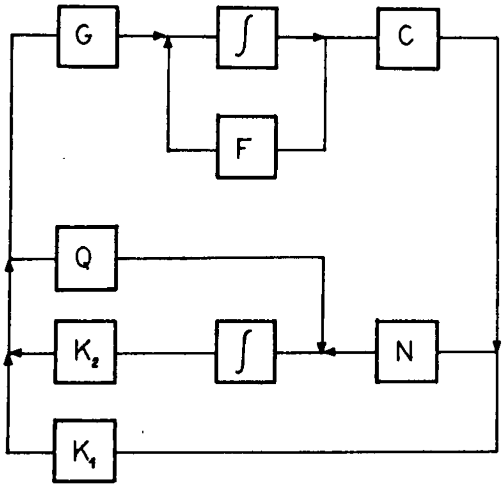


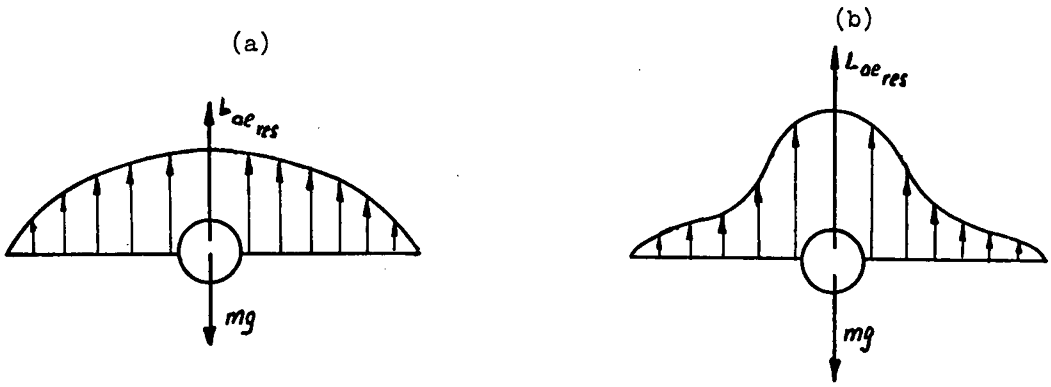
Fig. 3
Block diagram of "optimal control with incomplete state feedback"

In the case of the tracking problem control with incomplete state feedback is identical to that with complete state feedback. But under the influence of disturbances this control method will be different from complete feedback.

3. Manoeuver load control and gust alleviation

3.1 General remarks

The main idea of manoeuver load control and gust alleviation is to reduce dimensioning characteristics of the wing - for example the bending moment of the wing root - in the case of manoeuver or in flight through turbulent air. By this reduction an increase of lifetime of the wing or a decrease of structure weight can be achieved.
Given for example the lift distribution of case a to get a certain load factor it is certainly better to shift the lift distribution closer to the fuselage in order to reduce the root bending moment without changing total lift (case b).



To achieve the desired shift of the lift distribution one must deflect symmetrically the outboard ailerons (or other surfaces) to generate negative lift. Simultaneously the angle of attack must be increased to get the same total lift.

Generally the manoeuver load control and gust alleviation has the purpose to increase lifetime of the wing of an airplane. Therefore one must consider that the lifetime of the wing of a combat aircraft is determined by higher load factors and the control system must be designed as a menoeuver load control system. On the other hand transport airplanes will not experience these load factors and the lifetime of their wings is therefore determined by the influence of atmospheric turbulences and in this case the control system must be a gust alleviation system. These two design criteria have consequences in the size of the control system following from the different mode of system description. The manoeuver case is a relatively slow motion (slowly in comparison to the motion of the wing structure) and therefore the wing can be considered as a rigid body. In the case of gust alleviation this assumption can not be maintained any more, because the natural frequencies of transport airplane wings are much lower than those of combat aircrafts and the frequencies of the gust disturbances are nearly in the same range. In this case the gust disturbances may excite and amplify the wing structure movement and therefore the elastic behaviour must be included in the system description. Also the dynamic behaviour of the actuators and the non-steady aerodynamic lift generation must be included in the mathematical model.

In view of the remarks above the number of state variables will increase and the control design will become very complicated. For the example of gust alleviation for a STOL-transport aircraft it is shown, that in a relatively short time a good control design can be found by using design methods of optimal multivariable control theory.

3.2 System description of an elastic airplane

As demonstrated in chapter 2.2 and 2.3 a linear system description in the time domain is necessary for using these design methods. The flight control engineer will not be faced very often with the problem of description of elastic systems and therefore in the following part a method of system description will be shown.

The motion of a free elastic body may be composed by his modal modes. Each oscillation of a continuum can be divided in parts of modal modes. The absolute displacement of a point of the elastic continuum may be described as sum of its modal modes.

$$w\,(x,y,z,t) = \sum_{j=1}^{\infty} \phi_j\,(x,y,z)\, q_j\,(t) \tag{11}$$

$w\,(x,y,z,t)$ = abs. displacement of point (x,y,z) at time t

$\phi_j\,(x,y,z)$ = j - th modal mode

$q_j\,(t)$ = j - th generalized coordinate

The system description of an elastic system in the frequency domain, - neglecting disturbances which would be analog -, has the general form.

$$\left[-\omega^2 M_{gen} + j\,\omega\, K_{gen} + C_{gen}\right] q\,(j\omega) = L'\,(q) + j\,L''\,(q) \tag{12}$$

M_{gen}	generalized mass matrix	$M_{ij} = 0$ for $i \neq j$
K_{gen}	generalized damping matrix	$M_{ij} = 0$ for $i \neq j$
C_{gen}	generalized stiffniss matrix	$C_{ij} = 0$ for $i \neq j$
q	vector of generalized coordinates	
L'	real portion of non-steady aerodynamic forces	
L''	imaginary portion of non-steady aerodynamic forces	
ω	frequency	

The non-steady aerodynamic forces may be described in derivative form.

$$L_i' = c_{ij}'\,(\omega)\, q_j \qquad L_i'' = c_{ij}''\,(\omega)\, q_j \tag{13}$$

To get a transformation into the time domain an approximation must be used of which the curve of the transferfunctions of the non-steady aerodynamic forces in the frequency domain will be approximated as follows.

$$c_{ij}\,(s) = a_o + a_1 s + a_2 s^2 \;\ldots\ldots + a_k s^k \qquad s = j\,\omega \tag{14}$$

The aerodynamic force i, generated by form j will be

$$L_i \left\{ q\,(s) \right\} = (a_o + a_1 s + \ldots + a_k s^k)\, q_j\,(s) \tag{15}$$

Using the Laplace - Transformation this approximation leads in the time domain to

$$L_i \left\{ q\,(t) \right\} = a_o q\,(t) + a_1 \dot{q}\,(t) + \ldots + a_k q^{(k)}\,(t) \tag{16}$$

Fig. 4 shows the curve of one non-steady aerodynamic force in the frequency domain from 0 to 15 Hz together with its approximation. The degree of the approximation was k = 2. With this degree of approximation acceptable results can be found if the natural frequencies of the modal modes are not to far away from the natural frequencies of the airplane motion.

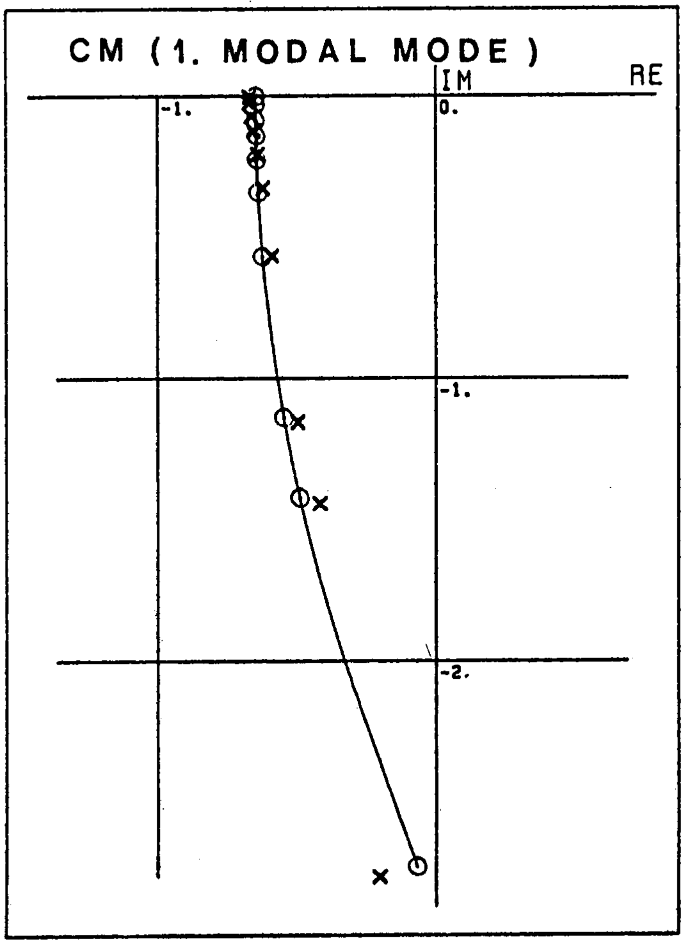


Fig. 4:
Curve of non-steady aerodynamic force in the complex plane with frequency as parameter (o...15 Hz)

+ calculated values
o approximation

With the approximation for aerodynamic forces a linear description in the time domain is found.

$$M \ddot{q} + K \dot{q} + C q = V(t)$$

M , K , C are the non-symmetrical (by the termes of aerodynamic forces) mass, damping and stiffness matrices of rank n (n is the degree of enclosed elastic modes). V (t) is a disturbance vector (that can be control surface deflection, gust or anything else).

3.3 Gust alleviation for a STOL-transport aircraft

3.3.1 General description for airplane

The efficiency of gust alleviation will be shown in the case of a transport aircraft. This aircraft is a prototype of a high velocity STOL-transport aircraft. Fig. 5 shows the main dimensions of this aircraft.

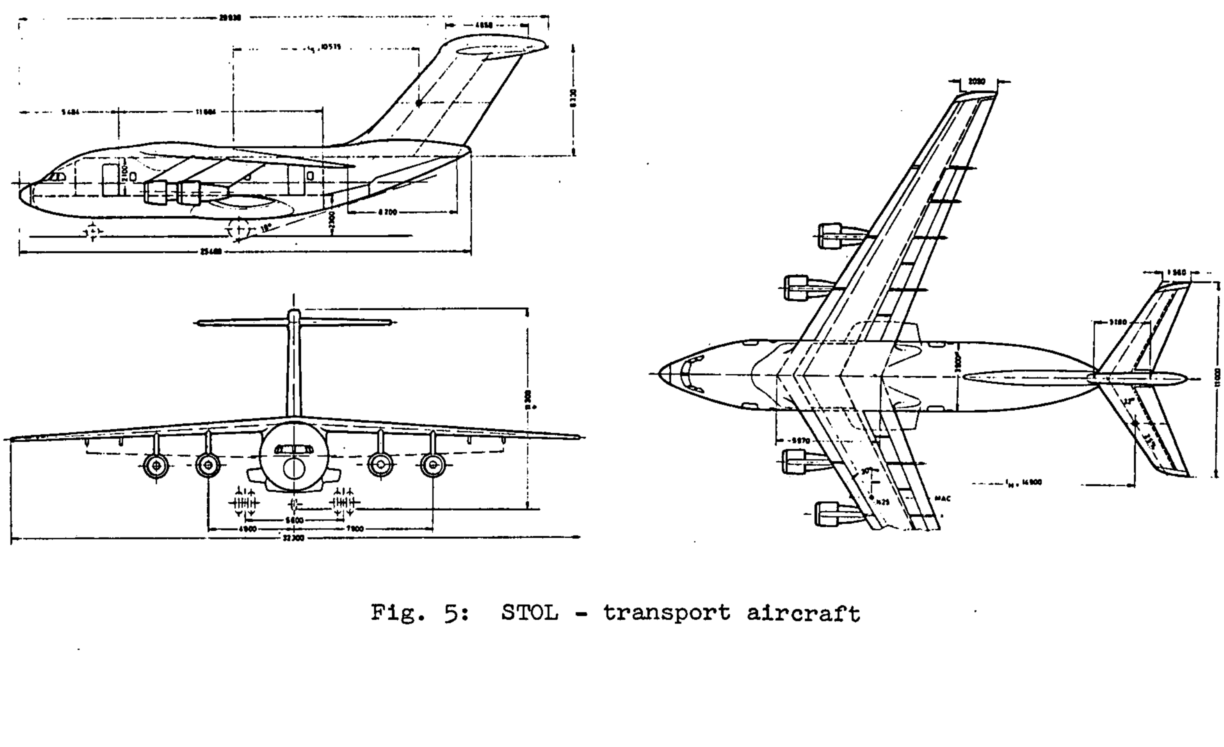


Fig. 5: STOL - transport aircraft

As manoeuver load control surface a combination of outboard rudder and leading edge flap (see fig. 6) has been chosen. By using this combination, the center of pressure will be on the elastic line of the wing and the control surface will not cause any torsion moment.

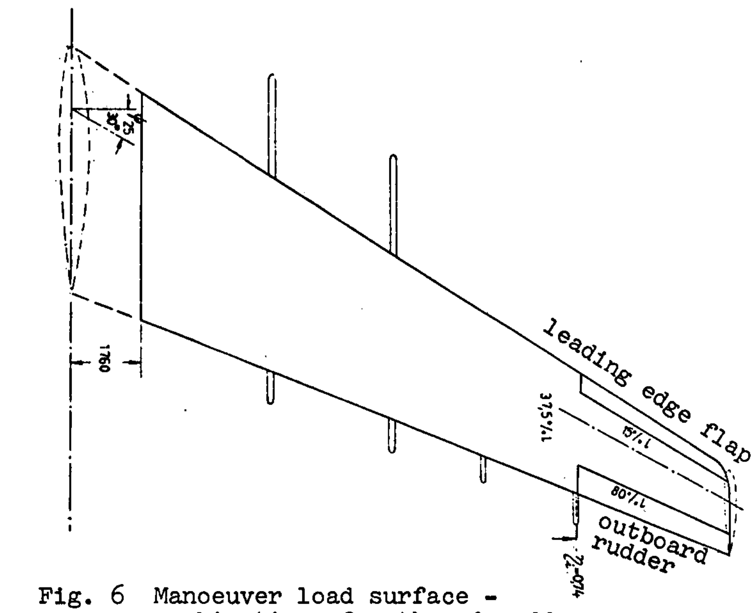
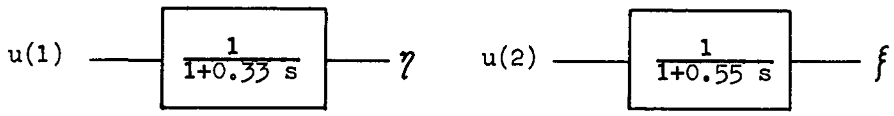


Fig. 6 Manoeuver load surface - combination of outboard rudder and leading edge flap

Previous investigations have shown, that the elastic behaviour of the wing can be represented by the first 2 bending modes.

1. bending mode　　　　2. bending mode

The dynamic behaviour will be approximated by first-order transferfunctions.



tailleron　　　　manoeuver load control surface

Admitting linear elasticity of the system, - that will mean linear dependance between deformation and forces - the linear system will be described by 11 state variables.

V_x	x-velocity	state variables of rigid motion (longitudinal motion)
$\dot{\alpha}$	rate of change of angle of attack	
α	angle of attack	
ω_y	pitch rate	
ϑ	pitch angle	
M_{root}	bending moment (root)	state variables of elastic motion
M_{hl}	bending moment (half wing length)	
$\dot{w}_{hl}$	deformation velocity (")	
$\dot{w}_o$	deformation velocity (outboard)	
η	tailleron deflection	
ξ	deflection of manoeuver load surface	

3.3.2 Control design with complete feedback for STOL-transport aircraft

Two control designs were made for the aircraft. The first design uses the taileron for control only and the second, the real gust alleviation system, uses the manoeuver load surface additionally. The first control design has been selected as reference in order to investigate the real influence of the gust alleviation.
The dynamic behaviour in the longitudinal motion has to be identical with both systems. By this postulation one will see, that a reduction of mechanical stressing will be achieved by the manoeuver load surface only and not by a slower natural motion. The investigations for gust alleviation were made in the flight case Ma = 0.7, H = 10.000 ft. The main stressing determing the lifetime of the wing will appear in this flight phase.

The control design has been established with CAD methods including graphical screen. With both control systems the transient of the angle of attack caused by a step disturbance of the angle of attack shall be identical and the bending moment shall be reduced by deflection of the manoeuver load surface. Just for these requirements the use of CAD will be very favourably, because one can define in relatively short time the desired control system layout.

A complete state measurement is assumed in that design. Fig. 7 shows the result. The transient of the angle of attack is approximately the same. On the other hand the influence of manoeuver load surface operation can be seen very clearly. It must be noticed that the values of all variables are deviations from the steady state values.

OPTIMAL CONTROL WITH COMPLETE FEEDBACK

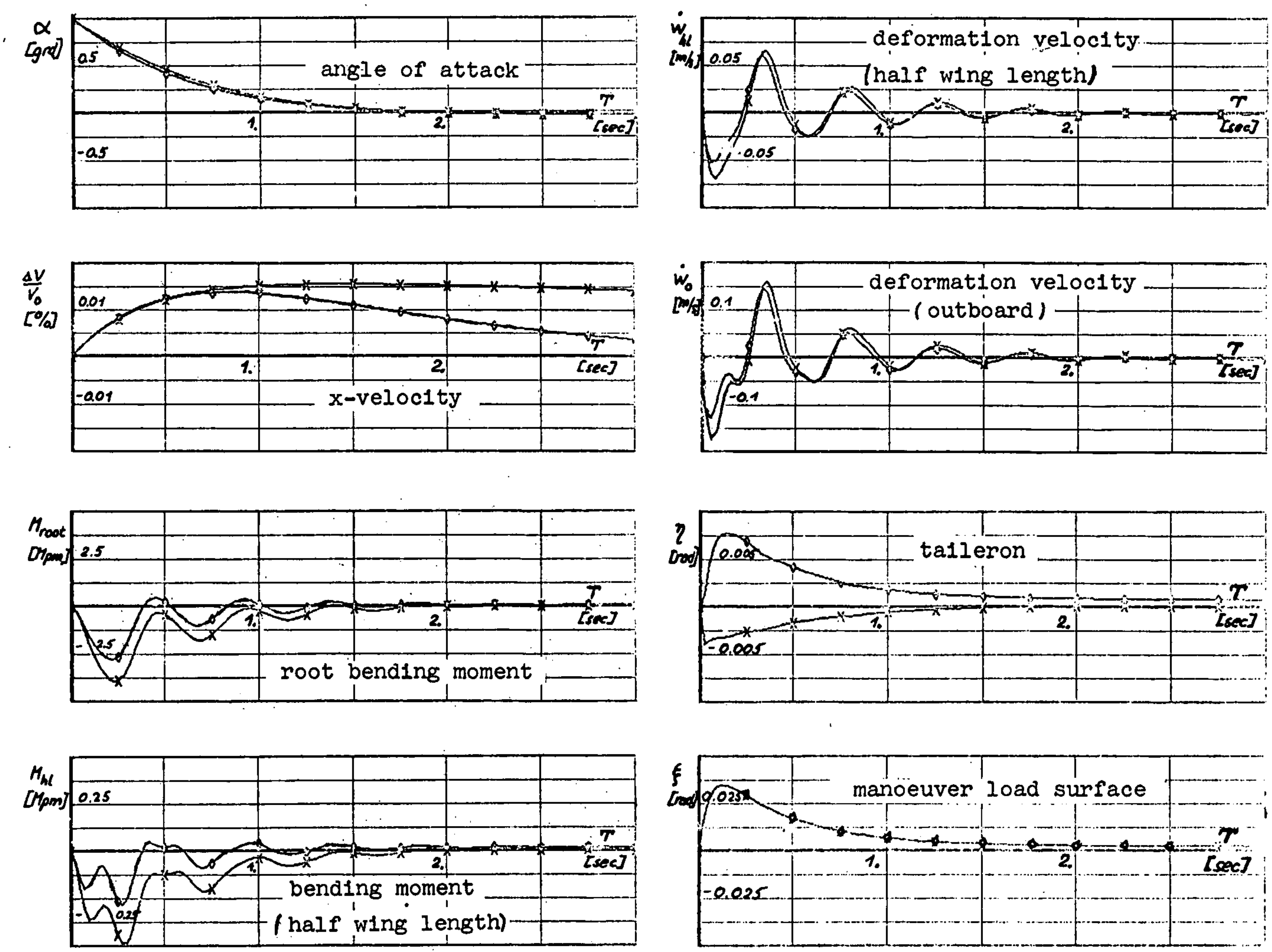


Fig. 7 Time history of state variables
× without gust alleviation
◇ with gust alleviation

3.3.3 Control design with incomplete state feedback

The design in chapter 3.3.2 served for principle investigations to estimate the possible increase in lifetime or decrease in structure weight by gust alleviation. In practice the measurement of α is impossible, and measurement of bending moments is very difficult. A new control design is made, which does not need these measurements. The simulation (see fig. 8) shows, that the result will be identical, if the disturbance affects a measured variable. Affecting non measured variables the time histories will diverge a little bit from those, created by the optimal controller with complete state feedback, but simulations have shown that the differences are not too extreme.

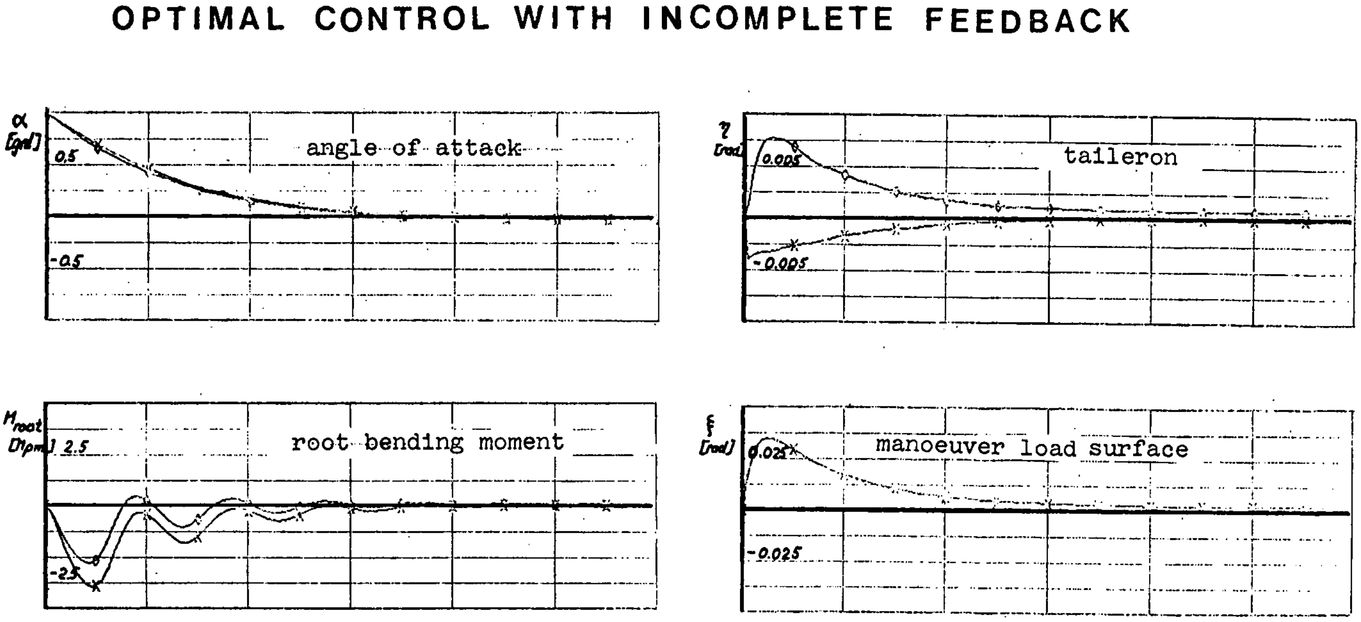


Fig. 8 Time history of state variables
◆ with gust alleviation
× without gust alleviation

3.3.4 Simulation of flight through turbulent air

The flight through turbulent air is simulated in the time as well as in the frequency domain. The model for gust simulation is the so called "Dryden-model" (MIL-00886). Fig. 9 shows a short time history and the influence of the gust alleviation can be seen very clearly for the bending moment.

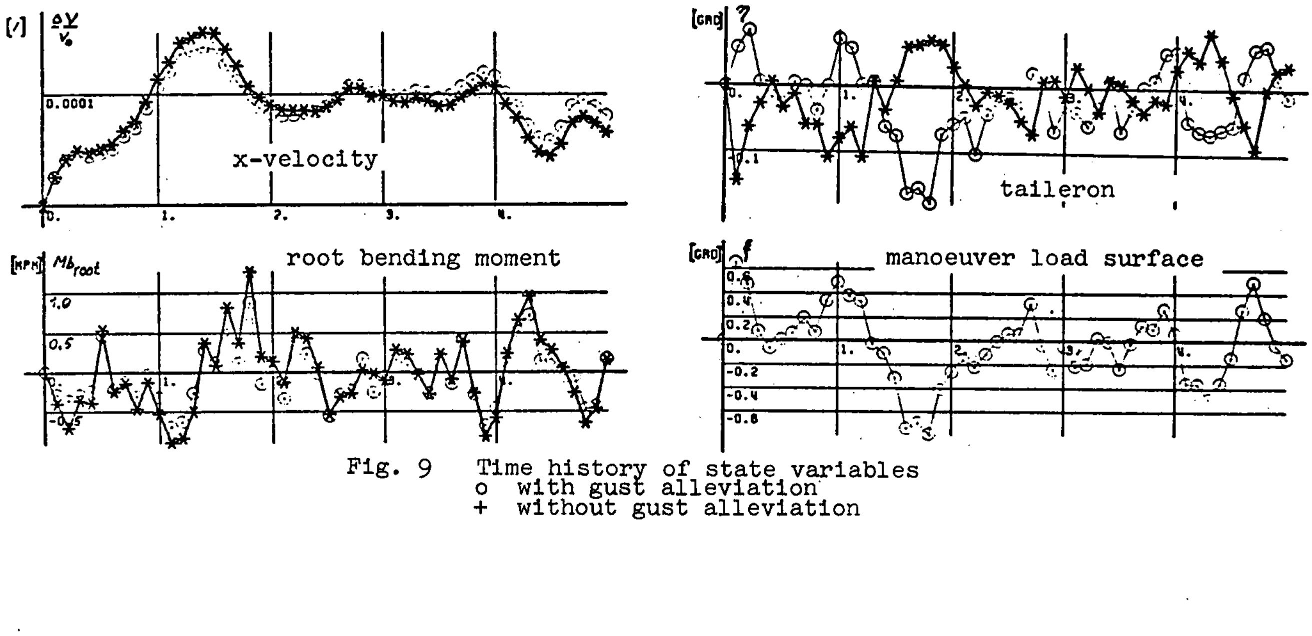


Fig. 9 Time history of state variables
o with gust alleviation
+ without gust alleviation

Exact statistical results can be found by a simulation in the frequency domain. Fig. 10 shows the distribution of the root bending moment power spectrum.

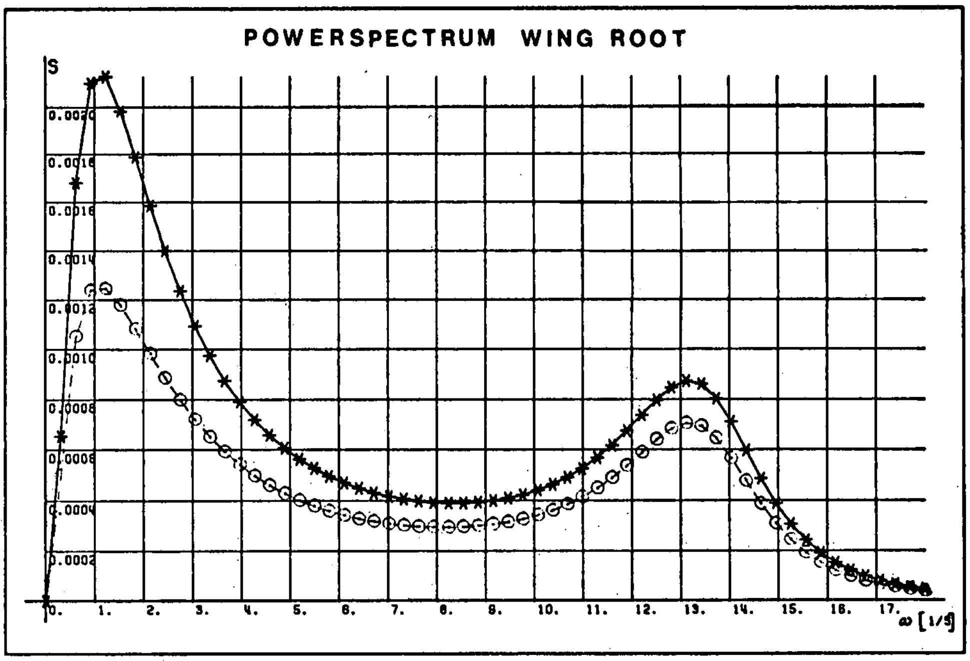


Fig. 10:
Powerspectrum of bending moment at wing root (Ma = 0.7, H = 10000ft)
+++ without gust alleviation
ooo with gust alleviation

3.3.5 Lifetime calculation

The first step for a lifetime calculation is an analyses of typical missions (MIL-A-00886). The result of this analysis is a distribution of load cycles (see fig. 11).

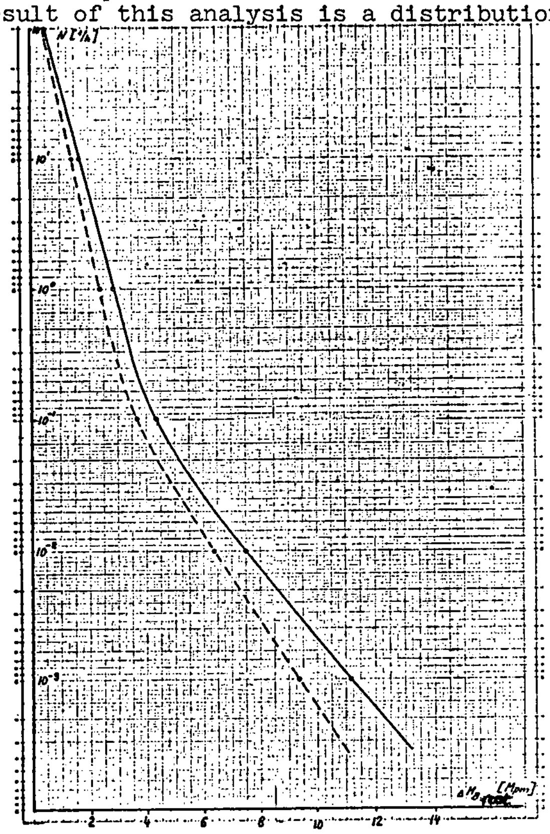


Fig. 11:
Cumulative loads per hour of wing bending moment
(Ma = 0.7, H = 10.000 ft)
—— without gust alleviation
--- with gust alleviation

The lifetime calculation was then made with a method called "Miner-rule". Assumption for this calculation was, that the total life of the aircraft will be 32 000 flights (1 flight = 1 hr.) As a result of that calculation the following is estimated

ca. 25% increase in lifetime of wing.

Converting this increase of lifetime into a reduction of structure weight, considering the additional weight caused by actuators, leading-edge flaps and sensors,

ca. 8% increase of payload

is estimated.

4. CONCLUSION

In this paper the efficiency of gust alleviation has been investigated. In the case of a STOL-transport aircraft it is demonstrated, that the gust alleviation will be an effective CCV-concept to improve the airplane performance. It is shown, that the gust alleviation system has to be integrated into the overall flight guidance control system. For the design of the control system the use of methods of modern multivariable control theory has been demonstrated. The results of the investigation only base on simulation, which have to be verified by real flight tests.

FORME EXPLICITE DE LA LOI OPTIMALE DE PILOTAGE D'UN AVION RIGIDE VOLANT EN ATMOSPHERE TURBULENTE

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RESUME

Le vol à grande vitesse et basse altitude des avions militaires pose des problèmes de confort, de manoeuvrabilité, de stabilité de plate-forme de tir, qui ne peuvent être pratiquement résolus que par l'utilisation de systèmes de pilotage automatique en turbulence. L'action de tels pilotes automatiques peut en général être limitée au domaine des fréquences associé à la Mécanique du Vol, du fait que les appareils militaires considérés sont en général très rigides. Les systèmes "en boucle fermée", qui réinjectent dans les gouvernes, après filtrage, certaines réponses de l'avion (par exemple la vitesse de tangage), sont le plus généralement utilisés ; leur application est cependant limitée, car ils ont tendance à augmenter les temps de réponse de l'appareil, ce qui peut être fort gênant pour les missions envisagées. Le système proposé ici est un système "en boucle ouverte", qui impose à la gouverne des ordres qui ne dépendent que de la turbulence rencontrée, qui est mesurée en temps réel à bord de l'avion. Un tel système a l'avantage de ne rien changer aux qualités de réponse de l'avion aux sollicitations du pilote.

L'objet de l'article est d'exposer comment l'utilisation de la théorie de Wiener permet de déterminer explicitement la loi de pilotage optimale, en fonction de paramètre réduits qui ne dépendent pas directement de la vitesse, mais seulement des caractéristiques massiques de l'avion, de ses coefficients sans dimension de portance et de moment, de la masse volumique de l'air et de l'échelle de la turbulence. L'obtention de la forme analytique de la loi de pilotage permet d'apprécier l'influence de ces différents paramètres, et guide dans la définition de systèmes auto-adaptatifs.

CLOSED FORM EXPRESSION OF THE OPTIMAL CONTROL OF A RIGID AIRPLANE TO TURBULENCE

SUMMARY

Flight of military aircraft at high speed, low altitude makes it necessary to use ride control systems to improve confort, handling qualities and combat ability. For the purpose of deriving such systems, flexibility can be omitted, due to the big difference between the frequencies associated with the Flight Mechanics and the frequencies associated with the first flexible mode. The closed loop systems that feed back some output information to the controls through proper filter, are widely used by the designers, but increases the time response of the aircraft to manoeuvre, which can be a source of trouble for some missions. The open loop system that will be described here senses turbulence which is used, after filtering, to act on the controls. Such a system does not change at all the handling qualities of the aircraft.

The paper explains how Wiener's theory makes it possible to derive in closed form the transfer function of the filter used for control. It shows that this transfer function can be expressed in autoadaptative form, the poles being proportional to the velocity of the aircraft. The influence of parameters like mass, scale of turbulence, is discussed.

Notations

$a(t)$ réponse impulsionnelle d'un paramètre $q(t)$ à la turbulence réduite $\frac{w}{V}$

$w(t)$ composante verticale de la turbulence atmosphérique

V vitesse de vol (en palier)

$\beta(t)$ braquage de la gouverne utilisée pour le contrôle

$b(t)$ réponse impulsionnelle de $q(t)$ au braquage β

$k(t)$ réponse impulsionnelle du système de contrôle

$*$ symbole du produit de convolution

$z(t)$ altitude de vol

M masse de l'avion

I inertie de rotation en tangage

ℓ, S longueur et surface de référence

ρ masse volumique de l'air

θ angle d'assiette

α angle d'incidence

C_z coefficient sans dimension de portance

C_m coefficient sans dimension de moment

$C_{z,\alpha}$ et $C_{m,\alpha}$ leurs dérivées par rapport à α

$$\xi = \frac{z}{\ell}$$

$$c = \frac{I}{\rho/2\, S \ell^3 C_{m,\alpha}}$$

$$m = \frac{M}{\rho/2\, S \ell\, C_{z,\alpha}}$$

$$\eta = \frac{C_{m,q}}{C_{m,\alpha}}$$

$$\lambda = \frac{C_{z,\beta}}{C_{z,\alpha}}$$

$$\nu = \frac{C_{m,\beta}}{C_{m,\alpha}}$$

$\omega_R = \frac{\omega \ell}{V}$ fréquence réduite

$$h = \frac{\eta}{2c}$$

$$\xi^2 = \frac{\nu - \lambda}{\lambda c}$$

$$\nu_R^2 = \frac{1}{c} + \frac{2h}{m}$$

$$\alpha_R = \frac{1}{2\nu_R} \left(2h + \frac{1}{M}\right)$$

$A(i\omega)$ fonction de transfert à la turbulence

$B(i\omega)$ fonction de transfert aux ordres de gouverne

$K(i\omega)$ fonction de transfert du contrôle

$S_w(\omega)$ densité spectrale de la turbulence

L échelle de la turbulence

$$r = \frac{L}{\ell}$$

σ_w^2 variance de la turbulence

$\tilde{\sigma}_{\ddot{z}}^2$ variance de la réponse de l'avion sans autopilote

$\sigma_{\ddot{z}}^2$ variance de la réponse de l'avion avec autopilote

$\gamma = \dfrac{\tilde{\sigma}_{\ddot{z}}^2}{\sigma_{\ddot{z}}^2}$ gain dû au système de contrôle

$F(\omega_R)$ loi de pondération filtrant les hautes fréquences

s variable de Laplace

s_K pôles de la loi de contrôle réduite.

INTRODUCTION

Un certain nombre de missions imposées aux avions militaires correspondent maintenant à des conditions de vol à basse altitude et grande vitesse. Comme la turbulence est particulièrement fréquente et intense au voisinage du sol, l'équipage se trouve, de ce fait, soumis à des conditions sévères qui, ajoutées aux difficultés de suivi de terrain, peuvent entraîner une fatigue marquée, une baisse des réflexes, et, à la limite, une inaptitude au combat. La structure, soumise à des facteurs de charge importants, peut par ailleurs présenter de graves problèmes de fatigue. C'est ce dernier point qui a particulièrement attiré l'attention du Structures and Materials Panel, qui consacre une certaine activité au problème, dans le cadre de l'étude de l'impact des systèmes C.C.V. sur le comportement des structures.

L'exposé se propose d'indiquer dans quelles conditions un avion militaire (par exemple un avion à aile delta), peut être doté, d'une façon relativement simple, d'un système de pilotage automatique qui réduit sa réponse à la turbulence. Le système utilise les gouvernes classiques et ne change en rien la Mécanique du Vol de l'avion en l'absence de turbulence. L'étude exposée ici ne porte que sur le comportement longitudinal d'un avion rigide, en présence d'une turbulence verticale, pour de petits mouvements au voisinage des conditions de croisière ; l'extension au comportement latéral ne présente aucune difficulté. Après une discussion des différentes approches possibles (système en boucle fermée ou système en boucle ouverte) on exprime, grâce à la théorie de Wiener [1], [2], la forme explicite d'un filtre qui, à partir d'une mesure de la turbulence, réalisée en temps réel à bord de l'avion, délivre à la gouverne de contrôle des ordres qui permettent de minimiser la variance d'une réponse quelconque, par exemple l'accélération au centre de gravité. La loi fait apparaître clairement l'influence de la vitesse et de la masse volumique, et, par conséquent, les conditions d'auto-adaptation du pilote automatique. Finalement, les gains que le système est susceptible d'apporter sont eux aussi explicités, et l'influence de différents paramètres (échelle de la turbulence, masse de l'avion, etc.) est analysée.

I - CRITIQUE DES DIFFERENTS SYSTEMES D'OPTIMISATION

Deux types principaux de systèmes peuvent être envisagés pour le contrôle du vol en turbulence : les systèmes en boucle fermée, qui réinjectent dans les gouvernes de contrôle, après un filtrage convenable, certaines réponses de l'avion, et les systèmes en boucle ouverte, qui agissent sur les gouvernes à partir d'une mesure de la turbulence que rencontre l'avion.

Dans le premier système (figure 1a), tout le comportement de l'avion est modifié, en particulier sa mécanique du vol, ses réactions aux ordres du pilote, etc. ; dans le second (figure 1b), toutes les propriétés et qualités de vol restent inchangées, à l'exception des fonctions de transfert à la turbulence. Ceci provient du fait que, dans les systèmes en boucle fermée, le système d'équations différentielles qui représente le mouvement de l'avion est modifié par l'apparition d'un "feed-back" destiné à amortir les différents modes de l'avion, alors que, dans les systèmes en boucle ouverte, le système d'équation reste inchangé, le second membre étant seul transformé par la présence du pilote automatique.

C'est d'abord dans le cadre du programme LAMS, puis, plus récemment, dans celui plus général des systèmes C.C.V., que la NASA et BOEING (Wichita Division), ont entrepris de nombreuses études sur l'optimisation du vol en turbulence par système en boucle fermée [3], [4]. Dans tous les cas, le principe repose sur l'utilisation d'un réseau de contre-réaction qui attaque un système complexe de gouvernes, à partir de mesures accélérométriques et gyrométriques sur la structure. Les constantes du réseau sont choisies de façon à modifier favorablement les fonctions de transfert à la turbulence de certaines réponses critiques.

L'étude a été poursuivie complètement, jusqu'aux essais en vol, sur un B52 équipé spécialement, à cette occasion, de canards horizontaux et verticaux. Le système de contrôle a été défini, théoriquement, en tenant compte de 30 modes élastiques, et vérifié sur calculateur analogique pour tenir compte des limitations d'efficacité des gouvernes ; l'emploi de servocommandes de grand gain et de large bande passante a nécessité des études de stabilité particulièrement délicates. C'est seulement après que l'étude théorique ait été confirmée lors d'essais en soufflerie sur une maquette dynamiquement semblable équipée du système que les essais ont été effectués en vol sur l'avion lui-même. Neuf heures de vol ont été faites en turbulence, au cours desquelles ont été mesurées les fonctions de transfert à la turbulence, avec et sans système de contrôle. La figure (2), tirée de la référence [4], présente ces fonctions de transfert, pour une condition de vol. Le gain obtenu grâce au système, c'est-à-dire le rapport de la variance de l'avion non piloté à la variance de l'avion piloté, est de l'ordre de 3.

Les excellents résultats obtenus s'insèrent dans le cadre d'une politique américaine très générale, orientée vers l'utilisation systématique de contrôles actifs, aussi bien pour le vol en turbulence que pour l'augmentation des vitesses critiques de flottement ou la réduction des charges de manoeuvre. Déjà, le projet de transport supersonique de BOEING faisait largement appel à ces techniques, et on sait que le bombardier B1 sera équipé d'un système de contrôle du vol en turbulence.

Pourquoi, dans ces conditions, s'intéresser encore aux systèmes à boucle ouverte ? Il y a, à notre avis, trois raisons principales :

1- la nécessité, le plus souvent, avec les systèmes en boucle fermée, d'équiper l'avion de gouvernes spéciales,

2- les limites de ces systèmes, qui, dans le domaine de la Mécanique du Vol, sont liées aux qualités de manoeuvrabilité (on ne saurait amortir à l'infini le mode de tangage),

3- les problèmes de stabilité liés aux contre-réactions.

Si les points 1 et 3 n'appellent aucun commentaire particulier, le point 2 peut être illustré, de manière spectaculaire, en montrant que l'on peut supprimer toute réponse d'un avion rigide à la turbulence, par un système en boucle ouverte, sans modifier sa Mécanique du Vol, pourvu que l'on dispose de deux gouvernes indépendantes indéfiniment efficaces. Il s'agit évidemment d'un exemple trivial, très simplifié, dont l'objet est simplement de montrer la possibilité de résultats qui ne sauraient être atteints par aucun système à boucle fermée.

Considérons donc un avion rigide, qui peut être piloté par deux gouvernes indépendantes : une profondeur classique, de braquage β , et un contrôle direct de portance, de braquage σ . Avec les notations indiquées en début de texte, le comportement longitudinal de l'appareil est traduit par les équations :

$$(1)\qquad \begin{cases} M\ddot{z} = \frac{\rho}{2} S C_{z,\alpha}\, V^2\alpha + \frac{\rho}{2} S V^2 C_{z,\beta}\,\beta + \frac{\rho}{2} S V^2 C_{z,\sigma}\,\sigma \\ I\ddot{\theta} = \frac{\rho}{2} S\ell V^2 C_{m,\alpha}\,\alpha - \frac{\rho}{2} S\ell V C_{m,q}\,\dot{\theta} - \frac{\rho}{2} S\ell V^2 C_{m,\beta}\,\beta - \frac{\rho}{2} S\ell V^2 C_{m,\sigma}\,\sigma \end{cases}$$

où α est l'incidence, reliée à l'assiette θ et à la vitesse verticale w de rafale par la relation cinématique :

$$(2)\qquad \alpha = \theta - \frac{\dot{z}}{V} + \frac{w}{V}$$

Supposons que la turbulence soit mesurée en temps réel, à bord de l'avion, par résolution analogique de l'équation (2), et que l'on donne respectivement, aux deux gouvernes, des ordres de pilotage de la forme :

$$\begin{cases} \beta = \mu_0 \frac{w}{V} \\ \sigma = \mu_1 \frac{w}{V} \end{cases}$$

reportant dans l'équation (1), compte tenu de l'équation (2), on aboutit au système :

$$(3)\qquad \begin{cases} M\ddot{z} + \frac{\rho}{2} S V C_{z,\alpha}\,\dot{z} - \frac{\rho}{2} S V^2 C_{z,\alpha}\,\theta = \frac{\rho}{2} S V (C_{z,\alpha} + \mu_0 C_{z,\beta} + \mu_1 C_{z,\sigma}) w \\ I\ddot{\theta} + \frac{\rho}{2} S \ell^2 V C_{m,q}\,\dot{\theta} + \frac{\rho}{2} S\ell V^2 C_{m,\alpha}\,\theta + \frac{\rho}{2} S\ell V C_{m,\alpha}\,\dot{z} = \frac{\rho}{2} S\ell V (C_{m,\alpha} + \mu_0 C_{m,\beta} + \mu_1 C_{z,\sigma}) w \end{cases}$$

Le premier membre des équations, qui traduit la Mécanique du Vol de l'avion, est le même, avec ou sans pilote automatique, alors que l'on peut complètement annuler toute réponse de l'avion à la turbulence en choisissant pour μ_0 et μ_1 , les solutions des équations :

$$\begin{cases} C_{z,\alpha} + \mu_0 C_{z,\beta} + \mu_1 C_{z,\sigma} = 0 \\ C_{m,\alpha} + \mu_0 C_{m,\beta} + \mu_1 C_{m,\sigma} = 0 \end{cases}$$

qui annulent complètement le second membre du système (3).

Il s'agit d'un exemple académique, puisque l'on a supposé les gouvernes infiniment efficaces, l'avion parfaitement rigide, et les forces aérodynamiques indépendantes de la fréquence réduite.

Sans ces réserves, l'exemple illustre parfaitement l'intérêt des systèmes en boucle ouverte ; il est théoriquement possible d'annuler toute réponse de l'avion à la turbulence, sans pour autant modifier en quoi que ce soit la Mécanique du Vol de l'avion.

Ces systèmes ont, malheureusement, un domaine d'application limité aux fréquences associées à la Mécanique du vol de l'avion rigide. Nous avons en effet supposé implicitement, dès le début, qu'il existait "une turbulence W " responsable des réponses de l'avion, turbulence que l'on mesurait en un point de la structure. Cette hypothèse est, dans l'absolu, en contradiction avec l'hypothèse d'isotropie, suivant laquelle la turbulence n'a pas plus de raisons d'être uniforme en envergure que suivant l'axe de vol ; une mesure ponctuelle de la turbulence risque dans ces conditions de n'apporter aucune information sur le champ de perturbation réellement rencontré par l'avion.

Une étude récente [5] permet de cerner le problème, en précisant jusqu'à quelle fréquence une information locale de turbulence renseigne d'une manière significative sur le champ de rafales rencontré par l'avion. On compare, pour ce faire, la demi-envergure b de l'avion à la longueur de cohérence transverse de la turbulence :

$$\Lambda = 1{,}403\,\frac{V}{\omega}$$

calculée pour un spectre de Karman, à une vitesse de translation V . On remarque alors que, chaque fois où la longueur Λ associée à une vitesse V et à un mode de pulsation ω est très grande par rapport à l'envergure $2b$, l'hypothèse d'ondes constantes en envergure sera acceptable ; au contraire, si le paramètre

$$p = \frac{\Lambda}{b} = 1{,}403\,\frac{V}{b\omega}$$

est de l'ordre de grandeur de l'unité, ou inférieur, l'hypothèse de constance en envergure ne pourra plus être retenue, et une mesure ponctuelle de la turbulence ne renseignera en rien sur ce que l'avion subit dans son ensemble. Le tableau 1, qui donne les valeurs du paramètre p pour le mode de tangage et le premier mode de flexion de quatre avions récents, montre à l'évidence que si une information locale sur la turbulence est significative dans le domaine de fréquences de la Mécanique du Vol, elle perd tout intérêt pour les fréquences associées aux modes de déformations.

Les réflexions auxquelles nous venons de nous livrer permettent, semble-t-il, de tirer quelques conclusions provisoires, et une philosophie de conception des systèmes d'optimisation du vol en turbulence : si les modes de déformation contribuent pour une part dominante à la réponse de l'avion, seuls les systèmes en boucle fermée pourraient être utilisés avec succès, car ils ne nécessitent pas la connaissance - illusoire - du champ de turbulence rencontré par l'avion ; si, au contraire, la Mécanique du Vol est le principal responsable du comportement en turbulence, les systèmes en boucle ouverte sont beaucoup plus séduisants, puisqu'ils permettent un contrôle sans modification des qualités de vol (supposées idéales), et puisqu'ils sont réalisables dans le domaine de fréquences considéré. Dans les cas mixtes, on peut envisager un contrôle en boucle ouverte pour les basses fréquences, complété d'un système à contre-réaction aux hautes fréquences, associées aux modes de déformation ; on jugera dans ce cas la qualité du système en boucle ouverte pour la gamme de fréquences sur laquelle il est défini.

II - OPTIMISATION EN BOUCLE OUVERTE

II.1 - Pose du problème

Comme nous l'avons vu au paragraphe précédent, l'optimisation par boucle ouverte d'un avion rigide peut être pratiquement parfaite, si l'on dispose de deux gouvernes indépendantes (profondeur classique et contrôle direct de portance). Bien que certaines réserves doivent être faites, dues à la schématisation abusive du problème, le système est probablement, tout compte fait, le meilleur que l'on puisse concevoir dans ces conditions.

Le problème que nous traitons maintenant est plus complexe, et vise à doter des avions existants, munis de gouvernes classiques, d'un système de pilotage automatique en boucle ouverte qui minimise la variance de leurs réponses à la turbulence. Il s'agit en fait d'un avion à aile delta (un Mirage III) que l'on désire piloter au mieux en turbulence, par manoeuvre des élevons classiques à partir de signaux déduits d'une mesure à bord, en temps réel, de la turbulence.

Le problème s'énonce alors ainsi sous forme mathématique :

" $a(t)$ et $b(t)$ étant respectivement les réponses impulsionnelles à la turbulence réduite w/v et au braquage β de la gouverne de contrôle, trouver une fonction $k(t)$ physiquement réalisable (c'est-à-dire réponse impulsionnelle d'un système stable) telle que les ordres de gouverne :

$$\beta(t) = k(t) * \frac{w(t)}{V}$$

minimisent la variance de la réponse :

$$q(t) = a(t) * \frac{w(t)}{V} + b(t) * \beta(t)$$

du paramètre considéré".

On suppose que l'avion est parfaitement rigide, que la mesure ponctuelle $w(t)$ de la turbulence est représentative, à toutes les fréquences, du champ rencontré par l'avion, et que les forces aérodynamiques sont indépendantes de la fréquence réduite. Les variances des réponses sont calculées pour les bandes de fréquences pour lesquelles on considère la turbulence comme uniforme en envergure.

II.2 - Fonctions de transfert de l'avion

Négligeant tout phénomène de "tamis", nous écrirons les équations de la Mécanique du Vol longitudinale d'un avion rigide, linéarisées autour de conditions de croisière à une vitesse V, sous la forme :

$$(4) \qquad \left\{ \begin{aligned} M\ddot{z} &= \frac{\rho}{2} S V^2 C_{z,\alpha}\, \alpha + \frac{\rho}{2} S V^2 C_{z,\beta}\, \beta \\ I\ddot{\theta} &= -\frac{\rho}{2} S \ell V^2 C_{m,\alpha}\, \alpha - \frac{\rho}{2} S \ell^2 V C_{m,q}\, \dot{\theta} - \frac{\rho}{2} S \ell V^2 C_{m,\beta}\, \beta \end{aligned} \right.$$

Compte tenu de l'équation cinématique

$$\alpha = \theta - \frac{z'}{V} + \frac{w}{V}$$

et avec les notations indiquées en début de texte, on aboutit sans difficulté à la forme non dimensionnelle :

$$(5) \qquad \left\{ \begin{aligned} & m \frac{\ell^2}{V^2} \ddot{\xi} + \frac{\ell}{V} \dot{\xi} - \theta = \frac{w}{V} + \lambda \beta \\ & -\frac{\ell}{V} \dot{\xi} + c \frac{\ell^2}{V^2} \ddot{\theta} + \eta \frac{\ell}{V} \dot{\theta} + \theta = -\frac{w}{V} - \nu \beta \end{aligned} \right.$$

On appellera respectivement $A(i\omega)$ et $B(i\omega)$ la fonction de transfert de l'accélération $\ddot{z}$ à la turbulence réduite $\frac{w}{V}$, et la fonction de transfert aux ordres β de gouverne (ce sont respectivement les transformées de Fourier de $\underset{a}{\overset{V}{}}(t)$ et $b(t)$).

Introduisant la fréquence réduite : $\omega_R = \frac{\omega \ell}{V}$

on mettra ces fonctions de transfert sous la forme :

$$A(i\omega) = \frac{V^2}{\ell m} A'(i\omega_R) \; ; \; B(i\omega) = \frac{\lambda V^2}{\ell m} B'(i\omega_R)$$

où $A'(i\omega_R)$ et $B'(i\omega_R)$ s'expriment uniquement en fonction des variables réduites :

$$(6)\quad \begin{cases} A'(i\omega_R) = \dfrac{\omega_R^2 - 2ih\omega_R}{\omega_R^2 - 2i\alpha_R\nu_R\omega_R - \nu_R^2} \\[2ex] B'(i\omega_R) = \dfrac{\omega_R^2 - 2ih\omega_R + \xi^2}{\omega_R^2 - 2i\alpha_R\nu_R\omega_R - \nu_R^2} \end{cases}$$

II.3 - Enoncé du problème d'optimisation

On appellera $K(i\omega)$ la transformée de Fourier de $k(t)$, c'est-à-dire la fonction de transfert de la loi de contrôle. Dans ces conditions, l'expression de la fonction de transfert de l'avion autopiloté à la turbulence est :

$$T(i\omega) = A(i\omega) + B(i\omega)K(i\omega)$$

soit, en introduisant la loi de contrôle réduite :

$$(7)\quad \tilde{K}(i\omega_R) = \lambda K\left(i\frac{V}{\ell}\omega_R\right)$$

l'expression :

$$T(i\omega) = \frac{V^2}{\ell m}\left(A'(i\omega_R) + B'(i\omega_R)\tilde{K}(i\omega_R)\right)$$

La densité spectrale $\phi_{\ddot{z}}(\omega)$ de l'accélération du centre de gravité s'exprime alors, en fonction de la densité spectrale $S_W(\omega)$ de la turbulence, par :

$$\phi_{\ddot{z}}(\omega) = \frac{1}{V^2}\,|T(i\omega)|^2 S_W(\omega)$$

et la variance $\sigma_{\ddot{z}}^2$ de la réponse, pour les fréquences pour lesquelles on la définit, sera :

$$\sigma_{\ddot{z}}^2 = \int_0^{+\infty} \phi_{\ddot{z}}(\omega)\,F(\omega)\,d\omega = \frac{V^2}{\ell^2 m^2}\int_0^{+\infty} |A'(i\omega_R) + B'(i\omega_R)\tilde{K}(i\omega_R)|^2 F(\omega)\,S_W(\omega)\,d\omega$$

où $F(\omega)$ est un filtre fictif, choisi de façon à limiter le domaine d'intégration sur lequel on calcule la variance.

On remarque alors que tous les modèles de densité spectrale de la turbulence peuvent se mettre, compte tenu de l'hypothèse de Taylor, sous la forme :

$$S_W(\omega_R) = \frac{\tau}{\pi}\,\sigma_W^2\,\psi_W(\omega_R)$$

où τ est le rapport de l'échelle L de la turbulence à la longueur de référence ℓ de l'avion. C'est ainsi que, pour le modèle spectral de Dryden :

$$\psi_W(\omega_R) = \frac{1 + 3\omega_R^2\tau^2}{(1 + \omega_R^2\tau^2)^2}$$

et, pour le modèle de Karman :

$$\psi_W(\omega_R) = \frac{1 + 8/3\,(1{,}339\,\tau\omega_R)^2}{(1 + (1{,}339\,\tau\omega_R)^2)^{11/6}}$$

Compte tenu de ces remarques, la variance $\sigma_{\ddot{z}}^2$ de la réponse se mettra sous la forme :

$$(8)\quad \sigma_{\ddot{z}}^2 = \frac{\tau V^2\sigma_W^2}{\pi \ell^2 m^2}\int_0^{+\infty} |A'(i\omega_R) + B'(i\omega_R)\tilde{K}(i\omega_R)|^2\,\psi_W(\omega_R)\,F(\omega_R)\,d\omega_R$$

et le problème d'optimisation s'énoncera ainsi :

Trouver la fonction de transfert $\tilde{K}(i\omega_R)$ physiquement réalisable telle que :

$$(9)\quad \delta\int_0^{+\infty} |A'(i\omega_R) + B'(i\omega_R)\tilde{K}(i\omega_R)|^2\,\psi_W(\omega_R)\,F(\omega_R)\,d\omega_R = 0$$

l'extremum devant, bien entendu, être un minimum.

Posé ainsi, le problème d'optimisation ne dépend plus explicitement de la vitesse de vol ; il aboutira par conséquent à une loi de contrôle réduite $\tilde{K}(i\omega_R)$ indépendante de V. La loi ne dépendra que de la masse volumique ρ de l'air, de l'échelle L de la turbulence, des coefficients sans dimension $C_{z,\alpha}$ et $C_{m,\alpha}$ (et, à travers eux, du nombre de Mach).

Une autre conclusion peut être tirée de l'étude du gain :

$$\gamma = \frac{\sigma_{\ddot{z}}^2}{\sigma_{z}^2}$$

considéré comme le rapport de la variance de la réponse de l'avion sans autopilote ($\tilde{\sigma}_{\ddot{z}}^2$) à la variance ($\sigma_{\ddot{z}}^2$) de la réponse de l'avion autopiloté. Du fait que :

$$\tilde{\sigma}_{\ddot{z}}^2 = \frac{\tau V^2 \sigma_W^2}{4\pi \ell^2 m^2} \int_0^{+\infty} |A'(i\omega_R)|^2 \psi_W(\omega_R) F(\omega_R) d\omega_R$$

on déduit que le gain, lui aussi, ne dépend pas explicitement de la vitesse de vol, mais seulement de ρ , L , et des $C_{z,x}$ et $C_{m,x}$.

II.4 - Expression de la loi de contrôle, gain et influence des paramètres

A partir de la formule (9) qui énonce le problème d'optimisation, la loi de contrôle réduite $\tilde{K}(i\omega_R)$ est calculée grâce à la théorie du filtre de Wiener. Cette méthode, appliquée pour la première fois par J. Boujot [6], est préférée à l'approche temporelle du filtre de Kalman, qui présente d'énormes difficultés dues au fait que les corrélations de turbulence et les forces aérodynamiques doivent être exprimées comme solutions d'équations différentielles.

Comme la méthode a été complètement développée dans une publication récente, on n'exposera ici que les principaux résultats. Prenant pour fonction de pondération :

$$F(\omega_R) = \frac{\lambda_R^4}{(\omega_R^2 + \lambda_R^2)^2}$$

on obtient, pour un spectre de Dryden, la loi de pilotage réduite :

$$\tilde{K}(s) = -\left[\frac{s(s+2h')}{(s-s_0)(s-s_1)} - \frac{a_0}{\lambda_R^2}\,\frac{(s^2+2\alpha\nu_R s+\nu_R^2)(1+s\tau)^2(s+\lambda_R)^2}{(s-s_0)(s-s_1)(s-s_2)(1+s\tau\sqrt{3})}\right] \tag{10}$$

avec :

$$s = i\,\omega_R$$

et :

$$s_1 = -h' - \sqrt{h'^2 + \xi^2}$$

$$s_0 = -s_2 = -h' + \sqrt{h'^2 + \xi^2}$$

$$a_0 = \frac{2h' s_0 s_1 (1 - s_0 \tau\sqrt{3})}{(s_0^2 + 2\alpha_R \nu_R s_0 + \nu_R^2)^2 (1 - s_0\tau)^2}$$

La loi de contrôle, écrite en variables physiques, a alors pour forme :

$$K(i\omega) = \frac{1}{\lambda}\left[-B_2\frac{\ell^2}{V^2}\omega^2 + i\,B_1\frac{\ell}{V}\omega + B_0 + \frac{C_1\frac{V}{\ell}}{i\omega - s_1\frac{V}{\ell}} + \frac{C_2\frac{V}{\ell}}{i\omega - s_2\frac{V}{\ell}} + \frac{C_3\frac{V}{\ell}}{i\omega - s_3\frac{V}{\ell}}\right] \tag{11}$$

les coefficients B_0 , B_1 , B_2 , C_1 , C_2 , C_3 ne dépendent pas de la vitesse de vol , mais seulement de ρ , L , $C_{z,\alpha}$, $C_{m,\alpha}$, et les pôles sont proportionnels à la vitesse. On a par conséquent parfaitement déterminé l'évolution de la loi de contrôle en fonction de la vitesse, dans les conditions d'adaptation du pilote automatique.

Le gain lui aussi s'exprime explicitement, après quelques acrobaties mathématiques. On trouve :

$$\gamma = \frac{\tilde{\sigma}_{\ddot{z}}^2}{\sigma_{\ddot{z}}^2} = \frac{\nu_R s_0}{2\alpha_R a_0^2}\psi'(\nu_R)$$

et on vérifie sur cette expression qu'il ne dépend pas de la vitesse de vol.

Connaissant le gain sous forme explicite, il est dès lors possible d'étudier l'influence des différents paramètres pour un avion donné (on a choisi ici un Mirage III). L'influence de l'échelle de la turbulence sur le gain est représentée sur la figure (3), et montre que, pour des échelles supérieures à 100 m, le gain est pratiquement indifférent à la valeur de L . Il s'agit là d'un résultat important, car l'échelle de la turbulence est un paramètre mal défini, et mal connu, dont il aurait été gênant qu'il ait une influence notable. La figure (4) représente l'influence de l'altitude sur le gain (en raison de la variation de la masse volumique), influence qui s'avère faible. La figure (5) présente l'évolution du gain en fonction de la fréquence de coupure choisie pour calculer la variance (la fréquence de coupure est définie comme $\lambda_R = p'\nu_R$). L'évolution est très forte, et montre l'importance d'un choix physiquement justifié de cette fréquence de coupure. Si l'on admet, ce qui semble raisonnable, que la coupure se produit quand la longueur de cohérence est égale à la demi-envergure b de l'avion, on trouve, pour p', une valeur voisine de 3, qui correspond à un gain de l'ordre de 12.

CONCLUSION

La communication qui vient d'être présentée ne prétendait pas à un exposé exhaustif du problème du contrôle actif du vol en turbulence. L'objectif était, tout d'abord, de montrer qu'à côté des systèmes en boucle fermée développés, en particulier, aux U.S.A., des systèmes en boucle ouverte présentaient un intérêt certain dans le domaine des fréquences associé à la Mécanique du Vol d'un avion rigide, mais qu'ils avaient eux-mêmes leurs limitations, dues à la nécessité d'une mesure représentative, du champ de rafales rencontré par l'avion. Nous avons ensuite tenté de présenter les caractéristiques principales d'un système en boucle ouverte susceptible d'être adapté à un avion existant, sans modification de la structure ou des gouvernes. Supposant la turbulence mesurée en temps réel, on a déterminé la loi de filtrage fournissant à une gouverne de contrôle unique des ordres permettant de minimiser la variance de la réponse de l'avion. Cette loi de contrôle possède des pôles proportionnels à la vitesse de vol, et aboutit à un "gain" indépendant de la vitesse, et peu sensible à l'échelle de la turbulence et à l'altitude.

Un tel système vient d'être installé sur un Mirage III et subit, au moment où sont écrites ces lignes, ses premiers essais en vol.

REFERENCES

[1] Wiener N. - Extrapolation, interpolation and smoothing of stationary time series. M.I.T. Press (1964).

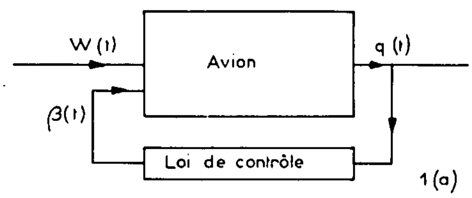
[2] Naslin P. - Introduction à la commande optimale. Dunod - Paris (1966).

[3] Newberry C.F. - Consideration of Stability Augmentation Systems for Large Elastic Airplanes. AGARD - CP N° 46 (1970).

[4] Johannes R.P. - Thompson G.O. - B 52 Control Configured Vehicle Program. BOEING Document D3 - 9169 (1973).

[5] Coupry G. - Effect of Spannise Variation of Gust Velocity on Airplane Response to Turbulence. J. Aircraft Vol. 9, N° 8, Aug. 1972, pp 569-574.

[6] Boujot J. - Sur le contrôle linéaire d'un système soumis à une excitation aléatoire. Application à la tenue d'un avion en atmosphère turbulente. Pub. ONERA N° 131 (1970).



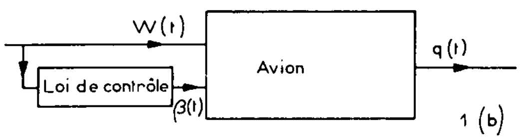


Figure 1. Comparaison des deux systèmes possibles de contrôle

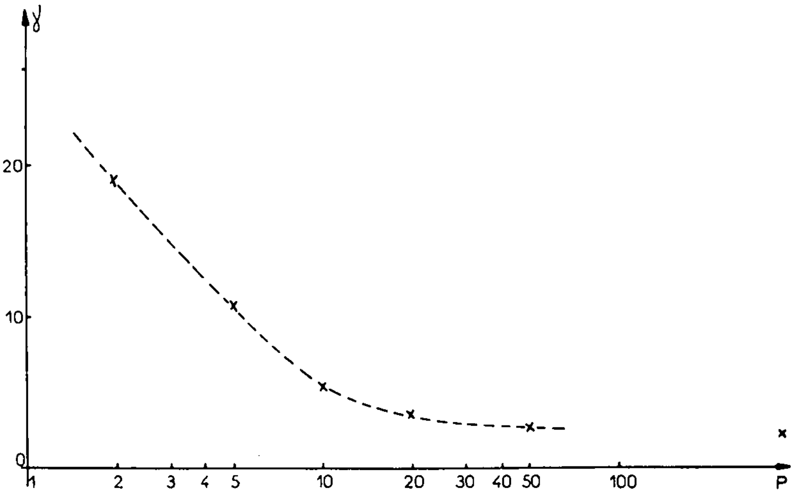


Figure 2. Influence de la fréquence de filtrage sur le gain

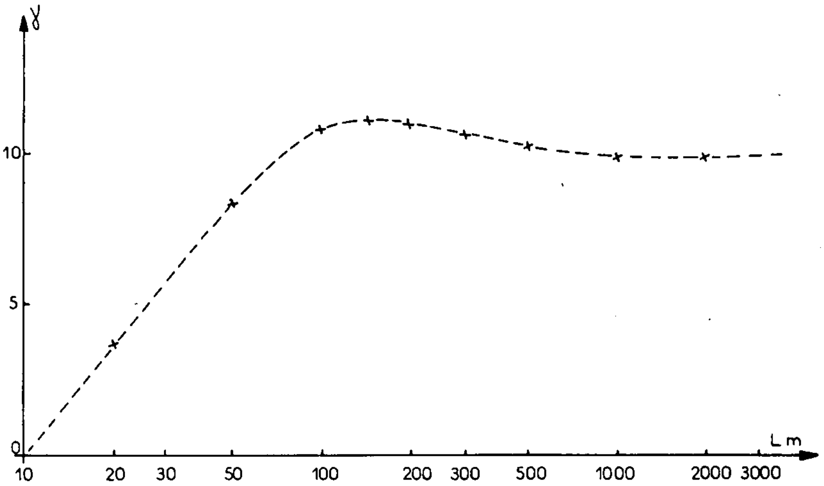


Figure 3. Influence sur le gain de l'échelle de la turbulence

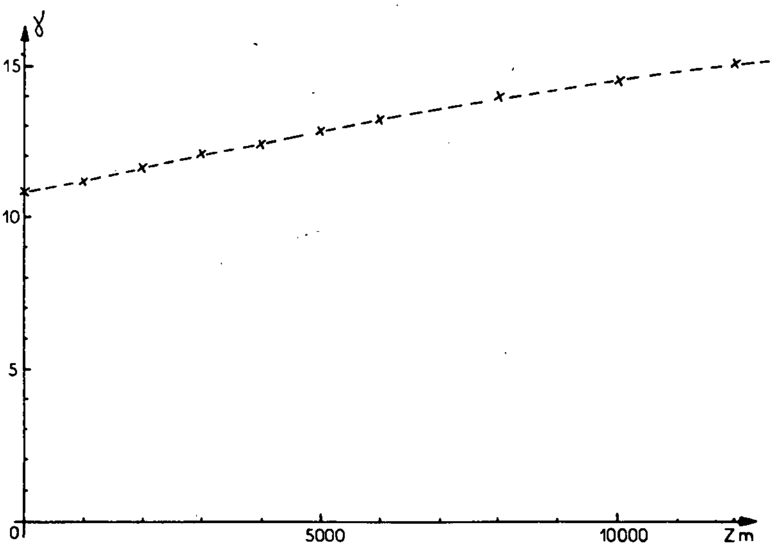


Figure 4 : Influence de l'altitude sur le gain

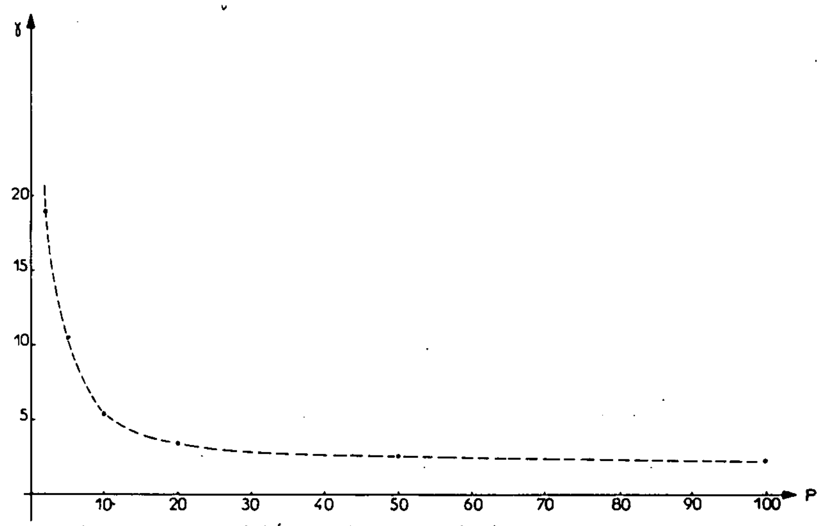


Figure 5 . Influence de la fréquence de coupure sur le gain

Type d'avion	Mode de tangage	Premiere flexion voilure
Caravelle	4	1
B 707	5	0,8
Concorde	10	1,3
B747	7	1,1

Tableau 1 . Valeurs typiques du paramètre P

APPLICATION OF ADVANCED MODEL-FOLLOWING TECHNIQUES TO THE DESIGN OF FLIGHT CONTROL SYSTEMS FOR CONTROL CONFIGURED VEHICLES

by

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SUMMARY

For the longitudinal motion of an aerodynamically unstable aeroplane a controller is designed by the methods of linear optimal control, the control actuators being thrust, elevator and landing flap. One main aim of this paper is to show how the design of such a rather complicated multivariable system is made straightforward.

After a review of optimal control the model-following concept is applied for approaching a desired tracking behaviour, especially concerning the airplane's response to a flight path angle command, in a systematic way. However, it turns out that the disturbance behaviour of the controlled system, represented by the airplane's response to an initial deviation in the flight path angle, is unsatisfactory. Therefore a new concept combining model-following and partial state-vector feedback is applied for designing disturbance behaviour and tracking behaviour separately, in each of both cases achieving a good compromise between the desired system trajectory and limited control action. It appears that the control system thus designed is very insensitive to variations in the most critical parameter, that is the location of the centre of gravity.

1. INTRODUCTION

In the last years investigations have shown that the flying qualities of an aeroplane can be essentially improved by completely resigning the aerodynamic stability of the aeroplane and instead of it generating stability by a control system; such an aeroplane is called "control configured vehicle", in contrary to modern aeroplanes, all of which use control systems for improving flying qualities but in addition are aerodynamically stable.

It is apparent that in such a development the techniques of flight control become increasingly important. Modern theory of multivariable control has to be applied to ensure that the airplane, representing a complex multivariable dynamic system is not only stabilized but shows up flying qualities which are prespecified and, hopefully, do not change much due to parameter variations in the airplane's dynamics.

The purpose of this paper, which is part of a more extensive investigation [2], is to show how modern control theory, especially optimal control, can be used to systematically approach some desired design objectives, concerning especially the tracking and disturbance behaviour of the longitudinal motion.

2. PROBLEM FORMULATION

We start from the set of linearized differential equations determined by the longitudinal motion of an aerodynamically unstable aircraft (destabilized F 104). Linearization is performed about some nominal flight conditions. In state space notation this motion is represented by a dynamic system of fourth order with three input variables, as thrust, elevator and landing flap are assumed to be admitted as control actuators:

$$\underline{\dot{x}} = \underline{A}\,\underline{x} + \underline{B}\,\underline{u} \tag{1}$$

$$\underline{x} = \begin{bmatrix} \frac{\Delta v}{v_o} \\ \gamma \\ \dot{\Theta} \\ \Theta \end{bmatrix}, \qquad \underline{u} = \begin{bmatrix} \frac{\Delta s}{s_o} \\ \eta_e \\ \eta_\ell \end{bmatrix}$$

where

$\underline{A}$: 4×4 system matrix

$\underline{B}$: 4×3 input matrix

$\underline{x}$: state vector consisting of

$\frac{\Delta v}{v_o}$: incremental horizontal speed

γ : flight path angle

$\dot{\Theta}$: pitch rate

Θ : pitch angle

$\underline{u}$: control vector consisting of

$\frac{\Delta s}{s_o}$: incremental thrust

η_e : elevator deflection

η_ℓ : flap deflection

This system description neglects the actuator dynamics, which are caused by the fact, that in first approximation thrust command and thrust variation are related for a fourth order transfer function, elevator command and elevator deflection by a third order transfer function, flap command and deflection by a first order transfer function. These transfer functions are fairly well known and without problems may be taken up into the state space model, yielding a total system order of 12 (4 from the longitudinal motion, 8 from the actuator dynamics).

The elements of $\underline{A}$ and $\underline{B}$ are, of course, dependent on the nominal conditions about which linearization is carried out; investigations showed that the parameters in $\underline{A}$ and $\underline{B}$ are only weakly dependent on the horizontal flight speed (nominally ≈ 0,7 Mach), however, strongly dependent on the location of the centre of gravity $\bar{x}_G$, which is expressed in fractions of the so-called "neutral chord". $\bar{x}_G$ = 0,3 for example means stability as the centre of gravity then lies in front of the "neutral point", however, $\bar{x}_G$ = 0,5 or $\bar{x}_G$ = 0,6 means serious instability as then the centre of gravity is behind the "neutral point". Fig. 1 shows the eigenvalues of $\underline{A}$ depending on the location of the centre of gravity. Note that in the unstable case one pole occurs on the positive real axis. Numerical values for $\underline{A}_{\bar{x}_G=0,5}$ and $\underline{B}_{\bar{x}_G=0,5}$ are:

$$\underline{A}_{\bar{x}_G=0,5} = \begin{bmatrix} -1,272\cdot 10^{-3} & 3,340\cdot 10^{-4} & 0,000 & -1,053\cdot 10^{-3} \\ 5,606 & -1,438 & 3,634\cdot 10^{-3} & 1,438 \\ -2,488\cdot 10^{-2} & -1,322\cdot 10^{1} & -1,134 & 1,322\cdot 10^{1} \\ 0,000 & 0,000 & 1,000 & 0,000 \end{bmatrix}$$

$$\underline{B}_{\bar{x}_G=0,5} = \begin{bmatrix} 6,026\cdot 10^{-3} & -1,802\cdot 10^{-4} & 0,000 \\ -3,725\cdot 10^{-3} & 2,8\cdot 10^{-1} & 2,872\cdot 10^{-1} \\ 5,186\cdot 10^{-1} & 3,008\cdot 10^{1} & 8,875\cdot 10^{-1} \\ 0,000 & 0,000 & 0,000 \end{bmatrix}$$

In the following the $\bar{x}_G$ = 0,5 case is referred to as the nominal case for which a controller is to be designed, the $\bar{x}_G$ = 0,6 case serving as a means for checking the parameter sensitivity of the controlled system, as the controller is required to work satisfactorily also in the nonnominal case.

The controlled system is required to show up the following features:

a) Tracking case ("γ-hold"):

The airplane should follow a flight path angle step command γ_c (for example transition from climbing to horizontal flight) within 2 seconds without overshoot (see Fig. 2).

b) Disturbance case:

When a disturbance has occurred, represented by an initial deviation γ_d (see Fig. 3) caused by a vertical windgust, this initial deviation should decay to zero as smoothly as the deviation of commanded and actual γ in case a).

Furthermore the controller should take in account that control action is limited. As it would be too tedious to invoke the control constraints explicitly in the control law resulting in a nonlinear controller, the requirement posed for the (linear) controller are:

For a γ-command or deviation of 1° the control actions causing the desired transient should not exceed:

15%	incremental thrust
2°	elevator deflection
16°/sec	elevator deflection rate
2°	flap deflection
10°/sec	flap deflection rate .

Finally it is assumed that the state variables are measurable or can be estimated by a filter.

In this context it should be mentioned that both design cases may be treated as "regulator" problems, that is in terms of bringing an initial state to zero. For this purpose one has to refer the actual state $\underline{x}$ to the desired stationary set point $\underline{x}_s$; regarding that in the stationary case $\dot{\underline{x}}_s = \underline{0}$, Eq. (1) may be referred to the stationary values:

$$\underbrace{(\dot{\underline{x}} - \dot{\underline{x}}_s)}_{\Delta\dot{\underline{x}}} = \underline{A}\underbrace{(\underline{x} - \underline{x}_s)}_{\Delta\underline{x}} + \underline{B}\underbrace{(\underline{u} - \underline{u}_s)}_{\Delta\underline{u}} \tag{2}$$

where $\underline{u}_s$ is the stationary control necessary to hold the stationary state $\underline{x}_s$. In this terminology there is no principal difference whether a new set point is introduced or a disturbance has occurred, as in both cases an initial $\Delta\underline{x} = \underline{x}_o$ is generated that has to be zeroed. In the tracking case this initial state is $\underline{x}_o^T = [0 \;\; \gamma_c \;\; 0 \;\; \gamma_c]$, as the pitch angle is desired to change the same amount as the flight path angle (Fig. 2), in the disturbance case it is $\underline{x}_o^T = [0 \;\; -\gamma_d \;\; 0 \;\; 0]$.

3. DESIGNING LINEAR CONTROL SYSTEMS BY OPTIMAL CONTROL

Design specifications for a control system in general are given in the time-domain, requiring that the controlled system should behave in some prespecified manner. In classical controller design people try to transform the design problem from time-domain to the frequency domain, because computations become easier there. However, the direct relationship to the desired time-behaviour is lost.

The introduction of state space methods in the last decades has facilitated the treatment of complex control systems in the time-domain enormously. Well-known design methods using state vector feedback are in particular pole-assignment and quadratic optimization.

The main disadvantages of assigning the poles of the closed-loop system may be seen in the following points:

a) especially in the case of multi-input systems the time-domain behaviour of a system is in general not sufficiently fixed by the location of the poles;

b) there are no simple methods by which nonuniqueness of the feedback gain - occurring in the multi-input case - might be exploited in some useful manner e.g. for minimizing the control activities.

For these reasons in the following only methods of quadratic optimal control are used for the controller design. Just as in most design cases, however, not the optimality of the controlled system represents the main design criterion, but the achievement of certain specifications by means of optimization as a design tool.

Let us start from the general state-space-description of a time-invariant linear system of order n:

$$\begin{aligned} \dot{\underline{x}} &= \underline{A}\,\underline{x} + \underline{B}\,\underline{u} \\ \underline{y} &= \underline{C}\,\underline{x} \end{aligned} \tag{3}$$

where $\underline{x}$ represents the n×1 state vector, $\underline{u}$ the s×1 control vector, $\underline{A}$ the n×n system matrix and $\underline{B}$ the n×s input matrix; the output $\underline{y}$ is formed by multiplying the state with the output matrix $\underline{C}$. This system description is a very general one; it may not only contain the plant to be controlled but may imply some given dynamic models or, e.g. in the case of an aeroplane, the dynamics of the actuator system. Accordingly there are many possibilities to achieve the design specifications by minimizing a quadratic performance index of the form

$$J = \int_0^\infty (\underline{x}^T\,\underline{Q}\,\underline{x} + \underline{u}^T\,\underline{R}\,\underline{u})\;dt \tag{4}$$

where $\underline{Q}$ and $\underline{R}$ are some weighting matrices to be chosen. Eqs. (3) and (4) represents an "optimal regulator" problem. The integrand in Eq. (4) may, for example, contain the squared control error - referred to a constant set-point -, it may, however, also represent the squared deviation of the system output compared to some model trajectory - provided that an appropriate system description has been chosen. Before going into details a short review of the general solution of the optimal regulator problem is given. Usually two cases are of interest:

a) <u>all state variables are fed back</u>

a) <u>all state variables are fed back</u>

that is they are either immediately available or can be estimated by a filter. Minimizing the criterion in Eq. (4) then yields a linear constant feedback law:

$$\underline{u} = -\underline{R}^{-1}\,\underline{B}^T\,\underline{P}\,\underline{x} = \underline{K}\,\underline{x} \tag{5}$$

where $\underline{P}$ is the stationary solution of the so-called matrix-Riccati-equation:

$$-\dot{\underline{P}} = \underline{O} = \underline{P}\,\underline{B} + \underline{B}^T\,\underline{P} - \underline{P}\,\underline{B}\,\underline{R}^{-1}\,\underline{B}^T\,\underline{P} + \underline{Q} \tag{6}$$

Using all state variables for feedback guarantees optimality for all initial states. If the system matrix $\underline{A}$ contains only the plant to be controlled, then the feedback law determines the tracking behaviour as well as the disturbance behaviour (one-degree-of-freedom-system).

Mainly in order to give some feeling for the amount of the airplane's parameter variations due to $\overline{x}_G$, Fig. 4 shows the γ-tracking and disturbance transients of the longitudinal motion stabilized by an optimal feedback law (5) with

$$\underline{k}^T = [1.476 \quad -3.068\cdot 10^{-1} \quad 2.38\cdot 10^{-1} \quad 9.032\cdot 10^{-1}]$$

which uses only the elevator as a control input and which was found in a trial and error process. In the nominal case $\overline{x}_G = 0{,}5$ the design specifications including control activities are met, for $\overline{x}_G = 0{,}6$ however, considerable overshoot occurs. This feedback law is denoted "controller C_O" and will serve for comparisons later on.

b) <u>only part of the state variables is fed back</u>

(partial state vector feedback). In a) the feedback matrix $\underline{K}$ could be chosen arbitrarily. Occasionally however, one has to deal with the problem that only the immediately measurable state variables or a certain well-defined part of the states occurring in the global system description can be used for feedback. This problem is formulated in the following way:

For the time-invariant system

$$\dot{\underline{x}} = \underline{A}\,\underline{x} + \underline{B}\,\underline{u}$$

find that linear feedback law $\underline{u} = \underline{K}^*\,\underline{v}$, which uses only the linear transformation $\underline{v} = \underline{D}\,\underline{x}$ (for example the output) for feedback, that is $\underline{u} = \underline{K}^*\,\underline{D}\,\underline{x}$, and at the same time minimizes the performance index

$$J = \int_0^\infty (\underline{x}^T\,\underline{Q}\,\underline{x} + \underline{u}^T\,\underline{R}\,\underline{u})\,dt$$

Generally any controller $\underline{u} = \underline{K}\,\underline{x}$ causes the costs

$$J = \int_0^\infty (\underline{x}^T\,\underline{Q}\,\underline{x} + \underline{x}^T\,\underline{K}^T\,\underline{R}\,\underline{K}\,\underline{x})\,dt \tag{7}$$

Taking in account that the movement of the closed-loop system $\dot{\underline{x}} = (\underline{A} + \underline{B}\,\underline{K})\,\underline{x}$ is easily computed as

$$\underline{x}(t) = e^{(\underline{A}+\underline{B}\underline{K})t}\,\underline{x}_o = \underline{\phi}(t)\,\underline{x}_o \tag{8}$$

the value of the performance index comes out to be:

$$J = \underline{x}_o^T \int_0^\infty \underline{\phi}^T\,[\underline{K}^T\,\underline{R}\,\underline{K} + \underline{Q}]\,\underline{\phi}\,dt\,\underline{x}_o = \underline{x}_o^T\,\underline{P}\,\underline{x}_o \tag{9}$$

The cost matrix $\underline{P}$ can be computed as the solution of the bilinear matrix equation (see [1]):

$$\underline{P}(\underline{A} + \underline{B}\,\underline{K})^T + (\underline{A} + \underline{B}\,\underline{K})\,\underline{P} + \underline{K}^T\,\underline{R}\,\underline{K} + \underline{Q} = \underline{O} \tag{10}$$

As we suppose that in the matrix $\underline{K}$ only the factor $\underline{K}^*$ can be chosen (remember $\underline{K} = \underline{K}^*\,\underline{D}$), the optimal feedback matrix becomes dependent on the initial state $\underline{x}_O$. The same is true for the cost matrix. However for practical reasons it is desirable to have just one $\underline{K}$ as the optimal feedback gain. In order to evade this difficulty at least partially, two ways can be gone, the first of which is proposed in the literature ([1], [3]):

α) One assumes the initial state to be a random variable, uniformly distributed on the surface of the n-dimensional unit-sphere, and seeks the particular feedback matrix minimizing the expected value of the costs over all $\underline{x}_O$. One can show that this leads to minimizing the trace of the cost matrix $\underline{P}$, that is one has to minimize a scalar function $f(\underline{K})$, where

$$f = \text{trace}\,\underline{P}$$

β) One actually optimizes the feedback matrix with respect to some special initial state $\underline{x}_O$, which is for example the most probable one. The function f then to be optimized clearly is:

$$f = \underline{x}_o^T\,\underline{P}\,\underline{x}_o$$

In both cases α) and β) one has to minimize a scalar function f with respect to a set of parameters contained in the matrix $\underline{K}^*$. The gradient of f with respect to $\underline{K}^*$ can be computed analytically (see [1]):

$$\frac{\partial f}{\partial K^{*T}} = \underline{D}\,\underline{M}\,[\underline{D}^T\,\underline{K}^{*T}\,\underline{R} + \underline{P}\,B] \tag{11}$$

where the matrix $\underline{M}$ is solution of the following bilinear matrix-equation

$$\underline{M}(\underline{A} + \underline{B}\,\underline{K}^*\,\underline{D})^T + (\underline{A} + \underline{B}\,\underline{K}^*\,\underline{D})\,\underline{M} + \underline{W} = 0 \tag{12}$$

where

$\underline{W} = \underline{I}$ in case α)

$\underline{W} = \underline{x}_o\,\underline{x}_o^T$ in case β)

For finding the optimal gain matrix $\underline{K}^*$ one might zero the gradient in Eq. (11) and try to solve the system of Eqs. (10)-(12); for example iteratively as proposed in [1] by assuming an initial $\underline{K}_o^*$ which stabilizes the system, computing $\underline{P}_o$ and $\underline{M}_o$ from Eq. (10) and Eq. (12) and inserting these matrices in Eq. (11), from which a first iteration value $\underline{K}_1^*$ may be found. However, nothing can be said about the convergence of this method and indeed computational results showed that in many cases no convergence is obtained. Therefore it seems preferable to use an ordinary gradient algorithm for finding the optimum, as gradient and function value are explicitly computable by Eq. (11) and Eq. (10) referently.

Conditions for solvability

The solutions just given presume that certain conditions, the so-called regularity conditions, are satisfied. Especially in the case of complete state vector feedback (case α)) sufficient conditions for the existence of a unique optimal control law that guarantees asymptotic stability of the closed-loop system are:

a) $\underline{R}$ is positive definite

b) the system is "stabilizable", that is noncontrollable subsystems are asymptotically stable

c) the system is "observable from the performance index", that is all movement of the system must show up in the performance index; this guarantees that unstable subsystems are surely stabilized. Positive definiteness of $\underline{Q}$ for example assures this sort of observability.

When applying partial state vector feedback to a non-asymptotically stable system one has firstly to check whether stabilization is possible at all.

4. USE OF REFERENCE TRAJECTORIES FOR CONTROLLER DESIGN

The simplest method using optimal control clearly is to take up only the plant into the system description and then varying the weighting matrices $\underline{Q}$ and $\underline{R}$ by trial and error until a feedback matrix arises that yields a satisfying system behaviour. But it is this trial and error process that has brought a lot of criticism to optimal control. In fact no metnod is known which would allow to vary the weighting coefficients q_{ij} and r_{ij} in a systematic way thus that the dynamics of the controlled system would be influenced in some desired manner. This lack of insight into the physical relationships between weighting coefficients on one side and time behaviour on the other side gives a motivation for introducing reference trajectories. One then does no longer weight the squared difference of the state from the zero state, but the squared deviations of the interesting output variables $\underline{y}$ from desired model trajectories $\tilde{\underline{y}}$:

$$J = \int_0^\infty [(\underline{y} - \tilde{\underline{y}})^T\,\underline{Q}_1(\underline{y} - \tilde{\underline{y}}) + \underline{u}^T\,\underline{R}\,\underline{u}]\,dt \tag{13}$$

Progressing this way shows up some important advantages:

a) by restricting to the output, the number of weighting coefficients is in general essentially reduced

b) the physical relationship between the weighting coefficients in $\underline{Q}_1$ and the time behaviour of the controlled system is apparent. One can expect that the difference between actual and model trajectory decreases with increasing weighting coefficient.

What remains to do then is to approximate the desired model trajectories as given by the design specifications by differential equations, that is by dynamic models and thus to bring them into a form accessible to optimization. There are two basic possibilities when taking up a model into the system description:

a) The states of the model are used for feedback. In this case the controller contains a realization of the model that generates the reference trajectory.

b) The states of the model and thereby the reference trajectory are not used for feedback (application of partial state vector feedback). In this case the reference is an auxiliary means for the controller design. (The application of this method for finding a feedback matrix which gives some desired disturbance behaviour is treated in more detail in Chapter 6.2).

In the following section the fundamentals of model following control are derived; furthermore its application to the design of a controller for the longitudinal airplane-motion is shown.

5. MODEL-FOLLOWING CONTROL

5.1 Fundamentals

For the dynamical system

$$\underline{\dot{x}} = \underline{A}\,\underline{x} + \underline{B}\,\underline{u} \tag{14}$$

the quadratic optimization problem is extended thus that the deviation of the interesting output variables $\underline{y} = \underline{C}\,\underline{x}$ from their model trajectories $\underline{\tilde{y}}$ occurs in the performance criterion. For convenience it is assumed that $\underline{y}$ itself can be interpreted as the output of a dynamical system

$$\begin{aligned} \underline{\dot{z}} &= \underline{\tilde{A}}\,\underline{z} \\ \underline{\tilde{y}} &= \underline{\tilde{C}}\,\underline{z} \end{aligned} \tag{15}$$

Remark:

Inputs to the model are not explicitly taken in account in Eq. (15). If however, one restricts to the particularly important case, that the model input $\underline{u}_M$ can be modelled in a form analogous to Eq. (15) (for example step- and ramp functions) then Eq. (15) is easily extended to the case where inputs are implied

$$\begin{aligned} \underline{\dot{z}}_1 &= \underline{\tilde{A}}_1\,\underline{z}_1 + \underline{\tilde{B}}\,\underline{u}_M \\ \underline{\dot{u}}_M &= \underline{\tilde{A}}_2\,\underline{u}_M \end{aligned} \tag{16}$$

Combining these equations yields the standard form:

$$\underline{\dot{z}} = \underline{\tilde{A}}\,\underline{z}$$

$$\underline{z} = \begin{bmatrix} z_1 \\ u_M \end{bmatrix} \qquad \underline{\tilde{A}} = \begin{bmatrix} \underline{\tilde{A}}_1 & \underline{\tilde{B}} \\ \underline{0} & \underline{\tilde{A}}_2 \end{bmatrix} \tag{17}$$

Remember now the performance criterion introduced in Eq. (13):

$$J = \int_0^\infty [\underline{u}^T\,\underline{R}\,\underline{u} + (\underline{y} - \underline{\tilde{y}})^T\,\underline{Q}_1(\underline{y} - \underline{\tilde{y}})]\,dt \tag{18}$$

For reducing this optimization problem to a standard regulator problem we introduce the augmented state $\underline{x}$ and the augmented system matrices $\underline{A}$ and $\underline{B}$:

$$\underline{\dot{\hat{x}}} = \underline{\hat{A}}\,\underline{\hat{x}} + \underline{\hat{B}}\,\underline{u}$$

$$\underline{\hat{x}} = \begin{bmatrix} \underline{x} \\ \underline{z} \end{bmatrix} ; \qquad \underline{\hat{A}} = \begin{bmatrix} \underline{A} & \underline{0} \\ \underline{0} & \underline{\tilde{A}} \end{bmatrix} ; \qquad \underline{\hat{B}} = \begin{bmatrix} \underline{B} \\ \underline{0} \end{bmatrix} \tag{19}$$

From the definition of $\underline{x}$ we derive the performance index Eq. (18) to be:

$$J = \int_0^\infty [\underline{u}^T\,\underline{R}\,\underline{u} + \underline{\hat{x}}^T \begin{pmatrix} \underline{C}^T \\ -\underline{\tilde{C}}^T \end{pmatrix} \underline{Q}_1(\underline{C},\ -\underline{\tilde{C}})\,\underline{\hat{x}}]\,dt \tag{20}$$

$$\underline{J} = \int_0^\infty (\underline{u}^T\,\underline{R}\,\underline{u} + \underline{\hat{x}}^T\,\underline{\hat{Q}}\,\underline{\hat{x}})\,dt \tag{21}$$

where

$$\underline{\hat{Q}} = \begin{bmatrix} \underline{C}^T\,\underline{Q}_1\,\underline{C} & -\underline{C}^T\,\underline{Q}_1\,\underline{\tilde{C}} \\ -\underline{\tilde{C}}^T\,\underline{Q}_1\,\underline{C} & \underline{\tilde{C}}^T\,\underline{Q}_1\,\underline{\tilde{C}} \end{bmatrix} \tag{22}$$

Eqs. (19) and (21) define a standard regulator problem. Let us assume that the regularity conditions are satisfied, then the optimal control law is known to be:

$$\underline{u} = -\underline{R}^{-1}\,\underline{\hat{B}}^T\,\underline{\hat{P}}\,\underline{x} \tag{23}$$

where $\underline{\hat{P}}$ is solution of

$$\underline{0} = \underline{\hat{P}}\,\underline{\hat{A}} + \underline{\hat{A}}^T\,\underline{\hat{P}} - \underline{\hat{P}}\,\underline{\hat{B}}\,\underline{R}^{-1}\,\underline{\hat{B}}^T\,\underline{\hat{P}} + \underline{\hat{Q}} \tag{24}$$

Splitting up $\underline{\hat{P}}$ into

$$\underline{\hat{P}} = \begin{bmatrix} \underline{P} & \underline{P}_{21}^T \\ \underline{P}_{21} & \underline{P}_{22} \end{bmatrix} \tag{25}$$

with $\underline{P}$ a submatrix of the system's order, the result of Eq. (23) can be rewritten as:

$$\underline{u} = \underline{K}_2\,\underline{x} + \underline{K}_1\,\underline{z} \tag{26}$$

where

$$\underline{K}_2 = -\,\underline{R}^{-1}\,\underline{B}^T\,\underline{P} \tag{27}$$

$$\underline{K}_1 = -\,\underline{R}^{-1}\,\underline{B}^T\,\underline{P}_{21} \tag{28}$$

Splitting up the algebraic equation (24) in a similar way, we arrive at two equations for $\underline{P}$ and $\underline{P}_{21}$:

$$\underline{O} = \underline{P}\,\underline{A} + \underline{A}^T\,\underline{P} - \underline{P}\,\underline{B}\,\underline{R}^{-1}\,\underline{B}^T\,\underline{P} + \underline{C}^T\,\underline{Q}_1\,\underline{C} \tag{29}$$

$$\underline{O} = \underline{P}_{21}\,\underline{A} + \underline{\tilde{A}}^T\,\underline{P}_{21} - \underline{P}_{21}\,\underline{B}\,\underline{R}^{-1}\,\underline{B}^T\,\underline{P} - \underline{\tilde{C}}^T\,\underline{Q}_1\,\underline{C} \tag{30}$$

A third equation for the submatrix $\underline{P}_{22}$ is omitted, because it is of no great interest in this context. We make the following conclusions from Eqs. (27) and (29):

a) The optimal model-following-controller consists of a feedback part $\underline{K}_2\,\underline{x}$ processing the state of the plant, and a feedforward part $\underline{K}_1\,\underline{z}$ processing the state of the model. This structure is shown in Fig. 5.

b) For computation of $\underline{K}_2$ only the submatrix $\underline{P}$ is needed which in accordance to Eq. (29) is independent of the model. In other words the feedback matrix $\underline{K}_2$ is influenced only by $\underline{Q}_1$ and $\underline{R}$. This seems to be surprising, but it becomes intelligible when one bears in mind that the controller must be optimal for all initial states and therefore also for those where $\underline{z}$, the model state, is zero.

Here it should be mentioned that basically a model-following-system with the structure of Fig. 5 constitutes a so-called "2-degree-of-freedom" controller with feedforward and feedback part. The model can be considered as a forefilter. For the tracking behaviour both feedforward- and feedback part are responsible, whilst the disturbance behaviour is determined by the feedback loop only, as in case of a disturbance acting on the plant the forefilter is not excited. In accordance to these considerations it is principally possible with a 2-degree-of-freedom controller to design tracking and disturbance behaviour separately. The model-following design as described previously fixes both feedforward and feedback gain in the sense of optimal tracking. The problems which may arise hereby when disturbances occur are discussed later on.

It was mentioned above that the results of this section are to applied with care. The reason for this warning is that the matrices $\underline{A}$ and $\underline{B}$ representing the augmented system do not constitute a completely controllable system as the model represents a non-controllable subsystem. This may make transition to an infinite optimization interval problematic. The consequences that may arise depend on the eigenvalues of the model matrix $\underline{\tilde{A}}$. Different cases may occur, proposed the original plant is completely controllable:

1. Assume that all eigenvalues of $\underline{\tilde{A}}$ are contained in the set of eigenvalues of $\underline{A}$. In this case the problem can be reformulated thus that complete controllability is given. This suggests for the case when the assumption is not satisfied by the original plant that compensation be included so that the resultant modified plant satisfies the assumption.
2. At least the non-asymptotically stable eigenvalues of $\underline{\tilde{A}}$ are contained in $\underline{A}$. The augmented system then is "stabilizable" and no problems will arise.
3. $\underline{\tilde{A}}$ has one or more non-asymptotically stable eigenvalues, which are not contained in $\underline{A}$ and are not generated by a compensator; as an example take the case where in the model step or ramp inputs are included introducing one or two poles at s = 0, the system however, does not contain any integration. In such cases the stationary error does not tend to zero, thereby letting the integral costs in Eq. (18) grow to infinity. As is shown in [4] however, the submatrices $\underline{P}$ and $\underline{P}_{21}$ and thereby the gain matrices $\underline{K}_2$ and $\underline{K}_1$ remain finite, if any sum of an eigenvalue of $\underline{\tilde{A}}$ with an eigenvalue of $(\underline{A} + \underline{B}\,\underline{K}_2)$ has negative real part. One always can achieve this constellation by choosing $\underline{Q}_1$ and $\underline{R}$ appropriately. However, nothing can be said about the optimality of the gain matrices in this case.

5.2 Design of a Model-Following Controller for the Longitudinal Motion of an Aerodynamically Unstable Airplane (Controller C_1)

It was already emphasized in Section 3, that when designing a control system generally one has concrete ideas concerning a desired time behaviour of the controlled system. In particular this is true for the tracking behaviour characterized by the transient response after a step input has occured. In our design problem, this transient response is characterized by the requirement that *after a step command specifying a new γ-set point has occurred, the actual flight path angle should reach this new set point within 2 se-*

conds with an accuracy of 4%. Furthermore the transient should not show up any overshoot, its shape being properly generated by a second order model with two real eigenvalues. Numerical values for these two eigenvalues are easily obtained by requiring that

a) the just mentioned specifications are satisfied

b) the maximal load factor occurring during the transient becomes minimal (recall that the load-factor is proportional to $\dot{\gamma}$).

With these considerations one arrives at the second order dynamic model

$$\dot{\underline{z}} = \begin{bmatrix} 0 & 1 \\ -6,25 & -5 \end{bmatrix} \underline{z} + \begin{bmatrix} 0 \\ 1 \end{bmatrix} \gamma_c \tag{31}$$

$$\dot{\gamma}_c = 0$$

where the model state $\underline{z}^T = [\gamma_M \quad \dot{\gamma}_M]$ consists of the model flight path angle γ_M and its derivative $\dot{\gamma}_M$. Modelling the command signal γ_c which is assumed to be piecewise constant would make it necessary to extend the model to a third order one, containing a pole at s = 0. As this nonasymptotically stable pole is not contained in the plant, a stationary error would result (see case 3 of the last section). A way out of this dilemma can be found by extracting the stationary set-point values. For the model too we have in the stationary case $\dot{\underline{z}}_s = 0$ and thus:

$$(\dot{\underline{z}} - \dot{\underline{z}}_s) = \underbrace{\tilde{\underline{A}}(\underline{z} - \underline{z}_s)}_{\Delta z} + \begin{bmatrix} 0 \\ 1 \end{bmatrix} (\gamma_c - \gamma_{cs}) \tag{32}$$

Because of $\dot{\gamma}_c = 0$ we arrive at:

$$\Delta\dot{\underline{z}} = \tilde{\underline{A}}\, \Delta\underline{z} \tag{33}$$

The scalar output

$$\tilde{y} = \tilde{\underline{c}}^T \Delta\underline{z} = [1 \quad 0]\, \Delta\underline{z} \tag{34}$$

of this model is to be followed by the output y of the plant (that is the flight path angle γ), where the state components of the plant are also referred to their stationary values

$$y = \underline{c}^T \Delta\underline{x} = [0 \quad 1 \quad 0 \quad 0]\, \Delta\underline{x} \tag{35}$$

(see Section 2, where the state of the longitudinal motion was defined as $\underline{x}^T = [\frac{\Delta v}{v_0} \quad \gamma \quad \dot{\Theta} \quad \Theta]$

The three control imputs used are thrust, elevator and landing flap.

According to the results of Section 5.1, the control law has the form

$$\Delta\underline{u} = \underline{K}_2\, \Delta\underline{x} + \underline{K}_1\, \Delta\underline{z} \quad . \tag{36}$$

The structure of the controlled system is shown in Fig. 6.

A commanded set point γ_c produces - by a linear transformation - the stationary control $\underline{u}_s$ and the stationary state $\underline{x}_s^T = [0 \quad \gamma_c \quad 0 \quad \gamma_c]$. Note that the stationary pitch angle Θ is set equal to γ_c, because it is assumed that stationarily the pitch angle is to be changed the same amount as the flight path angle.

After these statements the results of the previous section can be applied immediately. The weighting matrix $\underline{Q}_1$ by which the difference $(\underline{y} - \underline{y})$ is punished degenerates to a scalar factor q_1 and the weighting matrix $\hat{\underline{Q}}$ occurring in the standardized formulation (Eq. (21)) turns out to be

$$\hat{\underline{Q}} = \left[\begin{array}{cccc|cc} 0 & & & & 0 & 0 \\ & q_1 & & & -q_1 & 0 \\ & & 0 & & 0 & 0 \\ & & & 0 & 0 & 0 \\ \hline 0 & -q_1 & 0 & 0 & q_1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \end{array}\right] \tag{37}$$

As the $\hat{\underline{Q}}$-matrix contains only one coefficient, which furthermore can be chosen arbitrarily (only the relative weights of $\underline{Q}$ and $\underline{R}$ are of importance), there are just the three diagonal elements of $\underline{R}$ to be varied. This can be done in a relatively systematic way, for increasing the control weights leads to a better approximation of the model but at the same time to greater control amplitudes. Thus one can increase the coefficients in $\underline{R}$ until all controls satisfy the constraints for a typical transient (see Section 2).

Numerical computations performed in this way showed that a weighting matrix $\underline{R}$ = diag [2, 12, 4] is appropriate (q_1 set equal to 50). The results of this design are shown in Fig. 7a and 7b (Controller C_1). The tracking behaviour turns out to be very insensitive to parameter variations. There is no overshoot when the centre of gravity has changed to $\bar{x}_G = 0{,}6$ (the single-input feedback controller C_O shows up 20% overshoot, see Fig. 4).

For checking the disturbance behaviour which is completely determined by the feedback matrix $\underline{K}_2$, an initial state $\underline{x}_0^T = [0 \ \ -1 \ \ 0 \ \ 0]$ is assumed as it may be caused by a vertical windgust (see Sec. 2). The deviation in γ turns out to decay to zero unnecessarily fast in connection with a disagreeable loadfactor (Fig. 7b). This behaviour is caused by the fact that as the airplane is unstable it tries to rear, thereby generating more lift and reducing the γ-deviation quickly. In the first moment this movement is even reinforced, mainly by the elevator action (see Fig. 10b); then suddenly all controls are used to counteract this virtual unstability. In addition to this shortcoming an overshoot of 16% occurs for $\bar{x}_G = 0{,}6$. For these reasons in a second iteration a separate design of tracking and disturbance behaviour is performed.

5.3 Implying the Actuator Dynamics

Until now it was assumed that the airplane's longitudinal motion may by represented by a fourth order system. However, it is not difficult to imply the actuator dynamics which are described by a dynamical system of order 8. The whole system together with the second order model is then of order 14. The control variables are no longer the actuator deflections, but the signals produced by the controller.

However, one has to take in account that in general the states of the actuator dynamics are not available for feedback and modelling them does not seem to be adequate. Thus one has to deal with a problem of partial state vector feedback, the general solution of which is given in Section 3. In the course of the inquiry it appeared that no essential improvement was achieved by implying the actuator dynamics. The same proved to be true for an additional feedback of those states which would be easily measurable, that is the actuator deflections.

6. SEPARATION OF TRACKING AND DISTURBANCE BEHAVIOUR (CONTROLLER C_2)

6.1 Motivation

Though a model-following controller has the structure of a 2-degrees-of-freedom controller, which would allow to separate the design of tracking and disturbance behaviour, the feedback matrix $\underline{K}_2$ which determines the disturbance behaviour, is independent of the model (see Sec. 5.1). One can therefore not expect that the disturbance behaviour might be as favourable as the tracking behaviour. This statement was proved in simulations as mentioned in the preceding section. On the other side the indpendence of $\underline{K}_2$ from the model encourages to first design of feedback loop for good disturbance response and afterwards compute the feedforward matrix $\underline{K}_1$ by which the model state is introduced in case of tracking.

6.2 Generating a Desired Disturbance Behaviour

Also when designing only the feedback-loop, one can restrict oneself to such methods that guarantee an immediate reference to a desired time behaviour of the closed-loop system. In the present design problem this ideal behaviour is specified by demanding that an initial deviation in γ, caused by some disturbance, should decay to zero exponentially with a time-constant of $T = 1/1{,}5$ sec, that is slowlier and smoother than in the preceding design (see Fig. 7b). Two different methods are applied to reach this aim. Only the first of which is known from literature:

a) model-in-performance-index

This method starts from desired closed-loop dynamics which the feedback controller should generate; that is one requires the closed-loop dynamics of the system

$$\left.\begin{aligned} \dot{\underline{x}} &= \underline{A}\,\underline{x} + \underline{B}\,\underline{u} \\ \underline{u} &= \underline{K}\,\underline{x} \end{aligned}\right\} \qquad (38)$$

to be as close as possible to the dynamics of a model

$$\dot{\underline{x}}_M = \underline{A}_M\,\underline{x}_M \qquad (39)$$

where the degree of approximation is steered by means of the performance index

$$J = \int_0^\infty [(\dot{\underline{x}} - \dot{\underline{x}}_M)^T\,\underline{Q}(\dot{\underline{x}} - \dot{\underline{x}}_M) + \underline{u}^T\,\underline{R}\,\underline{u}]\,dt \qquad (40)$$

In order to have to deal with an easily solvable problem, one usually substitutes $\underline{A}_M\,\underline{x}$ for $\dot{\underline{x}}_M$ instead of $\underline{A}_M\,\underline{x}_M$ (one should be aware however, that thereby the direct comparison of model trajectory and system trajectory is lost). With this simplification Eq. (40) comes

$$J = \int_0^\infty [\underline{x}^T\,(\underline{A} - \underline{A}_M)^T\,\underline{Q}(\underline{A} - \underline{A}_M)^T\,\underline{x} + 2\,\underline{u}^T\,\underline{B}^T\,\underline{Q}(\underline{A} - \underline{A}_M)\,\underline{x} + \underline{u}^T(\underline{B}^T\,\underline{Q}\,\underline{B} + \underline{R})\underline{u}]\,dt \qquad (41)$$

Note that the integrand contains a "mixed" term. The optimal controller in this case is (see [1]):

$$\underline{u} = - \overline{\underline{R}}^{-1} (\underline{B}^T \underline{P} + \underline{S}) \underline{x} \tag{42}$$

where $\underline{P}$ is solution of

$$0 = \underline{P}(\underline{A} - \underline{B}\,\overline{\underline{R}}^{-1} \underline{S}) + (\underline{A}^T - \underline{S}^T \overline{\underline{R}}^{-1} \underline{B}^T) \underline{P} - \underline{P}\,\underline{B}\,\overline{\underline{R}}^{-1} \underline{B}^T \underline{P} + \overline{\underline{Q}} - \underline{S}^T \overline{\underline{R}}^{-1} \underline{S} \tag{43}$$

and the following denotations were made

$$\left.\begin{aligned} \overline{\underline{Q}} &= (\underline{A} - \underline{A}_M)^T \underline{Q}(\underline{A} - \underline{A}_M) \\ \overline{\underline{R}} &= \underline{B}^T \underline{Q}\,\underline{B} + \underline{R} \\ \overline{\underline{S}} &= \underline{B}^T \underline{Q}(\underline{A} - \underline{A}_M) \end{aligned}\right\} \tag{44}$$

As in this investigation the aim is not to simulate the dynamics of the completely specified model airplane, only those elementes in $\underline{A}_M$ and $\underline{Q}$ are filled which concern the desired γ-dynamics:

$$\underline{A}_M = \begin{bmatrix} 0 & & & 0 \\ & -1,5 & & \\ & & 0 & \\ 0 & & & 0 \end{bmatrix} \qquad \underline{Q} = \begin{bmatrix} 0 & & & \\ & 50 & & \\ & & 0 & \\ & & & 0 \end{bmatrix}$$

Again only the control weights are varied.

With a weighting matrix $\underline{R}$ = diag[0.6, 10, 0.3] indeed the transient differs from that of the model for no more than 4%.

b) model-following without realizing the model

The model-in-performance-index-criterion only weights the difference of the derivatives of model and actual trajectory. An immediate comparison of the trajectories is possible by formulating a model-following-problem, however, excluding the states of the model from feeding in, as only the feedback loop is to be designed: (Note that in case a disturbance has occurred, the model does not get any notice of it.) Consequently we have to deal with a problem of partial state vector feedback. Regard that the optimization in this case should be referred to the most interesting initial state (see Section 3), or a combination of particulary probable states. In the present design specification (γ should decay as $e^{-1,5t}$) the model is immediately given as

$$\dot{\gamma}_M = - 1,5\ \gamma_M \tag{45}$$

For the augmented system of 5th order (airplane plus model) the interesting initial state due to a deviation γ_o would be:

$$\hat{\underline{x}}_o^T = [0 \quad \gamma_o \quad 0 \quad 0 \quad \gamma_o]$$

The square of the difference ($\gamma - \gamma_M$) occurs in the performance criterion and thus causes an initial γ deviation to decay in approximately the same way as the first order "disturbance model".

A weighting matrix $\underline{R}$ = diag[0.6, 10, 0.3] proved to be appropriate for generating a feedback matrix with the desired properties, the control activities being even smaller than in a). In the following, therefore, this feedback matrix is assumed to be fixed due to a desired disturbance behaviour.

As one would expect, actually the closed-loop system contains a real pole close to that of the model. It should be emphasized that the control loop thus designed does not show up any overshoot when the centre of gravity is shifted to $\overline{x}_G$ = 0,6 (see Fig. 8b).

6.3 Generating a Desired Tracking Behaviour

The feedback matrix $\underline{K}_2$ being fixed, one may compute the feedforward matrix $\underline{K}_1$ in the sense of an optimal tracking of the second order model introduced in Section 5.2. However, now only the states of the "tracking model" are admitted for "feedback", yielding a new partial feedback problem.

Whenever an input signal appears changing the set point in γ and Θ by an amount of γ_o, this corresponds to an initial state

$$\hat{\underline{x}}_o^T = [0 \quad \gamma_o \quad 0 \quad \gamma_o \quad \gamma_o \quad 0]$$

This is an excellent justification for performing the partial feedback optimization referred to this initial state. For computing $\underline{K}_1$ one has to be aware that some part of the total feedback matrix, that is $\underline{K}_2$ in the feedback loop, is fixed and therefore has to be taken into account in the performance criterion. In Eqs. (10) to (12) this is done by adding to the instantaneous gain matrix $\underline{K}^* \, \underline{D}$ ($\underline{K}^*$ represents the gain matrix $\underline{K}_1$ of the model state, $\underline{D}$ the corresponding output matrix) a fixed term $\underline{K}_2 \, \underline{D}_2$, where $\underline{D}_2$ is an output matrix for the states of the plant.

Numerical computations showed that a weighting matrix $\underline{R} = \text{diag}[2, 10, 1]$ leads to satisfactory results (see Fig. 8a). It is very interesting to note that only unessential deteriorations in tracking occur compared to the former case where the feedback gain $\underline{K}_2$, too, was optimized with respect to the tracking behaviour. The increase in overshoot from 0% to 4% when shifting the centre of gravity to $\bar{x}_G = 0{,}6$ is almost negligible.

For the three controllers C_0, C_1, C_2 Fig. 9a and 9b indiciate the pitch angle deviations occurring in the tracking and the disturbance case, Fig. 10a and 10b the elevator actions.

6.4 Simultaneous Design of Tracking- and Disturbance Behaviour

Not in each case the desired tracking behaviour might be achieved so satisfactorily by computing a feedforward matrix, the feedback loop being fixed, as it was the case with this γ-hold design problem. The way out of this possibly appearing difficulty is to compute the whole gain matrix $\underline{K} = [\underline{K}_2 \;\; \underline{K}_1]$ in one computer run, where the sum of costs is minimized arising for the different initial states in the disturbance and tracking case. Both models are taken up into the augmented system matrix $\hat{\underline{A}}$, by the output matrix $\underline{D}$ it is defined that the state of the model to be followed in a disturbance case is not available, as it is not realized.

The result of the optimization then yields a compromise between tracking- and disturbance behaviour, with the possibility of assigning more or less weight to one of these design aims by properly choosing the relative weights. Furthermore it would be possible to imply more than only one tracking task, by implementing additional parallel feedforward parts containing appropriate models. In the present design case an additional investigation showed that an almost completely decoupled direct lift control could be achieved by introducing a second "tracking channel" whose gain matrix was simultaneously optimized together with the other two gain matrices responsible for γ-hold and disturbances respectively. Of course achievement of these other two design aims was deterioriated by implying a third aim, but this deterioriation was almost negligible.

LITERATURE

[1] ANDERSON, B.D.O., MOORE, J.B. — Linear Optimal Control. Prentice Hall, 1971.

[2] HIRZINGER, G., KREISSELMEIER, G., PIETRASS, A. — Entwurf parameterunempfindlicher Systeme für ein instabiles Flugzeug. will appear as DLR-Forschungsbericht 1974.

[3] LEVINE, W.S., ATHANS, M. — On the Determination of the Optimal Constant Output Feedback Gains for Linear Multivariable Systems. IEE Transactions on Automatic Control, Vol. AC-15, February 1970.

[4] KREINDLER, E. — On the Linear Optimal Servo Problem. Int. Journal of Control, Vol. 9, No. 4, 1969.

[5] TYLER, J.S., TUTEUR, F.B. — The Use of a Quadratic Performance Index to Design Multivariable Control Systems. IEEE Transactions on Automatic Control, Vol. AC-9, No. 4, October 1964.

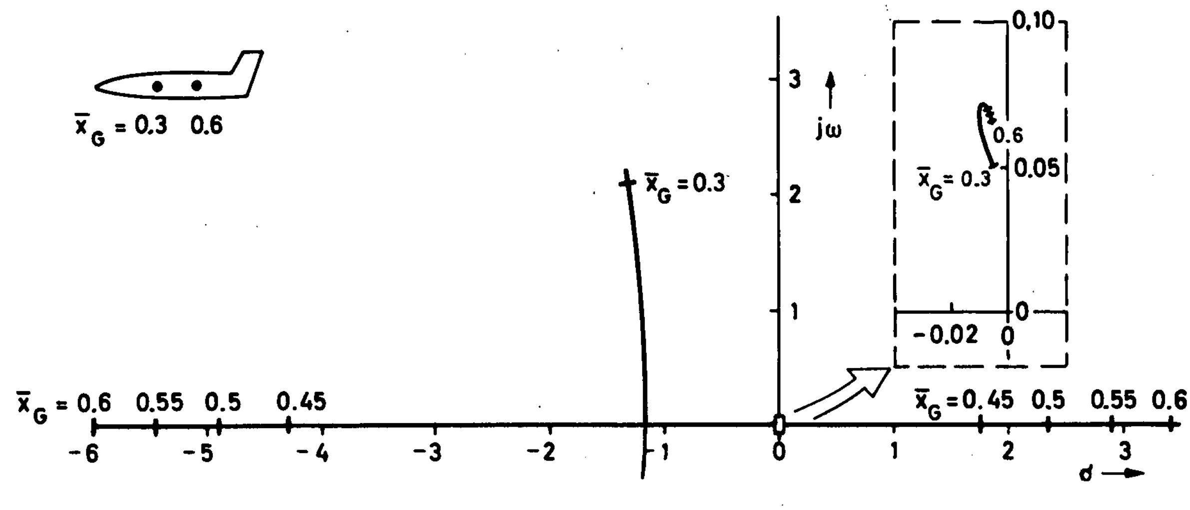

Fig. 1 Eigenvalue Configuration of the Longitudinal Motion Depending on $\bar{x}_G$

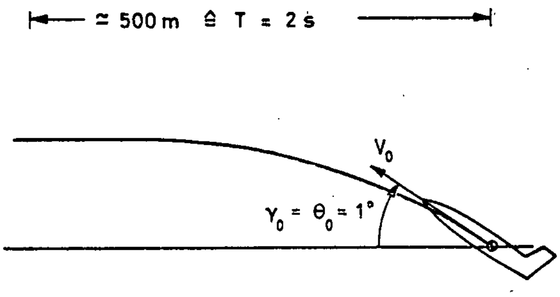

Fig. 2 Tracking Case

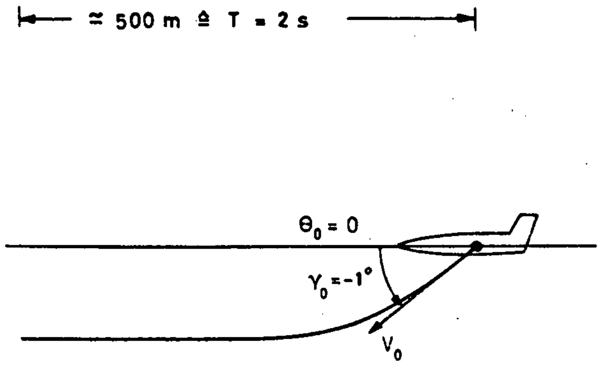

Fig. 3 Disturbance Case

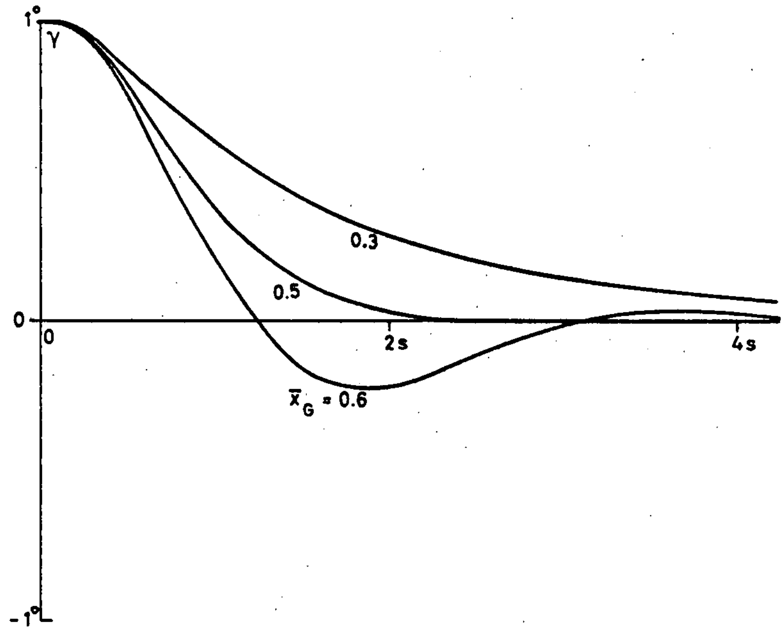

Fig. 4a γ-Tracking Response for Single-Input Feedback Controller C_o

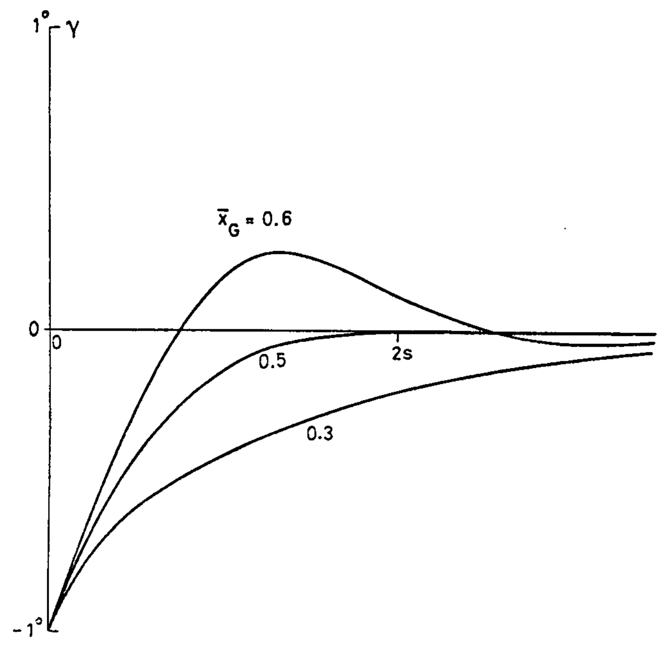

Fig. 4b γ-Disturbance Response for Single-Input Feedback Controller C_o

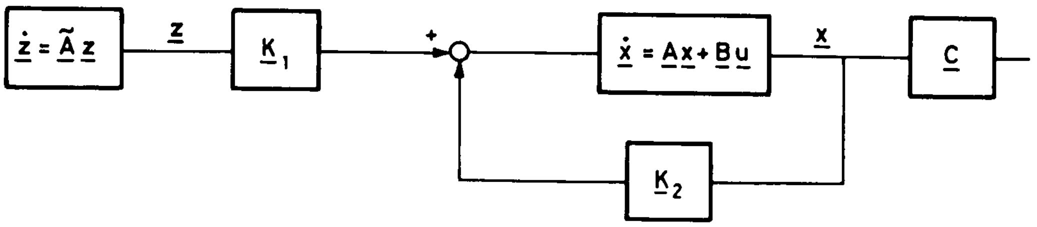

Fig. 5 Principal Structure of a Model-Following Controller

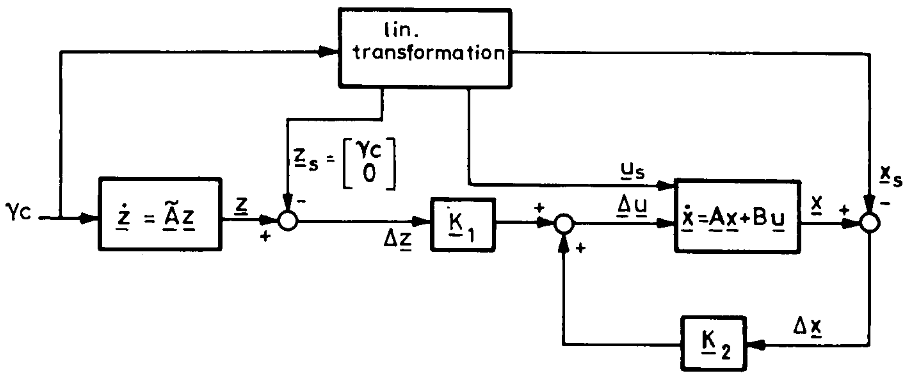

Fig. 6 Structure of Controllers C_1 and C_2

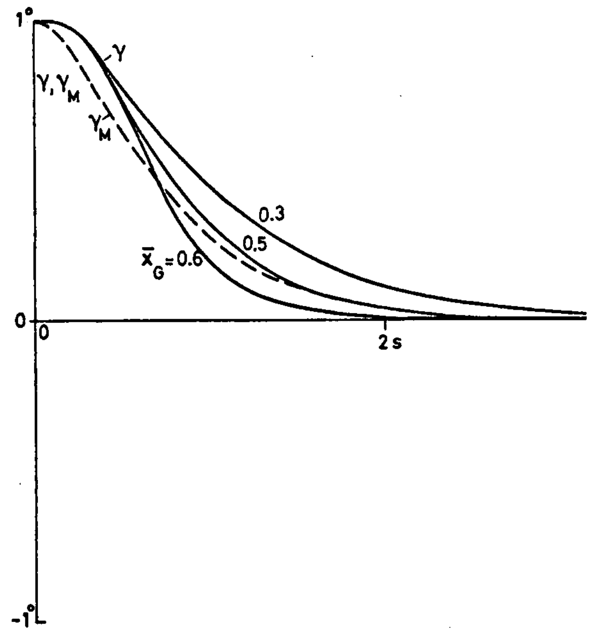

Fig. 7a γ-Tracking Response for Controller C_1

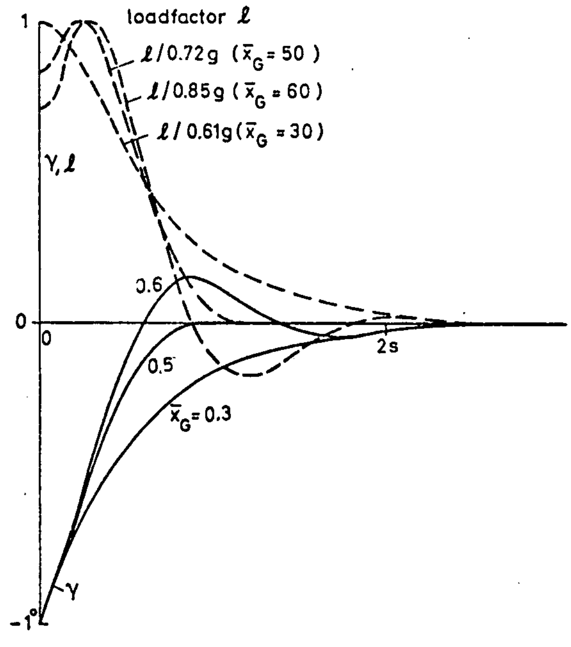

Fig. 7b γ-Disturbance Response and Load-Factor for Controller C_1

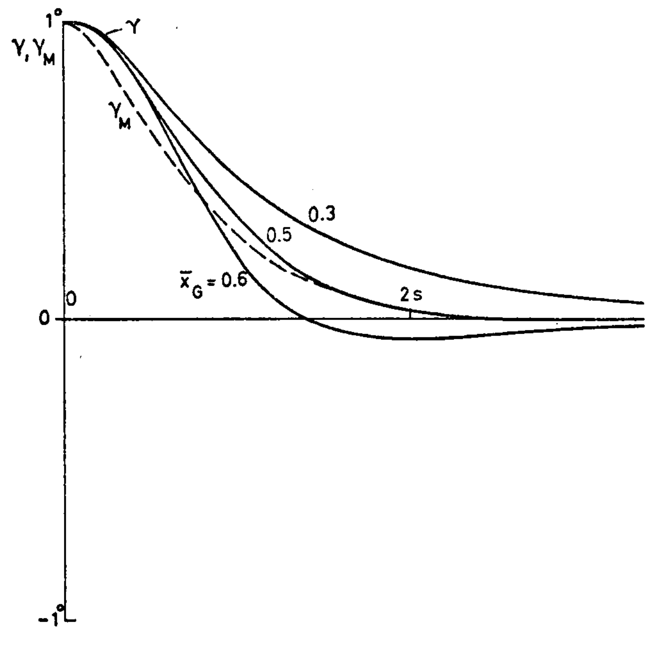

Fig. 8a γ-Tracking Response for Controller C_2

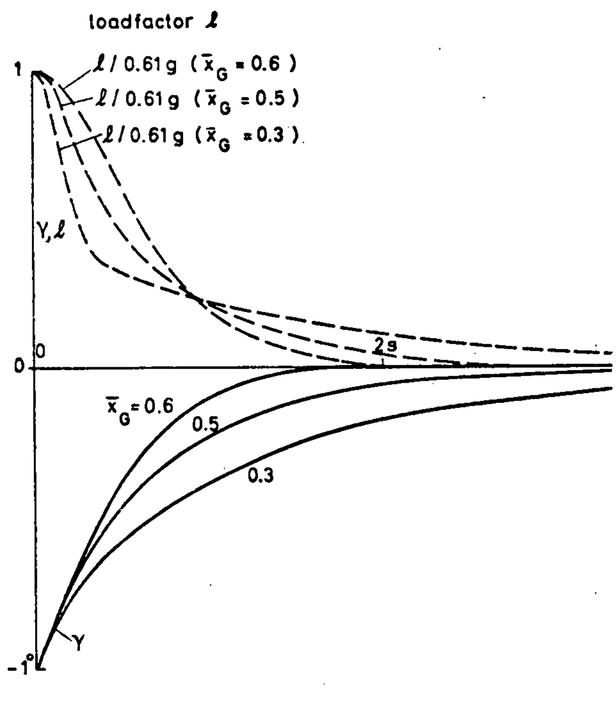

Fig. 8b γ-Disturbance Response for Load-Factor for Controller C_2

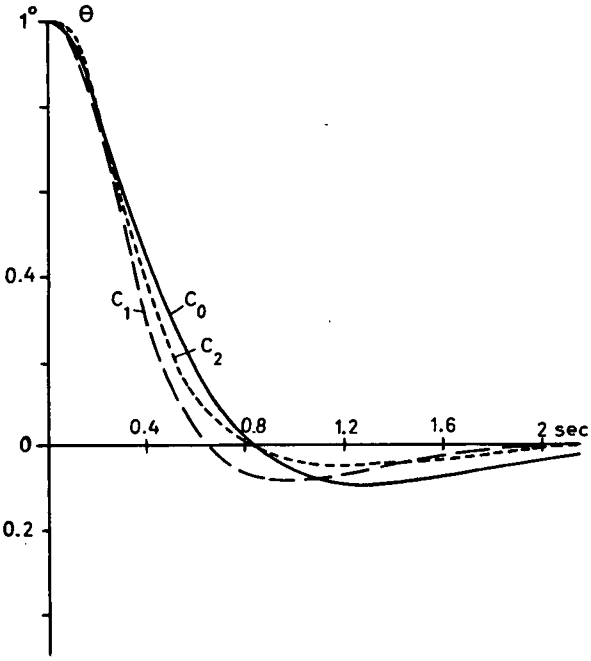

Fig. 9a θ-Response for Controllers C_o, C_1, C_2 in the Tracking Case ($\bar{x}_G$ = 0,5)

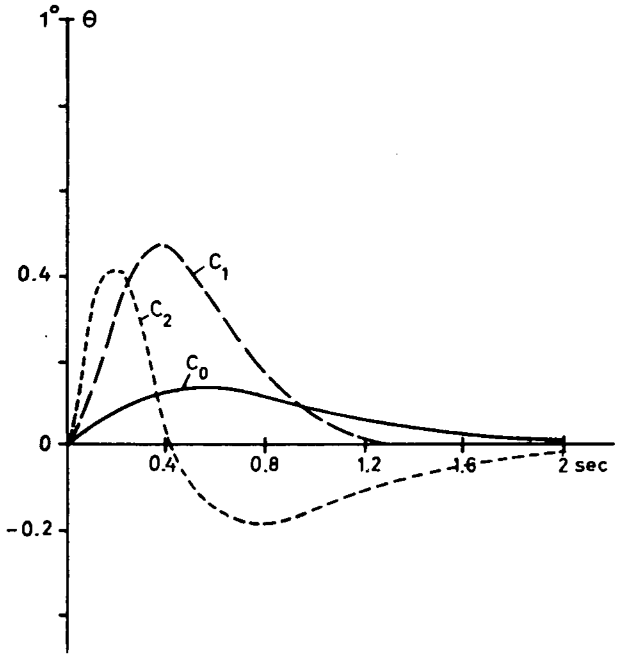

Fig. 9b θ-Response for Controllers C_o, C_1, C_2 in the Disturbance Case ($\bar{x}_G$ = 0,5)

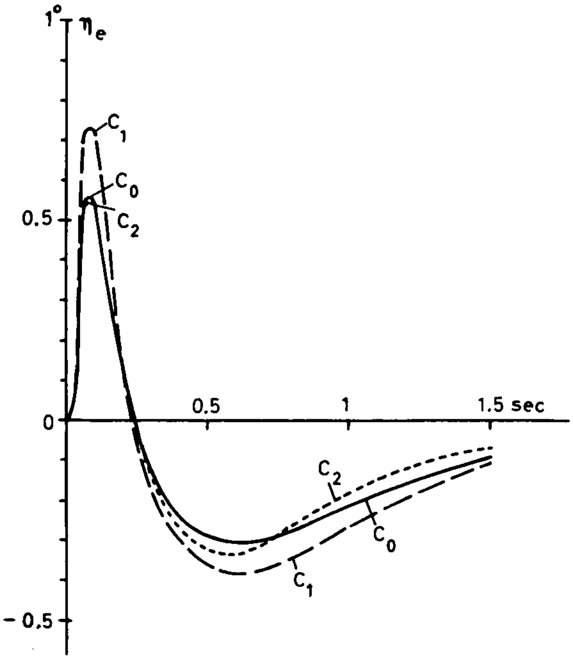

Fig. 10a Elevator Action for Controllers C_o, C_1, C_2 in the Tracking Case ($\bar{x}_G$ = 0,5)

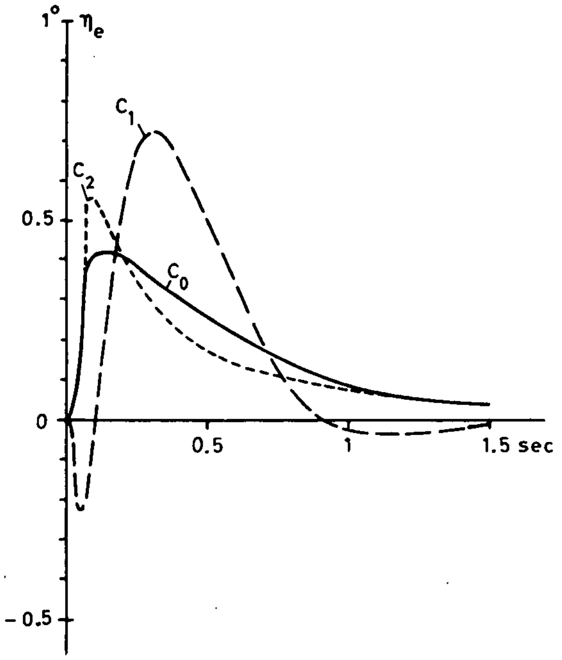

Fig. 10b Elevator Action for Controllers C_o, C_1, C_2 in the Disturbance Case ($\bar{x}_G$ = 0,5)

SURVIVABLE FLIGHT CONTROL SYSTEM

Active Control Development, Flight Test, and Application

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Summary

The major portion of the Survivable Flight Control System (SFCS) Program initiated by the United States Air Force in July 1969 was performed primarily by McDonnell Aircraft Company (MCAIR) over a four year period as a flight control advanced development program. The major objective of this program was to establish the practicality of active control concepts for use in future military aircraft. The SFCS quadruplex (four channel redundancy) primary flight control system is described. Incorporation of this type of control system in a tactical vehicle is expected to provide benefits in enhanced survivability, reliability, maintainability, cost of ownership, aircraft design freedom, and aircraft maneuvering performance. The simulations and ground-based system compatibility testing, performed to verify equipment performance and establish high level of pilot confidence prior to flight, are discussed. A summary of the flight test results obtained during 84 successful flights performed by MCAIR, USAF, USMC, and NASA test pilots is presented. Flight test results indicate that the F-4 with the SFCS installed exhibits greatly improved handling qualities over those characteristic of the production F-4.

The same F-4 SFCS aircraft, newly configured with close-coupled fully operable horizontal canards and fixed leading edge slats, was test-flown during the summer of 1974 over most of the aircraft's flight regime under the MCAIR sponsored Precision Aircraft Control Technology (PACT) Program. This aircraft incorporating control configured vehicle and maneuver load control conceptual features was successfully test-flown and evaluated by MCAIR and service test pilots in a flight test program consisting of 30 flights. The configuration evaluated resulted in unaugmented aircraft static margins ranging up to a minus $7.5\%\bar{c}$. Artificial longitudinal static stability was provided by the FBW primary flight control system with active feedback control. Results obtained from the pilot-in-the-loop simulations and actual flight tests are discussed. PACT flight test results verify that significant performance improvements in combat maneuvering envelope, buffet levels, and specific excess power are achievable in the F-4 with judicious application of control configured vehicle concepts.

Introduction

The United States Air Force initiated the Survivable Flight Control System (SFCS) Advanced Development Program in July 1969. One of the major objectives was to establish the practicality of the active control Fly-By-Wire (FBW) concept for use in military fighter aircraft. The active control development portion of this program was conducted by McDonnell Aircraft Company (MCAIR) as the prime contractor, with Sperry Rand, General Electric, and Lear Siegler selected as the principal equipment suppliers. This program's primary goal was to provide for the design, fabrication, qualification and successful flight test evaluation of a quad-redundant FBW primary flight control system in an F-4 aircraft. This portion of the SFCS program was completed in mid 1973 following a successful 84 flight test program performed by MCAIR, USAF, USMC, and NASA test pilots to evaluate the improved system and aircraft performance. The test aircraft to evaluate the SFCS is shown in Figure 1.

Major equipment elements of this full authority (motion command) FBW system include:

- o Quadruplex analog computer and voter units.
- o Adaptive gain and stall warning computer (duplex computations).
- o Master control and display panel.
- o Built-in-test computer (self test).
- o Centerstick and sidestick controllers.
- o Quadruplex secondary servo actuators.

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FIGURE 1
FBW AIRCRAFT IN FLIGHT

Results from this intensive FBW advanced development effort indicate significant improvements in overall flight control system performance and potential benefits in improved reliability, aircraft design freedom, safety, survivability, maintainability, and cost of ownership. Additionally, the strong and credible FBW technology base developed as a result of this program has paved the way for further aircraft design improvements through exploitation and application of advanced concepts such as Control Configured Vehicles and Multi-Mode Controls with potential system implementation advantages using digital design techniques.

One of the major follow-on development efforts using the F-4 SFCS was the MCAIR sponsored Precision Aircraft Control Technology (PACT) Program. The purpose of this effort was to flight test demonstrate selected Control Configured Vehicle concepts, such as relaxed static stability and maneuver load control provided by close-coupled horizontal canards and leading edge slats. Results and conclusions based on design studies, simulation and flight tests are presented.

SFCS Flight Control System Description

The SFCS is a three-axis fly-by-wire primary flight control system installed in a YF-4E (USAF S/N 62-12200) as shown in Figure 2. Each axis is configured with four channel redundancy, providing a two-fail/operate capability. A comprehensive built-in-test (BIT) system is utilized to ensure operational readiness of the flight control electronic and hydraulic equipment prior to flight. In addition, an in-flight monitor detects and disengages channels containing failed control system components during flight.

"Fly-by-wire" generally refers to the use of electrical signal paths, rather than mechanical linkages, to control the deflections of the aerodynamic control surfaces. In the conceptual design of the SFCS, aircraft motion, rather than control surface deflection, is the parameter to be controlled by pilot inputs. This is achieved by feeding back the outputs of suitable aircraft motion sensors. These are compared with pilot command signals in analog computers which act to provide electrical error signals. The latter, in turn, command the primary control surface deflections to produce the desired aircraft motion.

Pilot commands are generated using a center stick controller in the forward cockpit (Figure 3), side-stick controllers in front and rear cockpits, rudder pedals in both cockpits, and trim panels in both

FIGURE 2
SFCS EQUIPMENT LOCATION

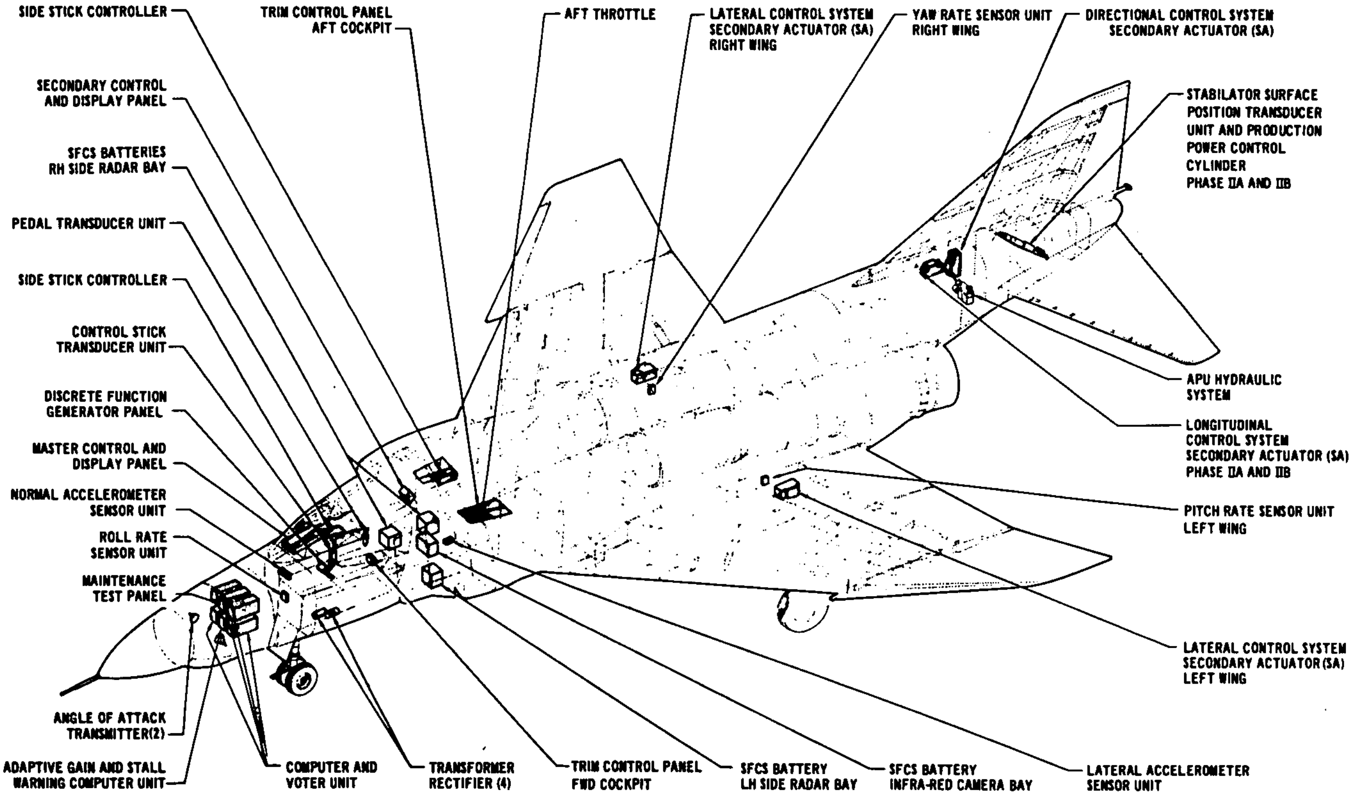
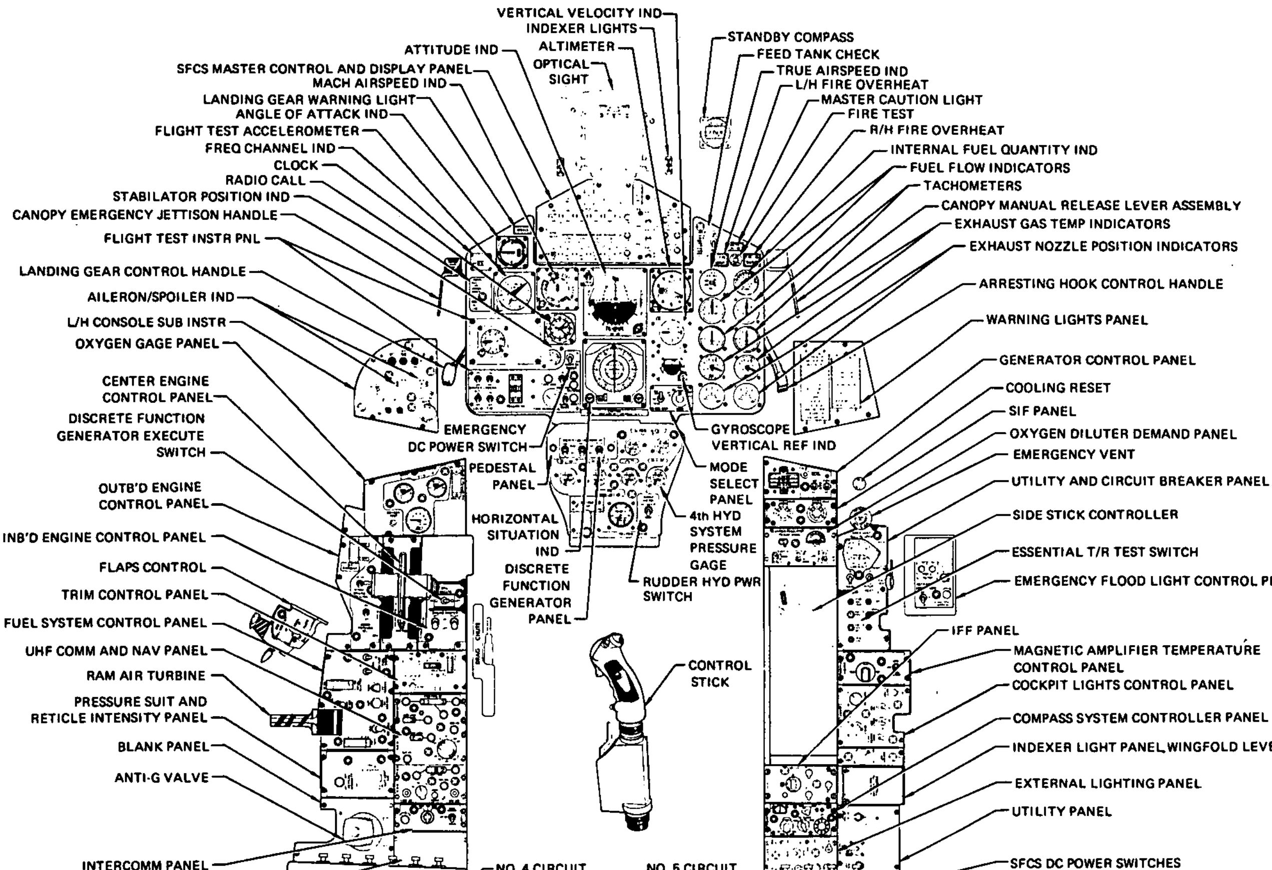


FIGURE 3
FORWARD COCKPIT SFCS



cockpits. The electrical centerstick force commands for pitch and roll are developed using quadruplex strain gages mounted just below the stick grip. Sidestick commands are provided by position transducers mounted across the feel springs within the sidestick controller unit. Both types of stick controllers are mass balanced to minimize crosstalk from one while maneuvering with the other. Aircraft trim is provided from trim wheels mounted on the left console of each cockpit. At the start of the SFCS program, longitudinal and directional mechanical back-up systems were provided to the pilot for "emergency" reversion from the fly-by-wire configurations. A Mechanical Isolation Mechanism (MIM) allowed pilot selection of either 100 percent FBW control or 100 percent mechanical control. Since the "coolie hat" on the centerstick was used for mechanical pitch trim in the mechanical emergency reversion mode, it was felt by the system designers that proportional trim wheels on the console would suffice for FBW operation. The mechanical backup system was removed after the 27th flight.

Each control axis employs four independent channels of electronics and four channel secondary actuators (the lateral axis had eight secondary actuator channels, four left and four right). Channel comparison and voting are provided at the output of the electronics and at the secondary actuators. The voting logic for the electronics selects the lower median value of the four channels. This value is then passed to the secondary actuators where individual channel outputs are again compared. These comparators cause any out of tolerance channel to be shut off. The outputs of individual secondary actuator channels are force-summed on a single output shaft to drive the slide control valve of the corresponding surface actuator.

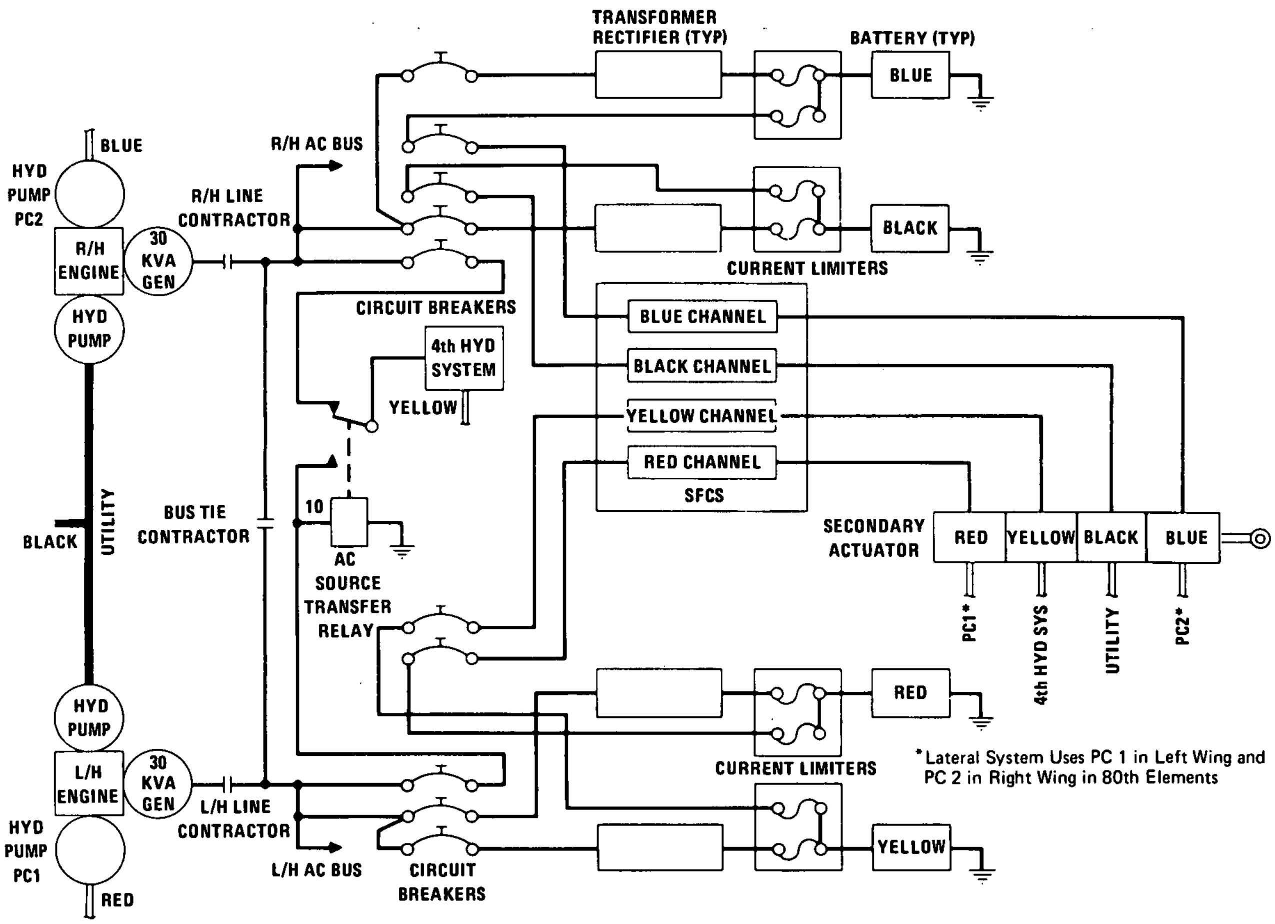
The quadruple redundancy and voting provide for essentially no performance degradation after first and second similar failures within the electronics or secondary actuator of the channels of each control axis. The faulted channels are voted out by means of the Inflight Monitor (IFM) and the system continues to operate with the remaining channels. The IFM detects, isolates, and shuts down failed electronics and secondary actuator channels. Appropriate display of the failures is presented to the pilot on the Master Control and Display Panel (MCDP) in the forward cockpit and on the Secondary Control and Display Panel (SCDP) in the aft cockpit. Upon a third similar failure, the system has no means of detecting which of the two remaining channels is operating properly. If the longitudinal channel experiences three similar failures, the stabilator will be locked in its last commanded position. In the lateral or directional axis, the corresponding surface will be returned to zero deflection and locked. For this condition caused by electrical failures, an electrical back-up mode path is provided for pilot selection. In the event the condition is a result of three similar failures in the hydraulic servo actuators, the pilot can select the Demand-On backup mode.

One of these back-up modes of control, termed the Electrical Back-up (EBU) mode, is provided in all three control axes. When EBU is engaged, the normal computations using motion feedbacks are bypassed, and each control system becomes a fixed gain direct electrical control system commanding surface position. The EBU is quadruplex using the same stick force command transducers, amplifiers, servos and actuators as in the Normal mode of operation. EBU is selected in the lateral and directional axes using a paddle switch on the centerstick, a trigger on the sidestick, or mode switches on the MCDP. Longitudinal EBU is selected by using the switch on the MCDP. In the EBU mode, the voters located ahead of the servo amplifiers are used, but the comparators used in the Normal mode are disabled, with all channels being turned on regardless of status. An equivalent mode of operation for the quadruplex hydraulic actuators, called the Demand-On mode, enables all elements which have hydraulic power, regardless of previous status. These two modes permit operation in the presence of failures in a voted section of the quadruplex system, thus providing a get-home-and-land capability.

The simplified SFCS power system schematic presented in Figure 4 shows the relationship between electrical and hydraulic power sources, electronic set channels and secondary actuator channels. The electrical and hydraulic power sources used in each channel were selected to minimize the effect of power source failures.

The primary sources of electrical power in the test airplane are two engine-driven alternating current (AC) generators. These generators power the left and right hand AC buses in a split-bus configuration. In the event of power failure of one of the sources, the bus-tie contactor automatically connects the buses together so they are powered from the remaining good source. To obtain quadruplex power sources for the SFCS equipment, two transformer rectifiers are connected to each of the two electrical buses. Each of the transformer rectifiers is connected in parallel with an aircraft battery and connected to one and only one SFCS channel. The batteries have sufficient capacity to power the SFCS for more than one hour. Use of the batteries assures continued electrical operation of all four SFCS channels in the event of total AC power failure.

FIGURE 4
SIMPLIFIED SFCS POWER SYSTEM SCHEMATIC



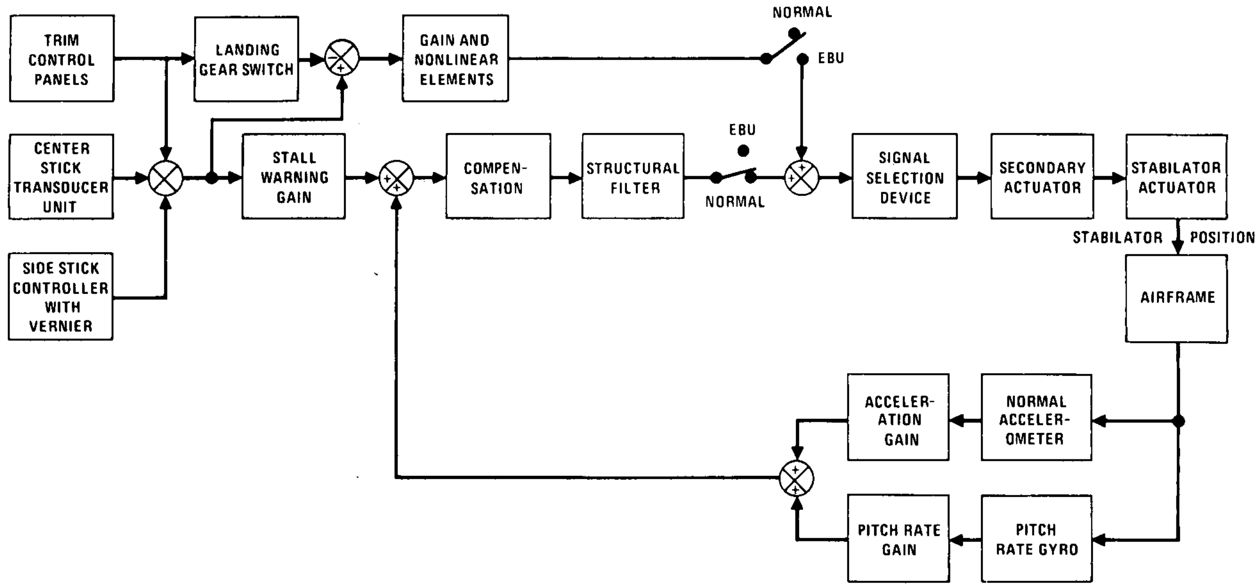
Three hydraulic power sources are normally available in the F-4 airplane. These hydraulic sources and their assigned colar code are: Power Control Hydraulic System No. 1 (PC-1) - Red; Power Control Hydraulic System No. 2 (PC-2) - Blue; and Utility Hydraulic System - Black. A fourth hydraulic system is required to maintain quadruplex redundancy for the SFCS in the test airplane. An auxiliary power unit containing an electric motor-driven hydraulic pump is utilized to provide the fourth hydraulic system, and is color coded yellow. Excitation for the auxiliary power unit is normally supplied from the left hand AC bus with automatic switchover to the right hand AC bus in the event of left hand bus electrical power failure.

Secondary actuators are used in all axes of control to convert electrical command signals to mechanical position commands for application to the surface actuators. Separate quadruplex electrohydraulic secondary actuators were installed, in preference to integrated secondary and surface actuator units, to permit utilization of existing F-4 surface actuators.

The four elements of the secondary actuators are physically isolated from each other, powered from separate hydraulic sources, and commanded through separate electrical channels of the Survivable Flight Control Electronic Set (SFCES). The outputs of the four elements of each secondary actuator are physically connected to provide the single mechanical input required for the surface actuator. The linkage between each secondary actuator and its associated surface actuator is designed with sufficient strength and integrity to maintain overall SFCS reliability.

The primary command loop in the longitudinal axis (Figure 5) employs a blend of normal acceleration and pitch rate for the feedback signal in the Normal mode. Pilot command inputs are prefiltered and then compared with summed normal accelerometer and pitch rate gyro outputs and the error signal produced is used to command stabilator position. The normal mode of operation incorporates Neutral Speed Stability (NSS) for all nonterminal flight phases. This is accomplished by the integrating action of the secondary actuator loop to provide a zero steady-state error control system. This essentially eliminates the need for the pilot to retrim the aircraft longitudinally as the flight condition changes. The forward loop integration is switched out when the gear is down to provide Positive Speed Stability (PSS) to the pilot when the aircraft is in the takeoff and landing mode.

FIGURE 5
SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM SINGLE CHANNEL OF LONGITUDINAL AXIS



The forward loop gain in the longitudinal control system, as well as the gain in the directional control system, can be either pilot selectable or automatically scheduled by an adaptive gain changer. The adaptive gain changer uses an interrogation signal to the stabilator to excite the airframe. The output of one of the four pitch rate gyros is then used to compute the stabilator effectiveness parameter M_δ. System gain levels are automatically changed as a function of the M_δ computation. Two gain levels are available in the longitudinal control system and four gain levels are available in the directional control system and the roll-to-yaw crossfeed path. No adaptive gain changing is used in the roll axis which is designed to function with fixed gains for all flight conditions. A structural filter is utilized in the longitudinal control system to reduce the loop gain at aircraft structural resonance frequencies and help eliminate the effects of structural bending.

The lateral axis (Figure 6) in the Normal mode employs a fixed high gain roll rate feedback loop to achieve a nearly invariant roll mode time constant and roll rate per stick force gradient throughout the flight envelope. Lateral stick commands are fed through a shaping prefilter and a triple slope gain gradient which varies with the magnitude of pilot roll rate command. This signal is then compared with the output from the roll rate gyro to generate an error signal to command aileron/spoiler deflections and the desired aircraft motion. The prefilter output also feeds the yaw axis through a roll to yaw crossfeed network to provide desired turn coordination. A structural filter is included to attenuate loop gain at structural resonance frequencies. The lateral stick command is also fed in a parallel path to command aileron/spoiler actuators directly. This path, which is engaged at all times, is the only path engaged in the roll axis when the lateral control system is in the EBU mode of operation. With gear down, the roll command gain is decreased somewhat to afford lower roll sensitivity during takeoff and landing.

The directional axis (Figure 7) also employs a direct electrical link, commanding rudder position with rudder pedal force. Parallel with this, the rudder pedal force command is shaped by a prefilter, then compared with a blend of lateral acceleration and cancelled yaw rate feedback. The error signal produces commanded rudder deflection in addition to that commanded by the direct link. The roll-to-yaw crossfeed

FIGURE 6

SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM SINGLE CHANNEL OF LATERAL AXIS

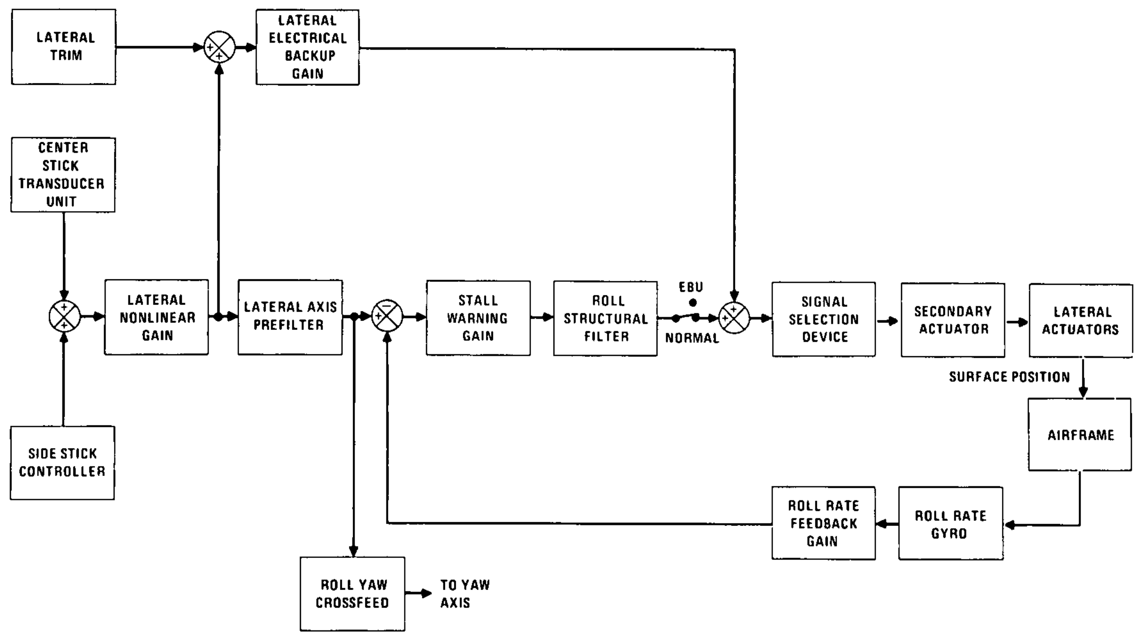
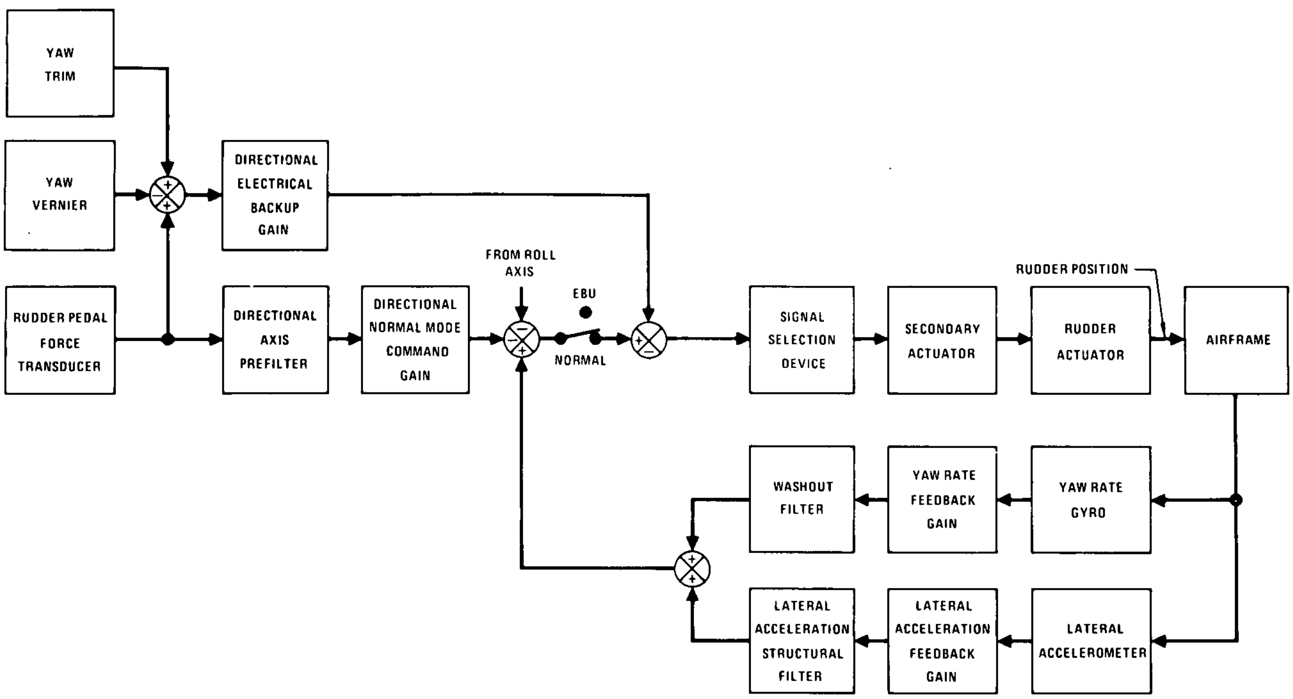


FIGURE 7

SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM SINGLE CHANNEL OF DIRECTIONAL AXIS



signal is shaped and then added to the yaw model command. Yaw rate and roll to yaw crossfeed gains are pilot selectable or automatically scheduled with M_δ by the adaptive gain computer. In the EBU mode, the yaw axis employs the direct rudder deflection command only.

A dual redundant stall warning function is provided through a blend of angle-of-attack and lagged pitch rate. Nose down pitch rates are rejected in the control law so that pilot push recovery from a stall condition is not impeded. The stall warning function is designed to reduce the command gain in the longitudinal axis, effectively increasing stick force per g, and to remove the roll rate feedback from the roll axis; both changes occur linearly as the stall region is penetrated.

The stall warning computer provides this increase in longitudinal stick force gradient and decrease in roll rate feedback as the angle-of-attack increases into the near-stall region. The basic longitudinal command loop attempts to prevent the nose of the aircraft from dropping through after a stall, as is common in the production F-4. The decrease in command gain in the near stall region originally magnified this effect by requiring high stick force for recovery. The design was modified so that the gain decrease is not active for nose down commands, alleviating this situation. This system does not provide sufficient clues to the approach to a 1g stall, and some method of achieving this will be required for future systems. A possible solution would be to include angle-of-attack as an on-line feedback and to compute a maneuvering limit using this additional loop.

A Discrete Function Generator (DFG) is available to provide precise, calibrated test inputs into each axis. Failure insertion switches are available for applying an electrical or hydraulic hardover or soft simulated "failure" into one channel of each control axis. The MCDP is located at the top of the front instrument panel and contains switches for manual and adaptive gain selection, and status indicator lights for mode (Normal or EBU), channel failures (electrical and hydraulic), stall warning and adaptive gain computer failure, BIT GO or NO-GO, and master caution. The extensive BIT capability was incorporated in the SFCS primarily for ground checkout of the flight control system to minimize the probability of takeoff with a failure existing in the system.

Simulation and Ground-Based System Compatibility Testing

A series of simulations was used during the development and test of the SFCS to assist in the design, verify equipment performance, train pilots, and correlate flight test data. The simulation effort was initiated with two three degree-of-freedom simulations to develop the longitudinal and lateral-directional control laws respectively. These control laws were then programmed on a CDC 6600 computer connected to a fixed base simulator for piloted studies of the total system. These studies were closely coordinated with the equipment suppliers to assure math models which could be converted to physically realizable hardware. The SFCS flight simulator is shown in Figure 8.

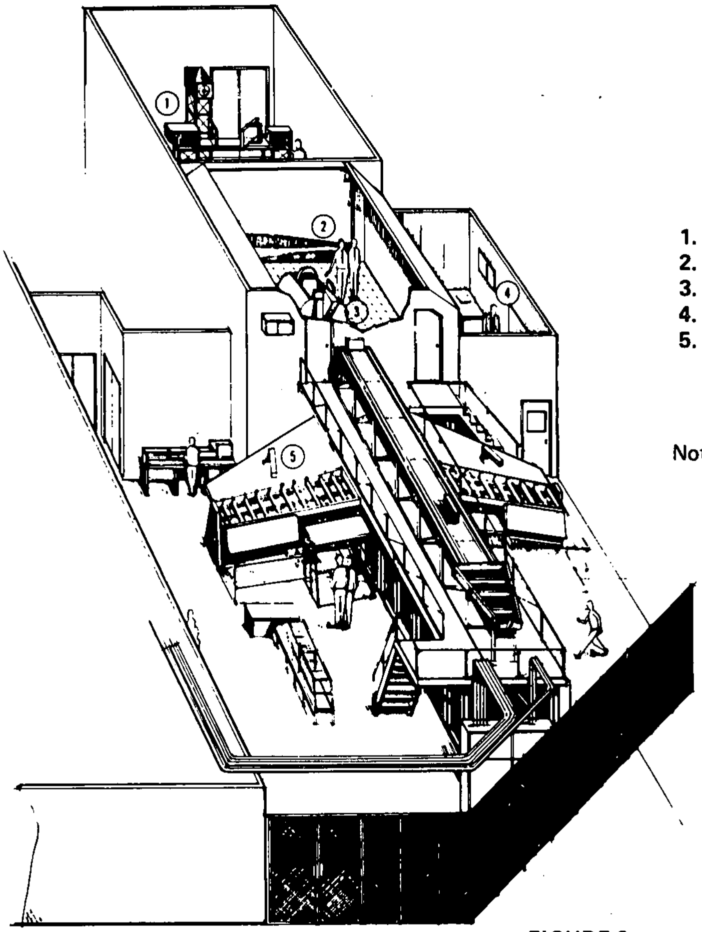
When hardware was available, it was integrated into the Iron Bird so that only the airframe equations of motion were modeled on the computer. This simulation functional block diagram is shown in Figure 9. Static gains, frequency responses, thresholds, and mode switching and failure transients were measured. Pilots then "flew" various missions to verify system performance. During this time, the project test pilots had the opportunity to study system operation with various simulated failures. Such "survivability missions" helped the pilots to thoroughly understand the fault detection and isolation techniques and to understand system operation during such occurrences. This simulation was used to train pilots for flight, and was rated as a very high fidelity simulation by the pilots. Some system anomalies were detected during this effort and were corrected before flight testing was initiated. Pre-flight checkout procedures and flight test techniques were worked out in advance using the simulator. As a result of this extensive simulation and ground test program, a high level of pilot confidence was established prior to the initiation of the flight test program.

SFCS Flight Testing Summary

The initial portion of the flight test program consisted of a progressive checkout and verification of all modes and functions throughout the flight test envelope. During this portion of the flight test program, longitudinal and directional mechanical back-up systems were available for emergency use. When confidence in the Fly-by-Wire system had been established, the airplane was ferried to Edwards Air Force Base for continued development and a more thorough evaluation of system performance. After a total of 27 flights, the mechanical controls were removed from the aircraft, and the last 57 program flights were accomplished with pure Fly-by-Wire control.

FIGURE 8

SFCS FLIGHT SIMULATOR LAYOUT

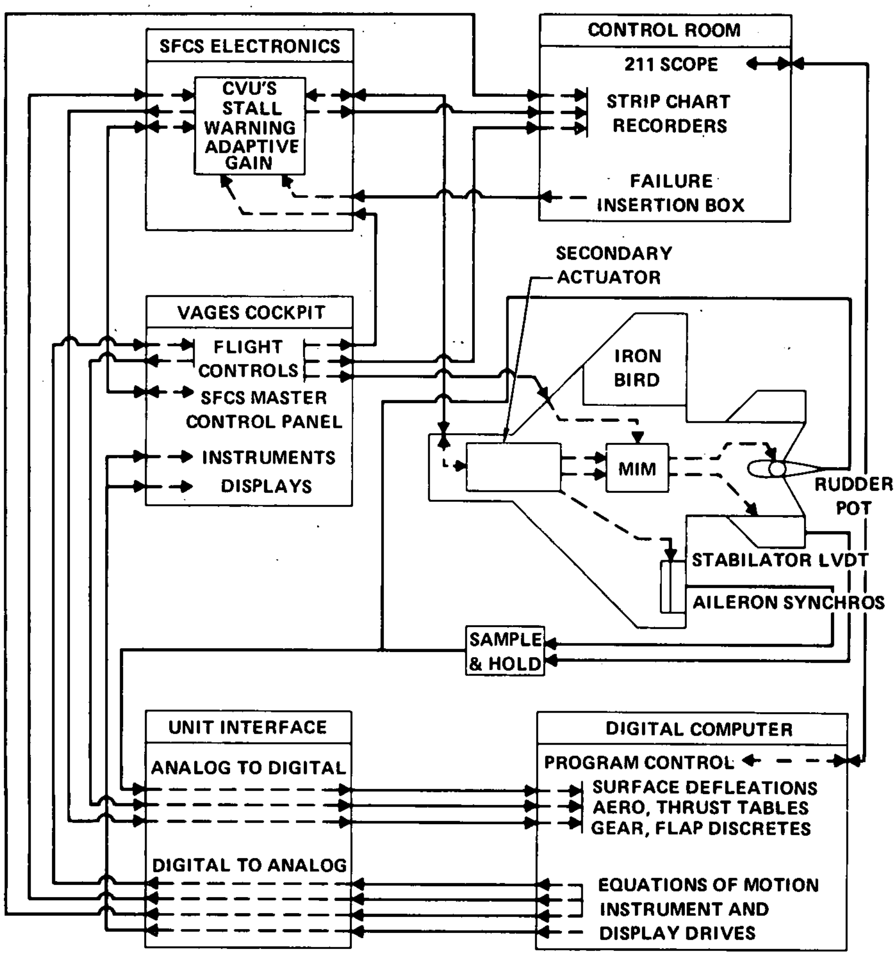


1. TARGET AND HORIZON GENERATOR
2. TRANSLUCENT SCREEN
3. VAGES COCKPIT
4. CONTROL ROOM
5. IRON BIRD AREA

Notes: VAGES - Variable Geometry Simulator

FIGURE 9

SIMULATION FUNCTIONAL BLOCK DIAGRAM



Two problems were uncovered early in the program: a sustained oscillation in the pitch axis due to structural feedback through the pitch rate gyro during flight in the Power Approach (PA) configuration, and an oversensitivity in lateral control. The structural coupling oscillation was eliminated by using a suitable notch filter applied to the pitch rate gyro signals. Attempts to reduce the lateral sensitivity met with varying degrees of success, with one gain reduction resulting in a lateral Pilot Induced Oscillation (PIO) near touchdown. The final solution flown was a three slope gain curve, using successively higher gains as the amount of command roll rate was increased.

The pilots reported that the response and damping of the aircraft had been significantly improved over the basic F-4. The pitch axis showed improved tracking capability, while exhibiting such characteristics as the virtual elimination of a pitch transient during a rapid deceleration through the transonic region (Figure 10). The uniformity of centerstick maneuvering force gradients for the longitudinal axis over the entire flight regime was much improved with the SFCS and is compared to the basic F-4 for several flight conditions in Figure 11. The data substantiates that the SFCS provides a more comfortable stick gradient throughout the flight envelope, allowing more precise control of pitch rate and normal acceleration as shown in the example of Figure 12. Improved pitch control at high g values is also attributed to the overall linearity of the stick force gradient versus g.

When operating in the Normal mode, no trim input is required by the pilot to compensate for the change in stabilator position necessary to trim the aircraft for changes in speed or flight conditions. Pilots felt that this greatly reduced their workload in handling the aircraft. It was found however, that some trimming of the long period phugoid motion was required by the pilot on the ferry flights. Notably, the adaptive gain changer operated satisfactorily throughout the program; however, it was felt that a less complicated device, such as using a highly reliable air data system for providing gain scheduling, would likely be sufficient for controlling most fighter class of aircraft.

FIGURE 10

TRANSONIC DECELERATION IN 3g WIND-UP TURN

30,000 FT

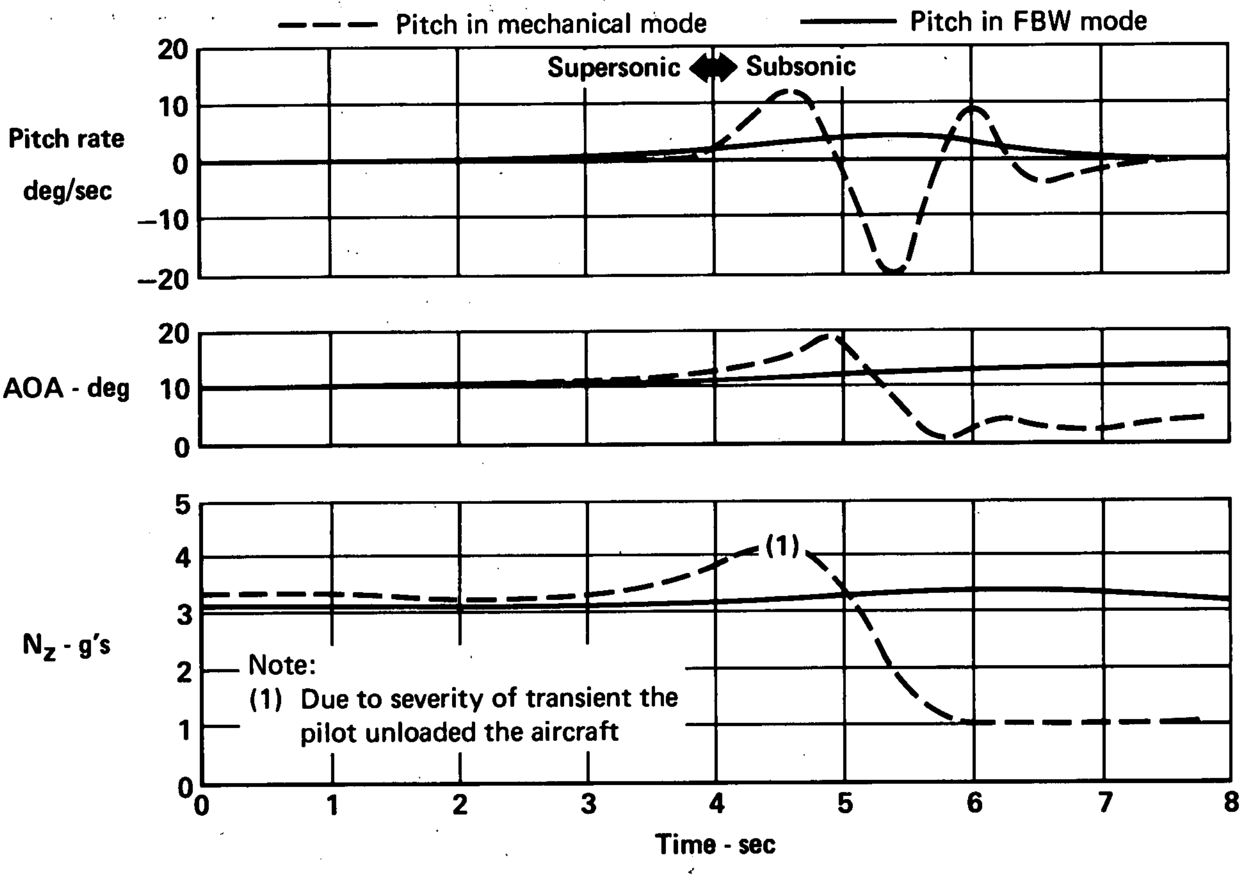


FIGURE 11

STICK FORCE PER G vs MACH NUMBER

ADAPTIVE GAIN - NORMAL MODE

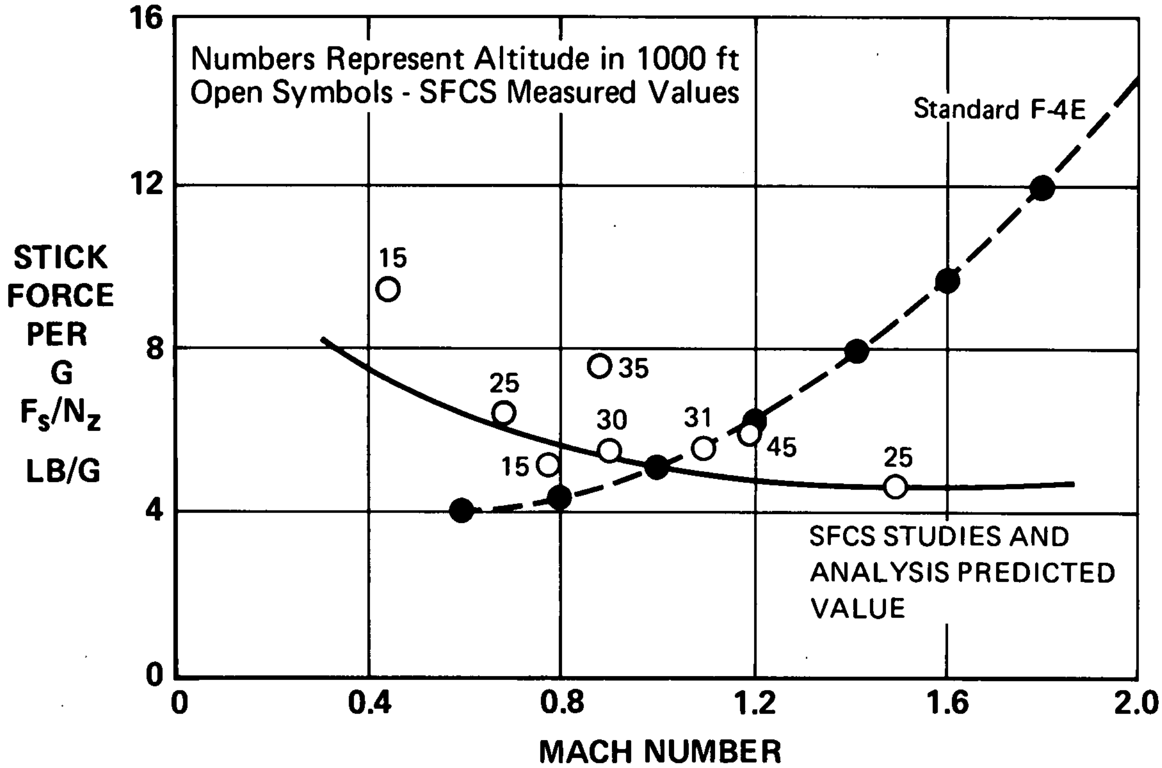


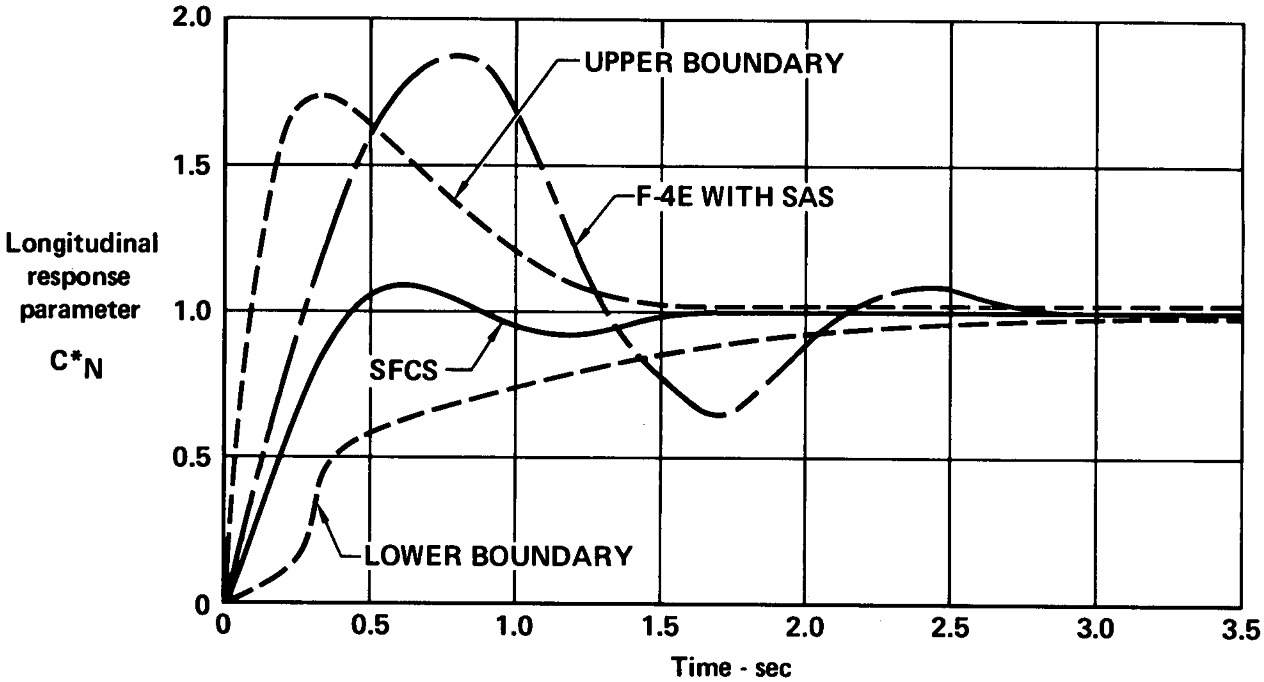
FIGURE 12

SFCS FLIGHT TEST DATA — C* CRITERIA COMPLIANCE

Flight No. 642
Altitude . . . . . 34,900 ft
Weight . . . . . . 38,715 lb

Run No. 7
Airspeed . . . . 310 kts (≈ 0.9M)
Type of input . . . . DFG

Pitch gain . . . . medium
C.G. (%MAC) 31.87



The roll axis was very responsive. Since pilot stick force commands roll rate directly, the aircraft was found to respond with much higher roll accelerations than the pilot would normally command for the same force applied with a mechanical control system. This faster response may be desirable under some combat situations (such as an evasive maneuver), but can also be too sharp in conventional flight. An optimum response for each segment of a mission may require a multi-mode control capability. Future Fly-by-Wire systems will probably incorporate such multi-mode control. Figure 13 and 14 present the roll rate time constant and roll rate to stick force ratios, respectively, of the SFCS and the basic F-4. These data show that the SFCS provided a more nearly uniform response over the flight envelope than is evident with the basic F-4.

One of the more significant determinations during the flight test program was in the area of command transducers. The SFCS included both a centerstick and a sidestick controller in the front cockpit. The pilot evaluation of the sidestick was that such a device could be developed for full-time use; however, the SFCS installations had a few drawbacks. The controller was mounted on the right console, with the choice of position constrained by existing cockpit structure, and was not in an optimum position for all control tasks. The input pivot was below the grip, and coordinated maneuvers were difficult to accomplish at high load factors. It was possible to produce very high roll rate maneuvers by twisting the wrist, an input motion aggravated by the basic system design sensitivity discussed previously.

The centerstick controller used a force transducer mounted just below the grip. The transducer was responsive to torque inputs as well as to linear forces. This could result in such effects as the pilot twisting the grip to the left while moving it to the right, thus generating a left roll while moving the stick in the direction which would command right roll with a mechanical system. On one flight, the pilot attempted a constant roll rate maneuver by restraining the stick against his knee. He inadvertently applied a nosedown pitch torque, and even though he attempted to pull the stick slightly aft to hold the nose up, his body coupled with the negative g motion and he increased the forward torque on the stick grip and commanded excessive negative load factor. It will be mandatory for future systems to require an input sensor which is sensitive only to the desired force commands and not to commands which can result from such inadvertent inputs. Double pivot point installations for the control stick transducers should be avoided.

FIGURE 13

ROLL RATE TIME CONSTANT SFCS vs F-4

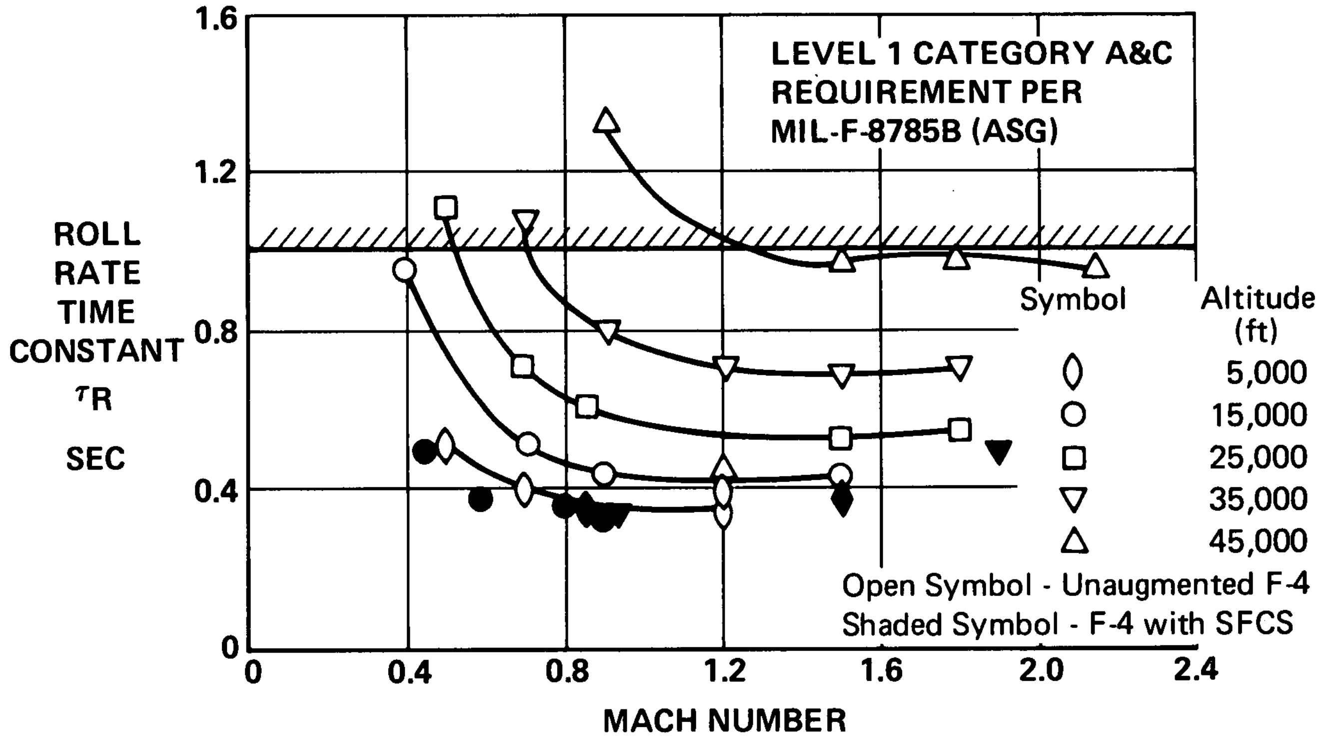
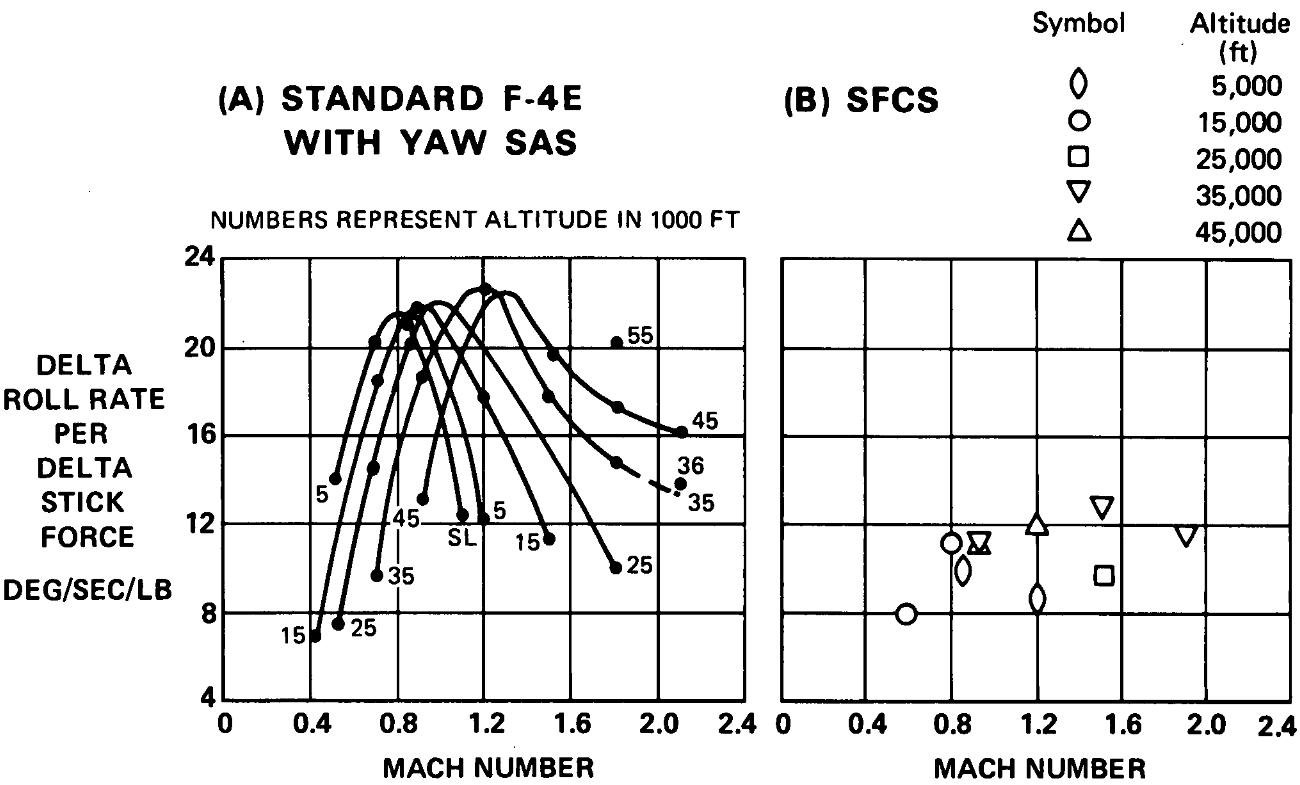


FIGURE 14

ROLL RATE PER STICK FORCE SFCS vs F-4



One point definitely made by the SFCS pilots was that stick position is used as a cue in anticipating impending aircraft response to pilot input command. This was demonstrated when an "out-of-harmony" condition, due to unequal stick position breakout and stick force transducer deadzone, caused aircraft response to occur before stick motion. Adverse pilot comments resulted and several modifications of the pitch control stick feel system were necessary before a satisfactory combination was obtained. Pilot opinion at the conclusion of several flights was that the most desirable condition was one of having no disparity between breakouts.

F-4 Pact Program

In January 1971, MCAIR began a USAF funded research program entitled "Compatibility of Maneuver Load Control and Relaxed Static Stability Applied to Military Aircraft". The purpose of this program was to determine the performance benefits which can be obtained for a fighter aircraft through the application of maneuver load control and relaxed static stability. The aircraft selected for study was the SFCS aircraft. This vehicle was considered a good test bed because extensive aerodynamic data for the basic aircraft and the configuration variations existed from flight tests and/or wind tunnel tests. Even more important, the fly-by-wire flight control system was available for providing artificial aircraft stability and desirable handling qualities. Subsequent studies were performed from March 1972 until November 1972 under the "Control Configured Vehicle Concepts Applied to Fighter Aircraft" program.

Study results obtained during the "Maneuver Load Control" and "Control Configured Vehicle" programs show that considerable performance benefits can be obtained through the use of horizontal canards. With these canards operating symmetrically, performance benefits of the following magnitude were predicted:

a. Energy Maneuverability ------------ plus 200 FPS
b. Control Limited Load Factor ------- 50% improvement
c. Lift Limited Load Factor ---------- 20% improvement

It was also indicated that some amount of side force can be obtained by operating the horizontal canards differentially.

The SFCS Aircraft has recently been fitted with horizontal canards as part of the MCAIR sponsored program "Precision Aircraft Control Technology" (PACT). The PACT SFCS aircraft configuration is shown in flight in Figure 15. The canards, shown in Figure 16, have a total exposed area of 40 square feet. The canards were designed so that the outer panels can be removed, thus permitting the flight testing of partial span canards having an exposed area of 17 square feet. At the time aircraft modification was started, a set of fixed wing leading edge slats became available. In view of the added benefits attainable with slat incorporation, the modification scope was increased to include the fixed slat installation.

The horizontal canards were electronically "geared" to the stabilator so that the minimum drag combination of canard and stabilator deflections will exist for straight and level as well as maneuvering flight conditions. The schedule for positioning the horizontal canards as a function of stabilator position is given in Figure 17. This schedule combines the approximate minimum drag points for subsonic and supersonic load factors from 1 to 5, maximum nose down deflection, and maximum nose up deflection on a simple schedule with no Mach number bias. This was found to be possible because the subsonic (unstable) points and the supersonic (stable) points diverge and never overlap.

As shown in Figure 17, during subsonic flight the stabilator and canard trim positions move in the aircraft nose down direction as load factor increases, a characteristic due to the fact that the unaugmented airframe is aerodynamically unstable. Supersonically, the stabilator and canard trim positions shift in the aircraft nose up direction since the airframe is stable supersonically. At all flight conditions the canard deflections are such that the canards aid the stabilator in generating the required pitching moments.

Each canard is driven by a production F-15 stabilator actuator. Structural clearance was provided for +20° (canard leading edge up) and -30° (canard leading edge down) deflections so that direct lift and direct side force capabilities could be added in the future. However, present electronics limit canard travel to the +10° to -10° range (with respect to the wing chord plane) required to implement the canard/stabilator schedule for obtaining minimum aircraft drag.

FIGURE 15
PACT AIRCRAFT IN FLIGHT

FIGURE 16

F-4 PACT GENERAL ARRANGEMENT

FIXED SLATS

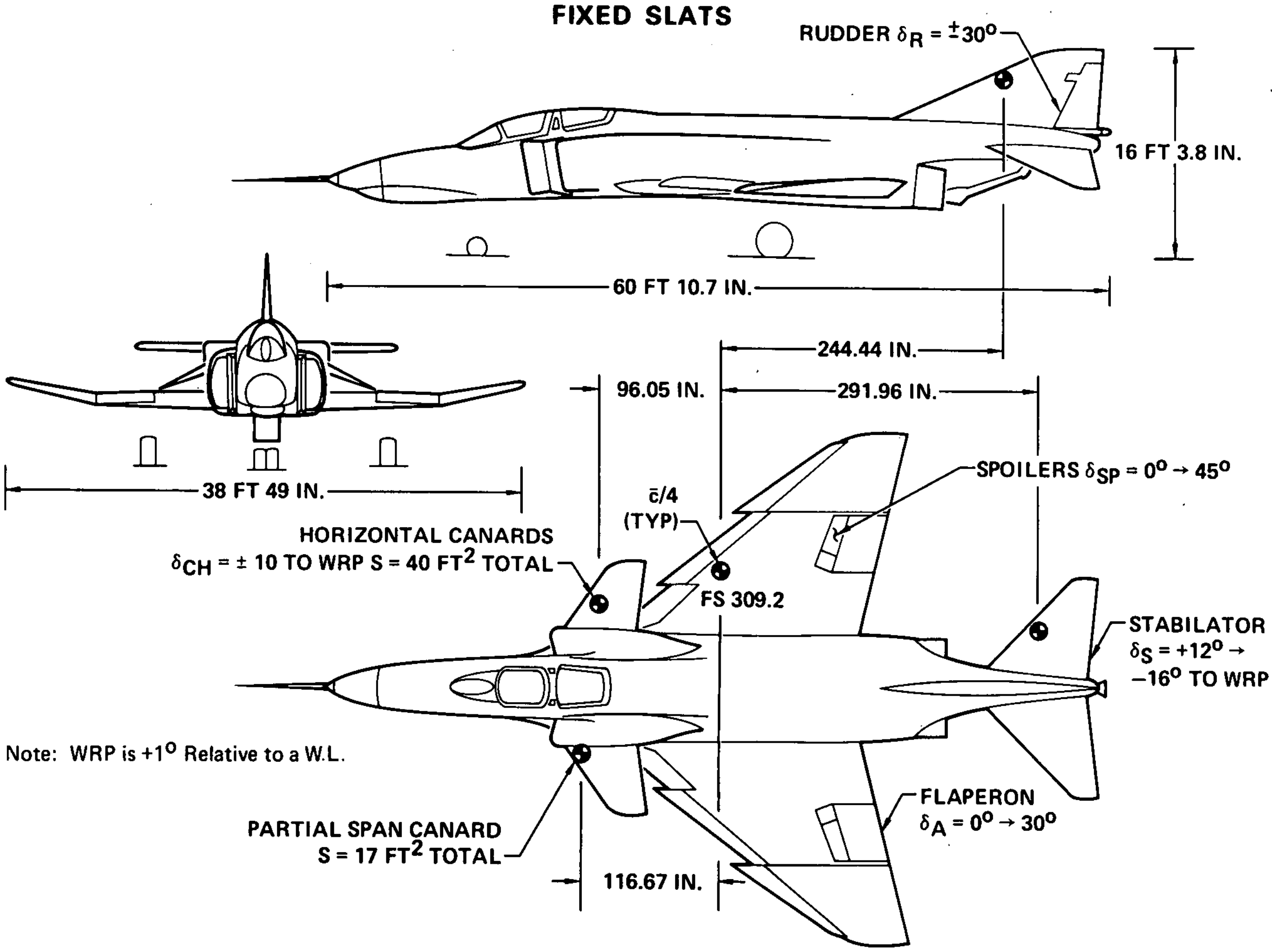
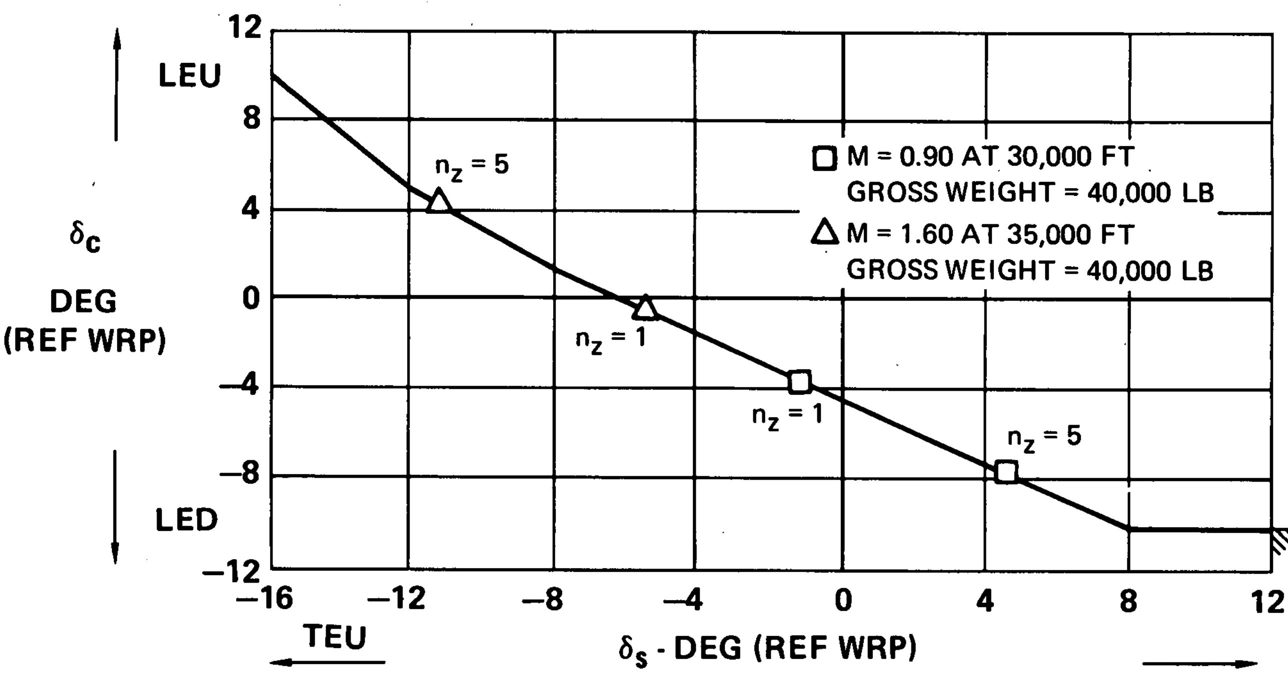


FIGURE 17

CANARD (C) AND STABILATOR (S) DEFLECTION SCHEDULE



Adding the canards to the aircraft moved the neutral point forward on the aircraft and caused the longitudinal axis of the unaugmented airframe to be unstable subsonically. Therefore, a highly reliable, full authority flight control system such as the SFCS is required for controlling the F-4 PACT. A control system design goal in the PACT program was to achieve good handling qualities and satisfactory stability margins with as few modifications in the basic SFCS flight control system design as possible.

Initial control system synthesis studies showed that longitudinal handling characteristics, similar to the SFCS handling characteristics, and adequate stability margins could be obtained by making only three changes to the SFCS longitudinal control system in addition to scheduling the canard as a function of stabilator position. First, the forward loop compensation network was changed slightly. Second, the normal acceleration feedback compensation was altered. Third, the stabilator travel range was rotated 5° in the aircraft nose down direction. A simplified schematic of the resulting flight control system is shown in Figure 18.

During the final stages of the PACT control law synthesis studies, SFCS flight test results indicated that the longitudinal and lateral control systems were sensitive at certain flight regimes. Some pilots objected to the sharpness of the SFCS roll responses at airspeeds above 250-300 KCAS. The longitudinal sensitivity was felt primarily during air-to-air tracking. In air-to-air tracking, the longitudinal and lateral flight control system responsiveness characteristics made it difficult for the pilots to null out elevation and traverse channel tracking errors.

The sharpness of SFCS roll responses was due in part to the presence of the electrical backup (EBU) path which is contained in the lateral channel electronics. This path, which is not switched out when the system is operating in the Normal mode (see Figure 6), permitted an unfiltered command signal to be sent directly to the aileron-spoilers. As shown in Figure 19, adding the filter in the lateral EBU path doesn't change the roll rate (p) response significantly but is very effective in lowering the initial "jerk" ($\ddot{p}$) which the pilot feels when he applies a lateral stick force command.

FIGURE 18

PACT SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM

SINGLE CHANNEL OF LONGITUDINAL AXIS

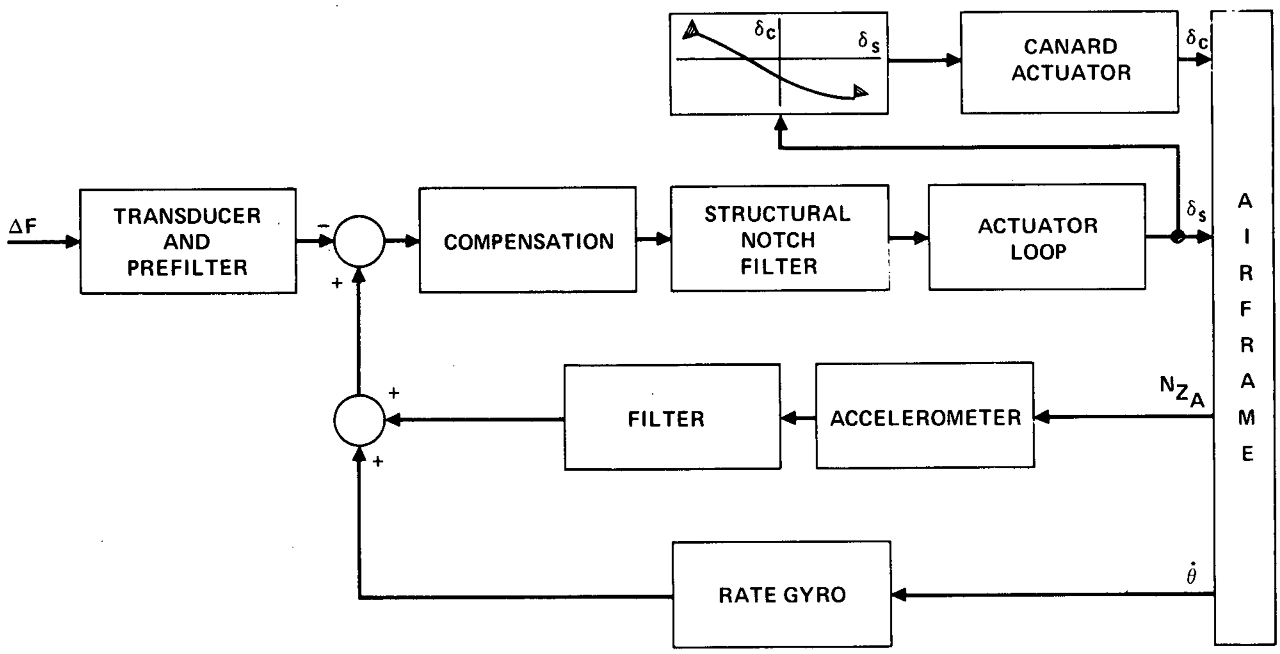
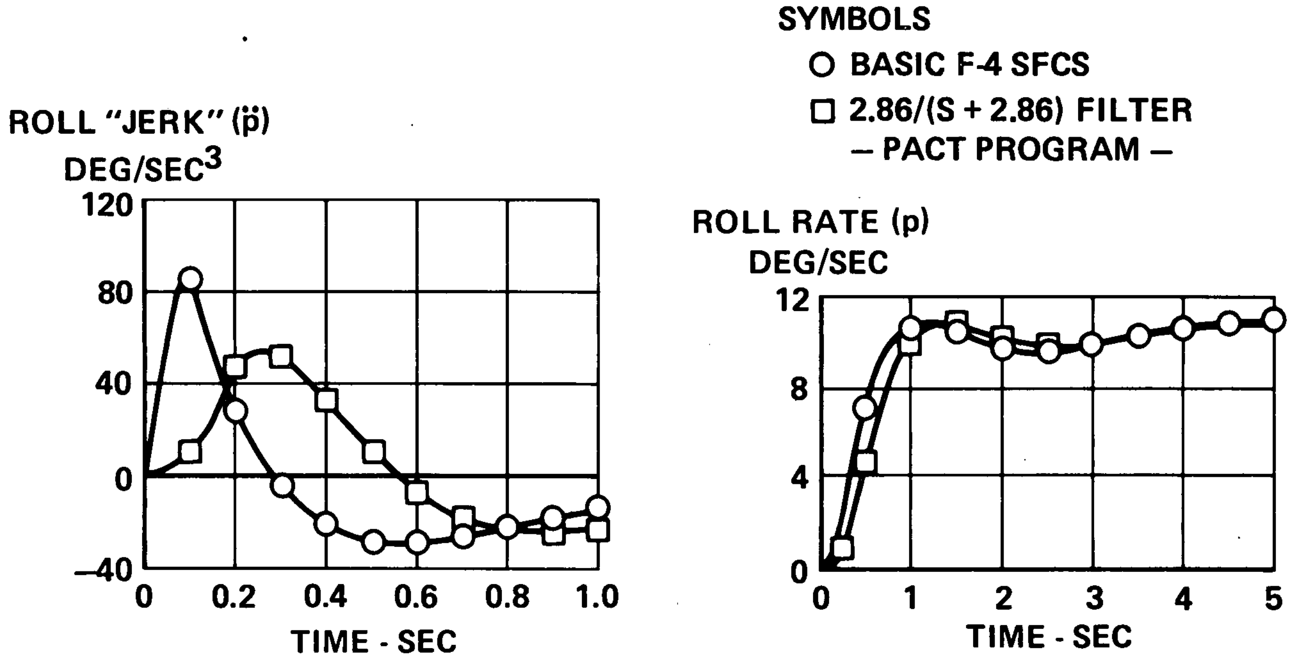


FIGURE 19
LATERAL AXIS STEP RESPONSES
0.6 MACH AT 30,000 FT



An early pilot-in-the-loop simulation study was performed to identify the proper longitudinal stick force prefilter and lateral EBU filter for alleviating the longitudinal and lateral sensitivity problem. On the basis of air-to-air tracking characteristics, the project test pilot selected time constants for the longitudinal stick force prefilter and for the lateral EBU filter. It was decided at that time to make the final filter selection at the time of the final PACT simulation since this early simulation utilized the SFCS aerodynamics (no canards).

The PACT pilot-in-the-loop simulation studies performed later in the program uncovered a potential aircraft divergence problem which was not anticipated in the original synthesis studies. At flight conditions at which the aircraft is particularly responsive, very large stick pulses could cause the simulated aircraft to diverge. Figure 20 illustrates this type of aircraft divergent response. This divergence tendency was caused by aircraft responsiveness due to the relaxed static stability. After some changes were made to the control system, the responses were strongly resistant to divergence even though the inputs were "abuse-type" stop-to-stop stick doublets. Therefore, for safety of flight considerations, modifications based on the simulation results were made to the longitudinal flight control system to limit the stick force which the electronics will accept, to increase the lag on the pilot stick force input, to reduce the normal acceleration feedback gain and to reduce the canard maximum rate capability. The rationale for these changes is:

(1) The magnitude of the stick force signal available for the SFCS was established to provide full stabilator deflection in the EBU mode. However, significantly lower stick force signals are needed to command aircraft motions requiring maximum stabilator deflections for the SFCS or PACT aircraft operating in the Normal mode. The Normal mode is required in the PACT longitudinal axis at all times since the unaugmented aircraft is unstable subsonically.

(2) The SFCS gains were established to provide a good maneuvering aircraft "across the board" with the stable basic airframe. The control system was considered by the project test pilot to be sensitive with these SFCS gains for the unstable PACT configuration when used for tracking tasks.

(3) Normal load factor feedback is destabilizing and was used to provide better response for SFCS. Since the PACT configuration is unstable at subsonic flight, less load factor feedback is required to properly shape the response.

(4) Lowering the canard rate limit effectively lowers the loop gain for large pilot inputs.

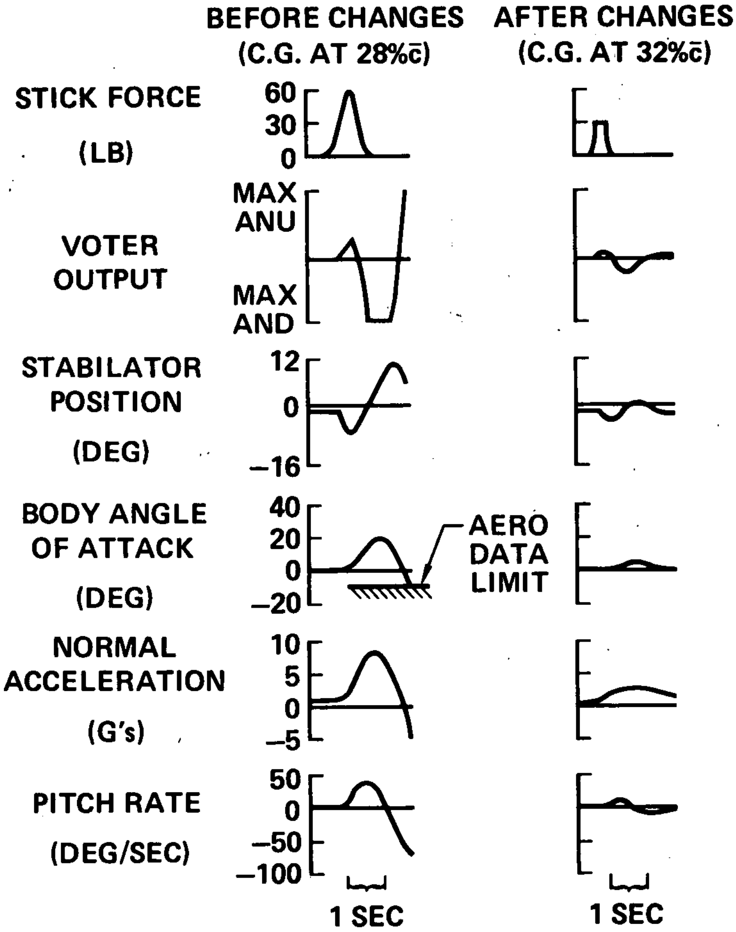
FIGURE 20

PACT CONTROL SYSTEM EFFECTS

- **LONGITUDINAL STICK RAPS**
- **M = 0.9 AT 15,000 FT**
- **SLAT AND CANARDS CONFIGURATION**

CHANGES

- **LIMIT PILOT FORCE INPUT (+30, −15 LB MAXIMUM)**
- **INCREASE STICK FORCE PREFILTER TIME CONSTANT**
- **REDUCE n_z FEEDBACK GAIN**

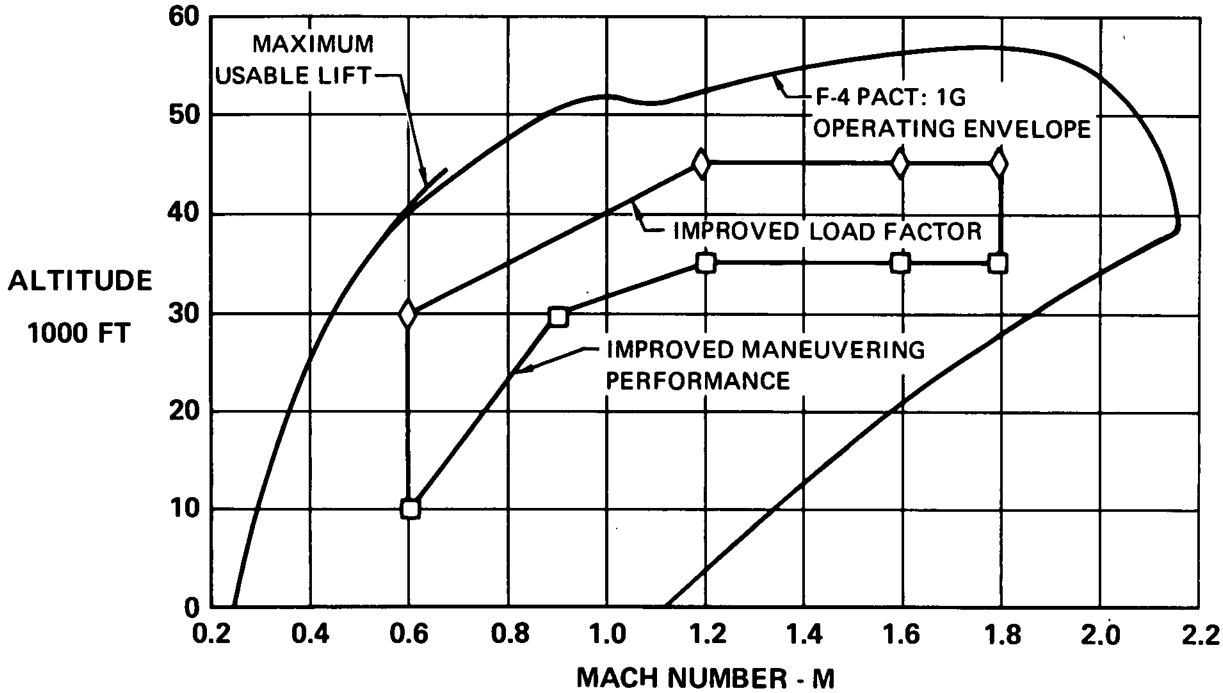


The initial shake-down flights of the F-4 PACT were flown in St. Louis without the canards, but with canard fairings in place on the duct inlets as well as fixed leading edge mid-board and out-board slats attached to the wing. After the shakedown flights, the aircraft was ferried to Edwards Air Force Base (EAFB) with wing tanks and centerline cargo pod installed. At EAFB the external store items were removed and the canards were installed. After completion of the flight test program, the aircraft was ferried back to St. Louis in the canards plus slats, wing tanks, and centerline cargo pod configuration.

The PACT flight test program consisted of 30 flights. Of these flights, 22 flights were made with full span canards plus slats, two flights were made with full span canards alone, and one flight was made with partial span canards plus slats.

Most of the F-4 PACT flight envelope shown in Figure 21 was explored. Good flying qualities were experienced throughout the flight envelope for each of the configurations. The longitudinal axis static margin ranged from roughly that of a production slatted F-4 (the configuration for the shakedown flights) to a minus 7.5%$\bar{c}$, reached in some of the canards plus slats configuration testing. Longitudinal and lateral-directional control pulses consistently indicated deadbeat aircraft damping. Buffet level was found to be very low during maneuvering flight, especially for the canards plus slats configuration. In general, the pilots preferred the flying qualities of the slats plus full span canards configuration.

FIGURE 21
F-4 PACT
FLIGHT CONDITION SUMMARY



One of the beneficial characteristics of the canards verified early in the flight test program was the favorable effect of the canards on approach speed as shown in Figure 22. For the canards plus slats configuration, pilots preferred full flap with Boundary Layer Control (BLC) on the flap for approach. In contrast, for the production F-4 with slats, the USAF prefers the half flap configuration without boundary layer control. Drooped ailerons were also added to the PACT aircraft to provide an additional 4-knot landing speed decrease when drooped ailerons plus full flaps were used instead of full flaps alone. Some further decrease in approach speed could be realized if the canard deflection were rescheduled to favor approach flight conditions.

Since performance results are currently being computed from the flight test data, extensive quantitative data are still not available. However, preliminary results indicate that the analytically determined performance predictions contained in Figures 23 through 27 will be verified by the flight test program.

Figure 23 shows the effect of the changes on lift coefficient. Subsonically, the basic F-4 was buffet limited (the point at which pilots complained about buffet level) at about 14° angle-of-attack, while the aircraft with slats can exceed 20° for a similar buffet level. In these estimates, the aircraft with canards was only given credit for the lift increment it developed in the wind tunnel tests at the angle-of-attack at which the comparison configuration (without canards) encountered limit buffet. In view of the slat-like characteristics evidenced by the canards in Figure 23, the method of prediction of limiting lift at buffet is considered conservative. As stated earlier, qualitative results from early flights indicate the aircraft with fixed slats and canards to be singularly buffet free. In the supersonic, control limited flight regime, the data of Figure 24 indicate the modified aircraft has capability to develop considerably higher lift than the unmodified aircraft can develop. Figure 25 relates the data of Figure 24 to aircraft ability to pull a given load factor at higher altitudes. As shown, canard surfaces increase the 4g maneuver ceiling capability by four to five thousand feet throughout the aircraft operating envelope. This converts into increased combat capability and versatility.

FIGURE 22

F-4 PACT APPROACH SPEEDS

C.G. @ 27%$\bar{c}$, α_{CP} =19 UNITS

SLATS EXTENDED

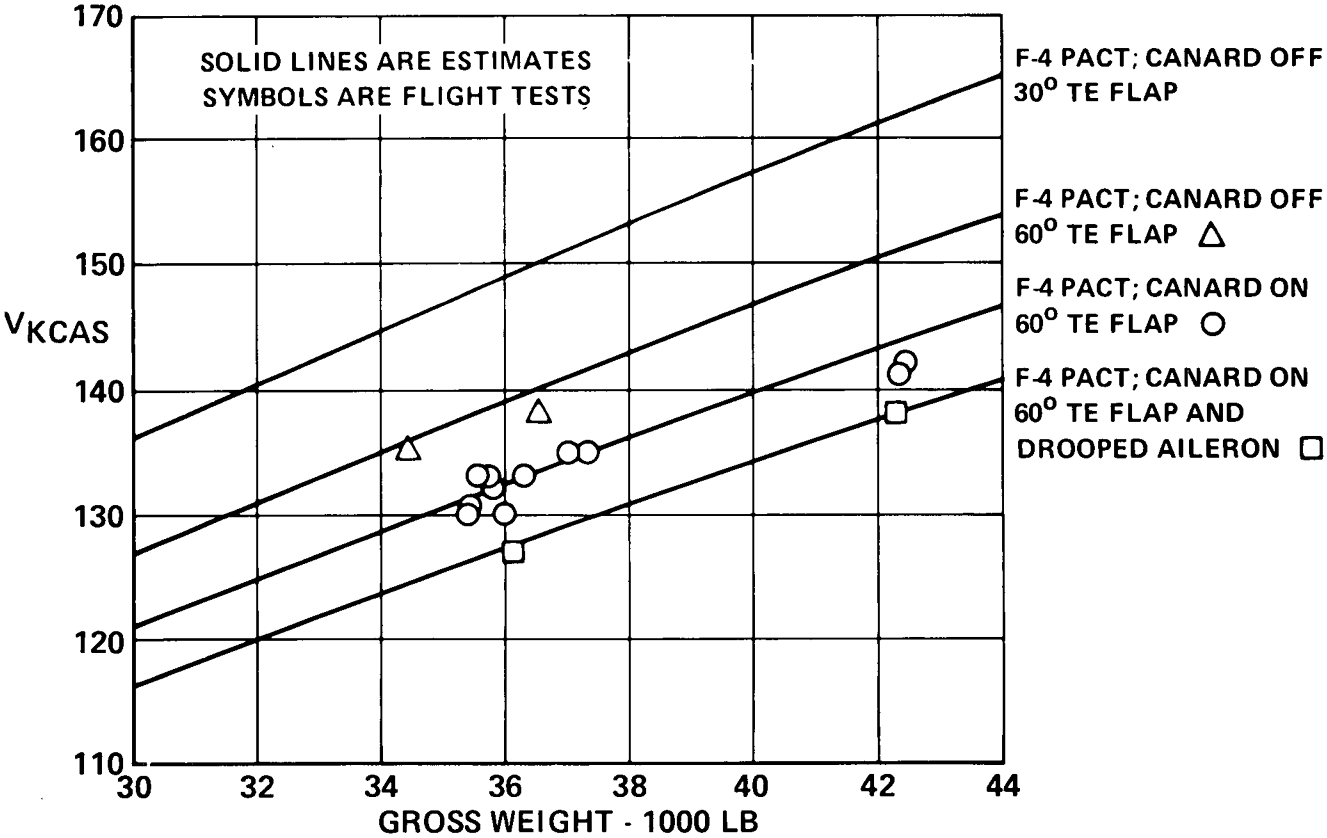


FIGURE 23

PACT TRIMMED LIFT CHARACTERISTICS

M = 0.9 C.G. = 33%$\bar{c}$

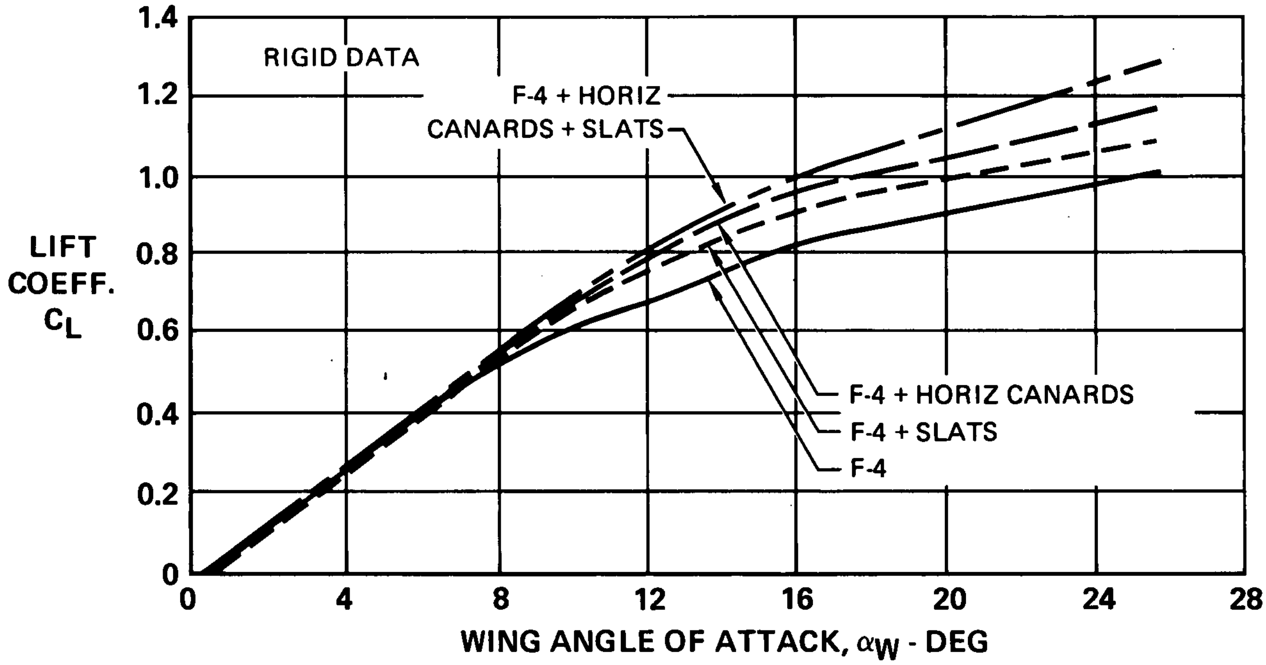


FIGURE 24

MODEL F-4
MAXIMUM USABLE NORMAL FORCE
C.G. = 31%c̄

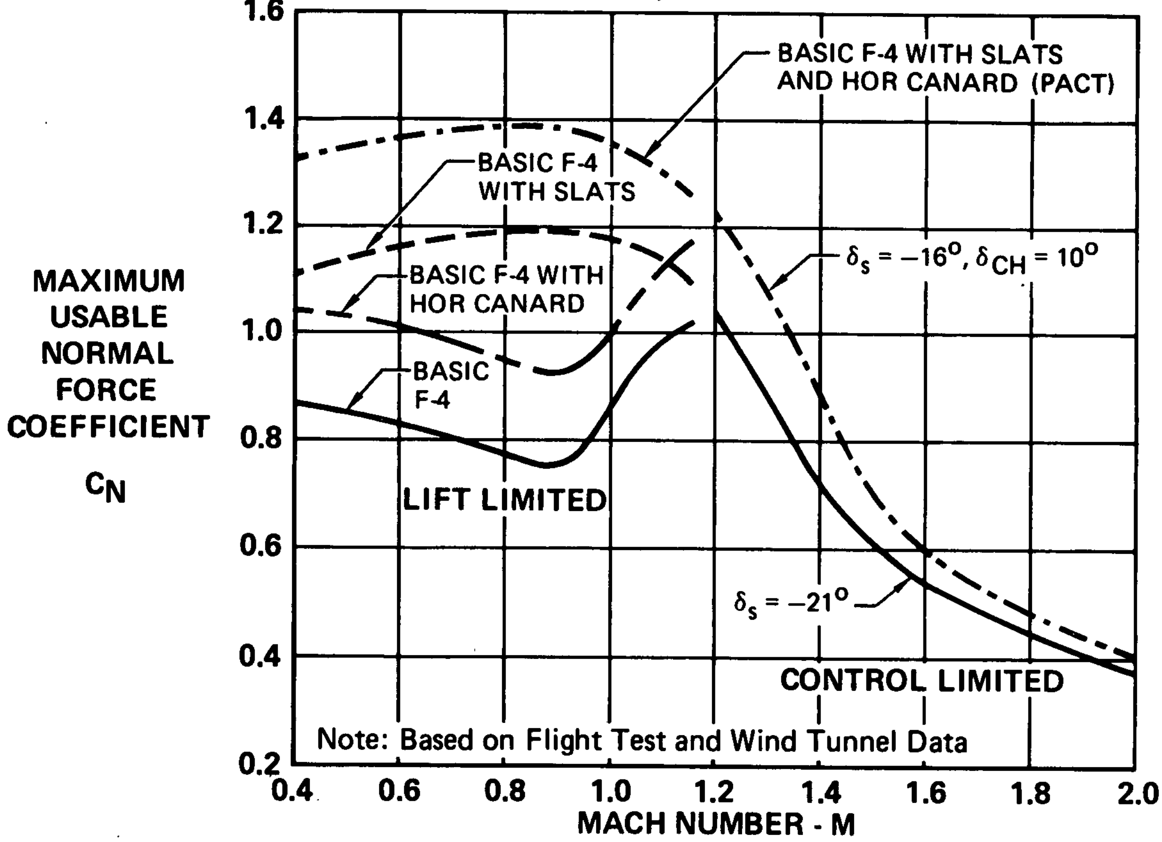
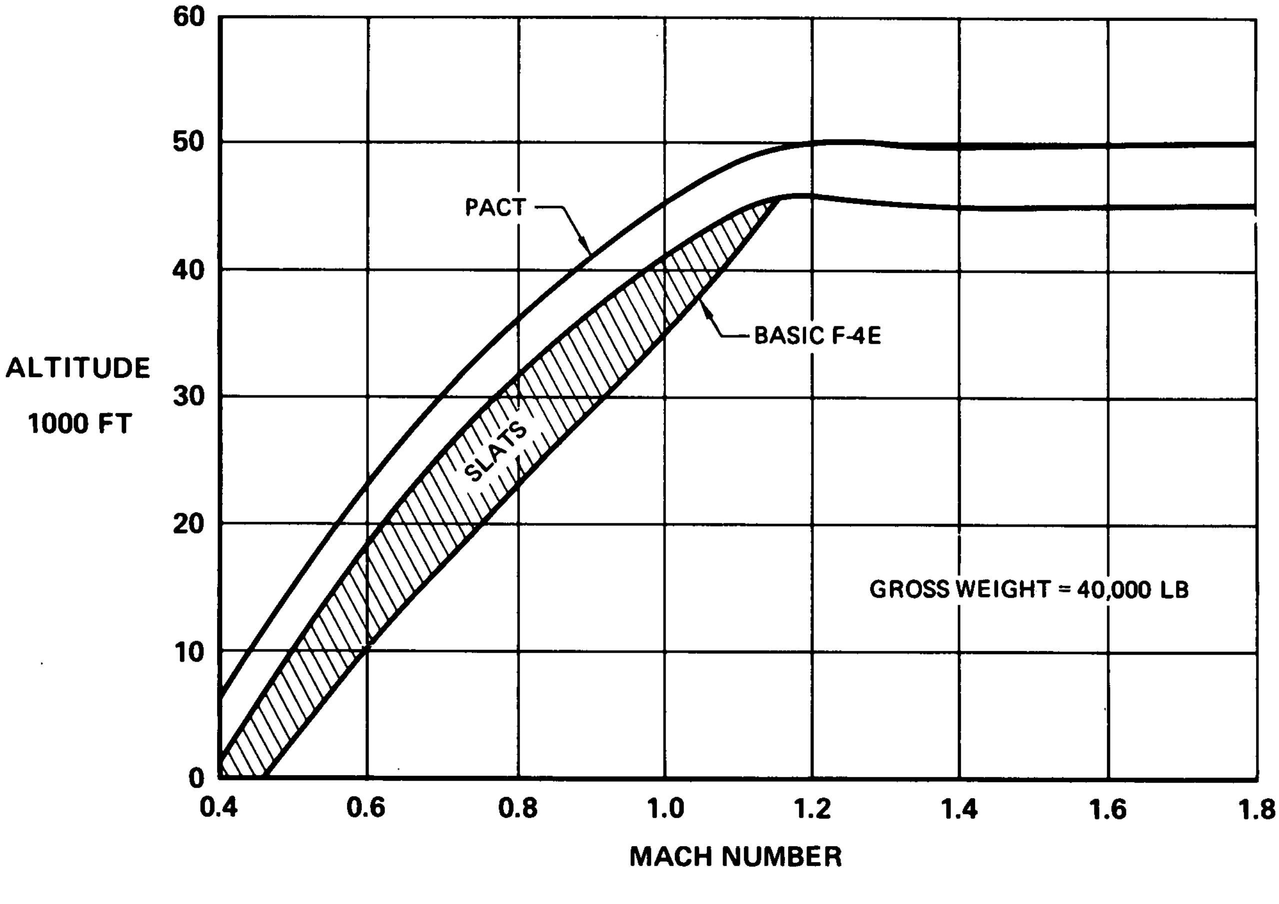


FIGURE 25

PACT EXPANDS THE FLIGHT ENVELOPE
CEILING FOR 4G INSTANTANEOUS MAXIMUM LOAD FACTOR



Another measure of combat capability and versatility is P_s, or specific excess power. This parameter is a measure of ability to climb while making turns or to accelerate while climbing in less tight turns. Increased g loading decreases P_s due to increased drag at high lift. If the drag is high enough, P_s becomes negative which means that altitude and/or airspeed will be lost in that particular maneuver. Incremental numbers of 100-300 ft/sec P_s improvement as shown in Figure 26 are considered significant enough to impress most fighter pilots.

Turn radius is of vital importance to fighter pilots. The increased lift capability illustrated in Figure 23 converts into improvements in turn radius as shown in Figure 27. While this sample is a supersonic one, the results are typical for lower speed conditions at the same altitude. At lower speeds, the absolute values of turn radius will decrease while the relative improvement in turning capability of the modified aircraft remains about the same. At lower altitudes, aircraft lift capability can exceed design structural envelopes; however, under those conditions, design placards control performance capability.

FIGURE 26

PACT EFFECT OF L.E. SLATS AND HORIZONTAL CANARD ON SPECIFIC EXCESS POWER

G.W. = 40,000 LB   C.G. = 31% $\bar{c}$ (MAXIMUM POWER)

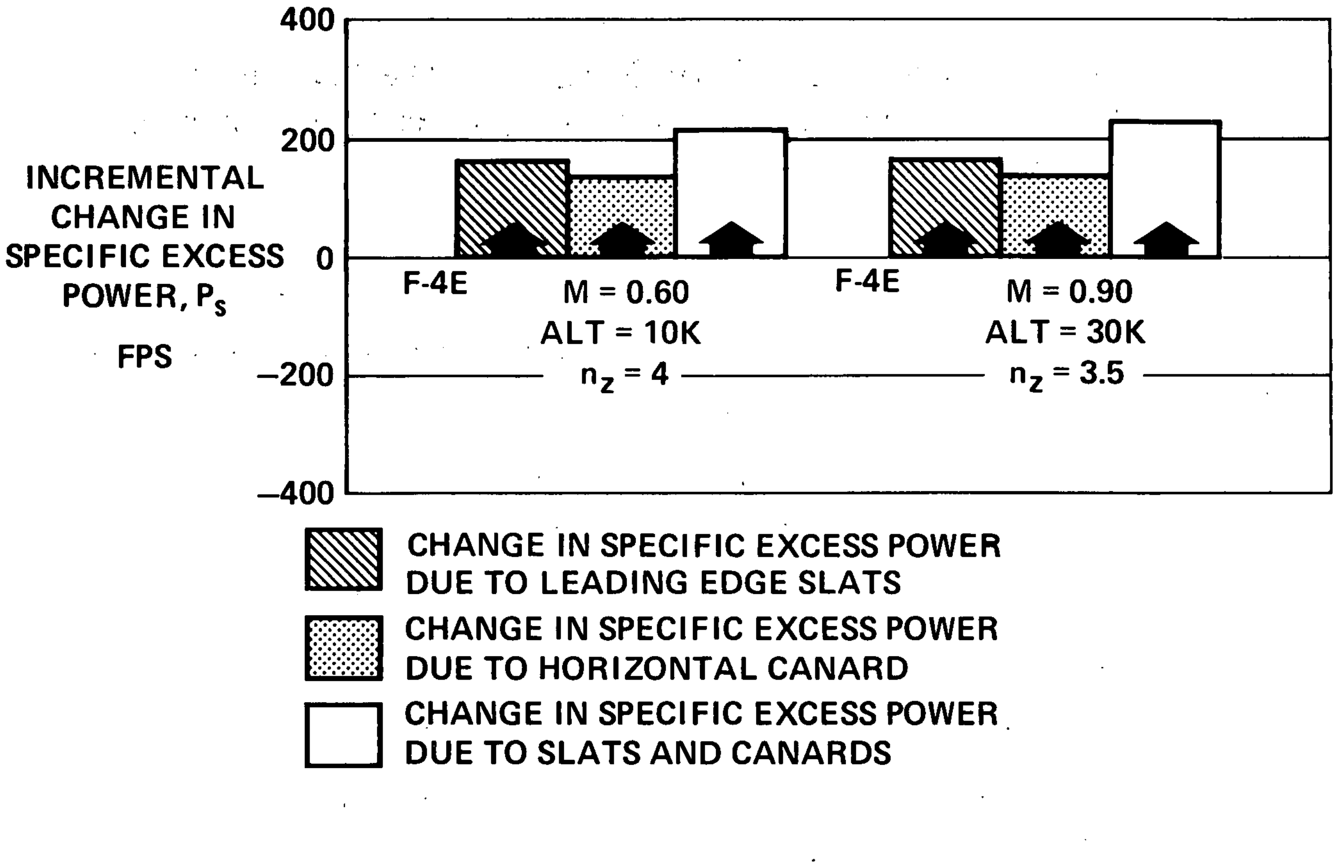
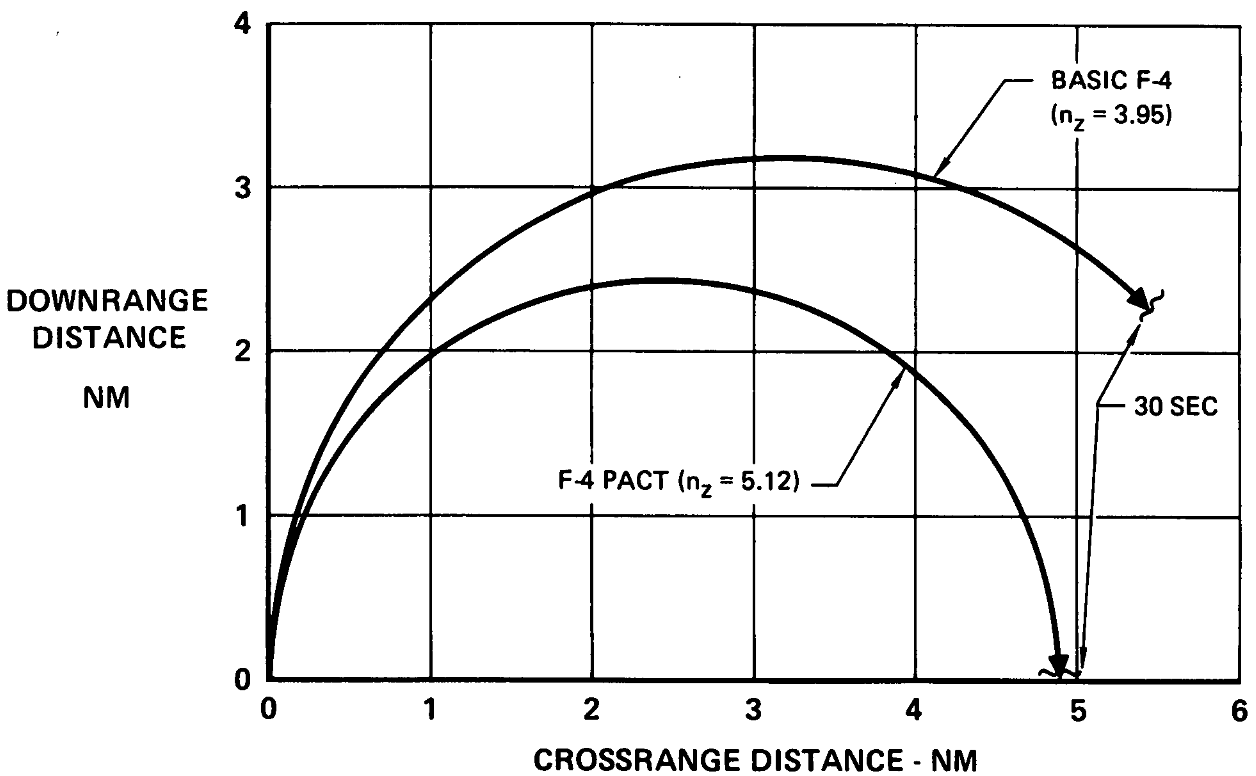


FIGURE 27

IMPROVED TURNING PERFORMANCE WITH CCV

M = 1.60
ALTITUDE = 45,000 FT
GW = 40,000 LB



Conclusions

During the SFCS and PACT programs, an active control Fly-by-Wire primary flight control system designed for controlling a longitudinally statically stable as well as unstable F-4 was developed, flown, and maintained. As a result of testing this system, the following more significant conclusions are offered:

- Fly-by-wire flight control of fighter aircraft is an existing engineering technology. Tailoring of such a system to an aircraft will provide significant improvements in aircraft handling qualities, and consideration of fly-by-wire control capabilities during preliminary aircraft design will provide still further advantages in total aircraft performance.
- The stability and performance characteristics of the SFCS demonstrated the feasibility of FBW systems for shaping aircraft response for general maneuvering and for precision flying.
- Fly-by-wire systems will require significantly fewer maintenance man hours per flight hour than will standard mechanical flight control systems.
- The SFCS flight test program provided a high degree of confidence in the safety and reliability of redundant fly-by-wire systems.
- A large scale, high-fidelity simulation which incoporates flight test hardware is considered necessary in the development of advanced flight control systems and for providing pilot training and confidence in such systems.
- The technology base developed as a result of the SFCS program has paved the way for further exploitation of more advanced flight control concepts such as control configured vehicle (CCV) and multi-mode controls for mission segment optimization.
- Digital computers offer potential advantages for implementing FBW control systems due to the complexity inherent with the redundancy and voting logic requirements.
- The PACT program has established feasibility for application of active control FBW technology through successful use of blended surface controls and relaxed static stability to achieve improved performance of a contemporary fighter aircraft throughout its flight envelope.
- Surface actuator position and rate limits need to be carefully defined and considered during active control FBW system design studies.
- FBW and CCV are key design tools since judicious application of these concepts can provide significant improvements in aircraft performance.

WEAPON DELIVERY IMPACT ON ACTIVE CONTROL TECHNOLOGY

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Dave Carleton, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, USA.

SUMMARY

This joint paper by the Air Force Armament and Flight Dynamics Laboratories emphasizes the need for cooperative efforts among the laboratories/test-organizations and users to improve and properly match aircraft pointing and armament component accuracies to achieve the maximum effectiveness with conventional weapons.

The Data Measurement Programs of the Armament Development and Test Center/Air Force Armament Laboratory are discussed, including the results and plans for the Instrumented Rack/Bomb and Gunnery Pipper/Fireline Trace and Impact Pattern Model Programs.

The Active Control Technology Programs of the Air Force Flight Dynamics Laboratory including objectives, designs, and results of the Tactical Weapon Delivery (TWeaD) Program are discussed. The objectives of the Multimode Control and the Control Configured Vehicle/Advanced Fighter Technology Integrator Programs are delineated.

It is concluded that incorporation of active control technology and matched armament component accuracies in future weapon systems shows promise for considerable improvement in the effectiveness of unguided weapons.

INTRODUCTION

The USAF must have a variety of weapon systems to meet all contingencies at minimum cost. Guided weapons are cost effective against targets that are heavily defended and/or are classified as small "hard targets"; however, unguided munitions are still the most effective for the majority of targeting situations. Also, the cost of a guided weapon is such that maximum effectiveness must be achieved from conventional guns, rockets, and free-fall weapons/bombs. Effective armament requires proper design matching of error sources; thus, as active control technology enhances aircraft pointing accuracies, armament system component accuracies must be adjusted accordingly. This interaction and its influence on tactics require close liaison among the Air Force Systems Command Laboratories and the using commands. This paper discussed two data measurement programs at the Air Force Armament Laboratory (AFATL) and Active Control Technology Programs at the Air Force Flight Dynamics Laboratory (AFFDL) which are directed toward achieving maximum effectiveness with conventional weapons.

DATA MEASUREMENT PROGRAMS

One of the AFATL roles within the Armament Development and Test Center (ADTC) is to develop the weapon delivery accuracy information that is included in the Joint Munitions Effectiveness Manuals (JMEM) used by NATO allies. Instead of the cut-and-try basis of decisions, operations planners can now determine the size of a flight of current aircraft/weapons required to accomplish a mission to any desired confidence level within a specified scenario. In conjunction with scenario/targeting studies, force level and effectiveness structuring can be determined from the given aircraft and weapon inventories.

In conjunction with the ADTC Test Wing and other USAF test organizations, AFATL has the responsibility of separating causes from effects and of proving or disproving claims and/or error budgets/estimates. The majority of the parameters currently available in JMEM, such as Circular Error Probable (CEP) and Range/Deflection Error components (REP/DEP), are gross measures of effect that have been derived from tests and combat assessments by the hopper statistics technique. The hopper statistics technique is defined to imply statistical analysis without due regard to all control variables. These measures are identified by aircraft/weapon combination and delivery tactic but obviously do not show the cause. Also, the rapid evolution of improved/new aircraft and weapon designs require continuous extrapolations, experienced opinions, and/or conservative estimates. The first part of this paper is directed to discussing some of the cause-identification programs which support the major role of ADTC/AFATL to design, develop, and test-validate more effective weapon systems.

Instrumented Rack/Bomb Programs

A large number of programs to improve bombing accuracies have been run with varying degrees of success. The majority have been concerned with deriving techniques/sensors for achieving the calculated geometric conditions for the aircraft at the time the bomb button is depressed. Static tests, analyses/budgeting, and simple drop tests are well established flight test procedures. However, store separation studies have been primarily concerned with the safety of the launch platform, and CEP has still been used to categorize results. The Instrumented Rack/Bomb Programs were devised to identify and to quantify the contributing error sources in bomb delivery systems (Figure 1).

The accurate prediction of the impact point of an unguided bomb is dependent on three distinct parameters:

a. Bomb position at release.
b. Bomb velocity magnitude and direction at release.
c. The net forces and moments acting on the bomb at each point in its trajectory.

The bomb position and initial delivery platform velocity and direction at release can be accurately determined by using 30-frames-per-second cinetheodolite tracking data. However, prior to 1972, no attempt had been made to measure the forces and moments that the bomb rack exerted on the bomb at release under dynamic delivery conditions nor the influence of the delivery aircraft flow field interaction with the bomb (Figure 2). In 1972, ADTC/AFATL and AFAL in a joint effort designed, constructed, and flight tested a TER-9A instrumented bomb rack. The measured rack parameters associated with the weapon release sequence are listed below;

a. Release signal.
b. Cartridge fire.
c. Front and rear initial store movement.
d. Ejector foot motion and force.
e. First mechanical linkage movement.
f. Sway brace strains.
g. Front and rear hook loads at separation.

In the TER-9A test (Reference 1), 43 productive missions were flown. Figure 3 gives the mean weight, moments of inertia, and center of gravity, with the associated standard deviation in each variable. Of particular interest is the fact that the rack ejector foot is striking the bomb about three inches behind the center of gravity, which should give a nose-up attitude at release. However, in the film of the releases, some bombs appeared to eject without any initial pitch angle. Figure 4 gives the measured mean, standard deviation, and range of magnitude of the initial yaw angle. As might be expected the results indicated that a bomb ejected from the slant angle of a shoulder station has a larger yaw angle than does one ejected normal to the wing. The ejection velocity was measured by two different techniques (linear potentiometer and chamber pressure measurement devices). Figure 5 shows the actual test data plotted for the two methods, as well as the nominal ejection velocity for both the centerline and shoulder stations. Figure 6 shows the mean and standard deviation of the ejection velocity for all bombs. Here again, the performance between the centerline and shoulder station is different, due mainly to the difference in gravitational forces acting on the bomb along the ejection axis.

Assuming that each measured release parameter acted independently on the bomb impact point, a sensitivity analysis (Reference 2) of the rack variables was performed. The results are presented in Figure 7. It should be noted that the assumption of independence is probably invalid, but the data base to date is too limited to even permit a conjecture on the true distribution of the various parameters, the correlation or the independence of these parameters. Since the independence of the variables is questionable, the influences of the various perturbations in the bomb-rack release sequence were not combined to give an estimate of total miss distance.

In an effort to obtain a data base sufficiently large to answer these and perhaps other questions, MER-10A and MAU-12B/A bomb racks have also been instrumented. To enhance the data obtained from the measurement of the forces of the MER-10A, 40 MK-82 bombs have been modified to allow the placement of a forward looking camera in the nose of the bomb. The camera is energized immediately prior to release and then ejected from the bomb prior to impact to preclude the necessity for bomb recovery. Utilizing a target grid layout as represented in Figure 8, roll, pitch, and yaw can be computed as a function of time throughout the trajectory.

Once a sufficiently large data base is established, the data will be analyzed to determine the empirical distribution of each of the variables and to examine the interrelationship between the variables. The eventual goal is to develop a model that will simulate the interface of aircraft flow fields and weapon release. All of the data collected from these tests should greatly improve the design of new rack systems. However, the immediate pay-off is anticipated to be in computerized release systems which will use on-board computers to partially control the station to station variation in some of the parameters. Such a system should certainly improve single drop accuracy and also allow for better pattern control when bombs are ripple released.

Gunfire Modeling Programs

When properly used, the aerial gun is the most versatile and efficient weapon available. However, due to the simple direct-fire, pull-the-trigger operation and the measured zero means and circular normal distributions from hopper statistics, the use of only the Central Limit Theorem and other large population probability laws can result in gun designs and tactics that give a high percentage of misses and/or overkill when hits do occur, or vice versa. Simiarly, in the past, considerable emphasis has been placed on linear autocorrelation coefficients. Zeroings simply show the cyclic nature of extended aiming processes and are not indicative of current short-burst tactics. Previously reported randomness is due to inadequate regression analyses of extended duration aiming and to unfiltered vibrations, drive speed variations, and tolerances of the recording instrumentation. It is also noted that the linear autocorrelation coefficients for long and short lines are identically unity. Further, it has been assumed that the gunfire pattern dispersion is random circular normal and that the dispersion can easily be adjusted without regard to the actual gun mechanization and installation. Thus, additional measures that will be meaningful to gun system design and tactics are necessary.

For about the last decade, AFATL has been engaged in investigating the causal nature of gunfire characteristics, through data collection, analyses, and modeling. Gunfire modeling efforts at AFATL consist of three major categories. The first category efforts are directed toward definition of man-machine aiming performance, with the characteristics of sight-cue/pipper relationships to the intended-aimpoint/target-center defining the processes. The second category efforts are directed toward definition of the nominal fireline/firepoints in the impact plane. The fireline can be visualized as a trace of the shot pattern that a zero dispersion gun would make in a plane perpendicular to the line of sight and through the target. For measurement purposes, the coordinate system in the plane has a target-centered origin, and the vertical and horizontal coordinates are defined by the vertical and horizontal plane references, respectively, of the gun. Thus, the fire trace differs from the pipper trace not only by considering the gun-to-sight transfer function but also by encompassing all firing errors, including those from uncertainties during the time of flight. The third category efforts are directed toward definition of impact points about each instantaneous firepoint.

Figure 9 (from Reference 3) shows schematically some of the additional parameters, such as draw magnitudes and velocities and the closest point of approach (CPA), originally developed in pipper analyses of gun camera film to define what the pilot believes he is doing. This has also been applied in fire trace analyses of simulator outputs. Figures 10 and 11 (from Reference 4) show the measured/original and time-series modeled firelines, respectively, for the indicated engagement conditions. Because fireline characteristics are the necessary inputs for gun system design, AFATL is establishing a data-bank of these time-series fireline models as a function of sight, pilot/signature, geometry, and target evasion characteristics. It is noted that the data bank technique is necessary to define fireline shapes due to the multiple driving functions, even though the gun-to-sight transfer function is explicit. CPA and fire timing are derived statistically from the data bank. Additionally, systematic errors--boresight and instrumentation mechanization accuracies--and target uncertainties that are not present in simulator outputs are treated statistically. Both deterministic and statistical properties are incorporated in the AFATL evaluation model, which can evaluate the influence of gun design details for specific engagement conditions. Maximum, minimum, standard deviation, and the probability-of-at-least-n hits are examined in addition to expected values. For example, a differentiation can be made between the case where one out of ten attempts causes an average of twenty hits and the case where five out of ten attempts cause an average of four hits. Both, of course, have an expected value of two hits per pass.

AFATL has recently developed an Impact Pattern Model which addresses mutli-barrel aimpoint, action times, rockback/droop, sidecock, mean-point-of-impact (MPI) variance, gun/mount vibrations and loadings, throw/coning if applicable, and ballistic dispersion characteristics to define impact points about the mean fireline. Figure 12 shows an AFATL plot routine output for one of the actual patterns used in the visualizations and determination of the model. Before firing (⊞) and after firing (⋈) boresight patterns, relative to the MPI (0,0 coordinates), are shown on the right. The shot impact coordinates are denoted by barrel symbols (0, ⊟, X, Y, +, and *), while the adjacent numeral (0 to 9) to the right corresponds to the first through the tenth shot from that barrel. Boldface numerals appear at the MPI of each barrel. Although representing the sequential order of firing, the number does not indicate which barrel fires first. It is noted that the first shot (*0) impact coordinates from barrel 6 are among the optical boresights.

The present data collection, analyses, and modeling were accomplished for a six-barrel Gatling gun; however, the model is quite simple and general. Shot groupings are used to account for motion trends in the total and single-barrel patterns. Vertical and horizontal standard deviations are derived to generate burst-to-burst variances of the total MPI about the intended/mean firepoint. The MPI distribution, with non-zero means and standard deviations, for each barrel is referenced to the total MPI. The pattern distribution is used to generate shot locations about the MPI of each barrel. At present, the part of the model that generates barrel patterns uses bivariate normal distributions, not necessarily circular normal, for those populations utilizing Analysis of Variance derived estimators. It is noted that present estimators have been derived from very limited data banks. Additional data are also required to establish the correlation of barrel MPI to optical boresight references. Variances between before firing and after firing boresights and between total MPI and average optical boresight complicate correlation analysis. However, inter-pattern and intra-pattern analyses should be helpful in explaining some of the observed phenomena. Recent instrumentation improvements, including acoustical scoring systems and automatic data management and retrieval programs at ADTC gun ranges, are not facilitating the accumulation of these data on USAF gun systems.

When these variances are added to the (mean) fireline model output, a realistic estimate of the spatial and sequential distribution of hits on a surface can be generated. Although quite detailed, such a model is essential when target sensitization must be addressed.

ACTIVE CONTROL TECHNOLOGY PROGRAMS

The Air Force Flight Dynamics Laboratory has both on-going and completed programs that exploit active control technology to enhance weapon delivery. Based on the assumption that aircraft handling characteristics do to some degree affect unguided weapon delivery accuracy, a major objective of these programs is to quantify the relative effects on vehicle response of various control concepts and the ultimate effect on weapon delivery.

Tactical Weapon Delivery (TWeaD) Program

A highly successful flight test development program recently completed was the Tactical Weapon Delivery (TWeaD) Program (Reference 5). In this program, the limited authority rate damper system in an F-4 Phantom II aircraft was replaced with a high gain, high authority Control Augmentation System (CAS). The major objective was to evaluate the improvement in weapon delivery accuracy that could be achieved with an improvement in aircraft handling characteristics.

The basic F-4 aircraft was designed as a high altitude, supersonic interceptor; but it has been used in a variety of tactical roles, such as air-to-ground weapon delivery and air superiority, including gunnery. The basic aircraft handling characteristics were compromised in these roles, so the F-4 represented a logical choice for evaluation. The mechanical portion of the flight control system was retained, and the CAS was installed with a minimum amount of airframe modification. Previously reported aircraft handling quality deficiencies that were believed to affect weapon delivery accuracy included low longitudinal short period damping, stick-force-per-g lightening and reversal, adverse yaw due to aileron deflection, and negative speed stability in the transonic region. The CAS was designed to eliminate these problems. Simplified block diagrams of the standard flight control system are shown in Figures 13 and 14.

The longitudinal portion of the CAS incorporated a three channel fail-operate/fail-neutral scheme. The functional block diagram appears in Figure 15. As shown in Figure 15, pilot stick force commands are compared with aircraft pitch rate and normal accelerations. If there is any difference between the pilot command and the aircraft response, an error signal is generated which displaces the stabilator via a series valve to drive the error signal to zero. The series valve had approximately 70% authority, which prompted the use of a triplex scheme for failure protection. This mechanization resulted in deadbeat short period damping, constant stick-force-per-g, and elimination of the transonic speed stability problem. The pitch CAS also contains a neutral speed stability mode which eliminates the requirements to re-trim the aircraft with airspeed changes. This feature proved to be very beneficial for air-to-ground weapon delivery, eliminating the need to trim the aircraft during the final approach dive. This allowed the pilot to maintain a constant dive angle with greater ease, so that more attention could be paid to monitoring airspeed, altitude, and tracking error. The tracking error was also reduced due to the deadbeat damping of the short period mode.

The lateral-directional mode includes the roll and yaw axis mechanizations of the CAS. Both axis mechanizations include two channel, fail-neutral operations using the production Stability Augmentation System (SAS) series servo valves which had ±5 degrees of authority, out of ±30 degrees of total control deflection. A functional block diagram is shown in Figure 16. The roll CAS converts lateral stick deflection into a dual-gradient, commanded-roll-rate signal. This is then summed with actual aircraft roll rate to generate a servo command signal. A dual gradient was required to reduce sensitivity for precision tracking while retaining maximum roll performance. The yaw CAS utilizes rudder pedal force and aileron position multiplied by angle of attack as command inputs. These signals are summed with aircraft lateral acceleration and stability-axis yaw rate to drive the rudder series-servo. Stability-axis yaw rate is obtained by multiplying roll rate by angle of attack and subtracting body axis yaw rate. Aileron position is multiplied by angle of attack in compensation for increasing adverse yaw from aileron with increasing angle of attack. The net effect of these modifications was a significant reduction in adverse yaw due to lateral stick deflection and an increase in apparent lateral control power. The CAS essentially decoupled the aircraft in roll and yaw. Response to a lateral input produced no residual oscillations (i.e., eliminated the Dutch Roll Mode), and a directional input resulted in a lightly damped, wings-level yaw oscillation.

Control harmony was enhanced by the addition of a feel spring in the pitch axis (replacing the production feel bellows) and identical lag filters in the pitch and roll forward paths.

The final system configuration was determined by conducting air-to-air tracking using a fixed, depressed gunsight at various flight conditions and by flying air-to-ground passes. Gun camera film, on-board instrumentation, and qualitative pilot commands were used in the selection of gains and models that resulted in the smallest tracking error and in producing the best flying qualities. The system was capable of in-flight gain changing so that the process of selecting the final configuration was accomplished rather rapidly. Once this was completed, stability and control and finally weapon delivery evaluations were conducted. The latter evaluation was completed by dropping inert ordnance and measuring the impact point or miss distance from the target. Other data obtained included aiming errors at release and theoretical impact point. The results of this program were compared with those obtained from a similarly conducted program that used a standard F-4 aircraft for the tests. A 27 percent reduction in impact miss distances was demonstrated with the CAS equipped aircraft. This percentage was also demonstrated in a later evaluation conducted by an independent test agency (Reference 6).

Air-to-air tracking tests were also conducted and the results were compared with those for a standard F-4 aircraft. In these tests, a fixed depressed reticle was used, and a 33 percent reduction in RMS (root mean square) tracking error was demonstrated for a benign targeting condition; i.e., a constant g-load turn.

The major conclusions from the program were that handling characteristics do influence weapon delivery accuracy and that several control law concepts which enhance handling characteristics also enhanced weapon delivery results. Specific findings that were of benefit in the longitudinal axis were: (1) deadbeat short period damping, (2) neutral speed stability, and (3) linear stick force gradients for maneuvering flight. All are desirable characteristics for an effective weapons delivery platform. Lateral-directional handling characteristics that were considered to be beneficial were: (1) decoupling of the roll and yaw axes, (2) reduced adverse yaw by incorporating rudder and stability-axis yaw-rate feedback, and (3) rudder to counter adverse yaw produced by aileron deflection. These factors, combined with harmonizing the pitch and roll responses and control forces, were the key factors resulting in the reduction of delivery errors.

Multimode Control Program

A current on-going program about to go into flight test is directed toward control system development to reduce weapon delivery errors. This is the A-7 Digital Multimode Flight Control Program (Reference 7). A major objective of this program is to demonstrate that specific aircraft response characteristics for a particular weapon delivery task, such as terminal aerial tracking or fine tracking for air-to-ground, can further reduce delivery errors. This concept is referred to as a multimode control system. The A-7D Corsair II was chosen as a test vehicle because the standard flight control system has a dual channel CAS which allows production sensors and series servo valves to be used without modification. The test hardware features dual digital computers providing fail-operational/fail-safe capability to investigate the feasibility of digital technology in future flight control systems.

The basic A-7D flight control system consists of mechanical linkages, two channel fail-neutal CAS in pitch and roll, and a SAS in the yaw axis. A functional block diagram of the production system is shown in Figure 17. As seen in the diagrams, the production control system configuration remains constant throughout the normal flight envelope, regardless of what mission task is being performed. This approach requires that the system be designed to yield acceptable handling characteristics throughout the flight envelope and that it can represent compromises for certain mission tasks. An example of this problem is that while neutral speed stability is very desirable for air-to-ground weapons delivery, the takeoff maneuver cannot be performed in this mode.

To develop a system that could be easily reconfigured for various tasks, the multimode system was designed to perform certain mode switching automatically, once a particular task to be completed has been identified by pilot control signatures.

The systems to be compared consist of the standard aircraft flight control configuration versus Flight Path (FP) and Precision Attitude (PA) modes. The objective of the FP mode is to provide increased normal acceleration response to applied stick forces to achieve more rapid flight path control. This is accomplished by utilizing pseudo neutral speed stability, increasing the normal acceleration response, and coordinating the aircraft in roll. A block diagram is shown in Figure 18. The PA mode produces a pitch rate response to applied stick forces to provide desirable dynamic response characteristics for precise control of aircraft attitude. The response is a minimum time response with near deadbeat damping. The mode also has the capability of coordinating the aircraft about the weapon fireline. Wind gust response is minimized for aircraft rotation in the PA mode and translation in the FP mode. A functional block diagram of the PA mode is shown in Figure 19.

Preliminary simulation results indicate that this multimode scheme should allow the pilot to make the necessary conversion maneuvers in a shorter time on target than the standard fixed-mode system.

Control Configured Vehicle Program

The concept of the control configured vehicle (CCV) is being explored in current study efforts, directed toward future flight test validations. The concept utilizes the principle of artificial stabilization on an unstable vehicle that has independently activated surfaces to produce pure forces or moments or a blending of the two (Reference 8). With this concept, a minimum time solution for changes in aircraft state can be realized. Implementation of this scheme for a typical future aircraft configuration is shown in Figure 20 for the cases of pure aircraft translation or pure rotation. Direct side force and direct lift control are possible with such a scheme. With these additional degrees of freedom available to control the aircraft, multimode control concepts and interface with the fire control system can be exploited to realize potentially large benefits in terms of reduced conversion time, increased tracking time and firing opportunities, and reduced pilot workloads.

The Advanced Fighter Technology Integrator Program has been recently undertaken to demonstrate and validate the payoffs in conventional weapon delivery and combat potential that such schemes offer.

CONCLUSIONS

In the past, many pilots have been unable to place the sight cue on the target, and even when the cue was reasonably close, the weapons missed by considerable distances. A few pilots could remember similar situations and after adjusting accordingly could hit more consistently. The large majority, however, either followed ideal procedures or became indifferent or both. Therefore, the incorporation of active control technology and of matched armament component accuracies shows promise for considerable improvement in the effectiveness of unguided weapons.

REFERENCES

1. AFATL-TR-73-111, Flight Test Results for An Instrumented TER-9 Bomb Rack, May 1973. (Unclassified)
2. AFATL-TR-70-110, Theoretical Sensitivity Analysis and System Delivery Accuracy Computations for High and Low Drag Weapons at Several Subsonic and Supersonic Delivery Conditions, October 1970. (Unclassified)
3. AFATL-TR-72-225, Volume III, Performance Analysis of Pilot Perceived Fireline (PAPPF), December 1972. (Unclassified)
4. AFATL-TR-73-241, Aerial Gunnery Methodology: Fireline Modeling Techniques, December 1973. (Unclassified)
5. AFFTC-FTC-TD-72-1, Development and Evaluation of the TWeaD II Flight Control Augmentation System, D. L. Carleton, R. E. Lawyer and C. W. Powell, November 1972. (Unclassified)
6. TAWC Project-72A-007 Final Report, TAC Evaluation of TWeaD Modified F-4C, B. L. Bowman, August 1972. (Confidential)
7. Proceedings of the IEEE 1974 National Aerospace and Electronics Conference. A Digital Multimode Flight Control System for Tactical Fighters, T. Yechout, et.el. May 1974 (Unclassified)
8. Office of Naval Research DISO-17508-I, An Investigation of the Potential Benefits of Direct Sideforce Control from a Mission Viewpoint, E. F. Carlson, July 1973. (Unclassified)

ACKNOWLEDGEMENTS

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FIGURES

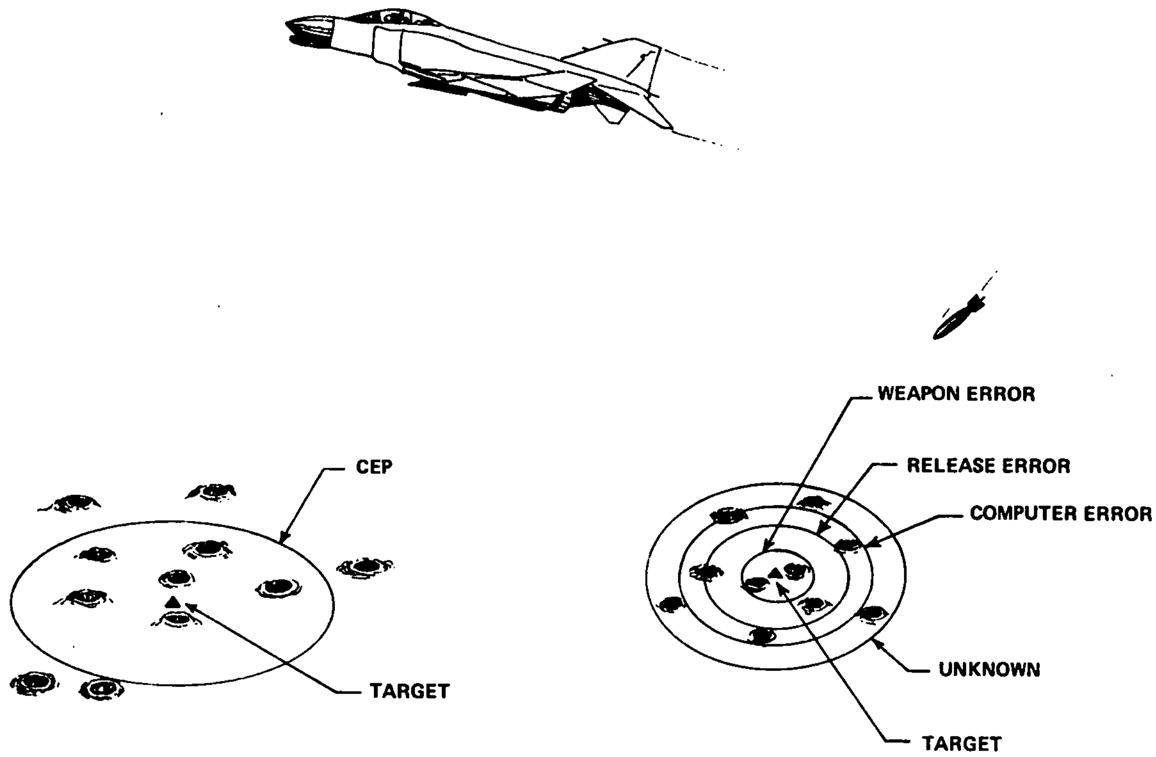

FIGURE 1. DETERMINATION OF DELIVERY ACCURACY

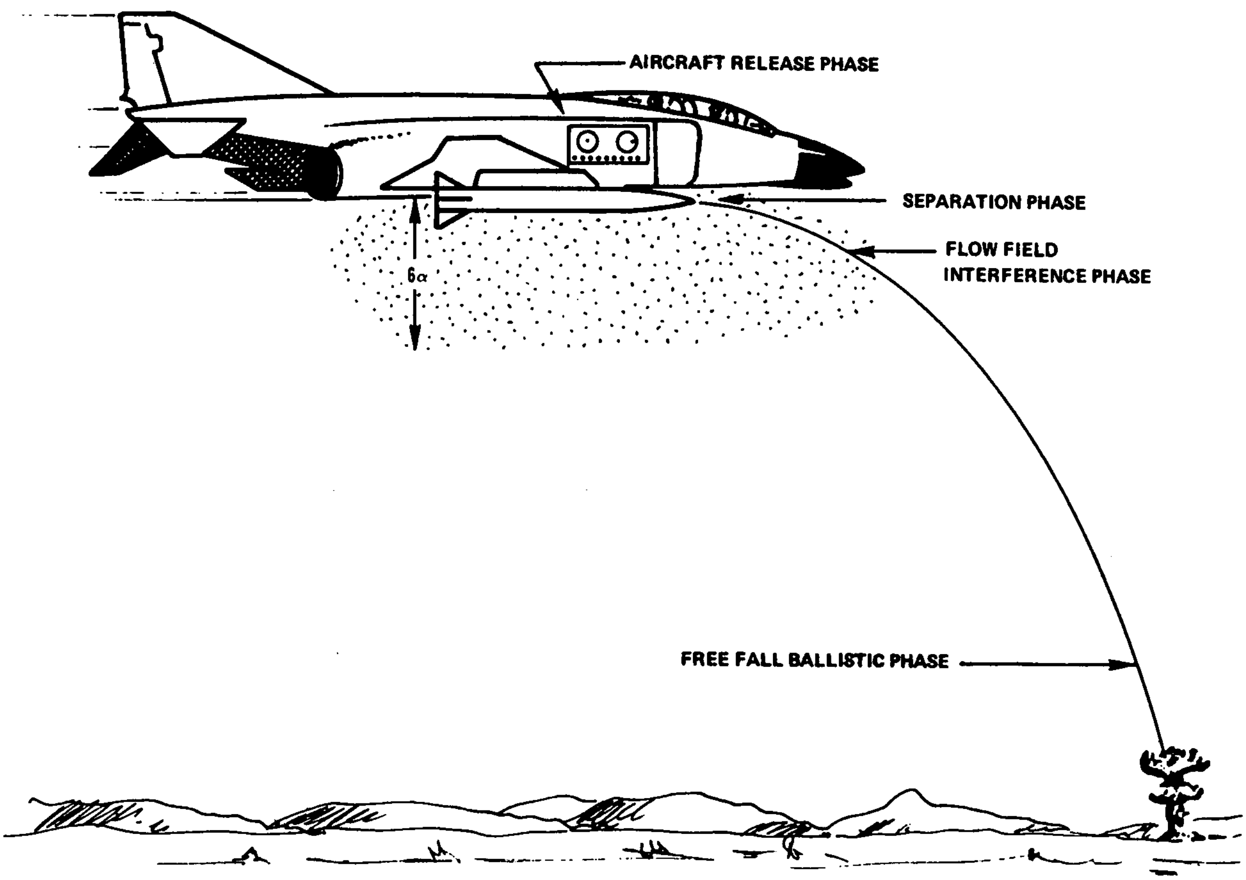

FIGURE 2. WEAPONS DELIVERY CYCLE

	DIMENSION A	DIMENSION B	MOMENT OF INERTIA ROLL SLUG FT^2	MOMENT OF INERTIA PITCH/YAW SLUG FT^2	WT(LBS)	CIRCUM-FERENCE (INCHES)
MEAN FOR 43 TER-9 BOMBS	35.77	3.12	1.71	37.65	508.06	34.03
STD DEVIATION FOR 43 TER-9 BOMBS	.29	.30	.04	.95	5.64	.09
MEAN FOR ALL CL BOMBS	35.66	3.20	-	37.25	508.46	-
STD DEVIATION FOR ALL CL BOMBS	.17	.27	-	.42	4.38	-

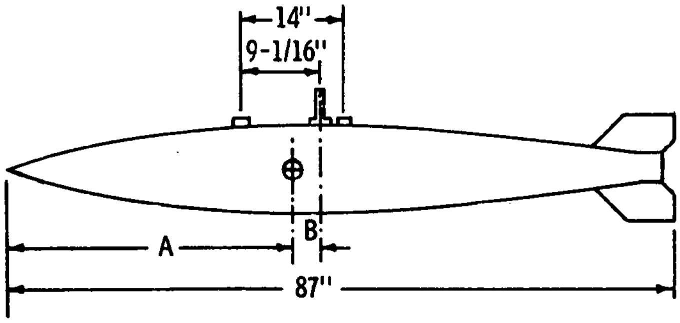

FIGURE 3. TER-9/CLASS MK-82 PHYSICAL DATA

RELEASE STATION	MEAN	STANDARD DEVIATION	RANGE MIN	RANGE MAX
CENTER LINE	4.8^0	2.9^0	0^0	12^0
SHOULDER	9.0^0	3.6^0	2^0	15^0

FIGURE 4. TER-9/CLASS
MK-82 BOMB (CLASS) INITIAL YAW ANGLE

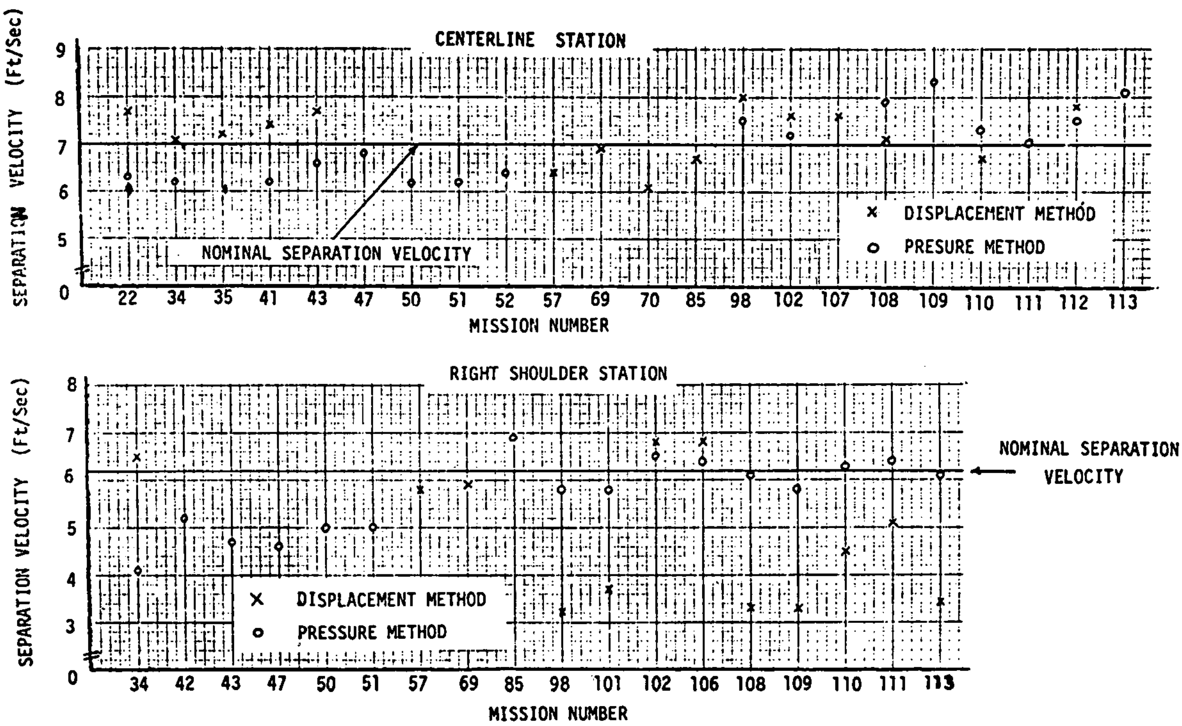

FIGURE 5. TER-9/CLASS MEASURED SEPARATION VELOCITIES

	CENTERLINE MEAN	CENTERLINE STD	RIGHT SHOULDER MEAN	RIGHT SHOULDER STD
Pressure Vs(ft/sec)	7.0	.7	5.7	.8
Displacement Vs (ft/sec)				
ALL (1)	7.2	.6	4.9	1.5
PARTIAL (2)	-	-	5.9	.9
Ejection Stroke (MS)	67.5	3.6	73.2	2.3

(1) All Missions

(2) Neglects Five Questionable Shoulder Missions

FIGURE 6. TER-9/CLASS SEPARATION VELOCITY

PARAMETER	MEAS VALUE	MISS DIST (FT)
RELEASE TIME DELAY		
CARTRIDGE FIRE TO PISTON 1ST MOTION (MS)	12.9	9.9
PISTON 1ST MOTION TO END OF STROKE (MS)		
CENTERLINE RACK	67.5	51.5
SHOULDER RACK	73.2	56.0
RELEASE TIME DELAY VARIATION-1σ (MS)		
CENTERLINE RACK	3.6	2.7
SHOULDER RACK	2.3	1.7
INITIAL YAW ANGLE (DEG)		
CENTERLINE RACK	4.8	8.8
SHOULDER RACK	9.0	34.0
MASS VARIATION-1σ (LBS)	5.6	2.5
DIAMETER VARIATION-1σ (IN)	0.03	1.4
TRANSVERSE MOMENT OF INERTIA VARIATION-1σ (SLUG-FT^2)	1.0	2.0
SEPARATION VELOCITY VARIATION-1σ (FT/SEC)		
CENTERLINE RACK	0.65	14.0
SHOULDER RACK	0.85	18.0
CENTER OF GRAVITY VARIATION-1σ (IN)	.30	7.0

FIGURE 7. TER-9/CLASS
MK-82 BOMB IMPACT SENSITIVITY TO MEASURED TER-9 RELEASE PARAMETERS

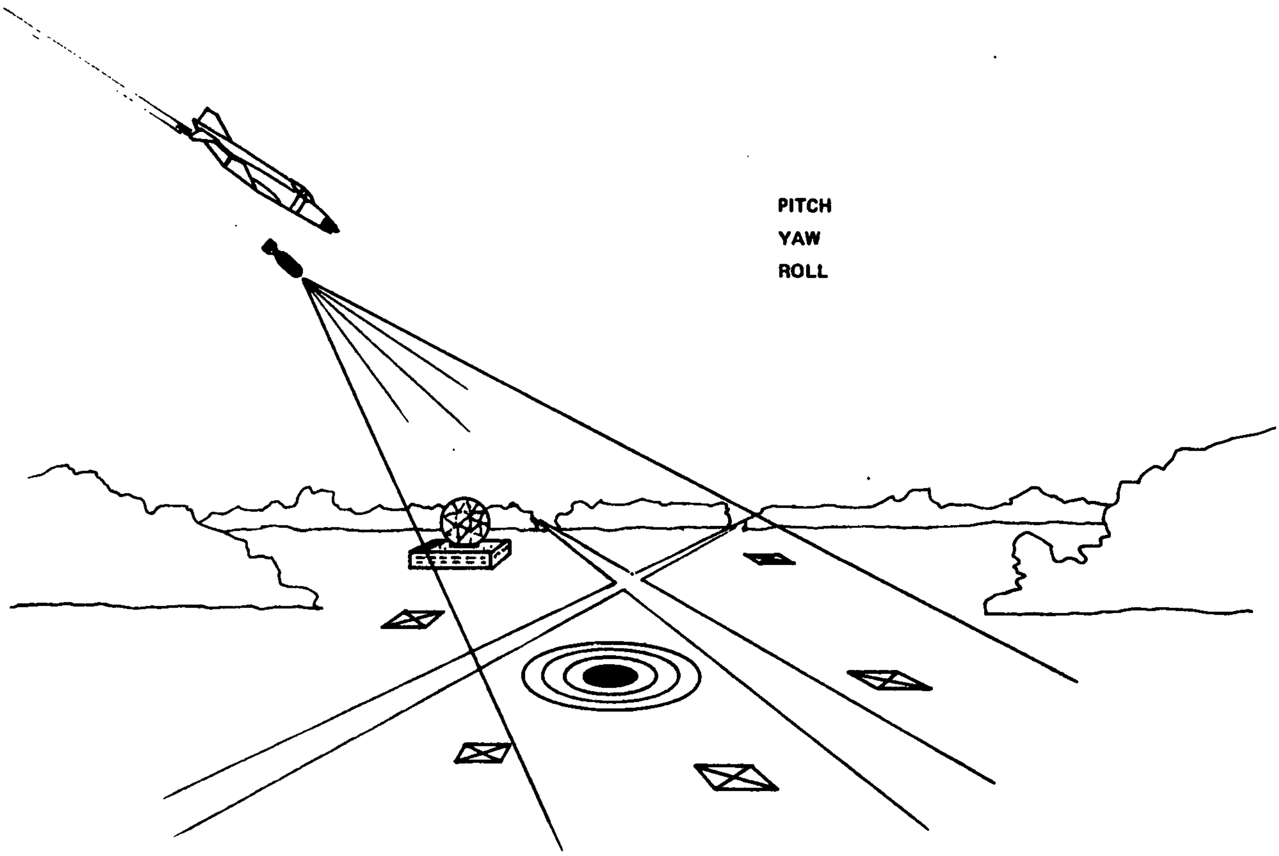

FIGURE 8. BOMB ATTITUDE

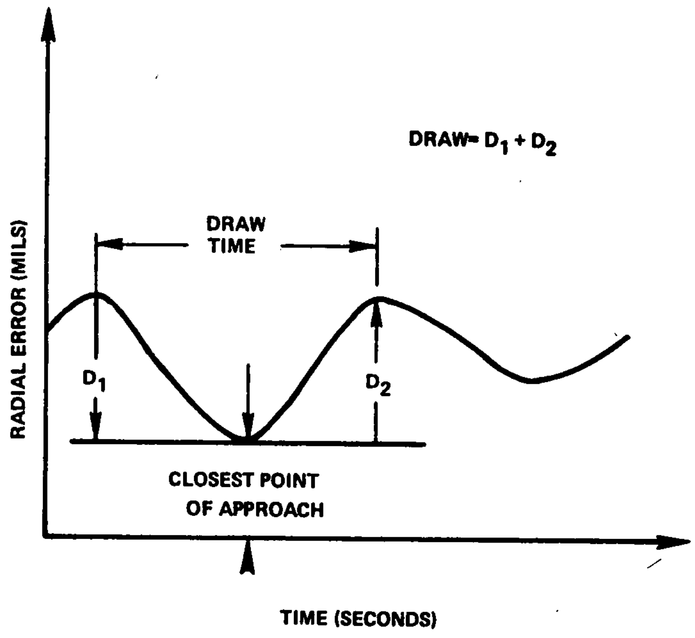

FIGURE 9. SIMULATED SMOOTHED DATA AND DRAW PARAMETERS

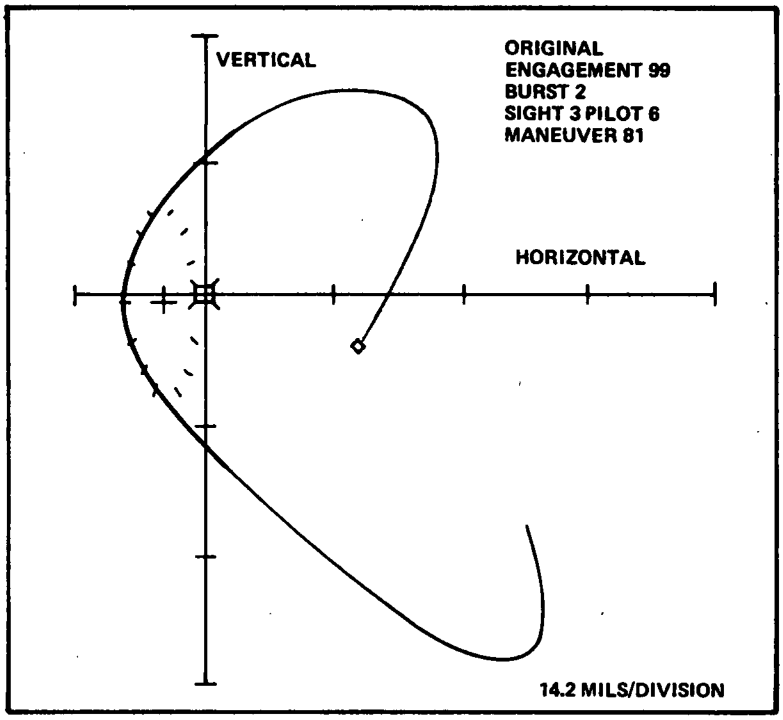

FIGURE 10. FIRELINE PATTERN

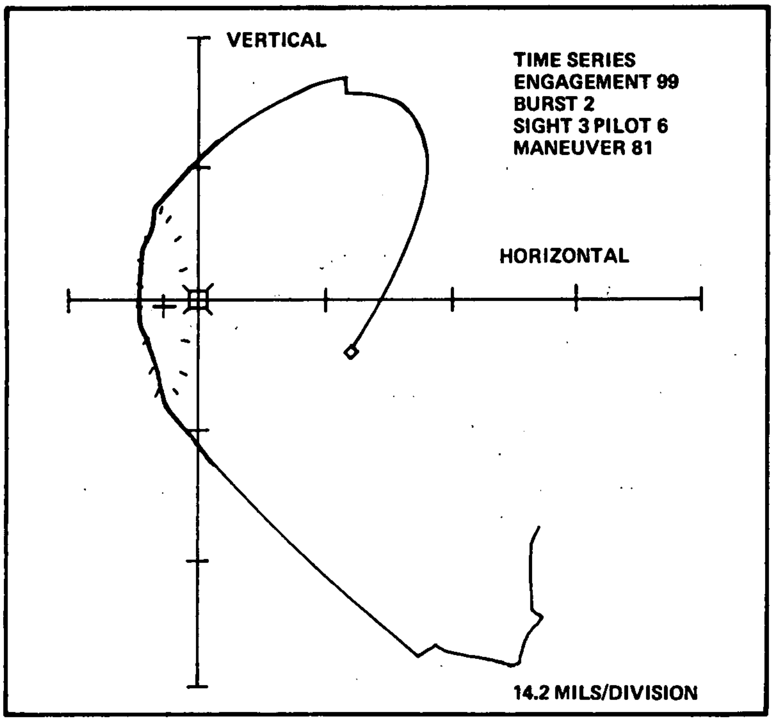

FIGURE 11. FIRELINE PATTERN

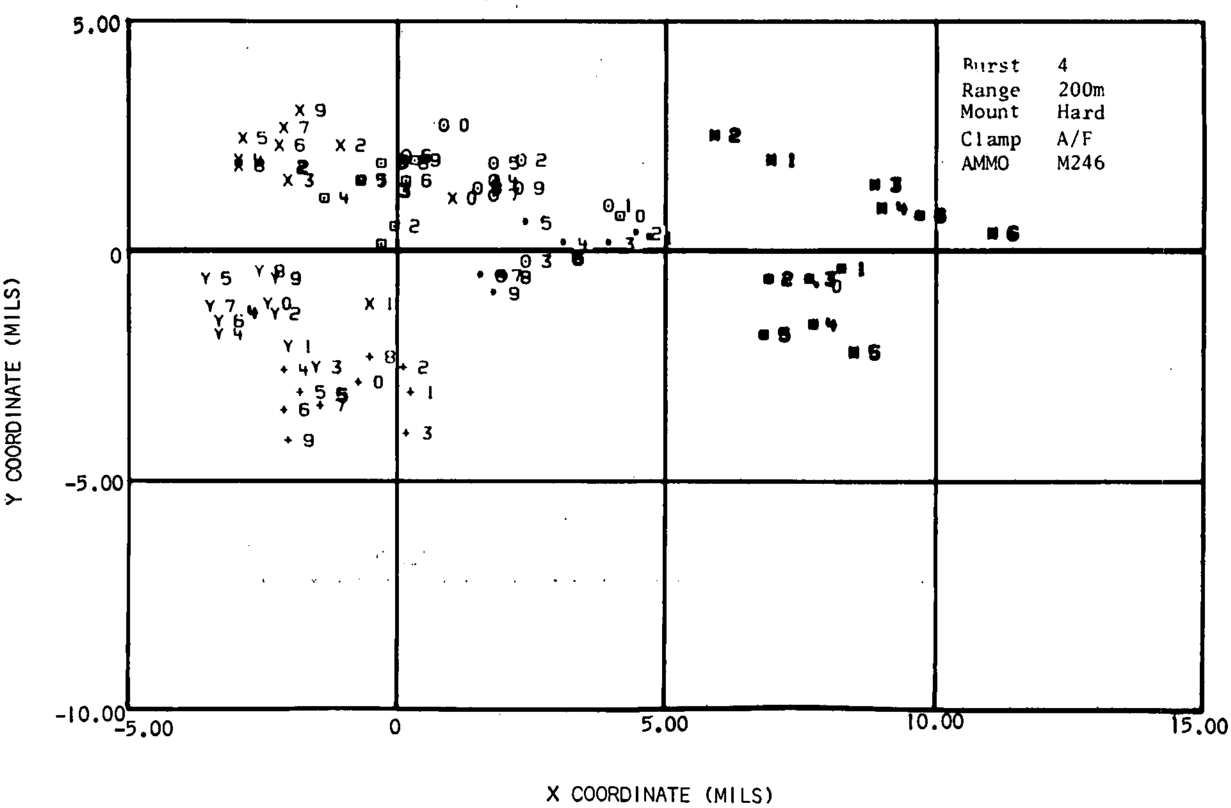

FIGURE 12. IMPACT PATTERN COORDINATES

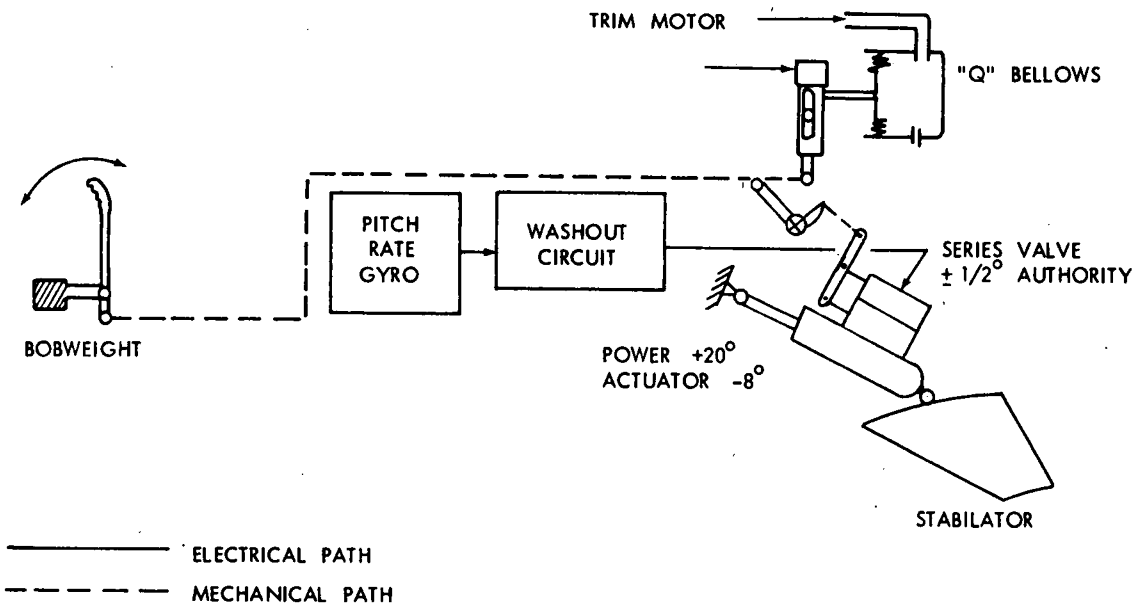

FIGURE 13. F-4 LONGITUDINAL FLIGHT CONTROL SYSTEM

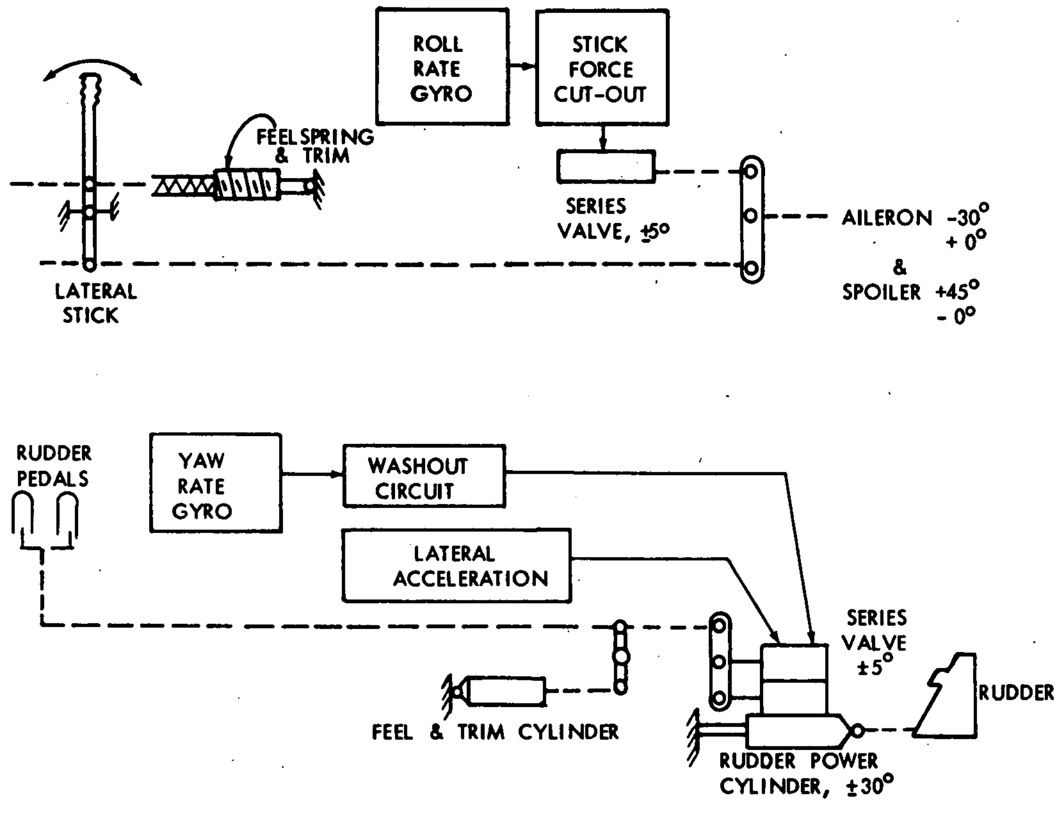

FIGURE 14. F-4 LATERAL-DIRECTIONAL FLIGHT CONTROL SYSTEM

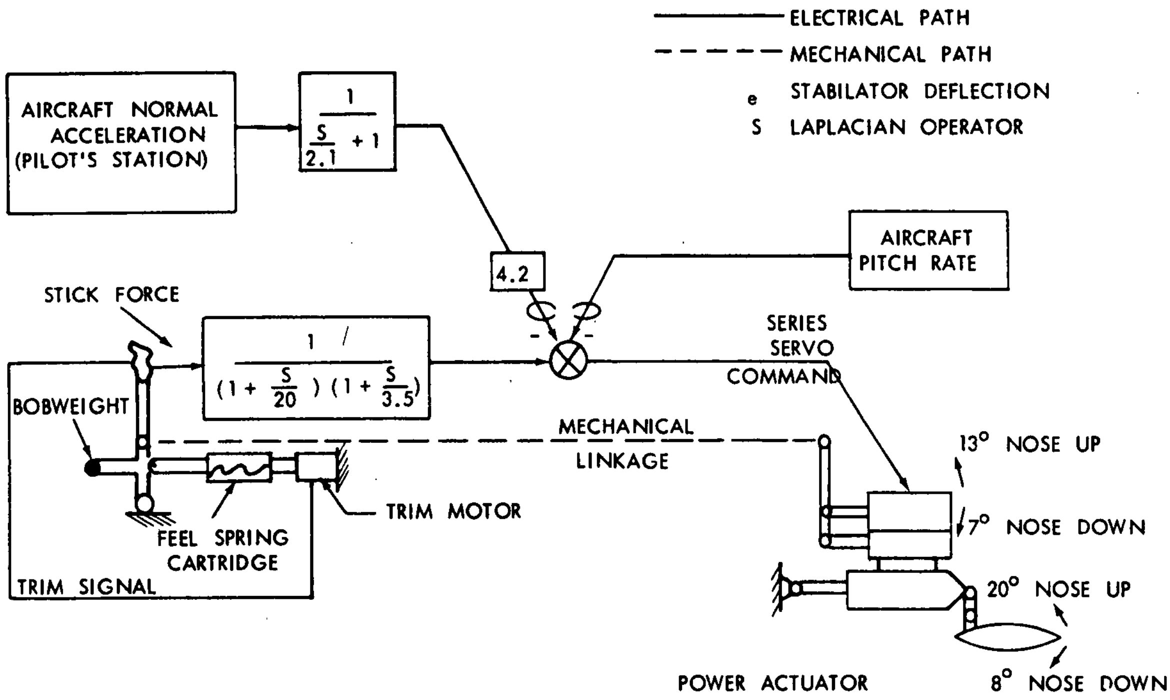

FIGURE 15. F-4 TWEAD LONGITUDINAL FLIGHT CONTROL SYSTEM

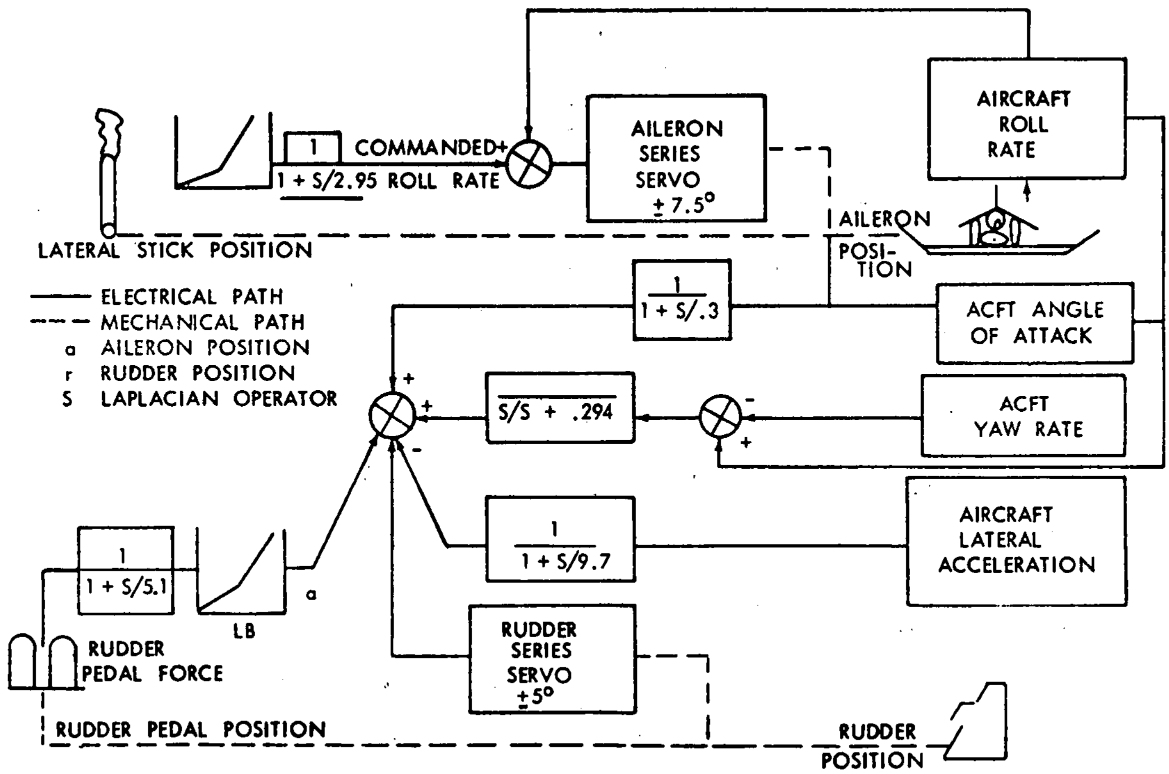

FIGURE 16. TWEAD LATERAL-DIRECTIONAL FLIGHT CONTROL SYSTEM

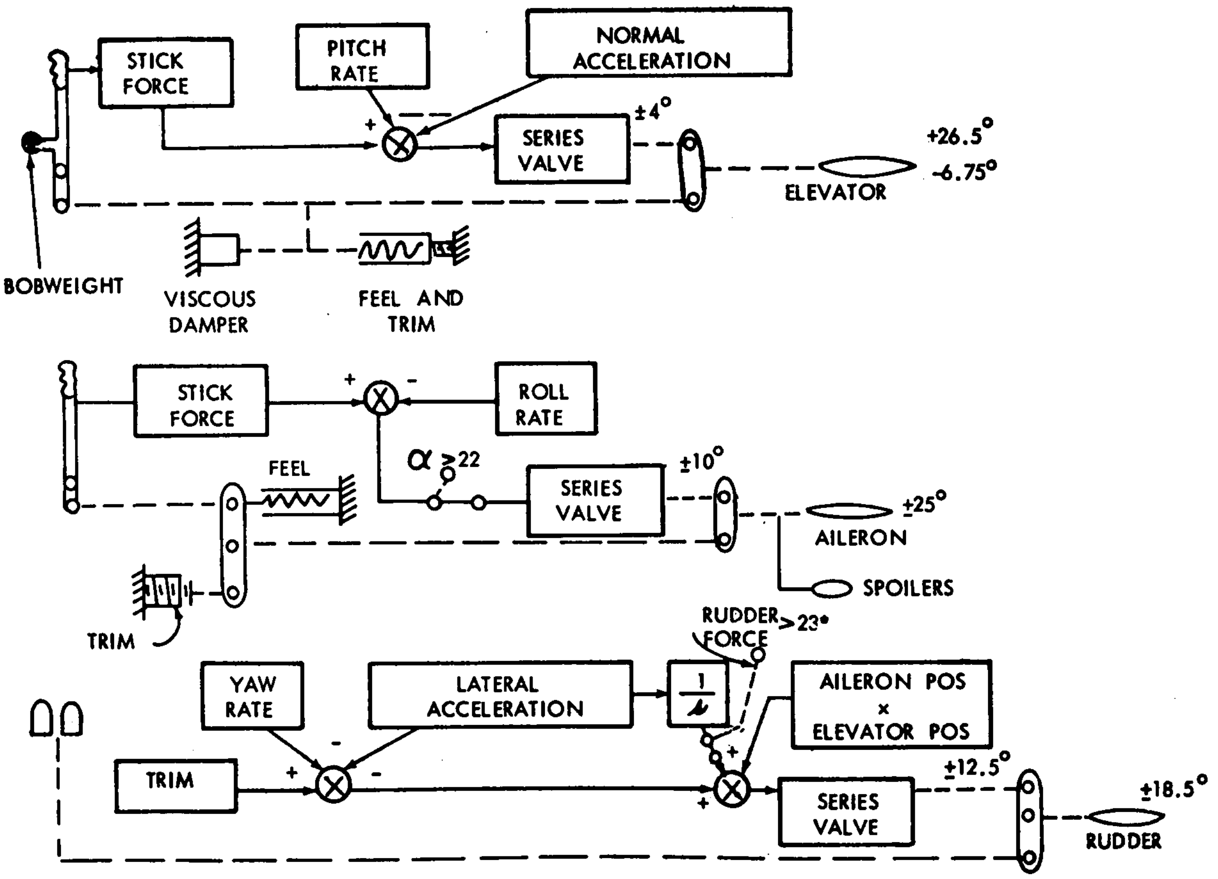

FIGURE 17. A-7D FLIGHT CONTROL SYSTEMS

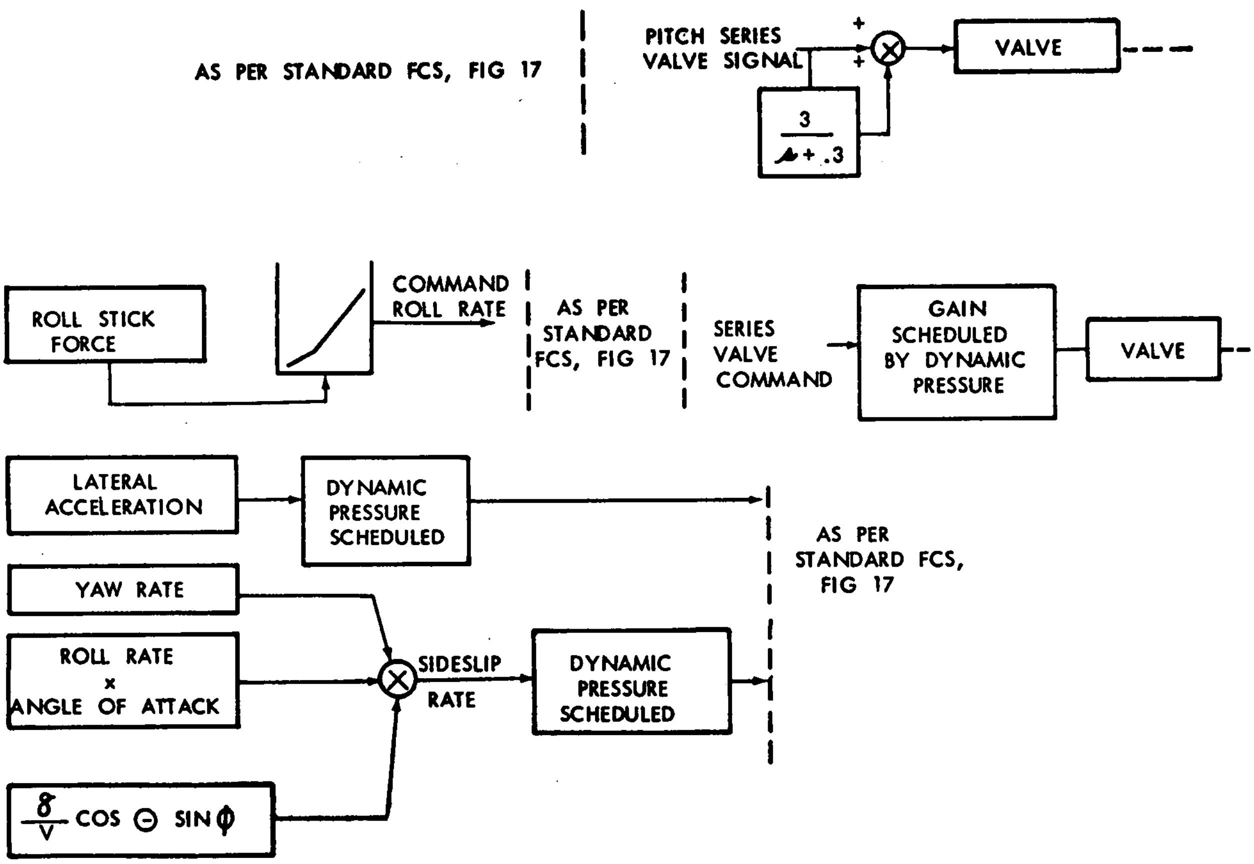

FIGURE 18. MULTIMODE FLIGHT PATH CONTROL DIAGRAM

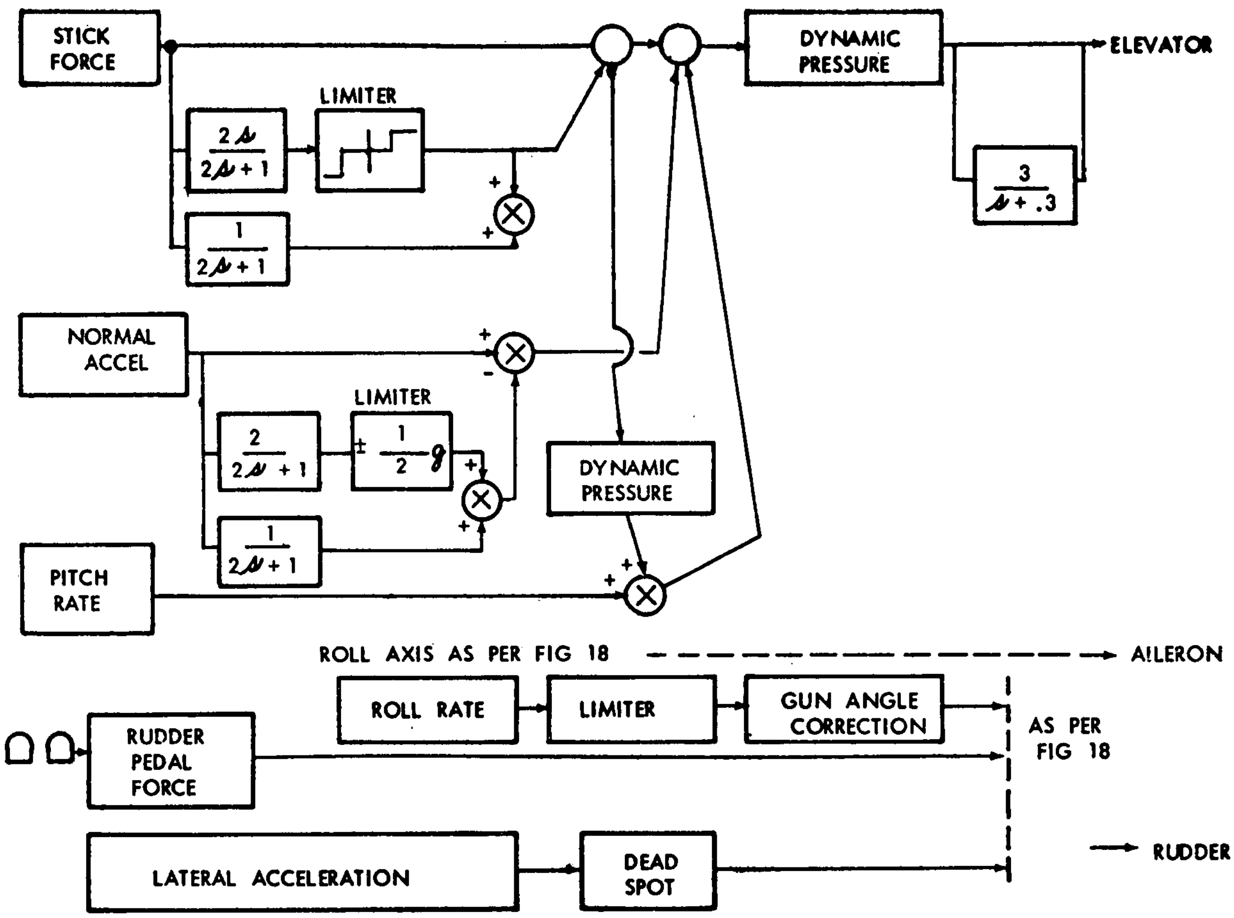

FIGURE 19. MULTIMODE PRECISION ATTITUDE FLIGHT CONTROL DIAGRAM

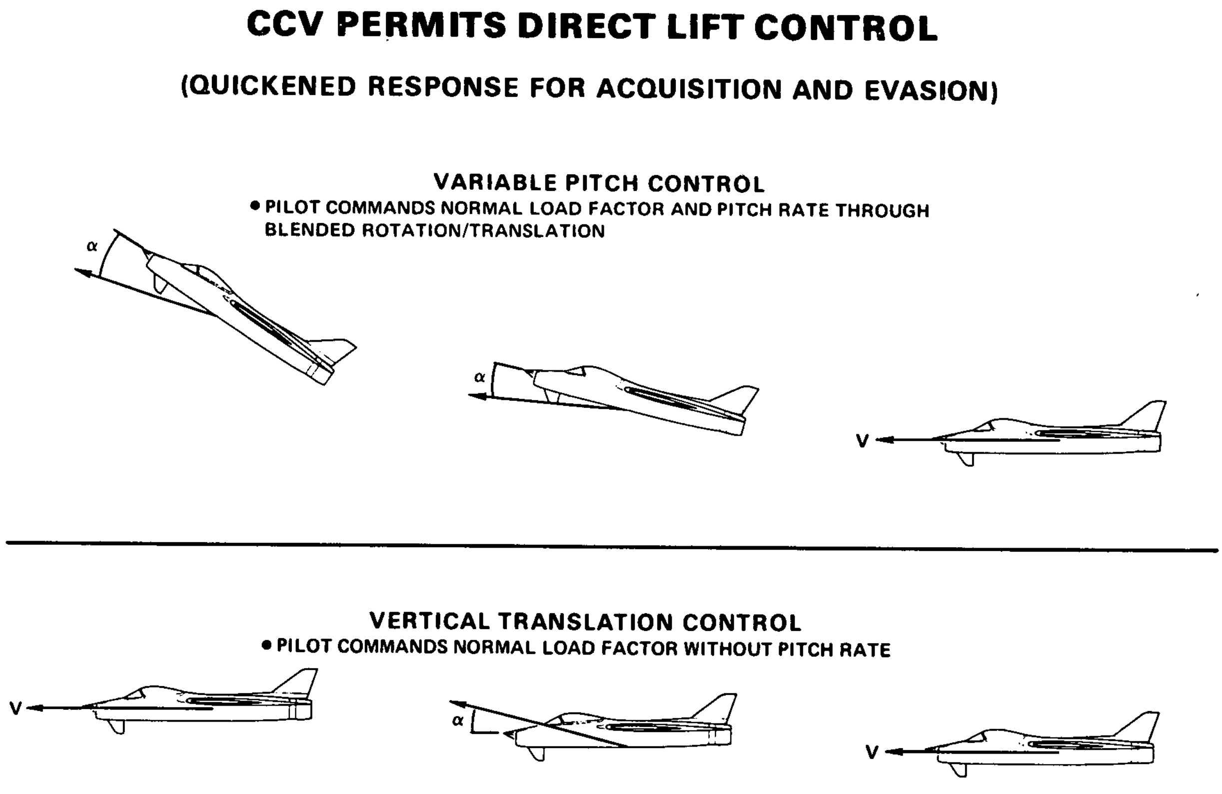

FIGURE 20. CONTROL CONFIGURED VEHICLE MECHANIZATION

CONTROL CONFIGURED VEHICLES B-52 PROGRAM RESULTS

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INTRODUCTION

The term Control Configured Vehicles (CCV) was coined to describe a design philosophy where modern control technology was allowed to impact total vehicle design. This is in contrast to the classic aircraft design approach where flight controls are included late in the design cycle after the design is fixed by aerodynamic,structural and propulsion considerations. The concept is an expression of a long history of growth in Active Control Technology (1). In the United States this growth is exemplified by such programs as XB-70 GASDSAS (2) and LAMS (3). The CCV program is in fact a direct outgrowth of LAMS and uses the LAMS aircraft as one of the test vehicles.

The Control Configured Vehicles Program being conducted at the Air Force Flight Dynamics Laboratory includes both fighter and bomber class aircraft. The fighter program is being conducted using the General Dynamics YF-16 Lightweight Fighter as the test bed. The Fighter CCV Program will demonstrate Direct Lift Control (DLC), Direct Sideforce Control (DSFC) and other advanced control techniques aimed primarily at improving mission effectiveness in both the air-combat and air-to-ground arenas. The Bomber Program on the other hand is aimed at proving concepts which will lead to overall range/payload improvements. These same concepts are also directly applicable to transport aircraft both military and commercial. The specific concepts considered on the CCV B-52 program and the expected benefits are described in the following paragraphs.

DISCUSSION

One classical design constraint is to provide the aircraft with inherent static stability; the reason is to provide adequate flying qualities for the pilot. This requirement is often difficult to meet all flight regimes and may impose severe drag penalties. The CCV design concept allows relaxing the requirement for static stability and provides the required flying qualities through an active control system. Such a system has been termed Augmented Stability (AS) and differs only in degree from a conventional Stability Augmentation System (SAS).

Aircraft structure is usually determined by peak loads encountered during maneuvers; for example, wing root bending moments during a max-g pull-up maneuver may well size the wing box and attaching structure. The bending moment is a function of lift and moment arm; therefore the bending moments can be reduced while maintaining the same lift by shifting the center of lift inboard (Fig 1). The benefits of such a Maneuver Load Control (MLC) system are reduced structural requirements or improved load factor capability for a given structure, or a combination of these to suit the design requirements.

Modern aircraft are usually as light as possible and removing structure because strength requirements are reduced may aggravate the problem of structural flexibility. A light, flexible structure may interact with the aerodynamics and produce structural instabilities or flutter. The classic solution to flutter is to increase the structural stiffness, add mass balancing, or limit aircraft operation to speed placards below the flutter speed. The first two solutions increase the empty weight at the expense of payload or fuel; all solutions degrade performance. The CCV solution is to use an active control system to prevent the flutter mode from occurring within the aircraft envelope, with satisfactory margins of safety.

Another aspect of a large flexible aircraft is the effect of flexible vibration on the crew and/or passengers. The lowest frequency structural modes for large aircraft often fall in the 2 to 7 Hz band at which the human is particularly sensitive. An active Ride Control System (RCS) can reduce the acceleration due to the flexible modes much more effectively than the classic solution of increased stiffness. The benefit of improved ride is improved man-machine effectiveness.

Although not new, the Load Alleviation and Mode Stabilization (LAMS) system is a CCV system. A slightly modified version of the LAMS system on the test vehicle, called a Fatigue Reduction (FR) system to differentiate it from the original, was also tested to demonstrate CCV system compatability. The purpose of the LAMS or FR system is to improve structural fatigue life by reducing structural dynamic loads.

Fundamental to the implementation of the above or any advanced control system is the concept of Fly-by-Wire (FBW). As used in the U.S., FBW means an electrical primary flight control system employing feedback such that the vehicle motion is the controlled parameter (4). This FBW capability was a prime consideration in selecting the LAMS vehicle as the CCV test bed. The left-hand pilot seat is a full FBW system yet the right-hand copilot seat retains the mechanical system for backuup (Fig 2). With this vehicle many of the test systems can be implemented without FBW redundancy (single-thread) and thereby separate the questions of system performance and hardware performance.

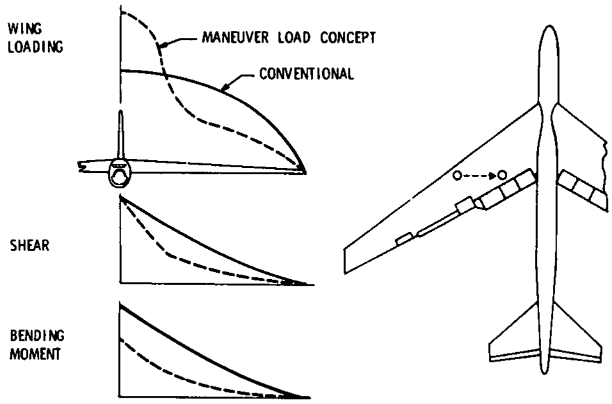


FIGURE 1. Maneuver Load Control Potential Benefit

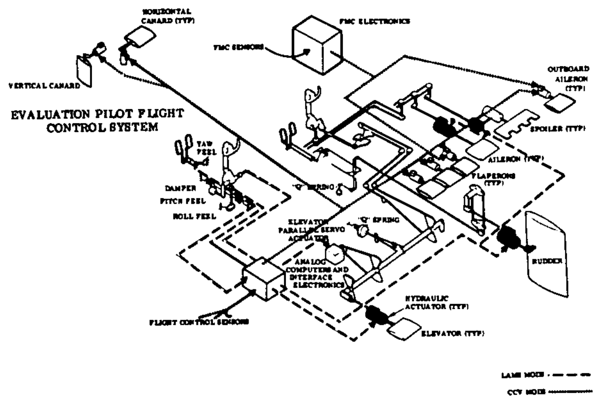


FIGURE 2. CCV Test Vehicle Control System

Equally important in the selection of this test vehicle was the abundance of test instrumentation and the flexibility of changing control laws implemented on the two onboard EAI TR-48 analog computers.

Before describing the CCV system design, ground and flight test programs and flight test results which follow, a schedule would be useful. The CCV Bomber Program was initiated in July 1971 after considerable exploratory research. The aircraft was modified with canards and associated equipment during October thru December 1972 for RCS flight tests in January and February 1973. The aircraft was then further modified until July 1973 when flight tests were conducted to evaluate MLC, AS, and FMC systems, demonstrate the compatibility of these systems operating simultaneously and with the RCS and FR systems, and evaluation of the systems at off-design configurations. The flight test program was completed in January 1974. The total flight test program required 35 flights totaling 122 flight hours. Data reduction and reporting was completed by June 1974, for a total program length of 36 months.

B-52 PROGRAM DETAIL

DESIGN CONDITIONS

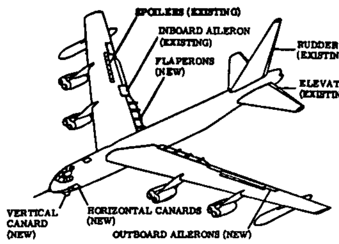
The various CCV systems were designed for a common flight condition except for the AS system which required an aft center of gravity and the RCS which required a low altitude for application. The nominal configuration is as follows:

Gross Weight	270,600 lbs	(122,500kg)
Center of Gravity	29 percent Mean Aerodynamic Chord	
Altitude	21,000 ft	(6400m)
Airspeed	300Kts CAS	(555Km/hr)

The gross weight for AS evaluation was approximately 300,000 lbs (136,000kg) with an aft cg at 42 percent mean aerodynamic chord (MAC), approximately the neutral point for this flight condition. The condition for RCS evaluation was 2000 ft (610 m) altitude at 330 Kts (611 km/hr) calibrated airspeed (CAS). The FMC, RCS and MLC systems were also evaluated at a heavier gross weight, approximately 364,000 lbs (165,000kg), to demonstrate system operation at other than the single design point.

The aircraft configuration includes a number of new control surfaces: one vertical and two horizontal canards mounted on the forward fuselage at the pilot station, three segments of flaperon on each wing replacing the inboard flaps, and a new outboard aileron located just outboard of the outboard flap on each wing. Standard flight control surfaces retained were elevator, rudder, five of seven spoiler segments (each wing) and the original ailerons. Figure 3 locates these surfaces on the aircraft and lists the surfaces used to implement the various concepts.

One further aircraft modification was required to provide the normally flutterfree B-52 with a flutter mode for FMC evaluation. This was done by adverse mass balancing in the wing external fuel tank. The now dry tank has 2000 lb (907 kg) lead ballast in the nose. This induces a mild, flutter condition with an analytically predicted flutter speed (V_f) of 335 Kts (620km/hr) CAS.



	SURFACE	SURFACE REQUIRED PER CONCEPT			
		AS	MLC	RC	FMC
EXISTING	ELEVATOR	X	X		
	RUDDER	X			
NEW	OUTBOARD AILERON		X		X
	FLAPERONS		X		X
	HORIZONTAL CANARDS			X	
	VERTICAL CANARD			X	

FIGURE 3. CCV Flight Control Surfaces

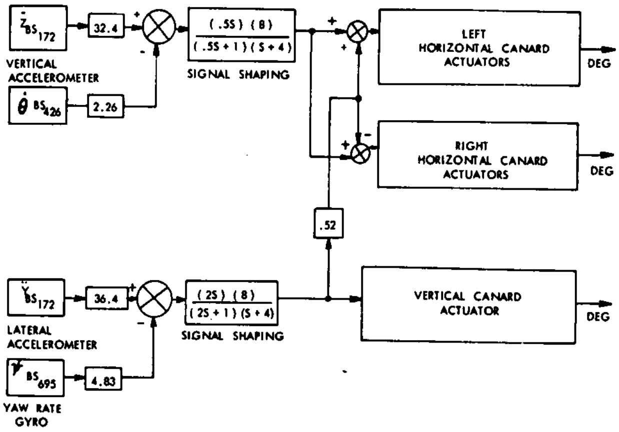


FIGURE 4. Ride Control System Block Diagram

SYSTEM DESIGN CRITERIA

Before discussing the CCV systems actually built and flight tested, the criteria used to design these systems will be described.

RCS Criteria The objective of the Ride Control System was to reduce the accelerations at the pilot's station due to atmospheric turbulence. The goal was to achieve a 30 percent reduction in A, r.m.s. acceleration per unit gust input, in the forward fuselage without degrading $\bar{A}$ elsewhere in the fuselage by more than 5 percent. The RCS was to be a completely automatic system without pilot input and was constrained not to degrade aircraft flying qualities.

MLC Criteria The goal for MLC was to reduce wing root bending moments by ten percent of the design limit during a 1 g incremental load factor pull up maneuver. At the selected design condition this requires a thirty percent reduction in bending moments. Since the system operates principally during maneuvers, it must also be compatible with pilot commanded inputs and provide adequate flying qualities.

AS Criteria The sole criterion for the AS system was to provide adequate flying qualities to the aircraft while flying at the neutral point. The objective was to provide piloting qualities at least as good as the basic aircraft in flight at a normal c.g. configuration.

FMC Criteria The purpose of the FMC system is to prevent flutter onset. This is accomplished by suppressing the bending or torsion motion of the wing or by altering the phasing of the two motions. Two specific goals were to increase the flutter placard by at least 30 percent and conduct flight tests ten knots (18 km/hr) above the unaugmented flutter speed.

In addition to the above requirements, each system was required to meet standard gain and phase margins typical of current day automatic flight control systems. Flight safety was a consideration in all systems and its impact can be seen especially in the FMC system where two systems are actually employed, either one of which can provide adequate performance should the other fail. Flight safety considerations also led to the installation of an aft body fuel dump system allowing the capability to return the c.g. to a more conventional configuration should the AS system fail to operate properly.

RIDE CONTROL SYSTEM

The concept employed in the RCS design was ILAF (Identically Located Accelerometer and Force producer). The ILAF concept, originally proposed for the XB-70 program (2), uses an accelerometer signal passed through a pseudo-integrator (first order lag) to produce a signal proportional to velocity. This signal commands canard deflections to produce a force proportional to velocity properly phased to increase damping.

Finally, to meet flying quality requirements, pitch rate feedback was provided to the vertical ride system, and yaw rate feedback was provided to the lateral ride system and each system was provided with a washout to reduce interference of the RCS to commanded maneuvers by the pilots. The final system block diagram is shown in Figure 4.

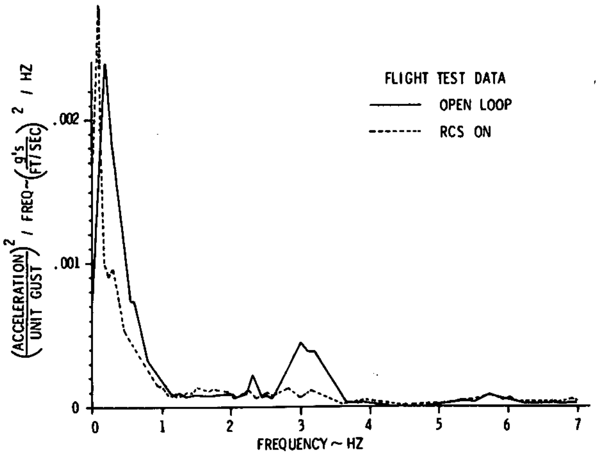


FIGURE 5a. Vertical Acceleration PSD

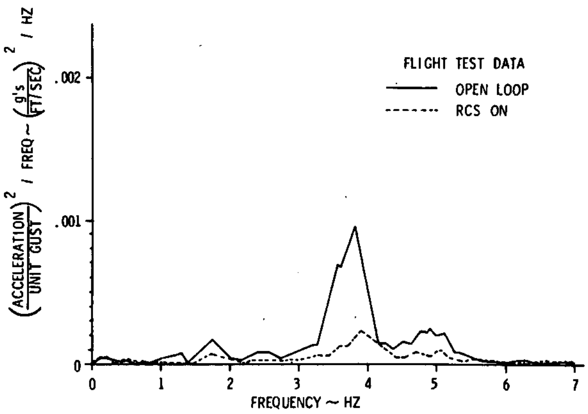


FIGURE 5b. Lateral Acceleration PSD

Performance of the RCS was evaluated at 330 kts (611km/hr) and 500 feet (152 m) above the local terrain peaks. From each run a 300 second continuous sample was selected for statistical analysis. The results are presented in the form of Power Spectral Density (PSD) and r.m.s. acceleration per unit gust input (1 ft/sec, .3 m/sec). Figure 5 shows a typical PSD of acceleration at the pilot station. Note the significant improvement in acceleration power in the range 2 to 5 Hz. This is also the band where man is particularly sensitive to vibration. The computed improvements in r.m.s. acceleration are listed in Table I. The figures represent an an average of several test conditions. The analytically predicted results are also presented for comparison.

TABLE I RIDE CONTROL SYSTEM PERFORMANCE

		Forward	Center	Aft
VERTICAL	Predicted	33%	15%	-2%
	Measured	15%	1%	-12%
	Corrected	18%	5%	-5%
LATERAL	Predicted	42%	5%	1%
	Measured	35%	13%	4%

After data reduction was completed it became apparent the vertical data included a significant contribution from pilot commanded elevator activity; r.m.s. elevator delections increased by three fold from systems off to RCS on. Rather than reaccomplish the tests, an analytic correction was applied to these data based on the measured elevator activity and analytic predicted response to elevator input. These corrected values are included in Table I.

During flight test, two variations of the RCS were also evaluated. The first was a twice nominal gain system; the second used accelerometer feedback only, also at twice nominal gain. In these tests elevator activity was not unusual, therefore a correction was not applied. The r.m.s. acceleration reductions are presented in Table II. Increased RCS gain degraded flying qualities somewhat and acceleration feedback only provided some improvement, as reported by the crew; however formal and extensive flying qualities evaluations were not made.

TABLE II MODIFIED RCS PERFORMANCE

		Forward	Center	Aft
2 x nominal gain	Vertical	25%	9%	-12%
	Lateral	33%	12%	0
Acceleration only, 2 x nominal gain	Vertical	35%	10%	-15%
	Lateral	45%	10%	6%

Note: the improved performance at the pilot station is somewhat offset by degraded performance at the aft fuselage station. The solution is to use more than just the canards for the ride control function. Such a system, termed MSS for Multi Surface System, has been analyzed and could provide total fuselage ride improvement as well as other CCV benefits (5).

MANEUVER LOAD CONTROL SYSTEM

The objective of the MLC system is to redistribute wing loading during maneuvers. The system must respond to pilot commands without adversely effecting flying qualities. The system designed to accomplish this function is depicted in Figure 6. Vertical acceleration sensed near the c.g. is used to command flaperon and outboard aileron deflection to redistribute wing loads and is also fed to the elevator for trim compensation. Pitch rate feedback to the elevator provides increased short period damping. The command augmentation or feed forward loop improves piloting characteristics. As is typical for flight control systems the feed forward loop is sensitive to flight condition and requires parameter adjustment as a function of airspeed and altitude (dynamic pressure).

Flight test results at the most difficult flight condition of 225 Kts (416 km/hr) CAS shows a reduction of peak and steady state bending moments by, respectively 10 and 9.6 percent of design limit load. Figure 7 shows that this is actually a 35 and 37 percent reduction at the test condition (MLC system ON compared to system OFF).

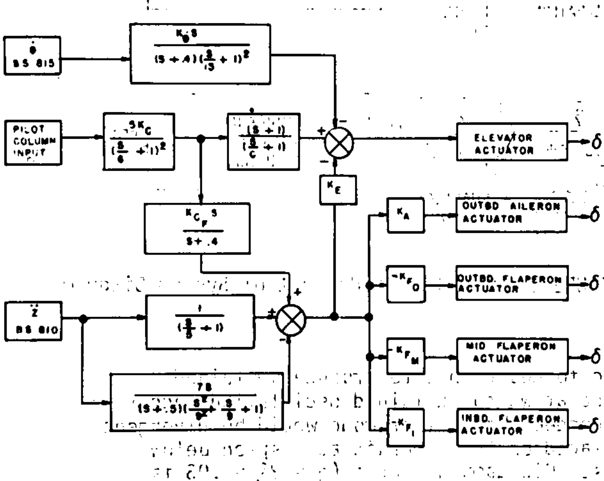


FIGURE 6. MLC Functional Block Diagram

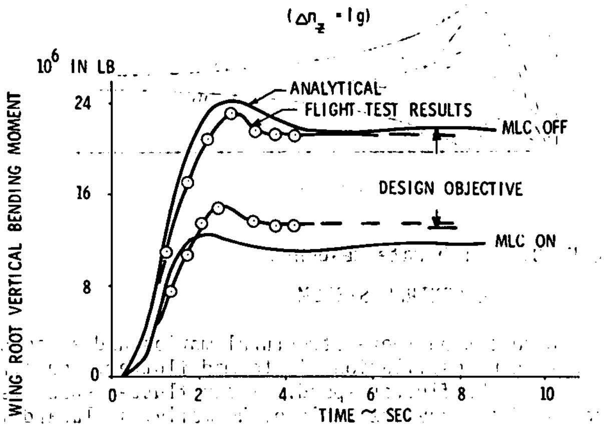


FIGURE 7. Maneuver Load Control System Performance

AUGMENTED STABILITY SYSTEM

The control system designed to provide adequate flying qualities during flight with aft cg locations is presented in Figure 8. The system uses acceleration sensed near the cg fed back with filtering to the elevator. The feed-forward loop from the pilot is common between AS and MLC systems.

Although the function of the AS system is the same as a conventional stability augmentation system (SAS), i.e. provides stability compensation, the AS makes an unacceptable aircraft have acceptable flying qualities whereas a SAS makes a marginal aircraft acceptable. The aircraft was actually flown with an aft cg, near the neutral point with the AS both ON and OFF. The crew reported the aircraft is flyable at the aft cg but requires constant attention to maintain a semblance of straight and level flight. With AS on the flying qualities were as good as a B-52 with a more conventional cg. Both results had been predicted by analysis and ground-based piloted simulation prior to flight test.

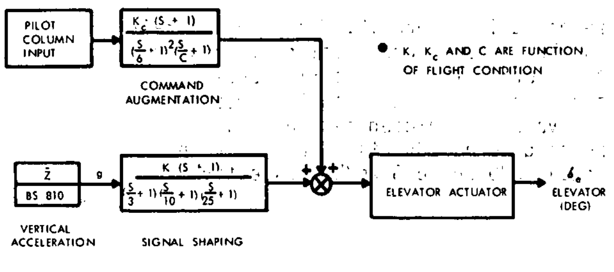


FIGURE 8. Longitudinal AS Block Diagram

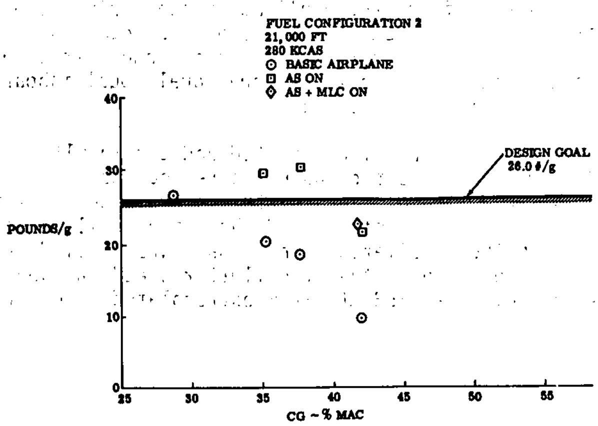


FIGURE 9. Effect of cg and AS on Stick Forces

In addition to the pilot's qualitative evaluation, quantitative measures of flying qualities were also obtained. Two of these are stick forces and pitch rate response. Figure 9 depicts the stick force per incremental "g" as a function of cg position. Note, the stick forces deteriorate rapidly as the cg is moved aft of the conventional B-52 cg limit of 35 percent MAC. The AS system provides adequate stick forces to the aft limit. At the most aft condition of 42 percent MAC, the objective of 26 lbs/g was not quite achieved; however, the crew was sufficiently satisfied with the flying qualitites and no further refinement was considered necessary. This merely underscores the fact that flying qualities are not defined by sharp and distinct boundaries.

Another measure of flying qualities, pitch rate response to a step control column input, is shown in Figure 10. The boundaries have been derived from frequency and damping requirements of the U.S. military specification on flying qualities, MIL-F-8785B. The pitch rate response $\dot{\theta}/\dot{\theta}_{SS}$ to a step input should, ideally, fall near the center of this band. As can be seen, both the analytic and flight test results fall within the limits, this indicates the requirements of the specification may be met.

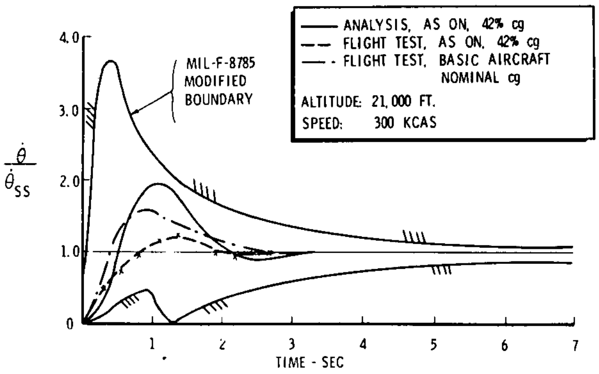


FIGURE 10. Pitch Rate Response

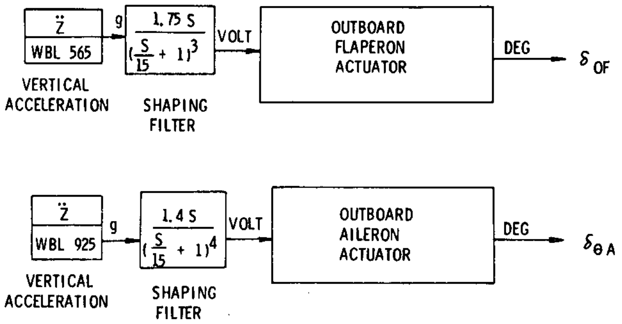


FIGURE 11. Flutter Mode Control System Diagram

FLUTTER MODE CONTROL SYSTEM

A condition where structural motion and aerodynamic forces couple to produce sustained or divergent oscillations is termed flutter. The airspeed at which sustained oscillations occur is termed the flutter speed. If the flutter speed is exceeded the flutter mode would be divergent and could destroy the vehicle. Normally, a placard is placed on the aircraft at a speed below the flutter speed such that the flutter mode has at least .015 damping ratio ($g \approx 2\zeta = .03$ is the term more commonly used by flutter analysts) provided a fifteen percent speed margin is also available.

Using the above criteria, a flutter placard was established from flight test at 307 Kts (568 km/hr) CAS with no CCV systems operating. This is based on an extrapolated flutter speed of 353 Kts (653 km/hr). The objective of the FMC system was to raise the flutter placard by at least 30 percent, to 399 Kts (738 km/hr), and to demonstrate the operation of the system 10 Kts (18 km/hr) above the unaugmented flutter speed, i.e. 363 Kts (672 km/hr) CAS. To do this, the system must clearly have safety designed in.

A simple block diagram of the Flutter Mode Control (FMC) system is given in Figure 11. Note there are two distinct channels. One channel senses vertical acceleration at the wing tip and feeds this filtered signal to the outboard aileron. The second channel senses vertical acceleration near the wing mid-semispan and feeds this signal to the outboard flaperon, also located near mid-semispan. The systems are designed such that either channel can provide adequate flutter suppression capability and satisfy the objectives described above. Thus adequate safety is provided should one channel fail.

Failure detection in FMC is provided by duplicating each channel. One system drives the right wing surface, the duplicate system drives the like surface on the opposite wing. Surface position is monitored by a threshold detector and if the surfaces differ by more than five degrees (.09 rad.) that channel, right and left wing, is disengaged and the crew is notified. Such a system has been termed dual-dual. Dual inboard surface to outboard surface channel and dual right wing to left wing.

The system was flight tested and the results are presented in Figure 12. Measured aircraft mode damping is plotted against airspeed. Note that up to the highest speed tested all systems off (basic airframe), each system is evaluated by itself. In order to assure performance of individual channels of the FMC the outboard flaperon channel gain was increased 25 percent and the outboard aileron channel gain was doubled from the values predicted by analysis. It should also be noted that the flutter speed had been predicted to be 335 Kts in analysis and was determined as 353 Kts from extrapolated flight test data (620 and 653 km/hr respectively).

On 2 August 1973, the aircraft was flown to 363 kts (672 km/hr) with the FMC preventing flutter onset. The system was again flown to 10 Kts CAS above the flutter speed on 8 October 1973 at a gross weight 100,000 lbs (45,360 kg) heavier using the same system. This demonstrates the system can be operateь over a wide range of flight conditions. The system also performed well at 6000 ft altitude (1829m) at both gross weights and at the aft cg, further verifying its operational practicality and compatibility with other systems.

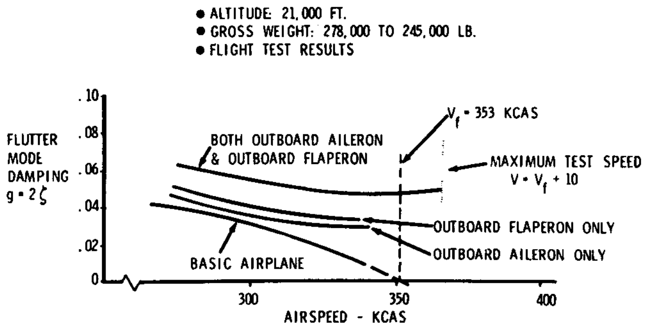


FIGURE 12. Flutter Mode Control System Performance

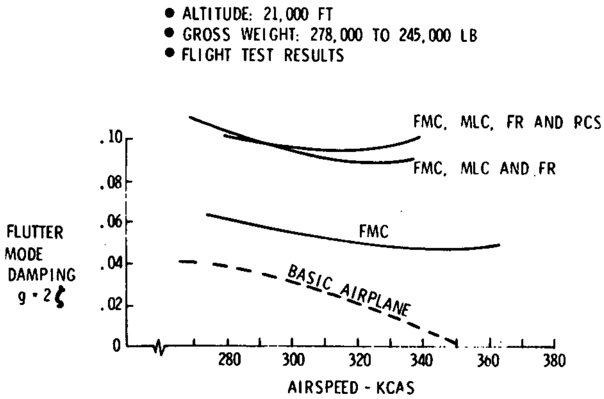


FIGURE 13. System Compatibility for Flutter

SYSTEM COMPATIBILITY

The basic premise of the CCV concept is that advanced flight control systems will be used to solve classic design problems. If this is the case, the various systems must operate in consonance in any combination. To demonstrate compatibility, the systems were evaluated in various combinations in flight tests. Of particular interest is flutter margins when all systems are operating. The results show, see Figure 13, that with all systems on, flutter damping is actually improved over FMC alone. Individual system flutter testing can be generally described as follows: AS and MLC had an insignificant effect on flutter damping, the FR and RCS being principally structural mode controllers add considerable damping to the flutter mode.

Another measure of compatibility is illustrated in Table III where wing root bending moment reduction normalized to a one "g" pull up maneuver, is evaluated with MLC only and MLC operating with other systems. As can be seen, MLC performance is improved with other systems operating.

TABLE III MLC SYSTEM COMPATIBILITY

System (s)	Incremental Bending Moment/Steady State Acceleration*
Basic Aircraft	14.13×10^6 in-lb/g
MLC	6.95×10^6
MLC, FMC, FR	6.75×10^6
MLC, FMC, FR, RCS	5.72×10^6

* Test Condition: 300Kts, 21,000ft, ramp-hold column input

OFF DESIGN CONDITION EVALUATION

Finally, in an effort to determine the operational practicality of these sytems, each was evaluated at an off-design configuration approximately 100,000 lbs. (45,360 kg) heavier and a more forward but conventional cg of 26 percent MAC. The FMC and MLC systems were tested as designed except for the MLC elevator/wing surface interconnect which reflects the new trim condition. The AS system was not evaluated at this condition since the flying qualities are quite adequate.

The Ride Control System evaluated at this flight condition consisted of acceleration channel only. To avoid flying qualities interference a second order wash-out or high pass filter was added in the pitch axis. The time constant for this filter was 1 sec. The principal reason for considering a new RCS configuration at this time was to take advantage of the opportunity to collect data in flight at a condition which also could be tested on an active model in a wind-tunnel test series. The FMC and MLC systems were evaluated on the tunnel model as well. The results of the FMC tests are reported on elsewhere (6). Generally, analysis,tunnel test, and flight test results show good agreement.

The results of FMC evaluation at this off design condition have already been described.

The MLC system actually exceeded the design objective of reducing the bending moment by ten percent of the design limit although at this condition the percentage reduction is less due to the higher, level-flight loads. The measured reductions varied from 11.7 percent of limit at 225 Kts (416 km/hr) CAS to 13.6 percent of limit at 280 Kts (518 km/hr) CAS.

The RCS Evaluation at low altitude indicated the same level of rms acceleration reduction as was obtained at the nominal flight condition using acceleration feedback only. The measured acceleration levels were lower (per unit gust) due to significantly less rigid-body response at this higher gross weight. Thus the percentage acceleration reduction is actually reported higher.

The tests performed at the aft cg limit principally for the purpose of demonstrating compatibility with AS, also serves as off-design condition evaluation. The flutter mode, although lightly damped, is not unstable in the aft cg condition. The FMC still provides additional damping to the mode, increasing its stability. Each system was shown to be compatible with AS at the aft cg configuration.

CONCLUDING REMARKS

The CCV B-52 program has successfully demonstrated the principal objective: CCV systems can be built and will perform to the degree predicted by analysis. In the process of demonstrating the above, a significant first was accomplished:

The aircraft was deliberately flown beyond its unaugmented flutter speed, relying solely on an active control system to provide flutter safety (2 August 1973).

The FMC was again evaluated at speeds above the unaugmented flutter speed on 8 October 1973.

The B-52 aircraft was flown in a statically neutral configuration with an Augmented Stability system providing adequate flying qualities (8 October 1973). The AS system was also flown at the aft cg limit several times in order to complete all the required data collection. Although exceeding the flutter speed appears more dramatic, adequate safety precautions were taken in both cases.

Safety was also a prime consideration in the choice of the B-52 as the test vehicle. Because of its long service this aircraft is particularly well known - including a known mild flutter mode. The dual controls allowed a conventional control system as backup to the test fly-by-wire system. The long endurance allowed step-by-step tests which gradually approached the critical conditions on one test flight.

Although the aircraft is well known, math models for the analysis were generated using state-of-the-art aeroelastic modeling techniques. These were checked with flight test data from prior tests just as wind tunnel data would be used to verify analysis of a new aircraft design. As a matter of fact, wind tunnel tests were performed to validate aerodynamic characteristics of the flaperons and outboard aileron. These tests provided corrections for downwash effects of the flaperons on the horizontal stabilizer, a notoriously difficult affect to predict analytically. A completely new vehicle design could be expected to do as well as the results achieved here; in fact a new vehicle could more fully capitalize on the performance benefits of CCV through incorporation during preliminary design.

REFERENCES

1. Holloway, R.B., "Introduction of CCV Technology into Airplane Design," AGARD Flight Mechanics Panel Symposium, Florence, Italy, 1-4 Oct. 1973.

2. Wykes, J.H. and R.L. Knight, "Progress Report on a Gust Alleviation and Structural Dynamics Stability Augmentation System (GASDSAS) Design Study," AIAA paper 66-999, Boston, Mass., 1966.

3. Johannes, R.P., P.M. Burris, and J.B. Dempster, "Flight Testing Structural Performance of the LAMS Flight Control System," AIAA paper 68-244, Los Angeles, CA., 1968.

4. Sutherland, J.P., "Introduction to Fly-by-Wire," Proceedings of the Fly-by-Wire Flight Control System Conference, 16-17 Dec 68, AFFDL TR-69-58, 1969.

5. Poyneer, R.D., "Multi-surface System for the CCV B-52", AIAA paper 74-126, Washington, D.C., 1974.

6. Redd, L.T., J. Gilman, Jr., D.E. Cooley, and F.D. Sevart, "A Wind Tunnel Investigation of a B-52 Model Flutter Suppression System," AIAA paper 74-401, Las Vegas, Nev., 1974.

A QUADRUREDUNDANT DIGITAL FLIGHT CONTROL SYSTEM FOR CCV APPLICATION

by

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ABSTRACT

This paper describes a parallel redundant digital fly-by-wire system. It will be tested in the near future on a CCV-test aircraft (modified F-104 G). Starting from a fail-op, fail-op requirement the reasons for the choice of a digital system are outlined. The system works with freely programmable identical airborne computers which run identical software. The computers perform the control laws and act also as central voters and monitors. Basic of the design is the principle of majority decision with elimination of a failed component. Finally the Quadruplex system represents a functional integration of Autopilot, Stabilisation, Air Data Computation and Built-In-Test-Equipment (BITE).

1. INTRODUCTION

The main objectives of the CCV-Aircraft approach are

- Improvement of performance
- Reduction of weight
- Increase of service life
- Improvement of guidance

Especially the first item leads to an aircraft configuration, which is basically unstable in the longitudinal motion and almost unstable in the lateral motion. The so designed aircraft is no longer manually flyable and a fly-by-wire control system becomes essential. On the other hand, regarding new Non-CCV-Aircraft we already find redundant control systems and the real question is not to say:

"We are going CCV, let's make a fly-by-wire control system for it"

but

"We are already using fly-by-wire, so let's make use of its full potential for the CCV-application".

In fact, there is no major difference between both ways from the control systems point of view except the following:
In a Non-CCV-Aircraft we normally have a mechanical back-up system, of which we hope never to be forced to make use. In a CCV-aircraft a mechanical back-up is not applicable, so the approach for the fly-by-wire control system here is to

- o demonstrate the reliability of fly-by-wire
- o fulfill the operational requirements

The following gives an overview, how MBB is going this way with the two additional main objectives:

- demonstrating CCV-technologie
- demonstrating a new control system approach

The described Quadruplexsystem will be used for the control of a CCV-Experimental Aircraft within the next few years.

2. OPERATIONAL REQUIREMENTS

2.1 Requirements

The operational requirements can be devided into two main requirements. One is a reliability number and the other is a technical requirement. Both lead to a technical solution, which presently is named: Redundancy. The described design is based on the following operational technical requirements:

- Continuation of mission without loss in performance of the weapon system "aircraft" after any first failure in the control system.
- Safe return to base after any further critical failure.

2.2 Possibilities

Regardless of the technology decision, which has to follow, the operational requirements can be fulfilled with

a) a three channel system with a failure self detection capability in each channel
b) a four channel system with majority decision and switching off (isolation) of the failed component or channel
c) a five channel system with continous majority decision between all five channels.

One word about the reliability number.
It is in fact an important requirement. With present electronic parts and components it can only be achieved in conjunction with redundancy. The following assertions are however addressed to the technical part of fulfilling the operational requirements by the use of a redundant control system.

3. TECHNOLOGY DECISION (ANALOG OR DIGITAL)

A few years ago when we started our investigations on digital control, this would have occupied the main part of the paper. Presently, the reliability is significantly increased. Weight, size and cost of digital airborne computers have gone down. That helps arguing.

We decided for a DIGITAL CONTROL SYSTEM because of the following arguments:

- Capability of the best possible adaption to the changes in the aircraft derivatives due to altitude, velocity, wing sweep angle, configuration, control modes etc.
- Simple realisation of nonlinearities
- Best possible implementation of new control theorie
- No drift
- Simple BITE-implementation
- Achievement of a higher degree of integration
- Lower cost for changes during the development phase
- Definition of hardware possible before software is finally defined

In fact are there some other characteristics which we expect to achieve, such as higher reliability and lower cost than former systems. But these are not used here as arguments. We will come back to this matter in the outlook at the end of this paper.

4. THE QUADRUPLEX CONCEPT

4.1 General features

The Quadruplex System is a parallel redundant control system which uses four freely programmable computers for the central data processing. Sensors and actuators are still analog. Nevertheless we call this a digital control system.

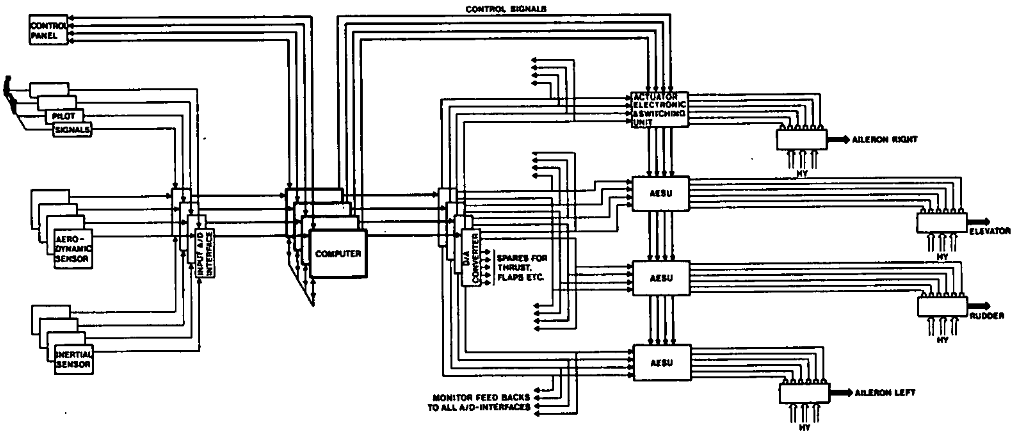


FIG. 1 SYSTEM OVERVIEW

The main features of this design are: It

- uses the principle of parallel redundancy with majority decisions
- works with available freely programmable airborne computers, which are identical and freely exchangable
- runs identical software in each computer
- uses the computers as central voters and monitors

The reason for this design are: It

- satisfies the operational requirements
- minimises the risk
- is cost effective
- allowes the transition to new failure detection and determination methods
- was available within a moderate time
- did not require major hardware developments

4.2 Functional description

4.2.1 Integration concept

The Quadruplex System integrates the functions of

- Stabilisation
- Autopilot
- BITE (Build in test equipment) and
- Air Data Computation

This is established by using Multi Variable (optimal) Control Methods which require that the state vector is measured, i. e. those variables which one gets as result of adding the variables normally needed for the CSAS and those for the autopilot. No further explanation is given here, because separate information about the used control method is available elsewhere. Obviously the capability of the computers forces one to integrate the BITE functions into this system.

4.2.2 Sensors

The experimental Quadruplex System has all sensors quadrupally redundant. A final system will discriminate between

4.2.2.1 Sensors Class I

These are such sensors, which are essentially necessary to stabilise the aircraft (i.e. α, q, β, p, r)

4.2.2.2 Sensors Class II

Which are necessary to fulfill the mission but not to stabilise the aircraft.

The redundancy degree of class I sensors will still be four, that of the class II sensors only three (or two if methods are available to detect the failed sensor)

4.2.2.3 Sensors Class S

are defined to be failure self detecting. They may replace some in both classes in the appropriate number. Presently there are no sensors of this type in sight for a reasonable effort.

4.2.2.4 Measured variables

The following variables will be measured or determined:

α	- Angle of attack
β	- Side slip angle
pt	- Total pressure
ps	- Static pressure
T	- Temperature
v	- Velocity
p	- Roll rate
q	- Pitch rate
r	- Yaw rate

a_{x_f} - X-Acceleration

a_{y_f} - Y-Acceleration

a_{z_f} - Z-Acceleration

ϕ - Roll Attitude

Θ - Pitch Attitude

Ψ - Heading

Additionally the pilot signals will be electrically picked off and the actuators have their internal position pick-offs. All signals will be made available to the computers in $(\pm)$ 10 V D.C.

In the experimental phase there are four sets of sensors. Each set is dedicated to one computer.

4.2.3 Data processing

The data processing except for certain interrupts and the BITE functions, is repeated every sampling period. The sampling period is not finally determined. It will be between 30 ms and 60 ms.

4.2.3.1 Interface control and data transmission

The analog signals are converted under program control via the A/D-converter and stored in dedicated memory locations.

4.2.3.2 Signal consolidation

Assuming an errorfree operation of the sensors, there nevertheless will be small differences in the signals because of the A/D-conversion, which cannot be avoided. This explains, together with the failure localisation possibility which would be lost, why a signal consolidation in front of the interface is useless. In order to detect and eliminate failures and consolidate the small differences in the errorfree signals, a communication between all four computers is necessary.

Each computer sends to each other the set of signals which it has received from its A/D-interface. After this operation, each computer has four sets of data available in its memory. Each computer now generates the average of those signals which are found to be free of error. After this consolidation, each computer possesses an identical set of data with which it can start the control law computations etc.

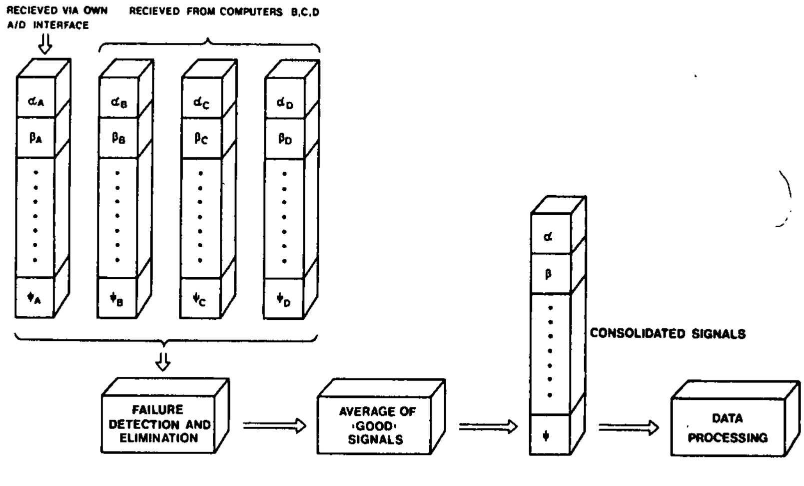


FIG. 2 PRINCIPLE OF SIGNAL-CONSOLIDATION

4.2.3.3 Direct Memory Access

The intercommunication between the four computers is mechanised as Direct Memory Access (DMA). DMA provides a high speed digital data transmission. DMA start address and block size determination occur under software control. All four computers can send and receive quasi-simmultaneously.

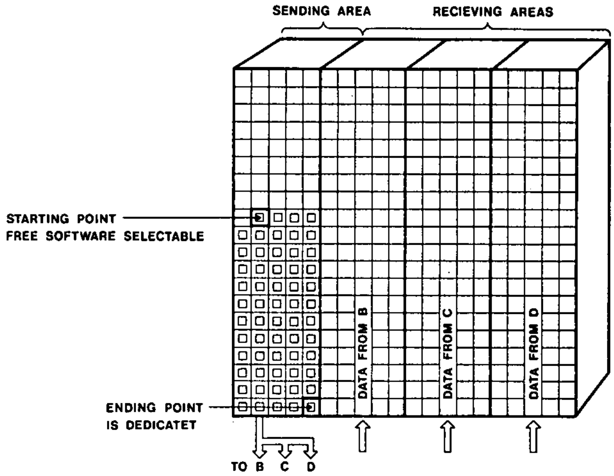


FIG 3. PRINCIPLE OF DMA-DATA-EXCHANGE

4.2.3.4 Failure detection, localisation and elimination

The task of failure detection, localisation and elimination has been dedicated to the computers as the central voters and monitors. Excluded are only the actuators which have their own failure detection.

4.2.4.4.1 Sensor failures

Since, via the signal consolidation (4.2.3.2) each computer is in possession of the signals of all four sets of sensors, each of the computers can detect, allocate and eliminate (by not using) a failed sensor. A/D-converter failures are detected and handled in the same manner.

4.2.3.4.2 Computer failure

The communication via DMA (4.2.3.3) allows one to exchange the computational results and compare them. Since they should be identical because of the signal consolidation (4.2.3.2) a detection possibility is given. Upon discrepancy no measure will be undertaken (except for a failure indication to the pilot) as long as the comparison of the D/A-outputs shows an agreement (4.2.3.4.4) because the reason for a discrepancy can be a

4.2.3.4.3 DMA failure

It is detected like a computer failure. As long as the D/A-check shows no failure, only a failure signal is given to the pilot.

4.2.3.4.4 D/A-Discrepancy

All four computers should output identical signals via their D/A-converters in the appropriate channel, if working properly. The D/A- (actuator input) - signals are feed back and cross strapped to the A/D-inputs of each computer, so that each computer via its A/D-converter knows the D/A-signals of itself and the three others. Independently each computer forms its "opinion" about each signal and sends its "opinions" via the digital parallel output interface to the Actuator Electronic and Switching Unit, where a failed signal will be switched off.

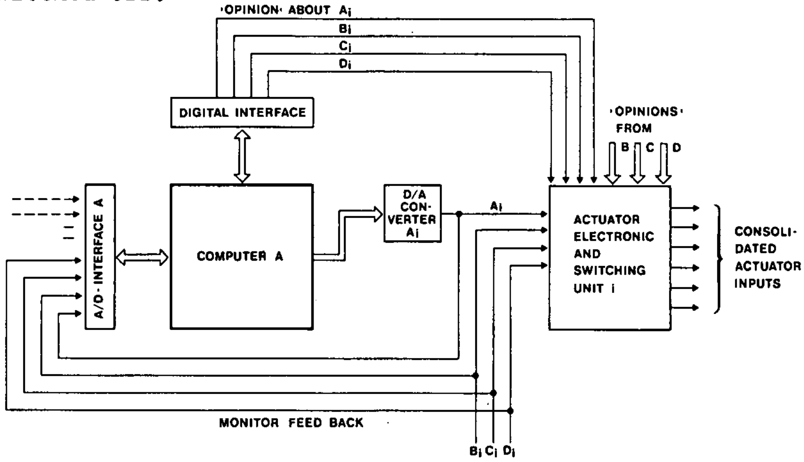


FIG. 4 MONITORING OF ANALOG OUTPUT SIGNALS

4.2.4 System constraints

Guide line for the design is the operational requirement (2.1). But there are some constraints given by the use of an existing aircraft as test vehicle, such as number of engines, existing power systems etc.
The design for the Quadruplex system has been made so, that these constraints do not effect the design itself but considerations have been spent on the electrical and hydraulical side of the aircraft.

4.2.4.1 Actuators

Series actuators will be used in conjunction with the existing power actuators. For the series actuators the operational requirement of surviving any two failures (electrically, hydraulically or combinations) can be observed by using a Trio-Duplex-Actuator Concept. The operational requirement will be violated (only in the test vehicle) with the

4.2.4.2 Hydraulic system

Only two systems are installed in the test vehicle. No additional installations are planed.

4.2.4.3 Electric power supply

The existing power sources are used in order to achieve four seperate 28 V DC power sources. These are battery buffered in order to guarantee at least 5 min. operation after a failure in the A/C-basic supply. This time is sufficient for the pilot to switch over to the

4.2.5 Safety system

Due to the system constraints of the given experimental aircraft a safety system has to be installed. It allows one to change the location of the aircraft center of gravity by dropping ballast. This brings the natural stability back to a manually flyable value. Simultaneously the control is switched over to the original control system. Thus the pilot can fly the "original" test vehicle. It should be noted that having a mechanical back-up is not our general intention but has to be resorted to due to the limitations in the redundancy of the hydraulic system.

5. SELECTED REALISATION PROBLEMS

5.1 Test vehicle

The Quadruplexsystem will be tested using an F 104 G Starfighter. The decision for this aircraft was due to suitability and availability. The following picture gives an impression about the available space for installation of the components of the Quadruplex system.

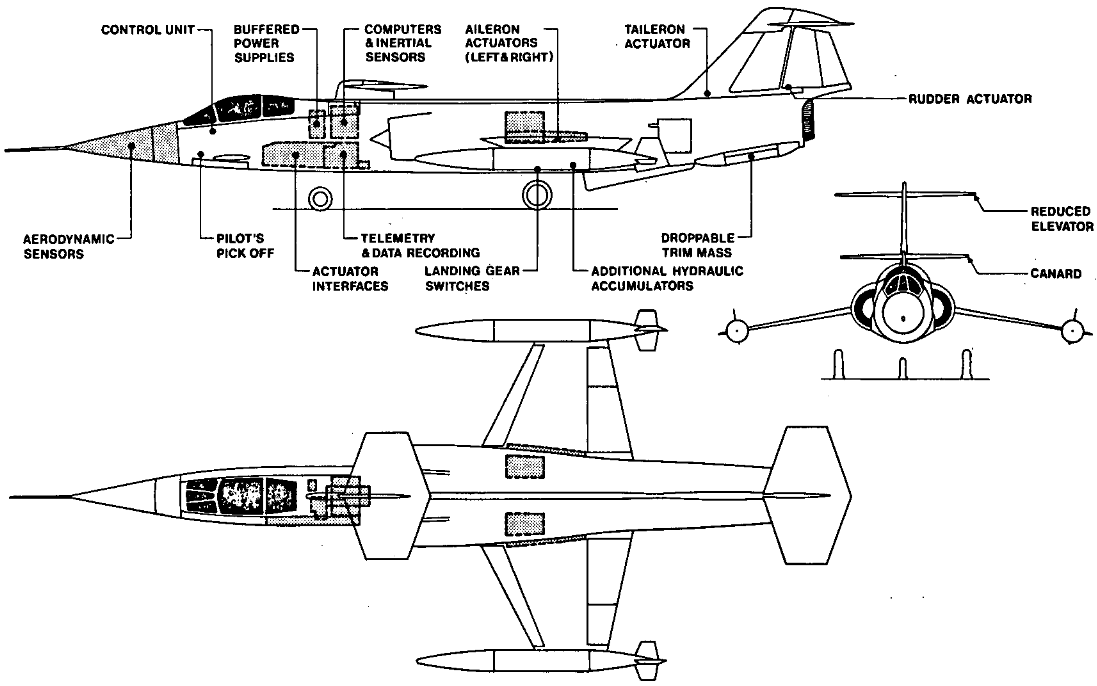


FIG. 5 **TEST AIRCRAFT F104G CCV-MODIFICATIONS**

5.2 Computers

Four Teledyne TDY-43 computers are used for the realisation of the Quadruplex system. Each of them is identical to the other and has the following

5.2.1 Configuration

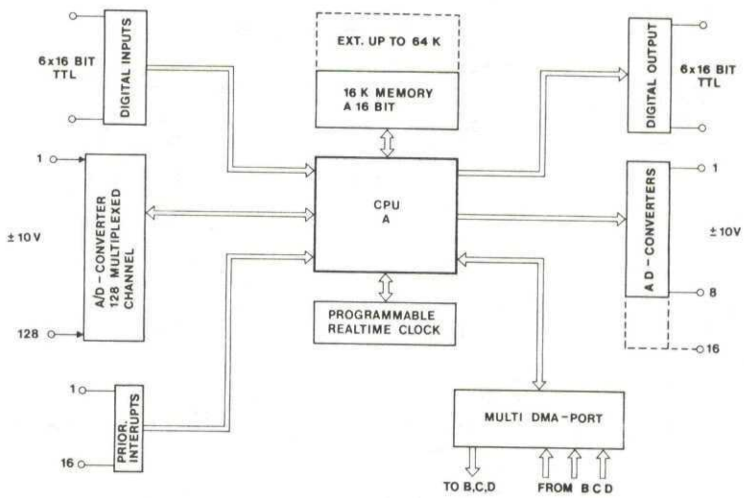


FIG. 6 COMPUTER CONFIGURATION

In order to give an impression about its performance a few computing times are listed below.

Cycle time:	1.33					
Add, Subt.						
or Load:	2.67	(Common)	1.67	(Immediate)	4.0	(Indirect)
Multiply:	6.0	"	4.33	"	8.67	"
Divide:	8.67	"	5.42	"	12.67	"
Time in μs						

The control laws used require a high number of multiplications, so the availability of a fast multiplication was of great importance.

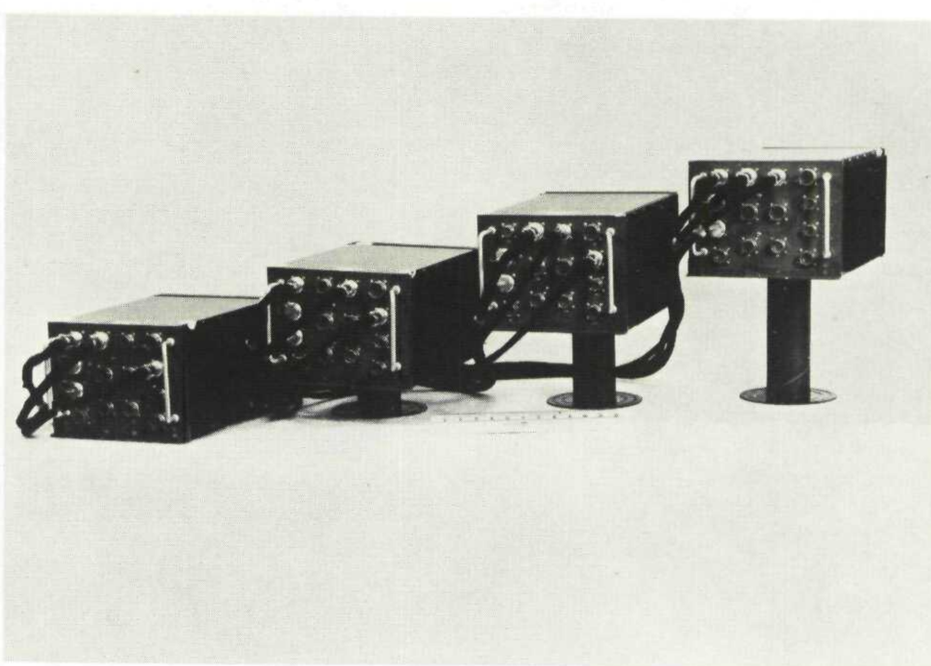
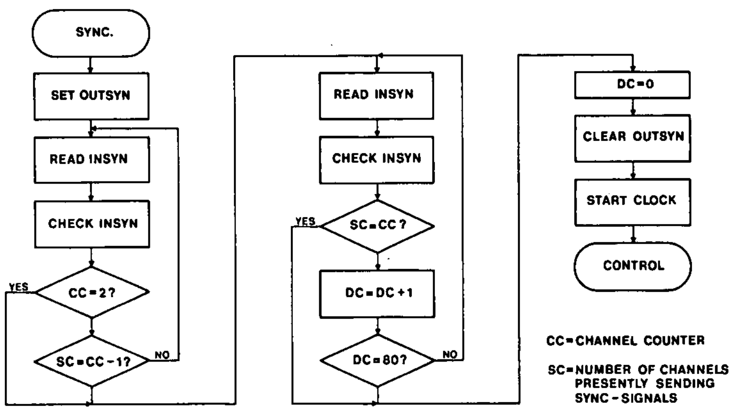


FIG. 7 COMPUTERS

5.2.2 Synchronisation

All four computers have their own precision time reference with an accuracy of 10^{-5}. However the need for a synchronisation, - may be only for the initialisation - shall be discussed. Synchronisation is needed of one integrates, uses non-linearities or wants to avoid problems which will occur because of nonsynchronous data sampling. Our solution for synchronisation of the four computers is shown in the following flow chart.

FIG. 8 COMPUTER SYNCHRONISATION



On the other hand, it generates additional effort and problems. Therefore an alternative solution is being discussed. The basic idea is to prove, that one can live with a maximum asynchronism of one sampling period (30-60 ms) with regard to signal consolidation (4.2.3.2) and failure detection (4.2.3.4). If this is true, the contents of integrators and nonlinear factors may become exchanged via DMA (4.2.3.3) and handled like common variables to be consolidated. The result is that all computers work with identical data and a maximum shift of one sampling period.

5.3 Actuator Electronics and Switching Unit

It has been shown that, if a failure occurs in the sensors, input-interfaces and DMA's these will be discovered and blocked by the computers. Failures in the computers and D/A-converters will also be discovered, but for blocking them an additional unit is provided for each actuator, the AESU. It has three functions:

5.3.1 The main logic

The task of the main logic is to form a "majority opinion" about each outgoing analog signal type of all computers. Basic of this decisions is, that all outgoing D/A-converted signals are fed back to all A/D-Interfaces (4.2.3.4.4). (Cross strapped). Every computer therefore is in possession of each outputted signal and can independently form an "opinion" about each signal. The "opinions" of all four computers form in the main logic of each actuator majority "opinions" on wether a particualr signal is bad or not. The failed signal will then be switched off in the

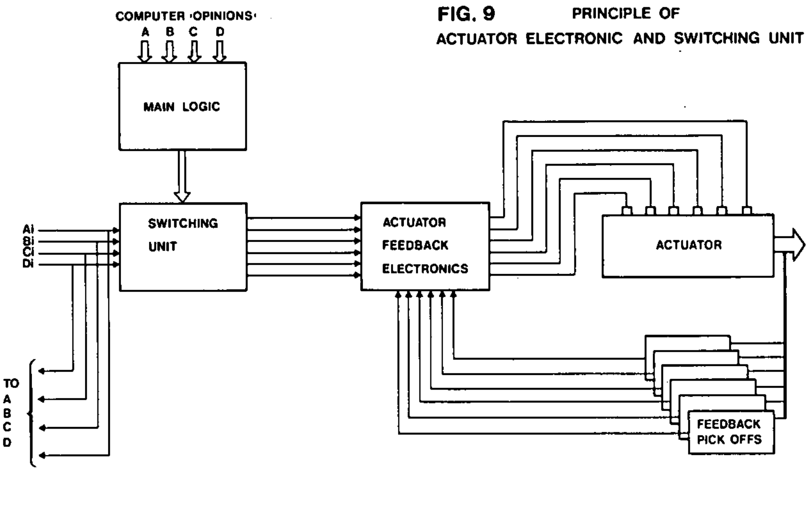
5.3.2 Switching unit

The switching unit is installed as many times as there are actuator inputs. In case of a trio-duplex-actuator six inputs are necessary.

5.3.3 The actuator feedback electronics

shows no difference to common actuator electronics, except that it is installed redundantly.

FIG. 9 PRINCIPLE OF ACTUATOR ELECTRONIC AND SWITCHING UNIT



6. SUMMARY AND OUTLOOK

A quadrupally redundant Digital-Total-Fly-By-Wire has ben described. The system provides fail-op, fail-op characteristics and a high degree of functional integration. It will undergo flight test within the next few years on an experimental F 104 G Starfighter. The objectives of this program are to demonstrate the CCV-technology in connection with a modern control technology.
Additional goals are defined as reduction of cost because of the benefits of a higher degree of integration (multifunctional use of the computers) and increasing of testability and reliability. In further program steps additional redundancy principles to the now used parallel principle will be investigated. Especially we are planning to use methods to monitor sensors by software and replace failed signals by observer techniques which we are already using in other projects. From the system theory point of view we are also preparing adaptive methods. The presently used method is a Multi Variable Optimal Control which uses a precalculated adaption to the changing aircraft parameters.

THE ASSET (ADVANCED SKEWED SENSORY ELECTRONIC TRIAD) PROGRAM

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SUMMARY

A redundant arrangement of angular rate sensors with skewed input axes, dispersed on an aircraft bulkhead, has been designed for fly-by-wire control applications. Compared to other redundant configurations, it best satisfied system reliability, survivability, and maintenance requirements. By also utilizing a high reliability "solid-state" angular rate sensor, expected maintenance costs will be decreased. The data management system designed for the ASSET configuration featured a parallel path failure detection and isolation algorithm. A unique method of selecting failure thresholds was developed to insure that false alarm probability and system errors were minimized. The results of this effort will contribute to the practical implementation of a digital fly-by-wire system, since a successful attempt was made to match proposed operational requirements. The ASSET concept will therefore provide a fail-operational and combat-survivable set of rate sensors designed to interface with all active control systems, regardless of redundancy requirements.

1. INTRODUCTION

Fly-by-wire control systems are rapidly becoming the cornerstone of the design of future high performance aircraft. Through redundancy, these electrical controls can be made safer and more survivable than the mechanical controls existing in a number of operational fighter/attack aircraft. Fly-by-wire system technology provides new freedom in the design of the aircraft; for example, relaxed static-stability margins and precise maneuvers not previously possible.

Analog fly-by-wire control techniques as well as equipment mechanization methods were verified during the U. S. Air Force 680J Survivable Flight Control System program. Currently, integrated digital flight control studies are progressing under the auspicies of a joint U. S. Navy/U. S. Air Force Definition Study for Advanced Fighter Digital Fly-By-Wire (DFBW) Control System. (1). These studies include the development of multimode control/display modes, multichannel digital fly-by-wire configurations, redundancy management techniques, multiplexing and lightning protection, and in addition, the improvement of survivability, reliability and maintainability. As part of this effort, the Naval Air Development Center, Warminster, Pa. (NAVAIRDEVCEN), in conjunction with the Grumman Aerospace Corporation, Bethpage, New York, is currently investigating arrangements of fly-by-wire sensors and their associated redundancy data management systems. For the initial investigation, the ASSET program is underway to develop an angular rate sensor system that satisfies the reliability as well as the survivability requirements of fly-by-wire, uses the minimum number of components and allows full interchangeability of the pitch, roll, and yaw axes sensors.

Previous to this program, redundancy concepts regarding the optimum use and configuration of the sensor subsystem portion of fly-by-wire systems have not undergone extensive scrutiny. Some work has been done in the analog implementation of monitoring and voting of in-line quad-redundant inner loop rate gyros and lateral and normal accelerometers, but this has been limited to configurations where the survivability was not considered and the redundant sensors were in close proximity to each other. In the allied field of advanced Strapdown Inertial Measurement Unit (IMU) design, however, work has been underway to develop redundancy data management techniques and configuration methods that permit the reduction of weight, power and volume of IMUs by skewing the input axes of sensors with respect to each other (2). The approach taken in this program was to apply these skewed sensor concepts to reduce the number of angular rate sensors required for a reliable and survivable fly-by-wire system.

2. ANGULAR RATE SENSOR RELIABILITY

Apart from current efforts directed towards developing fly-by-wire systems, the U. S. Navy has always been intensely interested in improving the reliability of the Automatic Flight Control Systems (AFCS) used in fighter/attack aircraft. Studies of AFCS component failure data for U. S. Navy operational squadrons, obtained from Navy Failure, Unsatisfactory or Removal Reports (FUR) and Electronic Failure Report (EFR) programs (3), and a recent study of field data of Navy fighter/attack aircraft (4) revealed that the three largest single failure producing elements in the AFCS are the Electro-Hydraulic Actuator, the Air-Speed Compensator, and the Rate Gyro Package. Table 1 summarizes the AFCS and the 3-Axis Rate Gyro Package failure rates from these sources. The failure rates of the parts and subassemblies of the 3-Axis Rate Gyro Package, obtained from the Reliability Monitoring Program (RMP), are presented in Table 2. The latter table reveals (as expected) that the major source of failure of 3 Axis Rate Gyro Packages is due to the gyros themselves.

Table 1 shows that the 3-Axis package failure rates from all sources are quite consistent. This indicates that the results can be used with a high degree of confidence. The data also shows that the percentage of gyro package failures to AFCS failures, ratio B/A, varies between 4% and 7%, a value large enough to be of concern in the maintenance of Navy aircraft.

From Table 1 the average failure date for the 3-Axis rate package is 408 failures/10^6 flight hours, Using a ratio of equipment operating hours per flight hour of 1.72, (5), and dividing by a factor of 3 to account for 3 gyros per package, a failure rate of 79 failures per 10^6 operating hours, for a packaged single gyro, is obtained. This value was subsequently used to calculate the reliability of redundant rate sensor configurations using state-of-the-art (conventional) gyros.

As expected, the RMP data indicates that the combined failure rate of the spin bearings and spin motor of the gyros are almost a factor of 6 greater than the failure rate of the closest competitor. Specifically, the spin bearing contributes 54 percent of the total failures, and the motor, 11 percent. It is the portion of the gyro configuration where most improvement can be gained with a design change. Consistent with this need, the General Electric Company has developed a promising "solid-state" rate sensor called the VYRO. The VYRO's design eliminates the rotating mass with its associated bearings, motor and gimbal and replaces them with a vibrating beam supported by two wires and driven by piezo-electric transducers. Without rotating parts, the VYRO is potentially more reliable than the conventional rate gyro. The predicted Mean Time Between Failure (MTBF) for the unit is 45,000 hours (22.2 failures/10^6 hours). Performance of the VYRO has been demonstrated on an F-4J airplane, and in laboratory test. (6). Figure 1 shows F-4J stability augmentation mode flight-test recordings of the production pitch rate gyro and VYRO for a pitch step input. The traces are almost identical. Because of its promise, the VYRO should be considered one of the leading candidates for future fly-by-wire applications.

3. RATE SENSOR CONFIGURATION SELECTION

Reliability Goals for Fly-By-Wire

The recent study of flight safety data of Navy fighter/attack aircraft referred to earlier, (4), also established failure rates for fly-by-wire control system components. These failure rates have been used to develop reliability goals for the DFBW program. The study showed that during the eleven-year period of 1960 to 1970, F-4, F-8, A-5, A-6 and A-7 aircraft showed a combined average of 5.5 aircraft losses per 10^6 hours, due to flight controls. With a three-to-one improvement in flight safety considered realistic for fly-by-wire over mechanical controls, a reliability goal for the DFBW inner loop augmentation functions was set to no more than 2 catastrophic failures (total loss of digital control function) in 10^6 flight hours. This value does not include the hydraulic or electrical power supply failures. Because the data showed that the power actuator was the major contributor to this failure rate, the reliability allocation for the electrical portion of the inner loop functions was set to 0.2 catastrophic failures in 10^6 flight hours. Table 3 was prepared, as part of this program, to reapportion this failure rate goal to the parts and subassemblies that make up the inner loop.

Table 3 shows that the equivalent Mean Time Between Failures for the DFBW rate sensor system should be 50,000,000 hours. This compares with the 3 Axis Rate Gyro Package reliability of approximately 4,000 hours MTBF. To satisfy these unusually low failure rate goals, various configurations of redundant components with In-Flight Built-In-Test (IFBIT) and comparison monitoring were analyzed during this program. This approach to meeting stringent reliability requirements has previously been used during a number of aircraft and spacecraft programs. For many of these programs, redundancy not only served the purpose of improving reliability but was also required to meet survivability goals; that is, satisfying system requirements after a sensor was lost due to accidental causes or battle action.

IFBIT

IFBIT takes on two forms for sensor systems as well as for other electronic and electromechanical equipment. The operation is called Self Test/In-Flight Built-In-Test (ST/IFBIT) when electrical or mechanical excitation is applied to the equipment, and its response to the excitation is compared to a 'model' of the output. To perform this type of test during a flight, a gyro may have to be taken 'off-line' so that the sensor response to the test excitation is not used for aircraft control. If the excitation is aircraft motion, then an estimate of the input to the sensor must be made. ST/IFBIT is generally complex to implement.

Externally Caused Failure/In-Flight Built-In-Test (EX/IFBIT) provides measurements to insure that the sensor has not been physically damaged or that changes in environmental or electrical excitation, that would fail the sensor, have not occurred. EX/IFBIT is usually less complex to implement than ST/IFBIT.

Redundant Sensor Configurations

To satisfy redundancy and reliability requirements, multiple rate sensors with the same maximum rate capability are usually oriented with their input axes coaxial or "in-line" with one another. In cases where comparison monitoring is used for malfunction detection, three or more coaxial sensors are provided. When comparison monitoring is employed, the output signal of each sensor is often compared to the average of the outputs of other sensors. If the absolute value of the difference is greater than a fixed threshold, the sensor is considered failed.

Specifically, a two-failure tolerant, comparison monitoring-malfunction detection requirement imposed on an aircraft rate gyro system has required the installation of twelve rate sensors with two different maximum rate capabilities. Typically, the sensors are arranged as follows: four high rate capability gyros with their input axes coaxial with the aircraft roll axis and eight of a lower rate capability coaxial with the pitch and yaw axes.

An alternate approach to satisfy redundancy and reliability requirements makes use of the principle that only three skewed non-co-planar rate sensors can be used to provide three axes of rate information, in essence an electronic triad. Gyro systems with six sensors skewed with respect to each other, so that any three could fail and the remaining input axes not be co-planar, have been manufactured and tested (2). Results to date indicate that a two-failure tolerant, comparison monitoring-malfunction detection skewed system is a viable alternative to redundant coaxial sensor systems.

To meet the reliability goals defined in Table 3, five candidate ASSET configurations were examined. Specifically:

- Configuration A - 3 sets of three coaxial rate sensors. Each set of three gyros includes comparison monitoring capable of detecting and isolating one failure. The Redundancy Capability is therefore 1 FAIL OP (sensor system operational after one failure). The configuration does not include ST/IFBIT. If ST/IFBIT was included, and was 100 percent effective, comparison monitoring would not be required. Under these conditions, 2 out of 3 sensors on an axis could failure (2 FAIL OP) and the system would remain operational.

- Configuration B - 3 sets of four coaxial rate sensors. Each set includes comparison monitoring that is capable of detecting and isolating two failures. No ST/IFBIT.

- Dodecahedron - 6 rate sensors oriented so that their input axes are perpendicular to the faces of a dodecahedron (2). The system of six sensors includes comparison monitoring that is capable of detecting and isolating two failures. No ST/IFBIT.

- Octahedron - 6 rate sensors oriented so that their input axes are perpendicular to the edges of an octahedron (7). The set of six sensors includes comparison monitoring that is capable of detecting and isolating two failures. Only 80% of the third failures are detectable. In this configuration, the sensor pairs are placed at 90° to each other. Potentially, this feature could offer a manufacturing and test advantage over other skewed configurations. No ST/IFBIT.

- Cone Configuration - 6 rate sensors oriented with their input axis symetrically arranged on the surface of a cone. The axis of the cone is placed along the roll axis of aircraft as shown in Figure 2. The set of six sensors includes comparison monitoring that is capable of detecting and isolating two failures. No ST/IFBIT. The cone configuration was developed to take advantage of the improved reliability and redundancy capability of skewed vector sensor systems in an environment where the maximum rate about the roll axis is considerably more than that of the yaw and pitch axes. The Cone Configuration allows one relatively low maximum rate capability gyro to be used. With careful selection of the central half angle, based on a knowledge of the maximum rate requirements of the aircraft, it has been possible to maintain the accuracy of the Cone Configuration equivalent to that of the conventional 3-Axis Rate Gyro Package.

These ASSET configurations are listed at the top of the columns in Table 4. The vertical column on the left lists the system requirements used to compare the various configurations to each other. Table 4 shows that of the sensor systems examined, the Cone Configuration with the high reliability sensor best satisfied the requirements of fly-by-wire. The skewed configurations are favored over the coaxial arrangements from an economic standpoint since they require the least number of sensors. This is reflected in sensor system Procurement Costs, Spare Sensors Required, and Ratio of Mean Time Between Maintenance Actions (MTBMA). The entries in the MTBMA row show the ratio of the frequency of maintenance action expected for the various configurations compared to the Cone Configuration. For example, Configuration B, with conventional rate gyros, will require 7.16 more maintenance actions than the Cone Configuration with VYROs. The Cone was selected over the alternative skewed configurations because it was the only one to satisfy performance requirements.

Reliability of Candidate Configurations

The reliability (or probability of successfully providing roll, pitch and yaw rate data) of the five candidate configurations, compared to the DFBW goal, was determined. Table 5 compares these values for a 3 and 5-hour mission. This length mission is not uncommon for a fighter/attack aircraft with refueling capabilities. Data for a 10-hour mission, more applicable to the sensor system applied to a transport aircraft, is also included. The configurations are compared in terms of sensor system failures expected per million missions.

The results show that both Configuration B and the skewed sensor configurations satisfy the DFBW sensor system failure rate goal of 0.02 failures/10^6 flight hours with the high reliability sensor as well as with the conventional gyro. A skewed configuration with 5 sensors (1 FAIL OP with no ST/IFBIT) may also meet the reliability requirements. This configuration was not examined in detail since the original guideline of fly-by-wire system design included the requirement that the system be 2 FAIL OP.

The values in Table 5 are based on the assumption that the failure thresholds were adjusted to the values required to obtain the false alarm and false isolation probabilities noted in the table. A false alarm means that a non-failed sensor is categorized as failed. A false isolation is manifested when a failure occurs and a non-failed sensor is categorized as failed. It was also assumed that ST/IFBIT was not used to detect rate sensor failures. Although ST/IFBIT has been suggested as a means of reducing the number of redundant components required to meet reliability requirements (8), its quantitative value cannot be measured unless,1) the percentage of random failures that can be detected by ST/IFBIT is known, and 2) the reliability of the equipment associated with the mechanization of ST/IFBIT is available. Because the study showed that the goals could be satisfied with a six sensor system without the incorporation of ST/IFBIT, this additional information was not required.

The utilization of the EX/IFBIT is a function of (1) the vulnerability of the physical arrangement of sensors to battle damage, vulnerability of the sensors to electrical excitation change and (2) the capability of the data management system to provide satisfactory control outputs with simultaneous failure. For the systems analyzed it was assumed that the data management system was not capable of satisfying fly-by-wire system requirements with two simultaneous "hardover" failures. Therefore, the hardware was assumed to be protected by EX/IFBIT equipment for configurations that allowed two or more sensors to be lost simultaneously (to small arms fire or to the failure of a power source exciting two sensors, for example).

Skewed Sensor System Performance

As previously discussed, the Cone Configuration was developed so that

the fewest number of rate sensors with the same maximum rate capability could be used and

the rate sensor errors reflected into the roll, pitch and yaw rate calculations would be approximately equivalent to the errors expected from presently existing 3-Axis Rate Packages.

The Cone Configuration with a central half-angle of 77° was selected as the skewed system best capable of satisfying performance requirements. The three major elements effecting the performance are: (1) the maximum rates imposed about the roll, pitch and yaw axes of a fighter/attack aircraft, (2) the relationship between a sensor's errors and its maximum rate capability and (3) the sensor error amplification that occurs when non-orthogonal sensor outputs are used to determine vehicle axes solutions. The first two elements result in a vehicle axis error amplification factor called KR_p, KR_q, KR_r and the third element in a factor called KS_p, KS_q, KS_r. The product of these two factors result in the total vehicle error amplification which was used to compare the performance of the candidate skewed systems.

The error amplification factor KR was designated as the ratio between the maximum rate capability required of the skewed sensor and the maximum rate capability of the corresponding gyro in a 3-Axis Gyro package. Its development is based on the assumption that rate sensor errors, such as electrical excitation sensitivity, hysteresis, temperature sensitivity and scale factor accuracy increases proportionally with the maximum rate capability of the instrument. That is, the errors expected from a sensor with a maximum rate capability of $\pm$ 300°/sec would be five times higher than those expected from a $\pm$ 60°/sec maximum rate capability sensor. An estimate of the maximum rates that can be expected about the fighter/attack aircraft axis is shown in Table 6. The table indicates that when 300°/sec is expected about the roll axis, the rates about the other axis will be relatively low.

Table 7 shows the maximum rate capability requirements for the individual sensors of the various configurations and values of the error amplification factor KR, developed by combining these requirements with the maximum rate capabilities of the sensors in 3-Axis Rate Gyro Packages of present day aircraft.

The vehicle axis amplification factors (KSp, KSq, KSr) are defined by the following equations:

$$KSp = \frac{\sigma_p}{\sigma} \quad (1)$$

$$KSq = \frac{\sigma_q}{\sigma} \quad (2)$$

$$KSr = \frac{\sigma_r}{\sigma} \quad (3)$$

where: σ_p, σ_q and σ_r are the standard deviations of the sensor errors along the vehicle axes, and can be calculated by resolving the output of skewed sensors to vehicle axes. σ is the standard deviation of the errors of each skewed sensor. The average amplification factors KS, for all the configurations of six sensor systems, using 6, 5, 4 and 3 sensors at a time were determined in the manner described in (9). The total vehicle error amplification factor, KT_p, KT_q, KT_r for the candidate configurations, shown in Table 8, was determined by using this information and the values of KR shown in Table 7. The results in Table 8 show that when constrained to one maximum rate capability sensor, the skewed Cone Configuration with a central half angle of 77° is superior, in terms of total vehicle axes error amplification factor, to the Dodecahedron and Octahedron configuration. For this reason it was selected as the primary configuration to be tested in the ASSET laboratory program.

4. PACKAGING AND SURVIVABILITY

The location and packaging of redundant angular rate sensors in fighter/attack aircraft have been influenced by two factors. First, comparison monitoring-data processing practices require that redundant sensors measuring the same quantity be packaged on a rigid mounting surface, in the vehicle, so that structural vibration inputs to these sensors are the same. Placed at different locations in the vehicle, the comparison monitoring equations are not able to distinguish between differences due to sensor errors and differences due to dissimilar vibration inputs. To account for different vibration inputs, the failure threshold must be raised, which in turn increases the probability of missing failures.

The second factor derives from the desirability of placing the angular rate sensors at the structural bending antinodes of the vehicle to diminish the possibility of regenerative interaction between the control system and structural bending. (Regenerative interaction means that the control system in combination with the bending aircraft results in undesirable control characteristics.) At the antinodes, the angular rate amplitude of vibration is minimized. Placement of sensors at or adjacent to these locations minimizes the need for output signal filtering and diminishes the residual vibration signal output, if filtering is employed.

Unfortunately, these packaging and placement constraints, coupled with the multitude of sensors required to satisfy the redundancy needs, are inconsistent with the desirability of separating redundant sensors in an aircraft to enhance the fly-by-wire control system equipment survivability. In a combat situation when an aircraft is under attack by small arms fire, it is desirable that the redundant components be separated so that a single round of enemy fire does not destroy all sensors.

The ASSET configuration that was developed as part of this program that best met all these requirements is shown in Figure 3. Each package contains two rate sensors (two-packs) placed to develop the 77° Cone Configuration. Each two-pack is interchangeable with any other two-pack and all sensors are excited by the same structural vibration and yet they are dispersed to satisfy survivability needs.

Two-packs were selected over one-packs (six separate sensors) three-packs (three in a common package), or six-packs (six sensors together), by first considering the probability of the sensor system surviving a small arms attack (30 or 50 caliber fire). Figure 4 shows the ratio of aircraft losses due to all causes and aircraft losses due to the redundant sensor system, with the aircraft under small arms fire. The results indicate that, for a fly-by-wire system applied to the aircraft used as a vulnerability model, (10), the one-pack, two-pack and three-pack configuration offer satisfactory survivability for a reasonable number of hits. Under attack, one aircraft will be lost due to the six-pack configuration being destroyed by fire, to every 56 aircraft lost due to other causes. This was judged as an excessive number of aircraft losses due to sensor system vulnerability and the six-pack was eliminated from contention.

The remaining one, two and three packs are compared in Table 9. From an aircraft installation standpoint, the three-pack appears best because only two aircraft mounting surfaces are required. Performance is adequate for all configurations. Manufacturing costs will probably be slightly less for the two-pack, and the capability of withstanding an additional failure is as indicated. Based on this comparison, the two-pack was recommended. The configuration envisioned is the two-pack designated as an Aircraft Weapons Replaceable Assembly (WRA) and the rate sensor, with any electronics unique to the particular unit, designated a Shop Replaceable Assembly (SRA).

5. REDUNDANCY DATA MANAGEMENT

Figure 5 describes the redundancy data management system designed to extract roll, pitch and yaw rate information from the Cone Configuration of skewed sensors. Signals from the sensors first enter the Transient Failure Removal Routine (TFRR). The TFRR places the sensor in a Temporary Failed (TF) category if the latest value of a sensor output is unreasonable compared to the previous value of the estimate of the angular rate along the sensor axis. When a sensor is in the TF category, its output cannot be used in subsequent calculations for one iteration cycle. At the completion of the TFFR operation signals from the remaining sensors simultaneously enter the Sensor Voting Computational Routine (SVCR), and the Failure Isolation Computational Routine (FICR). The SVCR and TFFR accept EX/IFBIT information as well as discretes from the FICR that indicate a Permanent Failure (PF) of a sensor has occurred. When a sensor is placed in this category, its output cannot be used to determine rate for all future iteration cycles. The SVCR provides angular rate signals along roll, pitch and yaw axes with the structural vibrations superimposed.

In operation, the SVCR removes the poorest performing sensor from further computation and then resolves the information, obtained from the remaining sensors, to provide roll, pitch and yaw signals. Then SVCR computations are performed every cycle while the permanent failures generated within the FICR are determined only after a number of cycles are complete. In this way, the control system can be provided with data free of transients and failed sensor outputs, at a high speed, and the FICR can function at a slower speed. The slow speed, non-time critical operation of the FICR improves the reliability of sensor failure identification because several iterations can be used to identify a failure, minimizing the number of false alarms and missed failures.

The Notch Filters shown in Figure 5 are used to attenuate the magnitude of the structural vibration inputs before the roll, pitch and yaw rate signals are transmitted to the control system. Only three signals are transmitted through the Notch Filters rather than six. This is possible because all the sensors are mounted on a rigid surface (one bulkhead), and are all sensing portions of the same three-dimensional vibration input.

Failure Thresholds

As failure thresholds are reduced, false alarm and false isolation probability is increased and the reliability of the system is diminished. As thresholds are increased, low magnitude failures are missed and the system performance can ultimately be degraded. To develop acceptable in-flight threshold levels for the FICR of the Cone Configuration, data from a Monte-Carlo analysis was used. The results of the analysis enabled the selection of the failure thresholds to values that provided an acceptable false alarm probability. (For ground testing, where false alarm probability is less important, the thresholds should be reduced to decrease the possibility of missing a failure.)

Using a model of good, in-specification sensors, a model of failed sensors and a description of the failure isolation law implemented in the FICR, the Monte-Carlo analysis technique provided false alarm probability and probability of missing a failure versus failure threshold. A sample of the results with no past failures is shown in Figure 6. Figure 6 shows that setting the threshold $T = 4.2\ \sigma$ causes the false alarm probability with no failures and the false isolation probability with one failure to be reduced to a negligible value. The probability of missing the failure varies as a function of the failure magnitude. With this threshold, the probability of missing a $3\ \sigma$ and $4.5\ \sigma$ failure is 100 percent, $6\ \sigma$ failure 99 percent, $7.5\ \sigma$ failure 81 percent, etc. With complete knowledge of aircraft control and sensor failure mode characteristics, these values can be translated to aircraft performance.

Advantages of Data Management System

This design approach, compared to other data management systems developed for skewed sensor systems (11), offers distinct advantages. These advantages are:

- Sensor outputs contaminated by transient noise are not used in the determination of roll, pitch and yaw rates.
- The outputs of failed sensors and the poorest performing sensors are not used in the determination of roll, pitch and yaw rates. This feature permits quantization and other gyro noise sources to be filtered in the FICR before the failure diagnosis is made, and allows the threshold to be set more precisely. This improved threshold adjustment capability enhances the reliability as well as the performance of the sensor system by minimizing the number of false alarms and missed failures. In addition, by removing the poorest performing sensors, the data manage-

ment system uses only the non-failed sensor outputs to calculate control signals, when moderate magnitude failures are missed.

The five types of simultaneous multiple failures possible with this hardware/software system are handled in the following manner:

(1) Multiple failures caused by physical damage, such as small arms fire or excessive localized environmental or electrical excitation changes, are detected as permanent failures by the appropriate EX/IFBIT (that must be included in the sensor/aircraft system design) before the data can enter the SCVR and FICR.

(2) Multiple transient failures are converted to the temporary failures category by the TFRR routine before the data can enter the SVCR and FICR.

(3) Two large magnitude random failures, that individually would be isolated by the FICR, have a very low probability of occurring in a six sensor system within the maximum time interval required to isolate a failure (say 3 minutes). The FICR was therefore not designed to isolate this failure mode. The probability of dual failures occurring within 3 minutes is only 1.232×10^{-12}, for sensors with a failure rate of 22.2 failures/10^6 flight hours and 1.560×10^{-11} for sensors with failure rate of 79 failures/10^6 flight hours.

(4) A moderate magnitude failure that is missed by the FICR, followed by a large magnitude failure does not present a problem. The FICR will detect and isolate a potentially dangerous large magnitude second failure because it considers the first sensor unfailed.

(5) A moderate magnitude first failure, that is missed by the FICR, followed by a moderate magnitude second failure of the same type, magnitude and polarity, presents the most difficult simultaneous failure condition to be handled by the FICR. Under these unlikely circumstances, the SCVR could use the outputs of the failed sensors to develop control signals. To counter this possibility, the accuracy and failure modes of the sensors, the mechanization of the FICR, the threshold selection process, and fly-by-wire control laws must be such that the flight safety will not be compromised.

6. SPECIAL VYRO CHARACTERISTICS

The use of the VYRO angular rate sensor in the Cone Configuration can potentially reduce fly-by-wire system maintenance costs. At present, however, there are some limitations in using the VYRO in an aircraft environment. Current models of the sensor have exhibited an excessive sensitivity to temperature change. To work around this problem and to make the sensor compatible with flight control systems, the General Electric Company has incorporated a 'wash-out' or high pass filter, in the instrument package. When the VYRO is used to provide damping, such as in a conventional stability augmentation system, the 'wash-out' does not compromise aircraft dynamic characteristics provided the time constant is properly controlled for the specific application. However, for command augmentation systems where the control laws may require steady-state rate data, the presently configured VYRO may not be compatible with the flight control system requirements.

To provide flexibility in using the VYRO, it is presently planned to remove the analog wash-out circuit from the VYRO and replace it with a digitally implemented filter that will both (1) provide steady-state rate by compensating for the relatively low frequency bias changes caused by temperature variations and in addition, (2) 'wash-out' the temperature induced bias for those control modes that do not require steady-state rate information.

7. CONCLUSIONS AND FUTURE PLANS

The results of this study have shown that the Cone of 6 skewed VYRO angular rate sensors, dispersed in two-packs, with comparison monitoring software, provides a system that can satisfy reliability, performance and survivability requirements of fly-by-wire without the complex forms of ST/IFBIT often employed to detect and isolate sensor failures. This analysis verified that the "brute-force" approach of installing redundant rate sensors, coaxial with the original roll, pitch and yaw rate gyros was not required. The study also showed how the use of the higher reliability VYRO angular rate sensor will reduce the MTBMA ... a factor that will play an important part in the economics of future redundant electronic systems used for active flight control.

A data management system for the Cone Configuration was synthesized. A method was developed to select comparison monitoring failure thresholds for the zero, one, and two, past failure system states. This method provides the system designer with false alarm probability and the probability of missing a failure as a function of threshold. The relationship between the threshold selected for ground testing and threshold for in-flight testing was also briefly reviewed. It is planned to examine this relationship and verify the sensor system performance and operational characteristics during the hardware development and flight test evaluation of the Cone Configuration with the VYRO.

While still in the exploratory development phase, the ASSET program will ultimately interface with the DFBW program and contribute to the practical realization of operational fly-by-wire by providing the sensory interface for all redundancy levels.

8. REFERENCES

1. Abrams, C. R., "Guidance and Control Programs at the Naval Air Development Center". Paper presented at the SAE Aerospace Control and Guidance Systems Committee Meeting #33, 15-17 May 1974, Paradise Valley, Arizona.

REFERENCES (Cont'd.)

2. Gilmore, J. R., McKern, R. A., "A Redundant Strapdown Inertial System Mechanization - SIRU", Charles Stark Draper Lab., Report #E2527, August 1970.

3. Johnston, D. E., and Weir, D. H., "An Assessment of Operational and Cost Tradeoff Factors for a Typical High Performance AFCS." STI Technical Report #119-1, June 1972.

4. Helfinstine, R. F., et al, "Reliability and Redundancy Study for Electronic Flight Control Systems", Honeywell Report No. 21718-FR, July 1972.

5. Klass, P., "Avionics Failure Rate Better Predictions", AWST, November 13, 1972.

6. General Electric Co., "Qualification Test Report VYRO Model CRS-1B", Aircraft Equipment Div., Binghamton, New York, August 1971.

7. Bejczy, A. K., "Non-Orthogonal Redundant Configurations of Single-Axis Strapped-Down Gyros", JPL Quarterly Technical Review, Volume I, No. 2, July 1971, N71-31119.

8. Borow, M. S., et al, "Navy Digital Flight Control System Development", Honeywell Document No. 21857-FR, December 1972.

9. Weinstein, W. D., "Optimum Skew Angle Between Redundant Inertial Systems" Proceedings of ION National Space Meeting, 23-25 February 1971. pp 255-276.

10. Dunn, A. A., "Damage to F4U, AD and F9F Series Aircraft in Korean Operations, August 1950 - July 1953" APL/JHU CF-2570, September 1956, Johns Hopkins University, Applied Physics Lab., Silver Springs, Maryland.

11. Wilcox, J. C., "Competitive Evaluation of Failure Detection Algorithms for Strapdown Redundant Inertial Instruments", NASA CR-124234, April 1973.

9. ACKNOWLEDGEMENTS

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Table 1 AFCS and 3 Axis Rate Package failure rates

AIRCRAFT PROGRAM	3 AXIS RATE GYRO PACKAGE FAILURE RATE (FAIL/10^6 FLT. HRS.) (B)	AFCS FAILURE RATE (FAIL/10^6 FLT. HRS.) (A)	RATIO $\frac{B}{A}$	REFERENCE MATERIAL
A-4C & F-4B	340	–	–	• RELIABILITY MONITORING PROGRAM (RMP) • 135,000 FLIGHT HRS.
A-4C	367 (PREDICTED) 508 (OBSERVED)	7,093	4.6% 7.2%	• 72,000 FLIGHT HRS. • FAILURE UNSATISFACTORY REMOVAL REPORTS (FUR) • ELECTRONIC FAILURE REPORT (EFR)
S-2E	416	10,580	3.9%	(4)

Table 2 Failure rates of parts and subassemblies of 3 Axis Rate Gyro Package

PARTS AND SUBASSEMBLIES	NUMBER OF ITEMS	ITEM FAILURE RATE (FAIL/10^6 FLT. HRS.)	TOTAL FAILURE RATE (FAIL/10^6 FLT. HRS.)
GYRO, RATE	3	99.8	299.40
CONNECTOR PIN	27	0.12	3.20
RESISTOR, COMPOSITE	6	0.09	0.54
SOLDER JOINT	150	0.023	3.45
TRANS., POWER	3	11.0	33.00
TOTAL			≈340

Table 3 Reliability apportionment for fly-by-wire inner loop electronics

SUBASSEMBLY	FAILURES PER 10^6 HOURS
COMPUTER	0.066
SENSORS/TRANSDUCERS/SWITCHES	
• TRANSDUCERS/SWITCHES/DATA BUS	0.037
• ACCELEROMETERS	0.010
• RATE SENSORS	0.020
SERVO/SECONDARY ACTUATORS	0.067
TOTAL	0.200

Table 4 Redundant rate sensor configuration comparison chart

SYSTEM REQUIREMENTS \ RATE SENSOR CONFIGURATION		COAXIAL		SKEWED		
		3 YAW 3 ROLL 3 PITCH (A)	4 YAW 4 ROLL 4 PITCH (B)	DODECAHEDRON	OCTAHEDRON	CONE CONFIGURATION
TOTAL NUMBER OF SENSORS		9	12	6	6	6
RELIABILITY	FAILURE RATE OF SENSOR $79/10^6$ HRS.	NOT SATISFACTORY	SATISFACTORY	SATISFACTORY	SATISFACTORY	SATISFACTORY
	$22/10^6$ HRS.	NOT SATISFACTORY	SATISFACTORY	SATISFACTORY	SATISFACTORY	SATISFACTORY
REDUNDANCY CAPABILITY	NO ST/IFBIT	1 FAIL OP. PER AXIS	2 FAIL OP. PER AXIS	2 FAIL OP.	2 FAIL OP.	2 FAIL OP.
	WITH ST/IFBIT	2 FAIL OP. PER AXIS	3 FAIL OP. PER AXIS	3 FAIL OP.	3 FAIL OP. *	3 FAIL OP.
MEAN TIME BETWEEN MAINTENANCE ACTION RATIO	FAILURE RATE SENSOR $79/10^6$ HRS.	5.39	7.16	3.6	3.6	3.6
	$22/10^6$ HRS.	1.5	2	1.0	1.0	1.0
SPARE SENSORS REQUIRED		SATISFACTORY	LEAST SATISFACTORY	BEST	BEST	BEST
PERFORMANCE		SATISFACTORY	SATISFACTORY	NOT SATISFACTORY	NOT SATISFACTORY	SATISFACTORY
PROCUREMENT COSTS		GOOD	LEAST SATISFACTORY	BEST	BEST	BEST

*20% OF THE TIMES FAILURES OCCUR THE SYSTEM IS ONLY 2 FAIL OP.

Table 5 Reliability of candidate rate sensor configurations vs DFBW sensor system goal

		FAILURES PER 10^6 MISSIONS					
SENSOR SYSTEM CONFIGURATION	3 COAXIAL (A)		4 COAXIAL (B)		6 SKEWED		DFBW SENSOR SYSTEM GOALS
INDIVIDUAL SENSOR FAILURE RATE x 10^6 HRS.	22.2	79	22.2	79	22.2	79	
HRS. PER MISSION 3	3.2 (x)	11.6 (x)	0.005	0.022	0.003	0.013	0.060
5	5.3 (x)	19.9 (x)	0.009	0.044	0.005	0.029	0.100
10	10.9 (x)	42.5 (x)	0.020	0.126	0.012	0.094	0.200

(x) MEANS RELIABILITY IS LESS THAN GOAL

ASSUMPTIONS

1. FAILURES OCCUR RANDOMLY IN TIME, ARE INDEPENDENT AND ARE NOT DUE TO EXTERNAL CAUSES.
2. FALSE ALARM AND FALSE ISOLATION PROBABILITIES ARE AS FOLLOWS:

CONFIGURATION	FALSE ALARM PROBABILITY	FALSE ISOLATION PROBABILITY WHEN FAILURE OCCURS	NUMBER OF PAST FAILURE
A	0.01%	0.5%	0
		–	1
B	0.01%	0.1%	0
		0.5%	1
		–	2
6 SKEWED	0.01%	0.1%	0
		0.5%	1
		–	2

Table 6 Maximum expected angular rates about vehicle axis

MAXIMUM RATES ON INDIVIDUAL AXIS (NON-SIMULTANEOUS)	MAXIMUM RATES THAT COULD EXIST SIMULTANEOUSLY
p (ROLL RATE) 300°/SEC. q (PITCH RATE) 60°/SEC. r (YAW RATE) 60°/SEC.	p 250°/SEC. q 40°/SEC. r 40°/SEC.

Table 7 Error amplification factors (KR)

	3 AXIS RATE SENSOR PACKAGE	OCTAHEDRON	DEDECAHEDRON	CONE (77°)
MAXIMUM RATE REQUIRE-MENT	+300°/SEC. = p, ± 60°/SEC. = q, r	212°/SEC.	256°/SEC.	110°/SEC.
KRp	1	0.71	0.85	0.37
KRq	1	3.54	4.27	1.84
KRr	1	3.54	4.27	1.84

Table 8 Total-vehicle axis error amplification factors (KT)

CONFIGURATION	NO FAILURES KTr KTq KTp	1 FAILURE KTr KTq KTp	2 FAILURES KTr KTq KTp	3 FAILURES KTr KTq KTp
CONE 77°	1.08 1.08 .66	1.24 1.24 .76	1.53 1.54 .96	2.39 2.37 1.62
OCTAHEDRON	2.50 2.50 .60	2.88 2.88 .69	3.53 3.53 .85	5.75* 5.75* 1.25*
DODECAHEDRON	3.02 3.02 .50	3.48 3.48 .57	4.19 4.19 .69	6.18 6.18 1.03
*(16 OUT OF 20 COMBINATIONS)				

Table 9 Comparison between one, two and three packs

	ONE-PACK	TWO-PACK	THREE-PACK
AIRCRAFT INSTALLATION	REQUIRES 6 ALIGNED MOUNTING SURFACES	REQUIRES 3 ALIGNED MOUNTING SURFACES	REQUIRES 2 ALIGNED MOUNTING SURFACES IN AIRCRAFT
PERFORMANCE AFTER LOSS OF ONE PACKAGE	SATISFACTORY	SATISFACTORY	SATISFACTORY (BEST COMBI-NATION OF SENSORS ARE RETAINED)
CAPABILITY OF WITHSTANDING ADDITIONAL FAILURES	CAN ISOLATE ADDITIONAL FAILURE WITH-OUT HELP OF ST/IFBIT	NEED ST/IFBIT TO ISOLATE ADDITIONAL FAILURE.	NO ADDITIONAL FAILURE ACCEPTABLE.
PROCUREMENT COSTS	HIGHEST	LEAST	ASSUMED HIGHER THAN TWO PACK BECAUSE OF THE MORE COMPLEX MOUNTING BRACKET THAT IS PART OF THE SENSOR ASSEMBLY

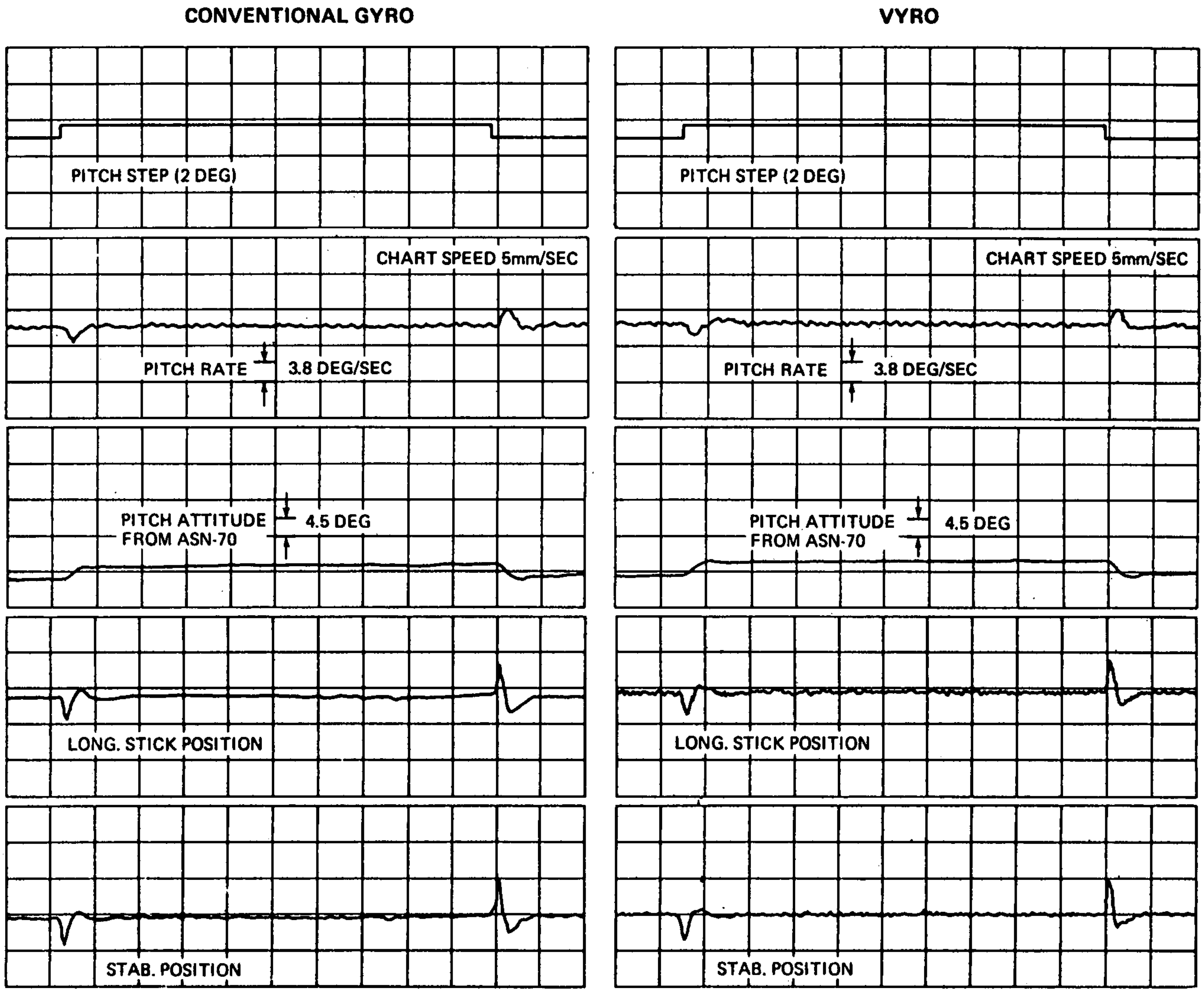

Fig. 1 Flight test comparison of conventional and VYRO angular rate sensor.

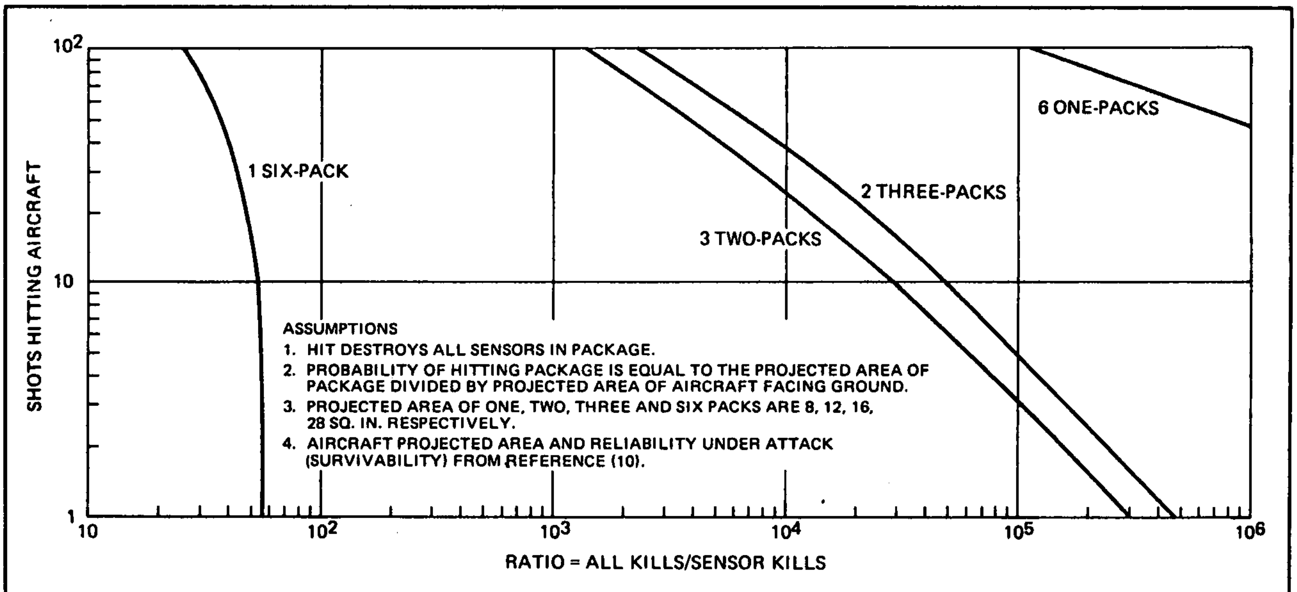

Fig. 4 Survivability of sensor packaging configurations

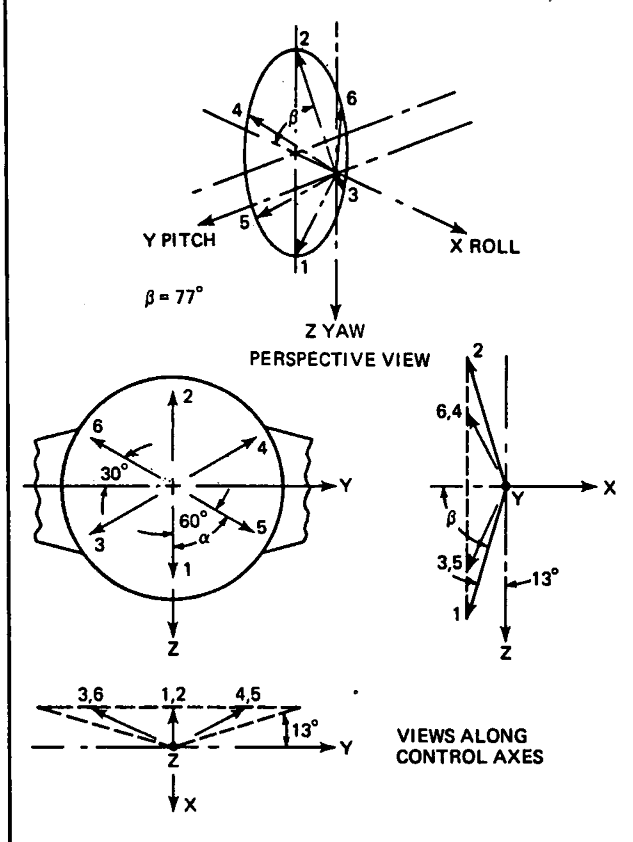

Fig. 2 Skewed cone configuration

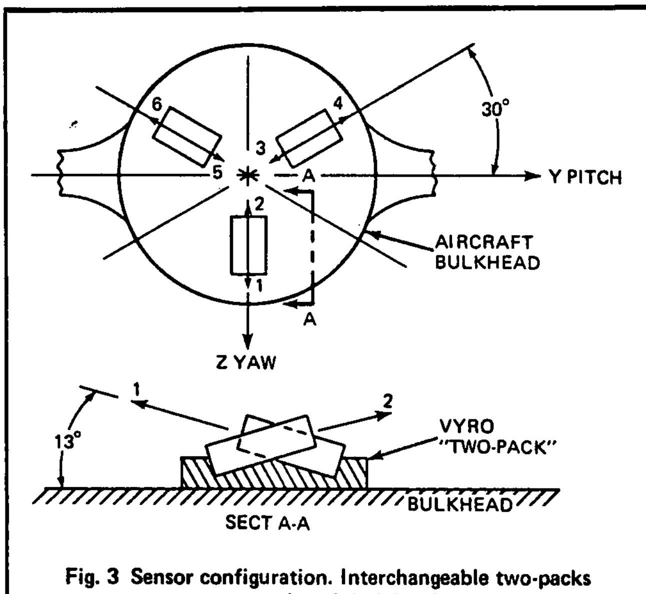

Fig. 3 Sensor configuration. Interchangeable two-packs on an aircraft bulkhead

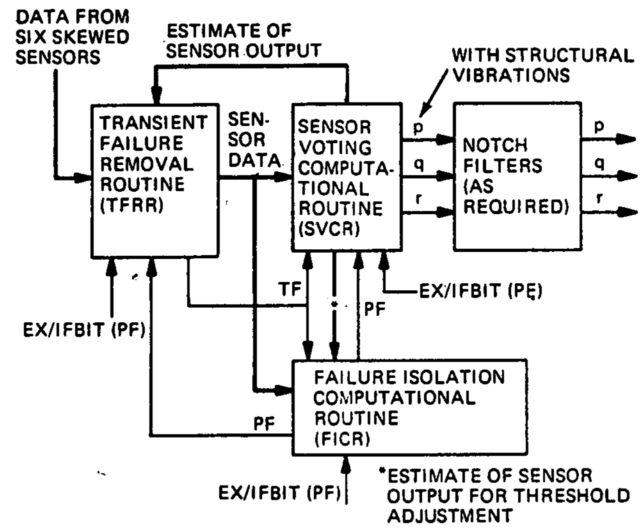

Fig. 5 Redundancy data management system

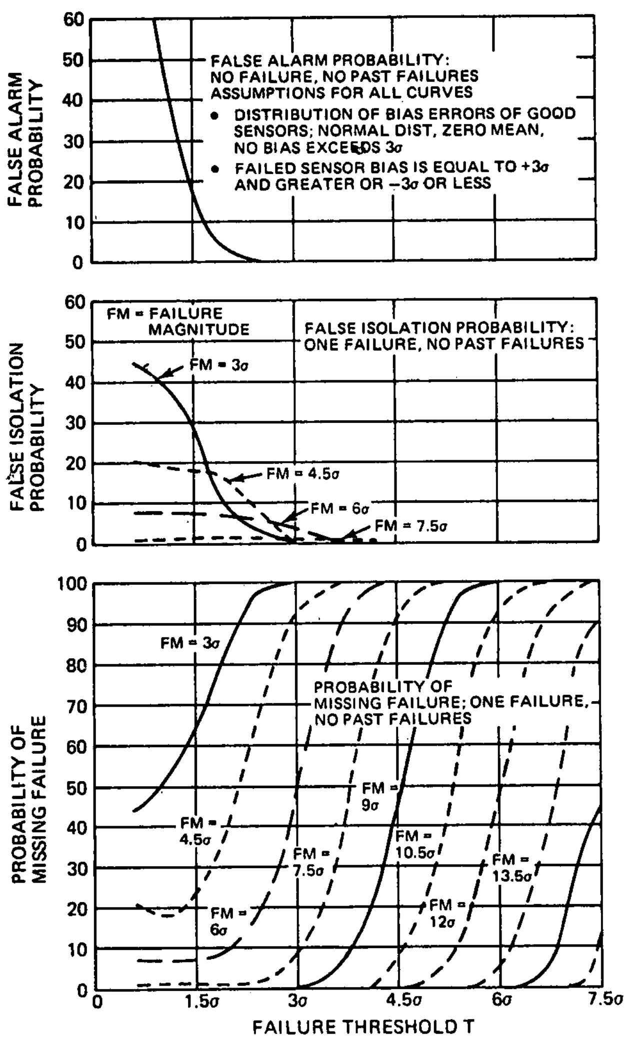

Fig. 6 Failure threshold selection charts

The Relevance of Existing Automatic Flight Control Systems to the Future Development of Active Control.

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SUMMARY

Some relevant examples of failure-survival automatic flight control systems are examined to show how the results of their design, implementation and operational usage can contribute to the successful introduction into full-time use of active control technology (ACT). Ground rules which were evolved some years ago for such redundant systems are re-examined in the interest of full-time ACT.

The important parameters affecting the successful design of a full-time ACT system are discussed. Some of the problem areas are mentioned and the use of some existing techniques for successful certification are suggested. The step from current fail-operative systems relying on some reversionary system to full-time ACT is examined.

The design requirements for the hardware and software for digital computations are detailed and some special problems of digital systems are highlighted and solutions are suggested. Some of the problems of system components such as sensors, computers and actuators are discussed.

1. INTRODUCTION

This paper deals with automatic flight control systems where there is no reversionary means of controlling the aircraft control surfaces or the aircraft if the system fails completely. In the paper this is referred to as full-time active control technology (full-time ACT).

The main purpose of the paper is to summarise some of the background and experience gained in failure-survival automatic flight control systems and to show that much of this is very relevant to the further development and proving of such systems to meet the increased demands of full-time ACT applications.

We believe that the disciplines and the methods which have been steadily developed to gain certification of many systems to date can be used or adapted to achieve the same success with full-time ACT.

2. BACKGROUND

During a period covering well over fifteen years we have been involved in the design, development, manufacture and test of many different types of failure-survival automatic flight control systems (AFCS). These have utilised most types of redundancy principles and have been applied to a wide range of military and commerical aircraft, large and small, subsonic and supersonic, fixed wing, rotary wing and VTOL. Some of these ended up as ACT applications even if they did not start that way and at least one is a true example of ACT.

In the context of full-time ACT they all fall short either because there was a limited time at risk during which the safety of the aircraft depended on the correct operation of the system or because there was a back-up system. Unscheduled automatic landing is an example of the first type and fly-by-wire with mechanical reversion is an example of the second.

The important point about these applications is the resulting design techniques which have been carefully built up. We believe that they are highly relevant to the development of future full-time active control systems. These techniques have evolved in both commercial and military aircraft systems out of joint efforts between the airframe manufacturers, the system manufacturers and the certification authorities to establish an adequate confidence level in the safe operation of aircraft in service. The various requirements may differ considerably but the experience and resulting discipline provide an invaluable background for both military and commercial aircraft applications.

3. GROUND RULES

In a previous paper on the application of redundancy principles to automatic flight control systems given at the 4th meeting of the Guidance and Control Panel in March 1967 a set of ground rules was put forward on the basis of experience up to that time. It was then thought that we were on the threshold of what was called electrical signalling in the UK. It was also thought that it would not be worthwhile just to replace mechanical control runs but that the real benefit was that it allowed the introduction of what we called manoeuvre demand. This has now become ACT.

All of these ground rules arose from what had been learnt from quite different types of redundancy and their application in the different areas of sensors, computers and automatic systems. We think that they have stood the test of time and they are repeated here exactly as presented in 1967.

- Redundant elements must be independent.
- The fault analysis of any common points which arise must be so simple that freedom from common mode failures is easily proved.
- All failures must either be detected and infallibly isolated or absorbed automatically with an efficiency which ensures that any output disturbance is acceptably low.
- The probability of nuisance operation of the failure-detection mechanisms must be much less than the probability of genuine single failures in one lane.
- All failures must be indicated immediately by some 'tell-tale' means.
- It must be possible to test individually each element of a redundant combination to ensure, before operation, that its full failure-surviving capability is available.
- The redundant system must be consistent with the aircraft electrical and hydraulic power supply arrangements and must be insensitive to specified normal variations and brief interruptions of these supplies.

Grouped slightly differently, the main headings and their relevance to ACT are:

Independent Redundant Elements

The need for more than one lane of control follows from the reliability requirement for the total system. It is essential to the integrity of the system that there are no common failures and that a failure in one lane should not be propagated to any other lane other than through the normal interconnection which is for instance by means of a voter monitor which immediately identifies and rejects the signal. In larger aircraft the lanes may be physically separated to minimise vulnerability to local damage. On smaller aircraft lanes should be physically segregated by using separate modules and connectors for each lane. Actuators are a special case and we believe that the fluid logic changeover solutions are potentially vulnerable in this area and will prove difficult to certify.

Simple Fault Analysis, Failure Detection Isolation and Indication

The fact that the lanes are separated and only come together at closely controlled consolidation points means that the failure mode of each branch only need be considered for hardover, fail to zero and oscillatory types of failure. The detailed failure modes and effects of individual component failures can be restricted to the voter monitor devices themselves and if necessary additional integrity can be added at these key points. The voter monitor device itself then provides the fault isolation and indication to the pilot and to the system failure logic. If these elements are allowed to become too complex their failure analysis and the demonstration of their integrity may become major problems in the successful acceptance and certification of the system.

Acceptable Output Disturbance Due to Failure

It is obviously desirable to minimise system output transients and errors resulting from the occurrence, detection and switching out of failures. This is particularly important in the low level high speed flight regime of military aircraft where 'g' per degree of control surface is high. This requirement implies that the thresholds of the monitoring comparators are set at the minimum value consistent with nuisance disconnect probability. The voter monitor or signal selection algorithm is particularly important. It may be a less demanding requirement in the case of large helicopters and large commercial aircraft.

Comprehensive Preflight Test

In a multilane system the monitoring functions used to detect failures often provide sufficient facilities to perform adequate preflight test. The monitors themselves are tested by introducing failures in each lane one at a time in a methodical manner to ensure that the monitor detects and correctly identifies a failed lane.

Low Nuisance Disconnect Probability

Acceptance by operators and pilots of ACT systems will only eventually be achieved if they have faith in the system, regardless of the theoretical reliability calculations that prove statistically the probability of a total system failure is extremely remote. The actual occurrence of nuisance disconnects must be demonstrated as being remote and this can only be assured by conservative design of comparator thresholds relative to actual working tolerances. On current systems a design aim of 6 sigma for the threshold value relative to the 1 sigma normal disparity has been used.

Compatibility with Aircraft Supplies.

For full time ACT the electrical and hydraulic power supply arrangements must take into account the required redundancy level for the ACT system. On single and two engined military aircraft the ACT supply requirements are likely to exceed the number currently used. Quadruplex ACT systems are likely to be required and there are various existing schemes for providing quadruplex stabilised electrical supplies from two or three aircraft dc or ac supplies plus a battery. Hydraulic power is more of a problem and three supplies are not compatible with a quadruplex system unless hydraulic changeover switches are used. If four supplies are made available it is undesirable because of vulnerability and common failures to route these into the small cross section area of an electrohydraulic actuator manifold. We should perhaps rewrite the last part of the last ground rule for ACT to state 'the power supplies should be specified to have output variations and interruptions which are consistent with the reliability and integrity requirements of systems vital to the safety of the aircraft.

4. RELEVANT SYSTEMS

Before looking ahead to the problems of successfully designing and gaining certification for full-time ACT systems we will look briefly at some of the systems which we consider relevant. We have restricted ourselves here to systems where we have direct experience, there are of course many others which are relevant and we hope that other designers share our views as a result of their experience. The first are military applications.

TSR2

In the early 1960s the British Aircraft Corporation TSR2 embodied relaxed lateral static stability as a basic design feature. The size of the fin was made as small as possible to minimise drag and gust sensitivity in the terrain following flight regime; this resulted in a statically unstable aircraft in yaw at high supersonic Mach numbers which further deteriorated with g. The solution to this was provision of a triple redundant control system based on lateral acceleration and yaw rate feedback to the all-moving fin as shown in figure 1. Operational philosophy was that a failure occurring in one lane would be switched out and the aircraft decelerated to a safe Mach number. The time at risk for a second failure was therefore at worst 30 seconds.

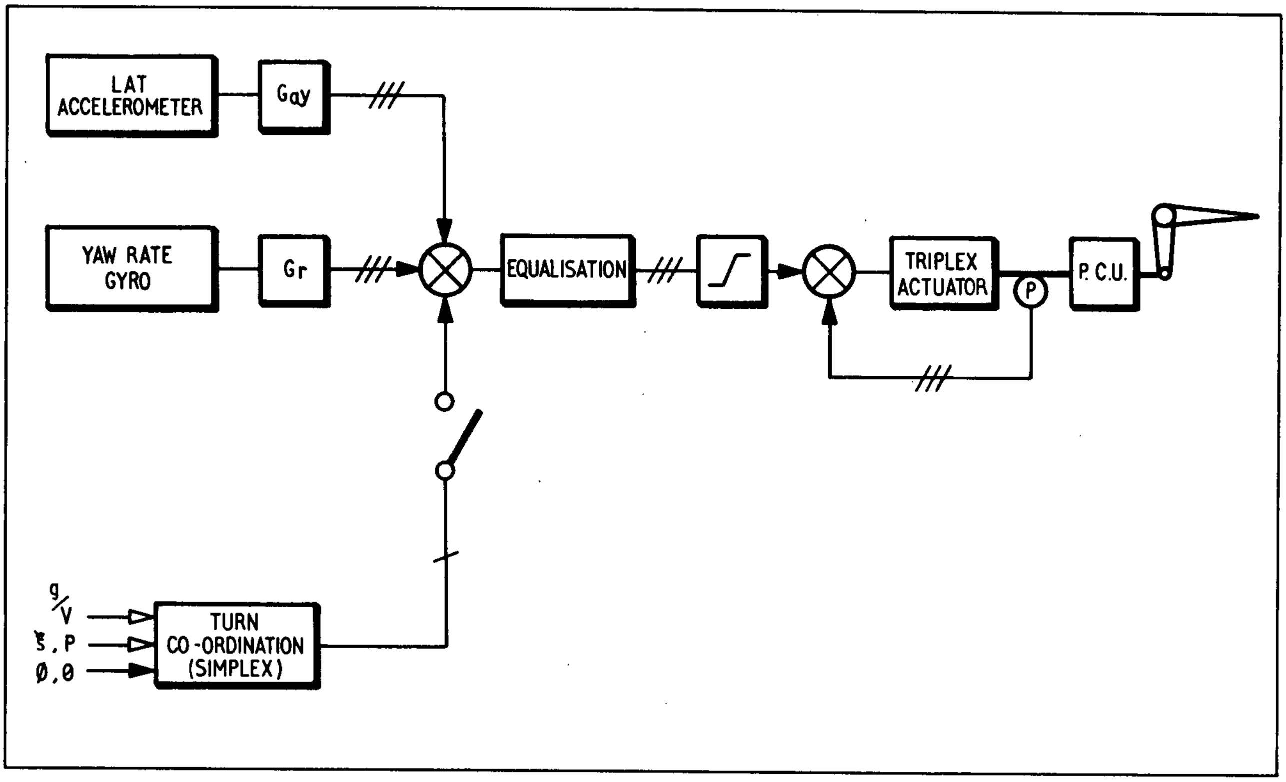

Figure 1. TSR2 Artificial n_y System

MRCA

The Panavia Multi Role Combat Aircraft (MRCA) uses a fly-by-wire command and stability augmentation system (CSAS) to give the required handling qualities over the full flight envelope. In addition to the failure survival capability of the CSAS the system can revert to a direct electrical link mode in pitch and roll before the final reversion to the mechanical control linkage. In yaw there is no mechanical reversion.

Lynx

In the Westland Lynx a dual-redundant collective axis stabilisation system is used to supplement the dual-redundant pitch axis system. This reduces the effects of pitch hardovers at the high speeds of which this rigid rotor helicopter is capable. This system was not part of the original design but was brought in to increase the intervention time which could be allowed for in-service use. It was preferred to other possible aerodynamic solutions or increased redundancy in the pitch system because it could be incorporated into the existing design with less impact to the programme. The resulting system has a high failure survival capability since it is effectively quadruple redundant with two lanes in the pitch axis and two lanes in the collective axis and it also therefore employs dissimilar redundancy with its benefits in system integrity. The performance of this system known as the "collective g compensator" in reducing the effects of failures in the pitch axis is shown in figure 2. With the pitch system inoperative, the full handling qualities are met over a large part of its flight envelope.

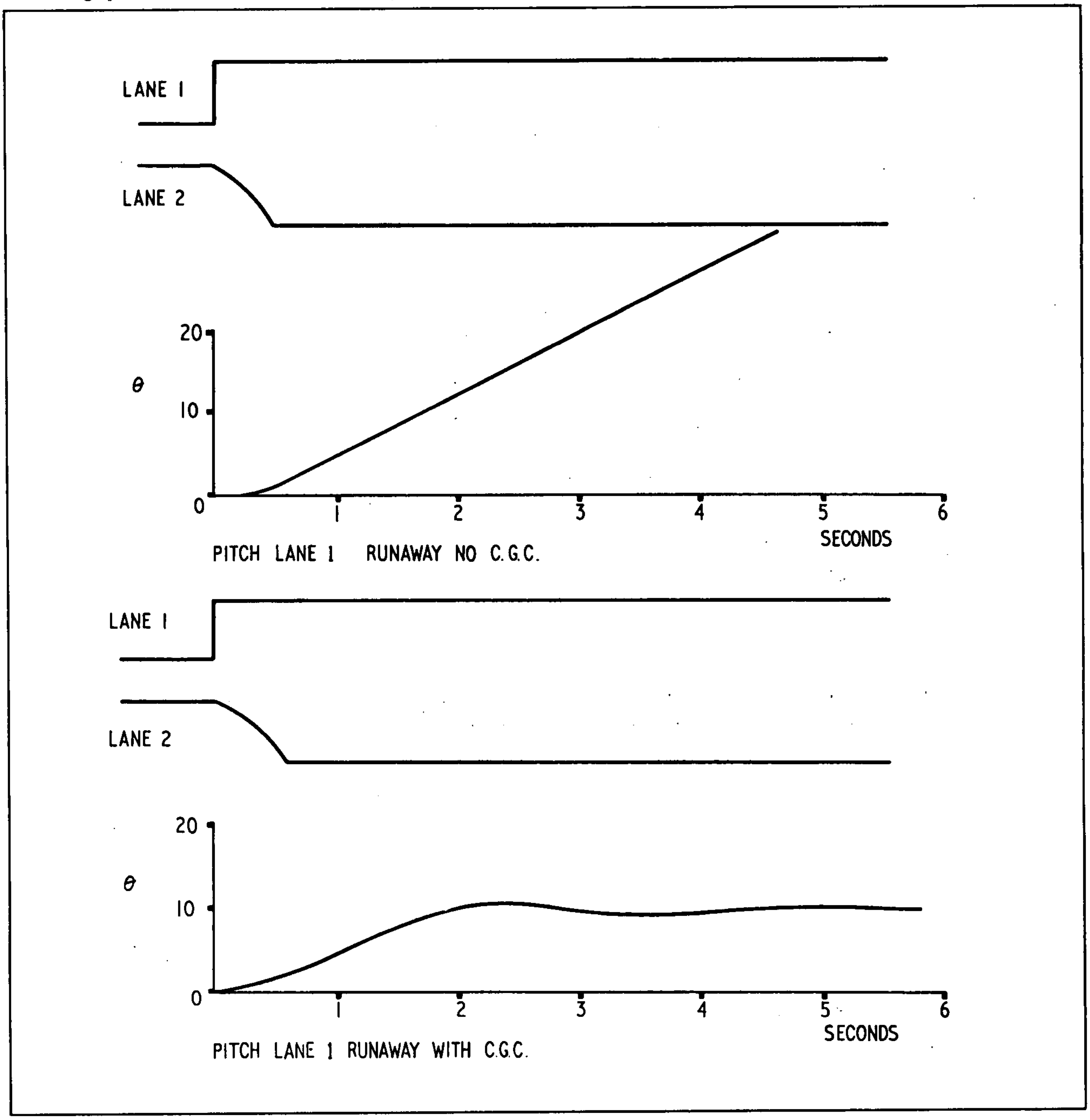

Figure 2 Lynx Pitch Axis Runaway

YC-14

The Boeing YC-14 is being developed to meet the United States Air Force Advanced Military STOL Transport (AMST) aircraft requirement. The automatic flight control system is required to work over the whole flight envelope but the fail-operative requirement is for the STOL mode. The design studies carried out on this system led to a digital triple redundant system. This system uses the background of considerable experience in failure-survival systems implemented with analogue electronics brought together with a background of the development of airborne digital processors specially tailored for flight control applications.

The requirement is basically for a command stability performance augmentation system of high availability to ensure a high mission success rate. Data is received from a large number of motion, guidance and scheduling sensors both analogue and digital. The outputs couple into series and parallel actuators of a highly redundant control surface configuration. One particular advantage of using digital techniques has been the utilisation of time multiplexed digital interlane data transfer. This removes one of the problems with analogue multiplex systems of many cross-connections for synchronising and consolidating various parts of the system.

A further step has also been taken in using fibre optic transmission lines for this purpose. This technology has been under development for about six years and can now be used as a practical technique in system design. It gives complete electrical isolation and has no electromagnetic interference problems.

Commercial Aircraft

On the commercial aircraft side Lockheed have already used ACT in their 1011 by reducing fin structure weight because of the existence, for other reasons, of redundant yaw dampers. These have generally been required in long-range large swept-wing jet commercial aircraft but on the first generation were not designed in initially and therefore were not truly ACT. Because of their inherent very low dutch roll damping in cruise, the resulting handling qualities of the unstabilised aircraft impose a pilot workload which is not acceptable to the certificating authorities. As a result this class of aircraft carries at least a dual-redundant yaw damper system and this then becomes a despatch item and requires preflight checking. The VC-10 has a triple-redundant system operating on three separate rudder surfaces. Each separate damper was ground tested and figure 3 shows the third (standby) yaw damper which does not include the lateral acceleration and aileron crossfeed signals used in the other two lanes which are part of the duplicate-monitored autopilot system.

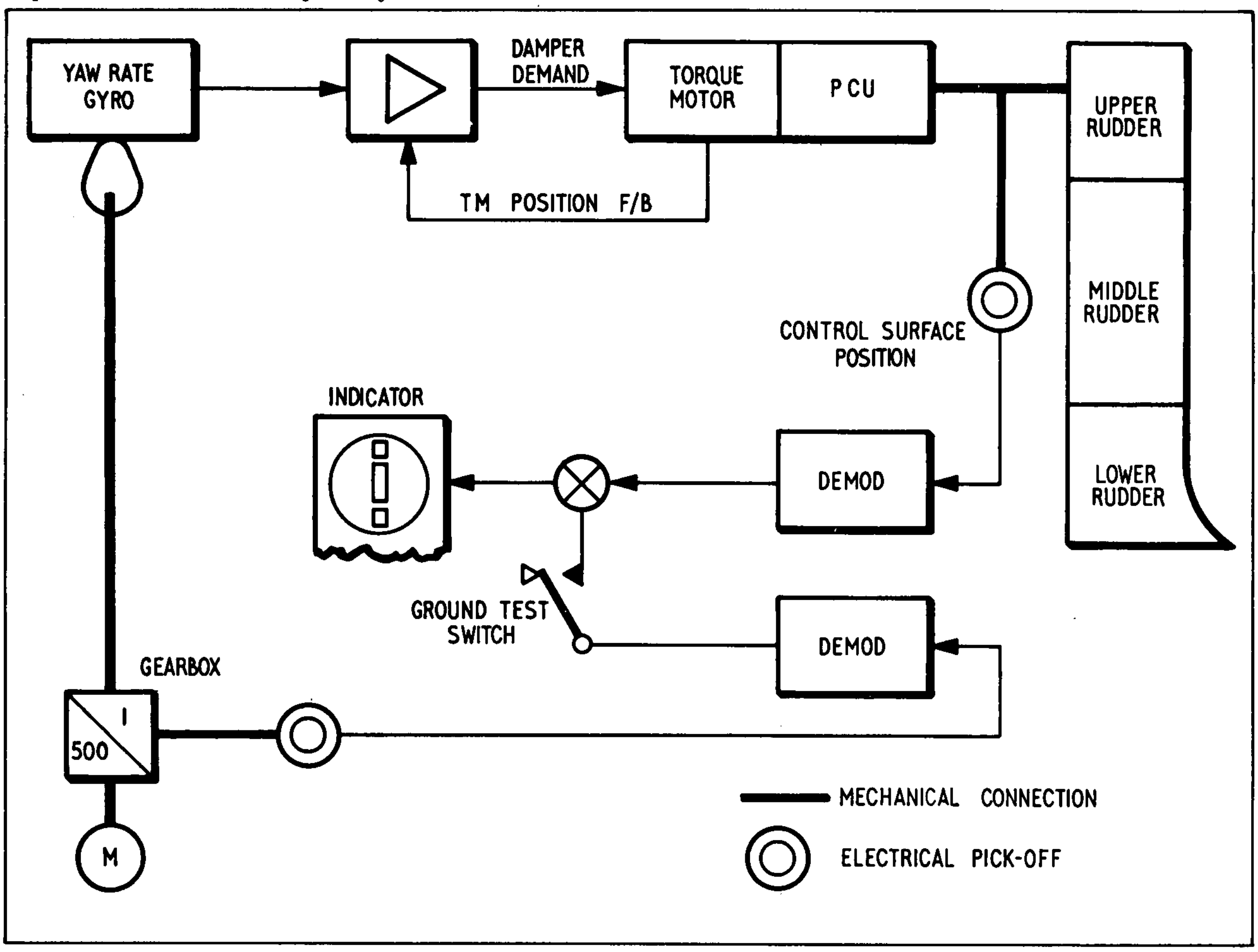

Figure 3 VC-10 Standby Yaw Damper

Prior to take-off, the pilot was required to press the ground test switch which caused the rate gyro assembly to be oscillated mechanically at the natural yaw frequency. The pilot then observed the null signal obtained on the triple yaw damper indicator obtained by subtracting the gyro and rudder oscillations. This indicator also enabled the pilot to observe the damper operation in flight.

The basic Concorde flying control system is fly-by-wire with mechanical reversion and is the subject of other papers at this meeting. The automatic flight control system which is closely linked to the basic flying control system includes automatic landing. The work involved in gaining certification for automatic landing systems is particularly relevant. The requirements of the Civil Aviation Authority (CAA) for automatic landing are that the average failure risk shall be less than one in 10^7 per landing. Because of the short time at risk for unscheduled automatic landings (approximately 30 seconds) this requirement can be met with single failure-survival systems such as triplex or duplicate-monitored. However to meet the requirements of the CAA and to gain certification a high degree of integrity is required.

The BAC VC-10 has been certificated to Category 2 weather minima with a duplicate-monitored automatic landing system. To meet the requirements, stringent design control and fault analysis procedures were followed. Rigorous segregation was obtained between the duplicate systems. Great care was taken from this point of view in the design of the autochangeover function. Control and monitor functions in each system were also carefully segregated. Particular care was taken in the design of monitor and signal consolidation functions. Vital comparators were duplicated and rigorous fault analysis of their circuits were carried out. All of these detailed procedures and techniques are relevant in the design of current and future ACT systems.

The experience gained from the development and successful certification of these systems forms the background to the development of the Concorde automatic landing system which will be certificated to Category 3 weather minima.

5. DESIGN PHILOSOPHY AND CERTIFICATION

It has been stated earlier in the paper that although many systems have been developed in the past that are relevant they fall short of the requirements for full ACT application because they were required to operate for a limited period or because there was a back up system. This meant that although they may have been potentially good enough for full time ACT application either they were not developed far enough or some compromise was allowable. However properly designed redundant systems affecting flight safety needed to conform to the ground rules stated in Section 3 of this paper.

Design Philosophy

At the start of the design process the system designer must allow for the requirements of development, manufacture, testing, certification and in-service use. In doing so he has five major design areas to consider -

- the basic performance
- redundancy management
- system implementation
- aircraft installation
- system usage by the operator

It is clear that there is a high degree of interdependence between these areas and therefore any incompatibility or weak link may have far reaching effects. The total cost of ownership must be acceptable to the user. For successful operation of the system it must meet the requirements in all of these areas and it must be seen to do so by the certificating authorities.

The basic performance is mainly determined by the system requirement for supplementing the aircraft handling qualities and providing the necessary response characteristics. These requirements will also determine the sensors to be used.

The redundancy management must take into account the type of aircraft and its operational requirements and limitations, the performance required from the system, the constraints imposed by the likely performance of the system components and the way in which the system will be used in service. The main interaction with the other areas occurs in redundancy management and its correct solution is vital to the integrity of the system and to its successful certification.

The system implementation in hardware or software must be chosen not only to give adequate margins in the basic performance of the system but also to ensure that the performance of the redundant elements is sufficiently well matched to give successful operation.

The installation of the system in the aircraft can have a major impact and for ACT applications special consideration must be given to the installation to ensure the best overall integrity for the system.

Usage by the operator must be considered in the widest sense - it embraces crew operation, maintenance techniques, preflight checking and crew actions as a result of system failures. Care must be taken to ensure that the integrity designed into the equipment can be maintained throughout its service life at an acceptable cost and level of maintenance.

Certification

Full-time ACT systems must have system failure probabilities so low that there is no practical way of demonstrating them at a system level during the development and flight test. To build up a confidence level that will enable such systems to be accepted by the authorities, the normal design process, including failure mode and effects analysis, must be supplemented by exhaustive lane testing and detailed analysis of the critical areas where common failures might occur. This process would include an extensive programme of rig operation to demonstrate the basic equipment reliability at a low level closed loop simulation of the system performance and checking this on an aircraft using real hardware. The simulation should be carried out so that a very large sample of the aircraft operation is represented and the effects of turbulence, system tolerances and failures etc. should be introduced on a statistical basis. If samples of pure simulation results can be matched with the results of the ground rig operation, and these can be further validated by comparison with flight test results, a confidence level in the system performance can be built up statistically.

Certification for Performance

A particular example of this approach to proving system performance comes from the certification work for automatic landing systems where considerable experience has been built up. This has resulted in BCARs and TSS Standards (references 1 and 2) being produced for the guidance of manufacturers and operators. These documents and the procedures and techniques behind them have been the result of a very considerable effort of collaboration by the manufacturers and the Certification Authorities. It is considered likely that a similar approach to proving system integrity will be required by the authorities for systems employing ACT in its various forms. European military requirements have already been indicated by the MRCA flight control system specifications and by R & D work being performed by or on behalf of the RAE, in connection with fly by wire systems. In January 1974 the Civil Aviation Authority issued Airworthiness Technical Note No. 108, "Certification of Fly-by-Wire and Control Configured Vehicles" (reference 3).

The common elements in all of these requirements are failure risk criteria given in terms of probabilities. A summary comparison is given in table 1. This covers the field of flight control from

FUNCTION	TYPE	FAILURE RISK
Cruise	Torque limited Hard-over monitored	$10^{-3} - 10^{-4}$/hr $10^{-6} - 10^{-7}$/hr
Approach	Monitored Duplex (Fail soft)	$<\frac{10^{-7}}{2}$ /Landing
Automatic Landing	Fail operative	$<\frac{10^{-7}}{2}$ /Landing
FBW/CCV	Military combat FCS Electronics	$<10^{-7}$/hr $\sim10^{-8}$/hr
	Future civil	$<10^{-9}$/hr

Table 1 Flight Control System Requirements

low authority systems to CCV in terms of failure risk. The figures given for automatic landing are less stringent compared with those for fly-by-wire and CCV which are likely to need to survive two failures and maintain full performance. However this comparison could be misleading, since it does not take account of the performance criteria specified for automatic landing. It has to be shown that the probability of the system performance distribution is within the limits laid down by the BCARs. This requires considerable design and analytical technique.

Safety Assessment

The heart of the certification procedure required for automatic landing systems is carrying out the safety assessment. This is another technique which can be taken from the certification process for automatic landing systems and applied to the certification of full-time ACT systems. The major requirements from the system designer's and manufacturer's point of view are listed in table 2. This table also indicates the failure risk associated with the failure effect rating according to BCARs. In addition the safety assessment for automatic landing is required to cover

- Statement of design principles
- Description of system, including safety analysis
- Limitations and crew procedures
- Nature and frequency of ground checks required

THE SAFETY ASSESSMENT	
Requires:- 1) Failure, event & error and effect analysis 2) Performance variation analysis 3) Ground, flight or simulator tests Define possible dangerous effects then trace how these could arise and are countered.	
BCAR	
Requires:-	
*Catastrophic effect –	Failure risk $\ll 10^{-7}$/hr
Hazardous effect –	Failure risk $< 10^{-7}$/hr
Major effects –	Failure risk $< 10^{-5}$/hr
Minor effects –	Failure risk $> 10^{-5}$/hr
*Needs substatiation beyond reasonable doubt.	

Table 2 Safety Assessment

The safety analysis covers

- Active failures
- Passive failures
- Nuisance events
- Performance assessment

The first two are dealt with by rigorous fault analysis of the whole system, emphasising areas of potential common failure. The third has given rise to considerable problems in the past and is mainly a function of the performance of individual elements of the system. In the Concorde AFCS accurate analogue computing and extensive consolidation and precision monitoring have been used to avoid this problem. Performance assessment has been a matter of considerable effort for automatic landing systems, involving thousands of simulated landings and various statistical techniques.

The performance assessment for automatic landing is relevant to the certification of full-time ACT systems and it is appropriate to note some of the detail. The estimated risk of exceeding performance limits must allow for aircraft configuration range, system and sensor parameters, atmospheric effects and ground characteristics.

Dissimilar Redundancy

For full-time ACT systems for commercial aircraft, it seems unlikely that forms of redundancy will be acceptable which cannot be shown to be free from common failure. The Civil Aviation Authority Technical Note No. 108 referred to earlier states under 'Catastrophic Effects' - 'A failure analysis can be used to demonstrate that taking the system as a whole, sequences or combinations of random component failures are indeed adequately improbable. However the common mode type of failure must also be shown not to represent a hazard............ The redundancy in the system must be such that it will continue to provide adequate service in the face of any conceivable disrupting influence in the environment, eg electrical noise, vibration, local fires, lightning strike or any likely combination of these. The redundant elements must not be vulnerable in the same way ...... The conclusion must be that for FBW or CCV where loss of the system would result in loss of the aeroplane, at least the sensing and computing elements must be essentially redundant and dissimilar.' Figure 4 shows some possible dissimilar computing arrangements for a quadruple redundant system. The first has 3 similar lanes and the remaining lane R with different electronic implementation. If there is a common failure in the 3 similar lanes R needs to be given priority and it must also be at least monitored. If a quadruplex redundant system is split into two dissimilar pairs there is a problem in identifying the failed pair. The possibility of making all four lanes dissimilar would clearly be extremely expensive in development and in production and would also introduce larger tolerances between lanes with their consequent problems. It is likely to be more fruitful to pursue the possibilities of multiple control surfaces and the use of dissimilar surfaces such as spoilers and ailerons. This seems to be the most likely way to avoid similar redundancy in the actuation system.

The Aircraft Environment

If it is assumed that the necessary level of performance can be provided then the most important features of the system are the reliability and integrity. The reliability requirement per lane may not be excessively high to meet the overall system failure probability target if say four lane redundancy is used. However if the equipment does not have a high standard of intrinsic reliability, it will not only be unacceptable from the maintenance viewpoint (and probably long term cost of ownership), but it will be unlikely to gain acceptability with the users since it will not engender a high confidence level. Any significant nuisance failure rate will also aggravate these effects. Integrity is the key to overall safety and for full-time ACT applications no compromises can be allowed. The biggest potential danger to integrity is the common failure hazard.

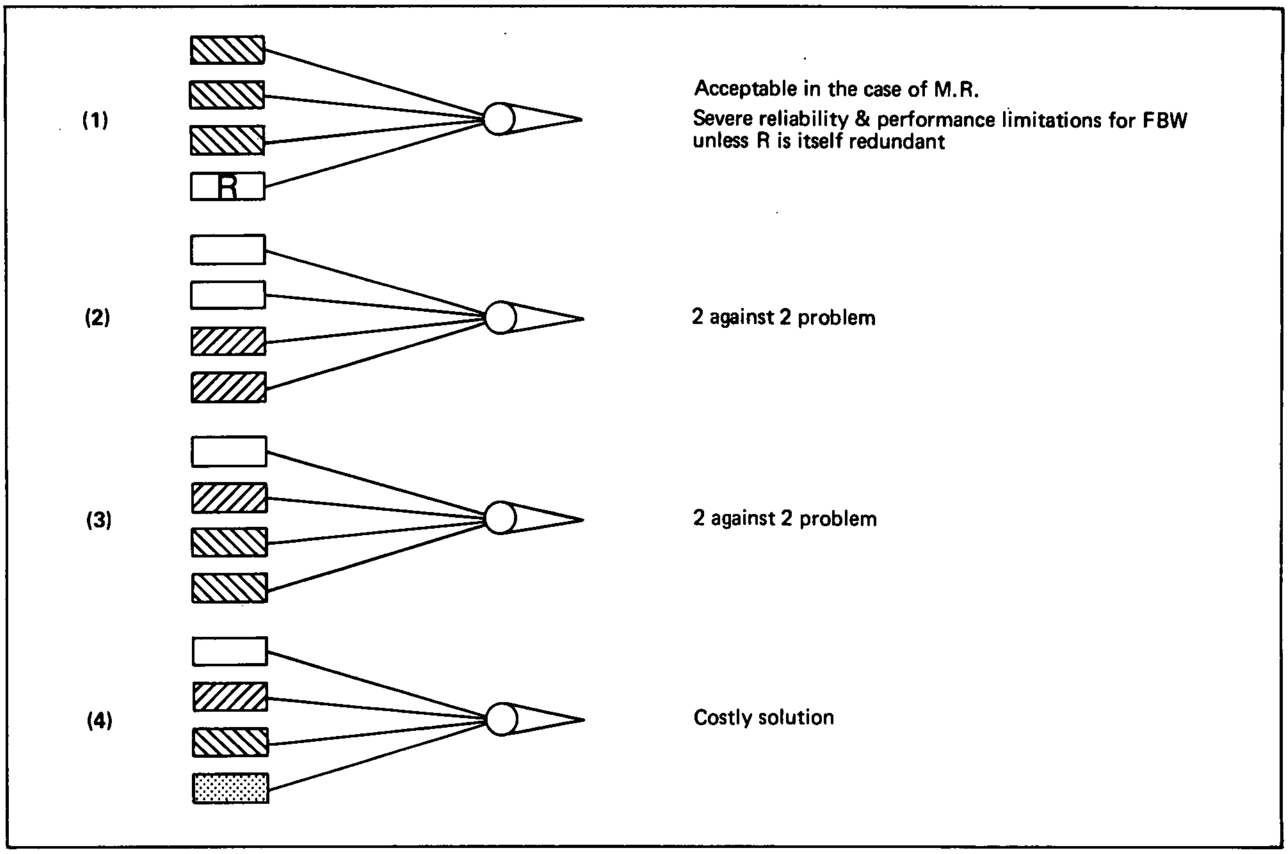

Figure 4 Problems of Dissimilar Redundancy

The recent trends in specifying airborne electronic equipment have been towards higher reliability, improved maintainability and less susceptibility to environmental effects. However the specified environments have hardly become easier to survive; the power supplies and the power transients have not become kinder to equipment; aircraft carry more and better sources of electromagnetic interference.

Improvements in these areas would have an enormous effect on the reliability and particularly the integrity of the system. It is here that the airframe manufacturers can make a main contribution. When systems are being asked to provide levels of integrity never before achieved it is surely inappropriate to call up the standard environmental specifications and allow the standard power supply variations. It is even less reasonable to call up temperatures of +95 deg C and vibration levels of 10g together with 28 volt supplies that may fall to zero for significant periods. Systems vital to the safety of the aircraft merit special consideration. For instance, typical component reliability figures suggest an improvement of the order of two if an operating temperature of +71 deg C is reduced to +40 deg C. It is also good practice to design equipment to withstand the specified vibration level with an adequate margin but the lower the real aircraft environment the higher the safety. Consideration should be given to isolating vital systems from the supply transients resulting from other systems.

6. EXTENSION TO FULL-TIME ACT

The current state of the art in fly-by-wire systems now in quantity manufacture for in-service use (as distinct from experimental prototype systems) is exemplified by the MRCA and Concorde. These have a level of system redundancy and aircraft electrical and hydraulic power supplies making them fully fail-operative after one failure and they have mechanical reversion. In each case the handling qualities of the aircraft are determined to a major extent by the AFCS and they were designed that way from the beginning. They must therefore qualify as examples of ACT technology and it is interesting and relevant to look at what must be done with this type of system if the mechanical reversion is to be deleted.

It is more relevant at an AGARD meeting to consider military applications so a generalised single fail-operative system of the type used on MRCA (see figure 5) will be looked at to see what needs to be done to extend this to a full-time ACT application by deleting the reversionary mechanical link as shown in figure 6.

The main areas for consideration are the level of redundancy, the signal selection and consolidation techniques in the computing, the sensors and the actuation system. The level of redundancy expected to be required is quadruple redundant or its equivalent which will give safe operation after two failures. The signal selection and consolidation have to detect and isolate failures without any significant nuisance

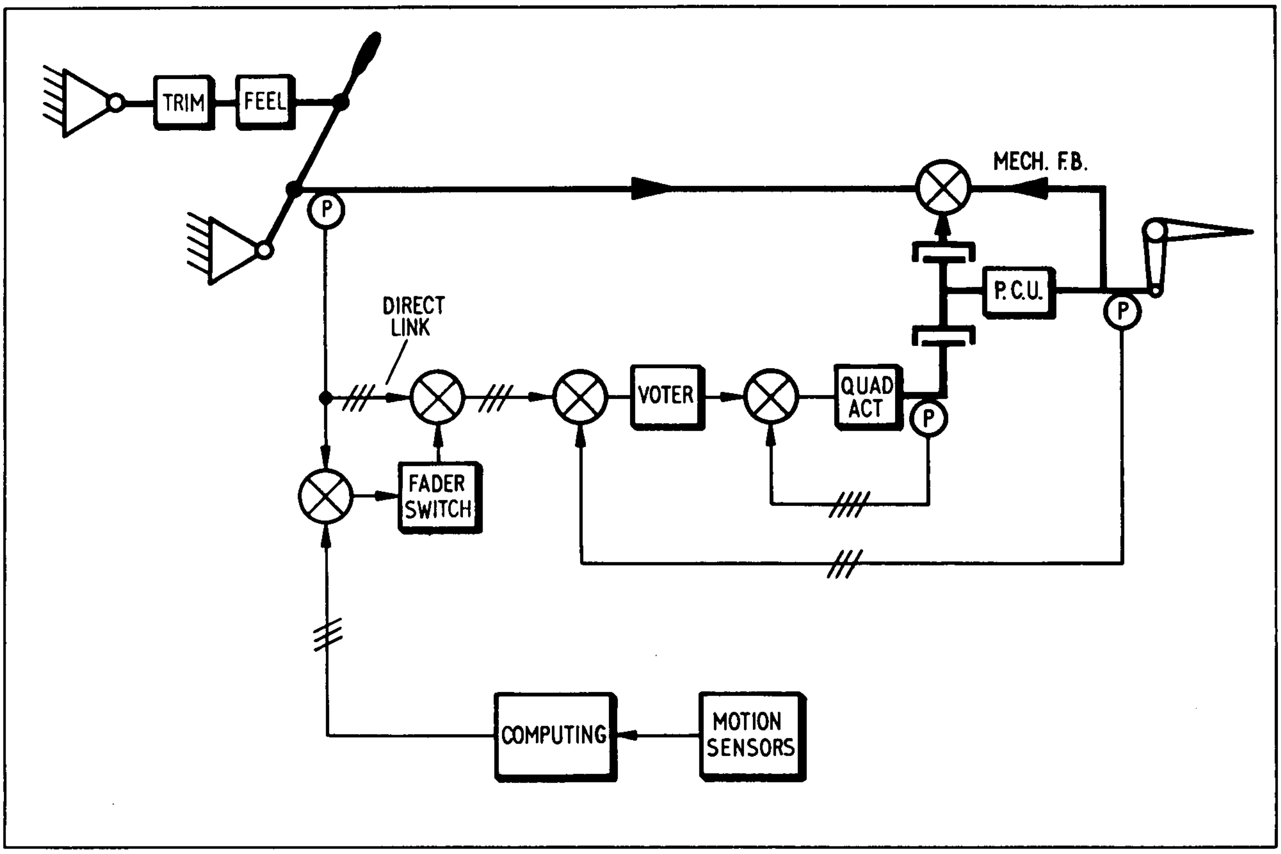

Figure 5 Fly-by-wire with Mechanical Reversion

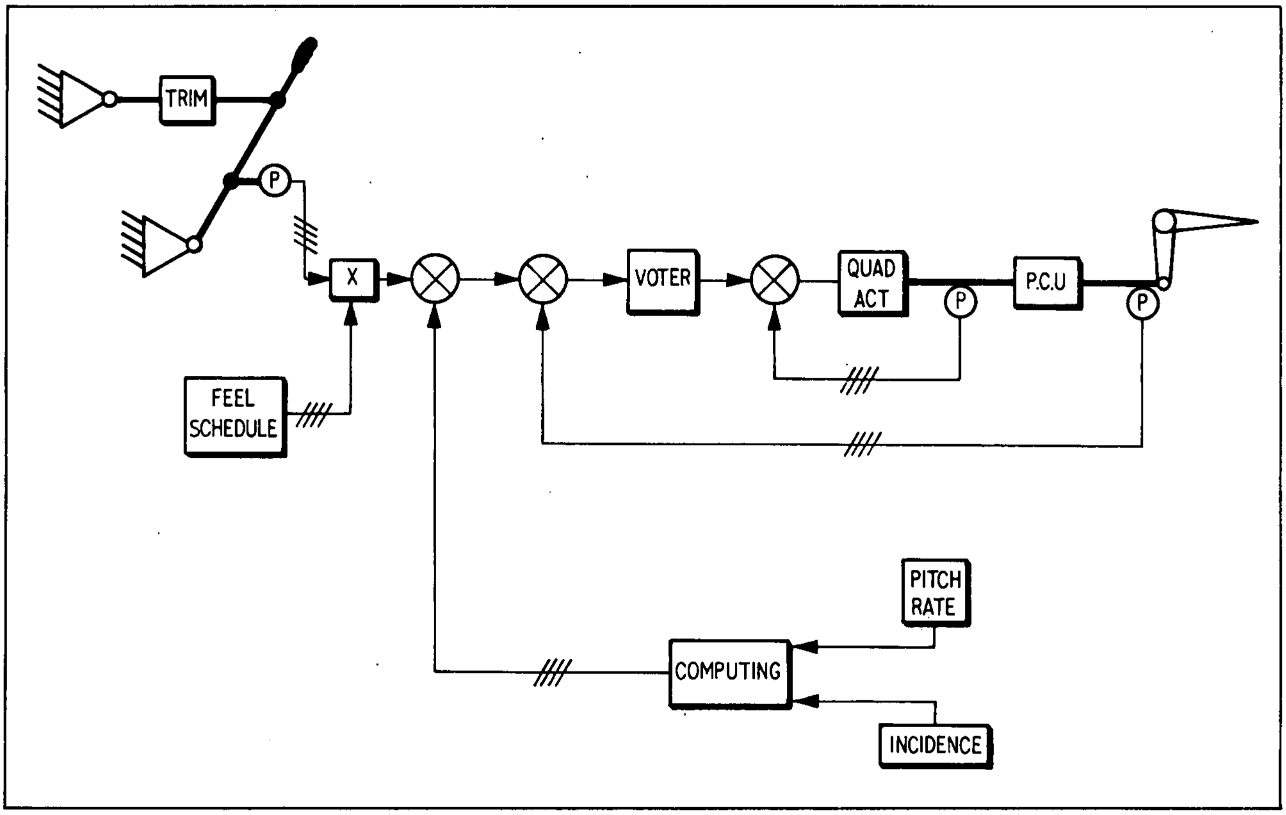

Figure 6 Full-time ACT System

disconnect probability and reduce tolerances between lanes. Most of the work to date has been related to voter-monitors in triple redundant systems.

The choice of algorithm depends particularly on the trade-off between transients resulting from failures and the nuisance disconnect probability - there is no unique solution. The high control effectiveness low-level strike aircraft may require a different solution to a heavy lift helicopter with its relatively low response to control inputs.

Sensors

The type of sensor for ACT are already in current use in military aircraft applications. Depending on the particular system the main elements of a high integrity ACT system will almost certainly include some of the following sensors:

- Rate gyros
- Accelerometers
- Position sensors
- Incidence/Sideslip sensors
- Air data capsules
- Attitude gyros

The main parameters in these sensors, necessary for a successful ACT system design, are performance, integrity, reliability and above all matching of outputs to ensure close tolerances.

In the case of rate gyros and accelerometers these have been developed over many years and the system designer can easily obtain the necessary bandwidth and resolution for ACT; however, for application to highly manoeuvrable aircraft, such as lightweight fighter aircraft, additional constraints are put on the sensor designer. The sensor outputs must track to acceptable accuracy, typically ±2·0%, both in steady state and dynamically changing situations when high angular rates and accelerations are being applied in axes other than the prime axis of measurement of the sensors. Thus a large number of performance parameters must be specified and controlled to ensure that the total tracking error between sensors is limited to an appropriate share of the system nuisance disconnect budget. The following parameters are among those specified and totalled to give an rms measurement error, with static tracking and dynamic tracking considered separately

- Scale factor error
- Linearity
- Zero offset
- Hysteresis
- Bandwidth (damping, natural frequency)
- Resolution
- Cross axis sensitivity to rate
- Cross axis sensitivity to accelerations
- Variation of parameters with temperature and vibration.

The main features of the sensor will be determined by system requirements, for example a self test facility is essential for preflight test and BITE, this may be extended to full-time on-line monitoring of gyro wheel speed and direction at the expense of increased system complexity. The difficulty is in obtaining and proving 100% monitoring and the required reliability may be more easily and certainly obtained by adding additional sensors to the system.

Position sensors must be carefully designed to avoid common failures; for example a multiplexed pilots stick position sensor with one drive shaft might dominate the reliability summation for an entire system. Although position sensors are relatively simple in principle, care must also be taken to control the output tracking since often there are many position sensors in one system, all contributing to interlane disparities.

In aircraft systems with relaxed static stability the provision of incidence sensors presents a special problem. The siting of the devices on the aircraft must be sufficiently close to minimise disparities due to asymmetric airflow during sideslip or flight refuelling/formation flying but sufficiently separate so that the probability of common failure due to bird strikes is vey remote. There is additionally the problem of measurement accuracy and tracking at low speeds and near the stall.

Similar care must be taken with provision of air data inputs to multiplexed systems which employ extensive gain schedules as functions of dynamic or static pressure. Simple capsules are usually specified which are connected to different pitot/static tubes. Pressure error correction is not applied since tracking is more important here than absolute accuracy, and the problems of dynamic tracking during manoeuvres and in the transonic region can be solved by suitable smoothing filters on the outputs. The resulting tolerances can be absorbed by the use of a voter immediately after the smoothing filter.

Actuation System

The actuation system is the last point in a redundant system and therefore must involve some form of consolidation and has to cope with any residual tolerances in the incoming and feedback signals. It also has to convert low level electronic signals into output movement at power levels up to 1000 kg. Compatibility with hydraulic supplies, integrity and vulnerability to battle damage are significant aspects.

The quadruplex electrohydraulic actuation system shown in figure 5 was designed to work from two hydraulic supplies since this has been normal for military aircraft. It was packaged into a small volume for ease of installation and integration with power control units. Figure 7 shows this actuator mounted separately on a manifold for rig testing. Figure 8 shows it integrated with a power control unit for an application with mechanical reversion after two failures. Figure 9 shows a similar installation where there is no mechanical reversion. Comparison of the last two figures shows the additional complexity in the unit to provide the changeover from electrical inputs via the quadruplex actuator to manual inputs via the control runs as well as the mechanical feedback required for manual operation. This quadruplex actuation system can be extended without major change to meet full-time ACT requirements.

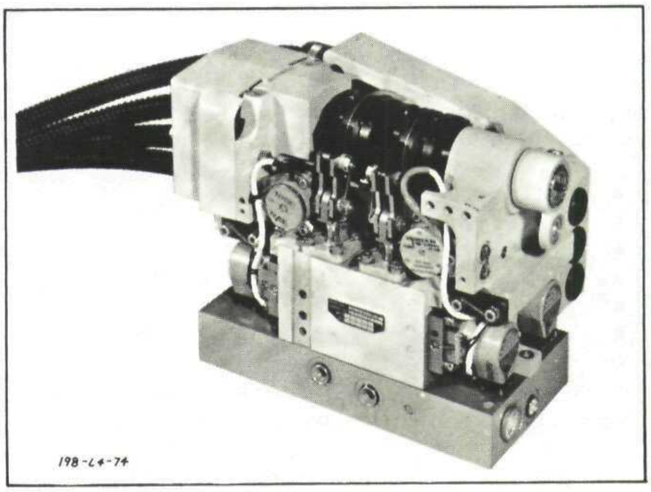

Figure 7 Quadruplex Electrohydraulic Actuator

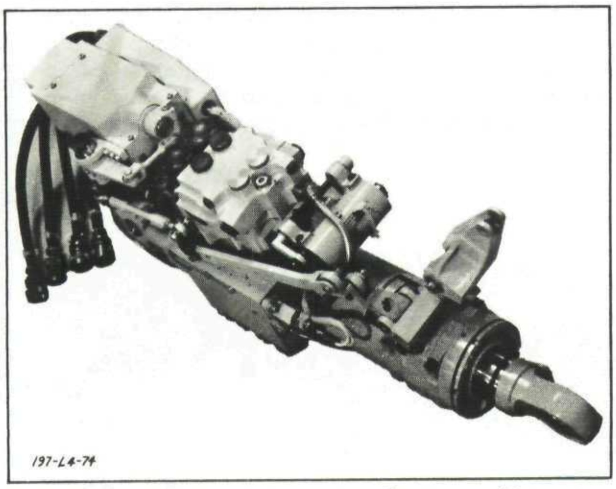

Figure 8 Integrated Actuation System with Mechanical Reversion

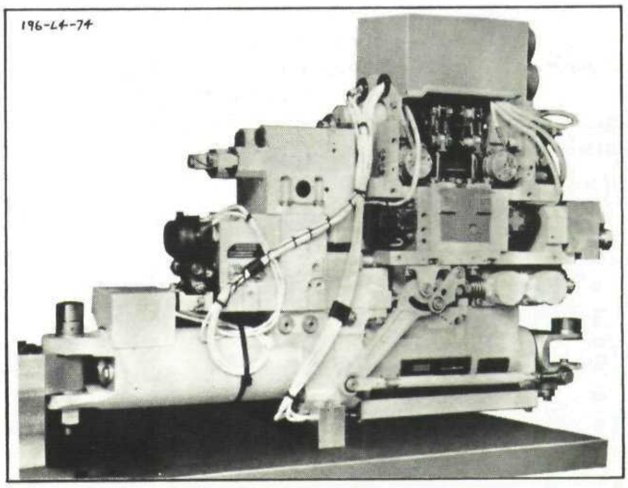

Figure 9 Integrated Actuation System without Mechanical Reversion

The choice of actuation system is dominated by the hydraulic supply arrangement. If in military aircraft the provision of two hydraulic supplies is adequate for the overall integrity of the aircraft then the combination of lane failures upstream of the actuator and an hydraulic supply failure must give a failure rate considerably lower than the probability of two hydraulic supply failures. On this basis a quadruplex actuator fed from two hydraulic supplies is adequate provided the pair of actuators remaining after a single hydraulic supply failure are monitored and a further individual actuator failure can be identified and isolated. The provision of a third hydraulic supply with changeover switching or the provision of four hydraulic supplies will give a reduction of between 10 and 50 in system failure rate, assuming a quadruplex actuation system. The ratio is determined mainly by the hydraulic supply failure rate which may differ considerably for different types of aircraft. For military aircraft, vulnerability to battle damage has become an important criterion which is closely related to integrity.

Integrity has been brought out several times in this paper as the prime design area affecting the final acceptability of a redundant system. This is particularly true for the actuation system. An interesting example is the British Aircraft Corporation VC-10 which uses split surface elevators and ailerons, spoilers and a trimming tailplane. Each split surface is separately and locally powered by electric motor/ hydraulic pump converters mounted integrally with the simplex jacks. This arrangement gives good integrity and its value for military applications in providing low vulnerability to battle damage was subsequently recognised, and R & D work and flight test was carried out on this 'power-by-wire' system. It is also interesting to note that on the VC-10 the duplicate-monitored autopilot provides output signals direct to these split surface integrated actuation systems and for integrity reasons the two halves of the autopilot were installed on opposite sides of the aircraft.

In this section we set out to establish what needs to be done to extend a current system to full-time ACT application. Quadruplex redundancy is expected to be required but it must have adequate integrity and meet vulnerability criteria for military use. Position, rate, acceleration and incidence sensors are available. The mechanical interface and integrity aspects of position sensors need improving. Rate and acceleration sensors need to be improved or redesigned to give reduced tracking errors between individual sensor outputs. Consideration needs to be given to the installation and vulnerability problems of incidence sensors. The actuation system needs additional pressure and position sensors to allow further monitoring of lane failures after loss of a hydraulic supply. The main task in the computing is to choose signal selection and consolidation algorithms suitable to the particular requirement. Perhaps the likely move to digital computation is the most significant step and this is considered in more detail in the following section.

7. DIGITAL ACT

The past work which forms the main background for the successful development of ACT used analogue and logic techniques. The experience of designing, manufacturing and using digital avionic systems is valuable but little or none of this has been critical to flight safety or has used redundant systems. Current automatic flight control systems are being implemented with digital techniques and already some rules are being established for the choice of processor, the software design and solving the environmental problems.

Task Oriented Processor Design

Over the past ten years we have developed a family of general purpose organised whole word digital processors designed specifically for avionics applications such as nav/attack, weapon delivery, head-up and head-down displays, air data systems and automatic flight control. These are referred to as a family because their instruction sets are subsets of a comprehensive master set which has been evolved in the design and use of ground based computers over many years of on-line control and computation experience. In this way good assembling, simulation and diagnostic facilities can be provided without substantial software investment for each new application. Each instruction subset is optimised for the particular application. Because the processors are designed for each specific application, the most appropriate component technology can be selected for a new design. The main design aim for each application is to minimise the hardware content to achieve high reliability and low cost, weight and volume. We refer to processors designed by this means as 'task oriented processors'. We believe that this approach should be further pursued for the development of digital ACT systems.

Design Aspects of Digital ACT

ACT systems require extensive signal selection, failure monitoring and automatic failure isolation facilities - these are usually a dominant factor in defining the processor to be used and its time allocation. A large number of input signals, computed integral values and output commands require consolidation and monitoring. A reasonably sophisticated signal selection and failure monitoring algorithm has to be employed in order to achieve the requisite failure transient performance while eliminating potential nuisance disconnects. If these algorithms are performed with hardware they have to be provided for each signal and therefore require a large amount of equipment. A non-task oriented general purpose computer would be most unlikely to provide an optimum solution.

The maximum iteration time allowed will be determined by system dynamics, while the minimum will be determined by the number of obeyed instructions and computing speed. It is therefore necessary to optimise, as far as is practical, the order code and store access time.

Analogue systems have many attractive properties which have become apparent during the development of failure survival automatic flight control systems and the detailed work carried out to achieve their acceptibility for in-service use.

These are -

- Ease of analysis
- Functional modularity
- Techniques widely understood
- Simple failure/effect analysis
- Straightforward failure path tracing
- Localised high integrity areas
- Limitations well understood

Considerable experience has now been gained of digital design in avionic and flight control systems and it is clear that these properties of analogue redundant systems are desirable in digital flight control systems. This is being sought by both software and hardware design techniques. An important principle is that of simplicity - the processors should be no more complicated than is necessary to perform the required functions. Consideration should be given to multi-processors in complex systems to enable specialised tasks to be performed by dedicated task oriented computers. Modular memory design is helpful and protection techniques such as overflow protection should be taken into account in both memory, register and arithmetic unit design. The design of the microprogram should also be rigorously checked for potential failure cases.

Software

The computational functions performed by an ACT flight control system may be categorised as follows:-

- Control laws
- Mode logic
- I/O control
- Signal selection and monitoring
- Synchronisation
- Self test
- Preflight test

This information may be derived from a system specification. An overall flow diagram can then be designed from which the specifications and flow diagrams for each sub-routine, program block etc. can be produced. Programming then proceeds with the aid of the following:-

- Host computer
- Flight control processor simulator
- Assemblers
- Compilers/compactors/interpreters

For high integrity redundant systems the following should be considered:-

- Common fault protection
- Programming control
- Fault path tracing
- Program monitoring
- Specialised programming methods
- Representativeness of host computer
- FCS computer simulator
- Integrity of support software

From these considerations and also to maintain the attractive properties and design visibility of analogue redundant systems in the software, various techniques have been developed. The program can be split into simplex analysable modules which can be thoroughly tested. These modules are rigorously tested according to specifications produced independently of the module design specifications. These tests are programmed independently of the module programming. Multiple methods of rigorously testing the flight resident program ensemble should be adopted. There should be minimum reliance on support software to avoid any hidden faults which these may contain. There must be strict configuration control and procedures.

Environmental Considerations

A particular problem area with digital equipment is electromagnetic compatibility (EMC) and it is a major potential common failure hazard. In current digital flight control applications where the computer units contain interface equipment and power supplies in addition to the processor and store elements considerable attention is paid in the detail design to EMC. Protection is achieved by widespread use of EMC gaskets, the avoidance of holes in the chassis or the restriction of their diameter, the introduction of a large number of filters and the choice and protection of data transmission lines. Another major common failure problem is the effect of lightning. A suitable choice of aircraft installation for vital equipment gives considerable protection against lightning effects. Equipment close to the centre line and away from extremities and junctions is less vulnerable and cable routeing and shielding is important. Internal filtering to avoid aliasing must be used in digital systems.

One valuable technique which is already being used in redundant digital flight control is the use of fibre-optic links for interlane connections. This provides a high integrity connection between the independent lanes of a redundant system which is impervious to electromagnetic effects and protects the redundant elements from common failures.

CONCLUSION

The intention of the paper was to show how the experience gained from the development of failure-survival automatic flight control systems, started in the 1950s and still continuing, is relevant to the future development of full-time active control systems. This has resulted in equipment and system design techniques and disciplines which, coupled with current further investigations and development, will enable full-time active control systems to be successfully developed in a reasonable timescale at an acceptable risk and cost.

References

1) BCAR 367 Airworthiness Requirements for Automatic Landing including Automatic Landing in Restricted Visibility down to Category 3.

2) TSS Standard No. 1-2

Supersonic Transport Aircraft Landing in Low Visibility Conditions

3) Airworthiness Technical Note No. 108

Certification of Fly-by-Wire and Control Configured Vehicles

PRODUCTION DESIGN REQUIREMENTS FOR FLY BY WIRE SYSTEMS

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SUMMARY

The problems of specifying design requirements for production Fly-By-Wire (FBW) flight control systems are addressed based on current state-of-the-art trends. The design goals and requirements of two development FBW programs are reviewed. Emphasis is placed on the impact of specific requirements on hardware mechanization complexity. Of particular interest is the sensitivity of FBW system design to safety, survivability and mission reliability requirements, and to related subsystem and interface concepts. Experience to date is used to provide recommendations and insight into specifying practical design requirements for production FBW systems.

I. INTRODUCTION

The synergistic characteristics of FBW flight control systems coupled with the potential performance advantages of advanced technology aircraft designs are dictating the use of FBW flight control systems for future aircraft. An FBW system is an airborne vehicle primary flight control system which provides the basic means of aircraft control and handling qualities with no mechanical interconnection between the pilot's station and the moment producing devices that control the aircraft. The system consists of primary power sources, aircraft motion and pilot input sensors, electronic control law computation, moment producing devices, and appropriate controls and displays. FBW systems can provide improved handling qualities over conventional control augmentation systems which are compromised by the characteristics of the primary mechanical/electrohydraulic controls. Furthermore, an FBW system's control laws can be changed in flight to provide optimum handling qualities (multimode control) for different piloting tasks, and they are easily integrated with other automatic functions such as outer loop flight control, fire control and propulsion systems. Unfortunately, loss of an FBW system results in loss of the aircraft and possibly loss of crew and/or passengers. This disadvantage might easily outweigh the previously cited advantages were it not for advanced vehicle designs that are not manageable without automatic control. Production FBW systems will therefore be required, and it is of utmost importance that design requirements be developed which will result in practical production systems in terms of size, weight, power and cost of ownership.

This paper attempts to provide some guidelines for the generation of design requirements by reviewing the results of two development FBW programs and the potential impact of requirements on system and hardware complexity. Specific design requirements will vary with many factors: commercial versus military application, mission time, technology employed, such as analog versus digital, etc. However, it is believed that the areas discussed here are generally applicable to all FBW system specifications.

II. OVERVIEW OF TWO U.S. FBW PROGRAMS

Although many programs have contributed to FBW technology, two programs with which the authors are personally familiar are the 680J F-4 Survivable Flight Control System (SFCS) and the U.S. SST. Experience on these programs, one military and the other commercial, form the basis for much of the discussions of this paper.

680J Program

The 680J F-4 SFCS (Reference 1) was sponsored by the USAF. This program had the broad objectives of improving combat survivability, proving FBW system performance, developing pilot acceptance, and demonstrating the practicality of the FBW flight control systems. The system employed quadruplex redundancy for all flight safety essential sensors and computation and actuation elements except for the surface actuators which were the conventional F-4 dual tandem actuators. Four hydraulic supplies were employed for the FBW secondary actuators. A dedicated transformer-rectifier (TR) and battery for each channel formed the electrical system. The system included Built In Test Equipment (BITE) which performed complete preflight test and flight-line maintenance functions. A summary of design requirements pertinent to the discussions of this paper is presented in Table I.

Mission reliability was not specified; however, based on safety considerations, a quadruplex - two fail operate - fail safe system was specified. With the complexity of functions as implemented for the development testing, quadruplex redundancy would have been required in any event to meet a probability of loss of the FBW system for a 1-hour flight of less than $2.3 \times x\ 10^{-7}$. This value was established as an objective in the early phases of U.S. FBW studies. The quadruplex system as designed had a calculated value of 1.24×10^{-10}. In addition to the normal FBW mode, which provided NSS characteristics in the air and speed stability during TOL, two backup modes were required. The pilot's emergency switch could revert the system to electrical backup which removed all closed loop aircraft state sensor inputs and converted the system to direct electrical links from pilot controls to surfaces. In addition, another set of pilot switches provided a means to remove all monitoring and logic, and to demand all channels "ON" independent of failure status. As discussed later in this paper, these provisions complicated the system, reduced normal mode reliability and were not considered effective.

Channel isolation was achieved by packaging each channel in separate line replaceable units (LRUs), physical separation of wire bundles, and dedicated channel connectors on each LRU. Some passive and some active buffering was employed to isolate interchannel logic and signal points after entering the LRU to meet wire-to-wire and wire-to-ground single connector shorting requirements.

TABLE I

SFCS SPECIFICATION
SUMMARY

Mission Reliability	Not specified ($< 2.3 \times 10^{-7}$ failures in 1 hour flight implied). MTBF = 400 hours
Redundancy	Quadruplex
Modes of Operation	Normal (NSS and TOL)* Electrical Backup (EBU) Demand ON
Channel and Axis Isolation	
Physical and Electrical	To the greatest Extent Practical
Connector Shorting Rule	Applicable to any one connector (wire to wire or ground)
Transients (pitch)	1st Failure .5g 2nd Failure .5g 3rd Failure 1.0g
BIT Failure Detection Probability	
Preflight	.995
LRU Isolation	.95
Special Quality Requirements	
Parts	Max Hi Rel
LRU Burn-In	100 hrs (failure allowed)
Acceptance	LRU and System Tests
Qualification	Full Spectrum
*NSS - Neutral Speed Stability, TOL - Take Off and Land	

Transient requirements, particularly on third (last) failure, complicated the system design by forcing extremely close channel tolerance control and disallowing the use of actuator equalization (balancing). This resulted in the necessity for interchannel signal selection circuitry at the servo command points.

Quality assurance requirements were necessarily severe and, in retrospect, would appear to be justified. Use of high-reliability parts, burn-in and system, as well as LRU acceptance, testing techniques were employed. System testing, in particular the automatic built-in self-test, proved invaluable in trouble-shooting and weeding out marginal equipment.

The four flight safety essential electronic LRUs for the SFCS, employing analog circuits (vintage circa 1970), weighed a total of approximately 116 pounds.

U.S. SST Program

The U.S. SST (Reference 2), cancelled in early 1971, was to incorporate a quasi FBW control system. At the time of cancellation, breadboard systems had been built and prototype equipment characteristics defined. On a functional basis the SST FBW system was perhaps 20 percent more complex than the 680J. However, the mission reliability and safety requirements to satisfy a commercial passenger environment dictated a much more sophisticated redundancy configuration. A summary of pertinent design requirements is presented in Table II. The safety requirements forced not only quadruplex redundancy but in fact two quadruplex systems. The Electronic Command and Stabilization System (ECSS) was the primary quadruplex FBW system. This system was backed up by a quadruplex Hardened Stability Augmentation System (HSAS) which, in normal operation, was negated by the ECSS. Upon loss of ECSS, the HSAS provided reduced, but acceptable, handling qualities. The relative complexities of the ECSS and HSAS mechanizations resulted in a mission reliability specification for the HSAS of less than three failures per 10^{10} flights or only 2 to 1 better than the ECSS. It is obvious that the HSAS was not justified on a mission reliability basis. Safety considerations and the psychological attractiveness of a backup system resulted in the requirement for the HSAS.

Isolation requirements were similar to those of the 680J SFCS except for connectors where all wires to any box could simultaneously short to each other and/or to ground. The impact of this requirement will be discussed later. Transient requirements were similar to those of 680J SFCS. Quality requirements were similar, except for a more severe burn-in requirement of 50 hours with no failures versus 100 hours with allowable failures. System test by the vendor was not required but would have been conducted by the prime.

Although the 680J and SST requirements would appear to be about the same, major differences in overall system complexity result from different safety requirements. For example, the SST mission essential electronics were estimated to weigh 309 pounds versus 116 pounds for the 680J. This order of magnitude of complexity propogates throughout the entire system, including interchannel wiring, cockpit real estate for controls and displays, power requirements, aquisition cost, increased maintenance cost, etc. These penalties should be recognized and justified on the basis of careful analysis and judgement of need.

TABLE II

SST FBW SPECIFICATION SUMMARY

Mission Reliability	
ECSS	7 failures in 10^{10} flights of 2.71 hours
HSAS	3 failures in 10^{10} flights of 2.71 hours
Redundancy	Quadruplex
Modes of Operation	Normal (NSS and TOL) HSAS BACKUP
Channel and Axis Isolation	
Connector Shorting Rule	Applicable to all connectors on an LRU (wire to wire or ground)
Transients (pitch)	1g
BIT Failure Detection Probability	
Preflight LRU Isolation	To meet mission reliability Not specified
Special Quality Requirements	
Parts LRU Burn-In Acceptance Qualification	Max Hi Rel 50 hours without failure LRU and System Tests Full Spectrum

III. DESIGN REQUIREMENTS

FBW design requirements must be explicit and realistic if practical system hardware is to result. Motherhood words such as "adequate" or "to the greatest extent practical," found in most specifications for conventional systems, cannot be used without explicit definition when specifying flight safety essential requirements. In both the 680J and SST programs the definition of word meanings, allowable failure modes of detail components, methods of analyses, etc, represented major program efforts. The ideal specification would avoid those interpretation problems. Perhaps more important is the fact that requirements must be realistic in terms of a-priori knowledge of the impact of requirements on mechanization complexity. It is not within the scope of this article, nor is it possible, to develop universal design requirements. Rather, the following discussions address those areas of design which are peculiar to redundant FBW systems, and attempt to provide some recommendations and/or insight into the specification of practical design requirements.

Channel Isolation

Isolation between individual channels of an FBW control system is required to prevent a failure in one channel from propagating to another. This isolation applies to all flight critical components from the basic power sources to the electronic subassemblies, and must include aircraft wiring. Isolation requirements may dictate separate channel connectors on each unit, separation of wiring bundles, physical separation of redundant electronic elements and buffering of all cross-channel logic and signal communication wires. Cross-channel signal and logic communication is necessary in a redundant system in order to perform monitoring and equalization functions. Both the 680J and SST systems employed Signal Selection Devices (SSDs) in each channel to perform these functions. An SSD collects the signals from all channels and selects one of the signals, (for example, the least mid value) as its output. Thus, the signals downstream of the SSDs in each channel are made essentially identical; whereas the input signals may be quite different due to tolerances and/or failures. Monitoring for failures upstream of an SSD is, of course, required. In addition, since all four channel signals are brought together at the SSD input, a potential single failure point exists, and isolation to prevent propogation of a failure is required. To demonstrate how the degree of isolation can affect the complexity let us look at two different specific requirements for channel isolation.

Case 1 (SST) - The system shall be designed to allow all input/output wiring to a single channel to be simultaneously shorted to each other, to ground, or to the input power source high side without affecting the remaining operational channels.

Figure 1 shows a simplified block diagram/schematic for the SSD which is employed to make the correct choice of the operational channels, process this signal to downstream circuitry, and provide the necessary cross-channel interface.

The SSD functions in the following manner:

- The input signals transmitted to the other three channels are buffered by active circuits.
- Power to each of the elements of the SSD is channelized, with power being shipped to other channels and received from other channels through current limits.

- Monitoring is performed through the combined action of the comparator, the ON/OFF switch and the oscillator/detector. The power switch is turned off if the comparator indicates a failure by stopping the oscillator for a preselected time period.

- A failure detected in one channel causes the power for that channel to be removed from the remaining operational channels through the action of the power switch.

After many interations to refine the design, the resultant SSD implementation used the following components:

10	Operational Amplifiers
40	Transistors
14	Capacitors
68	Resistors
15	Diodes
10	Zeners
157	Total Parts

Case 2 (680J) - The system shall be designed to allow all input/output wiring of any single connector of a unit to be shorted simultaneously to each other, to ground, or to the input power source high side without affecting more than one channel.

Figure 2 shows a simplified block diagram/schematic for the 680J signal selection device.

Since each of the individual channel signals and logic can be wired through separate connectors on each unit, the internal SSD isolation protection of Case 1 is not required. Resistor buffers can be used in place of active buffers, the current limiters may be eliminated, and the comparator can be simplified.

The 680J SSD implementation required only the following components:

5	Operational Amplifiers
4	Transistors
4	Capacitors
23	Resistors
4	Diodes
2	Zener Diodes
42	Total Parts

or a four-to-one improvement.

The complexities associated with redundant systems to isolate channels with crossties has long been recognized. Considerable effort has been put into minimizing crossties and thus avoiding the problem. Recent work on fiber optic data transmission lines represents a perfect solution to this electrical isolation problem. In addition, EMI susceptibility and generation problems are avoided as well as electromagnetic pulse-induced core currents caused by lightning or nuclear blasts. Although this technique is particularly applicable to digital systems, it may be applied in analog FBW mechanizations where multiplexing of interchannel data is deemed practical.

Electrical Power

The electrical power interfaces and channel power supplies are responsible for a significant portion of the design task and resultant hardware penalties associated with an FBW system. FBW systems are redundant and, in general, require independent sources of electrical power for each channel. In addition, traditional aircraft power sources are of such poor quality that the channel power supplies must not only generate and regulate computation and logic circuitry power, but must also generate high quality ac excitation power for sensors and position pickoffs. The complexity of the channel power supplies increases as the quality of aircraft power supplies decrease. For example, assume an FBW channel were to be interfaced with an aircraft dc power source which could have transients allowing zero voltage conditions for 50 milliseconds under normal operation. Furthermore, assume that the FBW channel must perform within specification without producing transients during the 50-millisecond power outage. If capacitors were employed to provide the required hold-over capability, several thousand microfarads would be required.

The power supply system for the 680J program is illustrated in Figure 3. This system isolated each channel from the normal aircraft bus systems through separate transformer-rectifiers to dedicated channel batteries. The batteries, in addition to providing power for engine-out conditions, provide a high quality transient-free voltage source. All channel ac and dc power requirements were generated by the channel power supplies. Incidentally, preflight test was required to verify the power system, including the battery status. The NASA F-8 FBW program followed a similar philosophy, except precise ac supply requirements were generated by dedicated ac inverters rather than being built into the FBW electronic units.

The SST power system was similar to that of the 680J system with all power supplied from a multiplexity of dc busses. Each ECSS channel was assigned two busses (Figure 4) which were ORed by the channel power supply which then generated all sensor ac power and all dc computation and logic power requirements. Loss of a single dc source did not result in loss of any function or channel. Loss of a second buss resulted in loss of only one channel. Each HSAS channel was assigned two dc busses and a dedicated battery which was of lower (but adequate) voltage than the dc busses. Loss of power to the channel required two bus failures and a battery failure. No single failure in the ECSS or HSAS power supplies could backfire and result in loss of two busses. This was accomplished by both circuit breaker protection at the busses and current limiting the channel power supply input circuits.

One cannot generalize requirements in the electrical power area; however, over-specification in this area will result in significant penalties.

Spatial Separation

The concept of isolation and, in particular, of spatial separation of redundant elements to improve survivability has been a major impetus in the development of FBW systems. Through this technique the system could be made less vulnerable, for example, to small arms gun fire in the case of the military, or to fire and Piper Cubs in the case of the SST.

Both the SST and 680J programs attempted to achieve maximum spatial separation. In both cases, electronics were packaged by channel, and aircraft wiring bundles were channelized and separated to the greatest extent practical. Redundant sensors were contained in common packages for practical reasons of sensor alignment and tracking. Other redundant items such as actuators and switches were also packaged as units for expediency.

The primary penalties of spatial separation are decreased packaging efficiency and increased aircraft wiring weight. In the case of the 680J, the intercommunication between channels required a total of 316 wires. However, the units were located in close proximity for practical reasons; it kept the wire weight down, but compromised the advantages of physical separation. In the SST the units could be separated by large distances, and interchannel wiring was numerically greater by a factor of four. The wire weight penalty was large enough to seriously consider multiplexing of interchannel data.

For fighters, where space is limited, serious consideration should be given to enhancing survivability by shielding redundant elements either by existing aircraft structure or armor plating rather than spatial separation. For large vehicles the principle of spatial separation of electronics units appears to be valid. In no case does the separation concept appear to be practical for sensors, actuators, switches, etc.

Axis Isolation

Separation of longitudinal and lateral axis flight controls, in terms of failure effects, has been a commonly imposed requirement in non-redundant systems. This separation is not deemed to be necessary in redundant FBW systems. In both the 680J and SST, separation was maintained to some extent. Axis functions were packaged in separate modules. In most cases computation, logic, sensor or servo failures in one axis did not shutdown the other axis. However, power supplies were common, and a power supply failure resulted in shutdown of the failed channel in all axes. The degree of isolation or separation attained was without penalty. A requirement to isolate would have forced duplication of power supplies which would have been a significant penalty. Even more significant, if and when digital FBW systems are specified, axis separation requirements would double or triple the number of computers required.

Backup System Requirements

The concept of using backup systems to assure or enhance safety of flight has a long and successful history, both in conventional aircraft and later in spacecraft. Initial FBW system specifications have included requirements for, and have been designed to include, something called backup. In the case of the 680J, both an electrical backup mode and a demand "ON" mode were implemented. The SST had the HSAS backing up the ECSS. In both cases, the backup provisions ended up complicating the primary FBW circuitry and logic, even though they required using a significant proportion of the primary equipments as part of the backup.

The general conclusion in this area is that backups should not be required unless they can be implemented essentially independent of the primary system. Ideally, separate surfaces should be employed. Otherwise it is probably more cost effective to increase the mission reliability of the primary equipment.

Built-In-Test (BIT)

Requirements for BITE to provide preflight test and LRU isolation capability are an essential part of any FBW system design specification. Mission reliability calculations for each flight must include factors to account for the potential failure status of the system prior to flight. The only practical approach to achieving the required high mission success probability is to require a high-confidence preflight test. In addition, the maintenance of an FBW system would be an impractical task without automatic BIT isolation of failures to an LRU with high confidence level. The problem for the specification writer is how high is high confidence.

The importance of BIT preflight test is illustrated in Figure 5 which shows the relationship between mission unreliability, preflight test thoroughness, and the time at which the flight is taken. The curves of Figure 5 are based on a typical quadruplex system with identical channels, each with an MTBF of 1600 hours, an in-flight failure monitoring capability of 90 percent, and a one-hour mission time.

Four conditions for the BIT system are shown, ranging from zero percent BIT system to one of 100 percent test capability. The 100 percent BIT tested system can demonstrate a constant probability of mission success for each one-hour flight throughout the life of the system. For a system in which a preflight BIT test is not employed, the probability of a mission failure is increased by three orders of magnitude over a 99 percent preflight tested system when the system has accrued 100 hours of on-time. A more dramatic difference is obtained at 1000 operating hours where the untested system has degraded mission reliability by five orders of magnitude. A significant conclusion can be drawn from this data. Preventative maintenance is required for FBW system components which cannot be 100-percent tested. Otherwise the system will have to be overdesigned by orders of magnitude.

The quantitative specification of preflight test requirements in a design specification is probably a mistake. Meeting the specified mission reliability is the real task for the designer. Preflight test is one means to this end result. The final SST specification recognized this point by simply requiring that BIT preflight testing be required, the resultant test system analyzed, and the results included in the mission reliability analysis. A preventative maintenance factory level test was allowed periodically at times equal to equipment MTBF. In the 680J program, where mission reliability was not specified, BIT requirements were quantitatively specified and required to be demonstrated via a BIT verification test. The initial requirements included:

Probability of Fault Detection - .98 with 95-percent confidence

Probability of Fault Isolation - .95 with 90-percent confidence

Probability of a false GO - $< .05$

The fault detection probability of 98 percent applies to faults which may or may not cause a channel to disconnect during the flight. The fault isolation probability is interpreted to mean that if BIT detects a fault, the faulty LRU will be identified 95 percent of the time. The probability of a false GO is a most serious requirement which was recognized to be inadequate and changed to .005. In addition, the requirement was redefined as follows:

"The probability of an existing undetected fault which, in combination with another fault, could result in system shutdown shall be less than .005."

The difficulty with these quantitative BIT specifications and mission reliability specifications is how do you prove that you meet them. The methodology of failure modes and effects analysis for redundant systems is not as precise as it is for single systems. For example, in the design of the preflight BIT for the 680J, all failures that could be identified by all persons involved which, in combination with another failure could result in shutdown, were 100 percent tested for. Therefore, was the probability of a false GO equal to zero? We think so, and the system was cleared for flight on this basis. The 680J system qualification tests included BIT verification testing to test for 98-percent fault detection and 95-percent fault isolation capability. This test consisted of insertion of actual faults, using a cumulative binomial test plan. The test results were 92 and 90 percent, respectively, which were considered acceptable for flight based on an analysis of the types of failures that were not detected and/or isolated. In actual use of the system, no failures were ever incurred that BIT did not identify, and confidence became so high in the BIT capability that BIT was used as the final test on the system.

The BIT equipment can become a significant percentage of the overall system. In the 680J, the BIT electronic parts count represented approximately 15 percent of the total system electronic parts count. In the SST system, the test concept used the automatic flight control system digital computers to program and analyze the test results on the ECSS and HSAS. At program cancellation, it was estimated that this testing would require a minimum of 8000 words and possibly as much as 16,000 words of memory to achieve the required mission reliability and maintainability requirements. In addition, the BIT interface electronics in the ECSS and HSAS would have been close to 15 percent of the total electronics.

Other aspects of BITE to be considered in the design specification result from requiring BIT initially. The BIT circuitry must be isolated from the flight control system so that it cannot inadvertently be initiated in-flight, and must also be sufficiently protected against failure through redundancy compatible with the mission safety and reliability requirements of the flight control system. The BIT system should also provide self-verification testing of sufficient detail so as to not detract from the reliability and safety requirements of the overall system. Self-verification test can be inherent within the design of the BIT computer, i.e., timed sequence, wrap-around verification, or a special test routine could be employed.

Transients

Allowed failure transient magnitudes following channel or total system failures not only can significantly affect complexity, but can actually dictate system redundancy structure. Most of the preliminary FBW specifications included requirements to limit normal acceleration to .05g on first failure. This number, which is equivalent to normal turbulence on a calm day, was soon recognized to be impractical, and a more reasonable .5 to 1.0g evolved in both the 680J and SST programs. Techniques that can be employed to limit transient magnitudes include:

- Flight channel tolerance control with low threshold monitoring and fast switching.
- System partitioning through use of SSDs which effectively block hardover transients and divide total channel tolerances in terms of failure transient effects.
- Equalization of SSDs (not applicable to last failure) to effectively smooth transition from a failed channel to a good channel.
- Use of force sharing/force fighting servos.

In practice, a combination of all these techniques were employed in the SST and 680J systems. The systems were probably overdesigned in this regard. During 680J flight tests where failures were inserted, the pilots could barely feel the transients.

Overdesign resulted from two factors. First, the transient specification was absolute and independent of whether the failure occurred during straight and level flight or during a 6g manuever. Surface effectiveness was assumed worst case. No probability considerations were allowed. Second, the 1g transient specification on last failure precluded the use of equalization of the force summing servo, and forced the use of SSDs at the servo command level to control channel and servo tolerances. The system thus had two voting points, with attendant channel crossties, where one at the servo would have been sufficient to meet other requirements.

In the case of the SST, the high mission reliability requirement dictated the use of SSDs at the servo command point to partition the system independent of transient requirements.

The ideal fail-operational transient specification should be tied to statistical probabilities of occurrence, with recognition that there really is no such thing as fail-safe on last failure for a flight essential FBW system.

Controls and Displays

Controls and displays are necessary elements in an FBW system. The redundancy structure with its multiple sensors, multiple servos, multiple channels, multiple functions and multiple axes opens Pandora's Box for the controls and display designer. In a development program, control and display of status of each element of a redundant system was found to be a valuable aid in troubleshooting the system. For example, the Master Control and Display Panel for the 680J program is shown in Figure 6. Status and reset control of each redundant element was provided. This allowed an experienced system engineer to interrogate the system in conjunction with BITE and isolate failures to a signal transistor switch in channel 3 or a broken wire between channels 1 and 4. To the pilots it represented a nightmare, and numerous complaints of excessive workload were registered.

The SST flight controls which included triplex general-purpose digital computers to perform outerloop control, cockpit management and flight safety essential test functions made available control and display of all these same functions and more, but in a more sophisticated manner. Failure status messages were available in the form of an alphanumeric display on a panel that also contained preflight testing controls. Failures were flashed as they occurred. The pilot could interrogate the display and get successive messages identifying each failure existing at the time. In addition, via another display, pilot required actions such as "divert" or "transfer fuel" were displayed. Several other panels provided warning, control and reset capability for the flight controls.

Space was not a serious problem in the SST, and a multiple crew divided the workload. In a fighter, space is severely limited and pilot workload is already high. Minimization of controls and displays is essential. Use of the aircraft's master warning flasher and display of failure warning messages on the aircraft's CRT vertical situation or multifunction display appears to be the ideal. A dedicated cockpit panel space to control BITE and display go/no-go results is required as a minimum. In addition, LRU failure isolation results must be displayed or recorded, but this need not be in the cockpit.

Quality Assurance Provisions

The traditional requirements for performance testing under laboratory conditions and design approval testing are still valid, but should be more stringent for FBW systems. For example, burn-in of each deliverable item under AGREE conditions, including temperature cycling and vibration for 100 hours, was required on the 680J equipment. Sufficient failures associated with uninspectable workmanship and detail part infant mortality were uncovered prior to LRU and system acceptance testing to more than justify the costs of this procedure. In the area of design approval testing, a new test to demonstrate survivability to Electromagnetic Pulse (EMP) effects due to lightning and/or nuclear threats should be added.

In addition, it is recommended that system testing as well as LRU acceptance testing be required on each set of deliverable hardware. This type of testing, which requires the individual LRU suppliers to have a system bench setup simulating all other interfacing hardware, is particularly important in the case of redundant electronic units. It is nearly impossible to design and refine an LRU test specification which assures that the unit will work in the system. This situation exists today in many LRUs that are in production. In order to have tight LRU test tolerances sufficient for redundant operation, one must first produce enough units to establish the statistical nominal values. Yet the specification is frozen far in advance of this information. Consequently, the tolerances are initially left large in order to accept units that will satisfy system requirements. These tolerances are then approved, and later it is found that the nominal value is off but a change cannot be made without undue effort and cost. Even if it were feasible, a system setup test would still be required to validate the built-in test circuits and operation as a system.

Other aspects of quality assurance for FBW/flight safety essential systems relate not to specific requirements imposed by the specification, but result from the potential legal responsibility or liability of suppliers of such equipment. It is estimated that the inspection, test and documentation associated with production of flight safety essential equipments is three-to-one more costly than these same activities for non-essential flight controls. These costs should be recognized by the specification originator who should exercise due judgement and avoid the imposition of unnecessary special quality assurance requirements.

IV. CONCLUSIONS AND RECOMMENDATIONS

FBW systems can be designed and built to meet almost any requirements that can be explicitly defined. The physical characteristics and cost of ownership of these systems can vary by orders of magnitude as a function of slight differences in design requirements. The problems in past prototype FBW design specifications have been associated with either overspecification or conflicting specifications or design goals. Overspecification leads to mechanization complexity. Conflicting specifications or goals increase design and development costs.

The primary design requirement for an FBW system is mission reliability. Mission reliability allocations should be made on an approximately equal basis for the various subsystems or elements which are in series in terms of safe flight. Overspecification of the electronics portions imposes penalties which may initially seem small but may in fact be large.

In general, safety, survivability, BITE, and even preventative maintenance requirements should all be referenced to mission reliability requirements. Requirements developed based on the question "What if...." should be avoided. Threats and their probability of occurrence must be defined and assessed in terms of effect on mission reliability.

And finally, the ideal specification would be explicit in all details, including term definitions, method of analysis, definitions of detail part failure modes, and probabilities of failure.

REFERENCES

1. Kissinger, R. L., Vetsch, G. J., "Synthesis and Analysis of a Fly-By-Wire Flight Control System for an F-4 Aircraft," AIAA Paper 72-880, Stanford, California, August 1972.

2. Tomlinson, L. R., "Problems and Solutions Related to the Design of a Control Augmentation System for a Longitudinally Unstable Supersonic Transport," AIAA Paper 72-871, Stanford, California, August 1972.

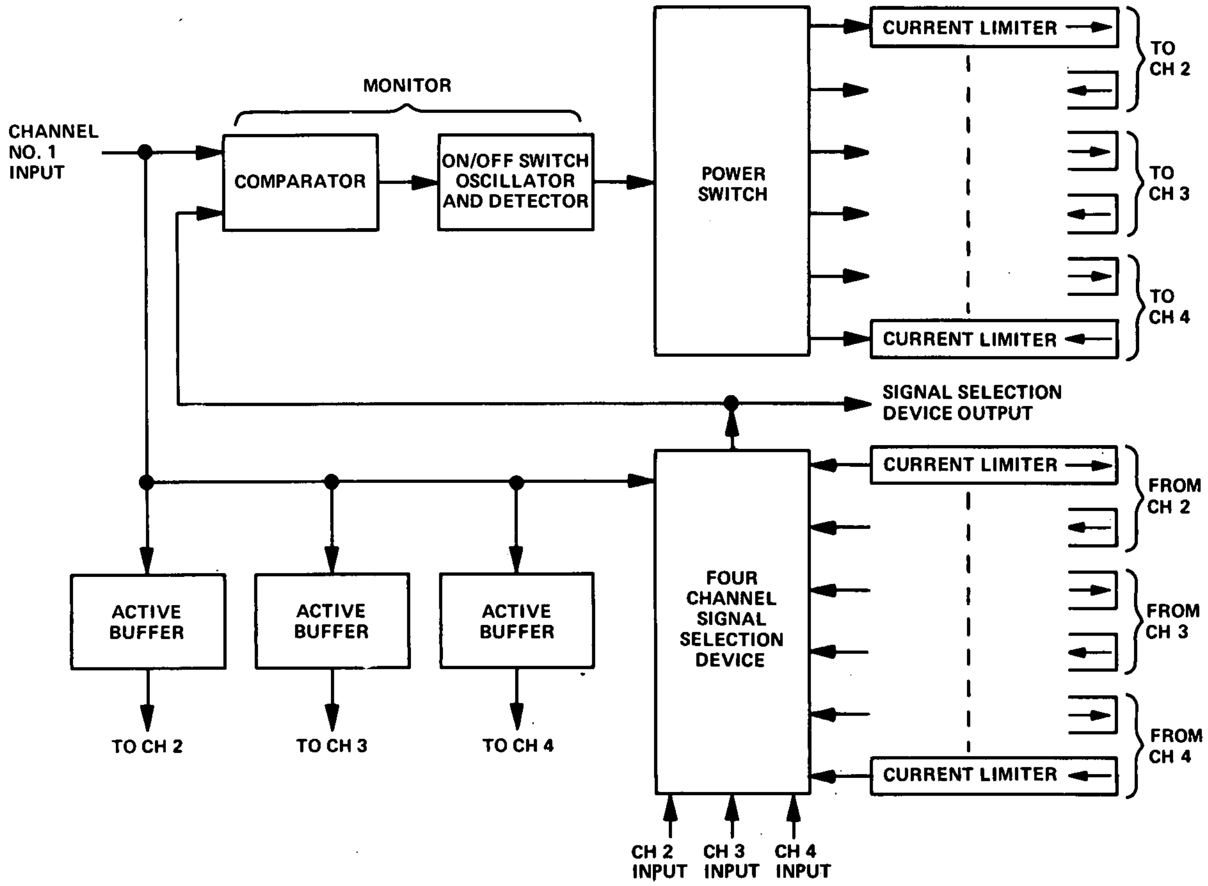

Figure 1
Signal Flow Diagram for Case 1 Signal Selection Device

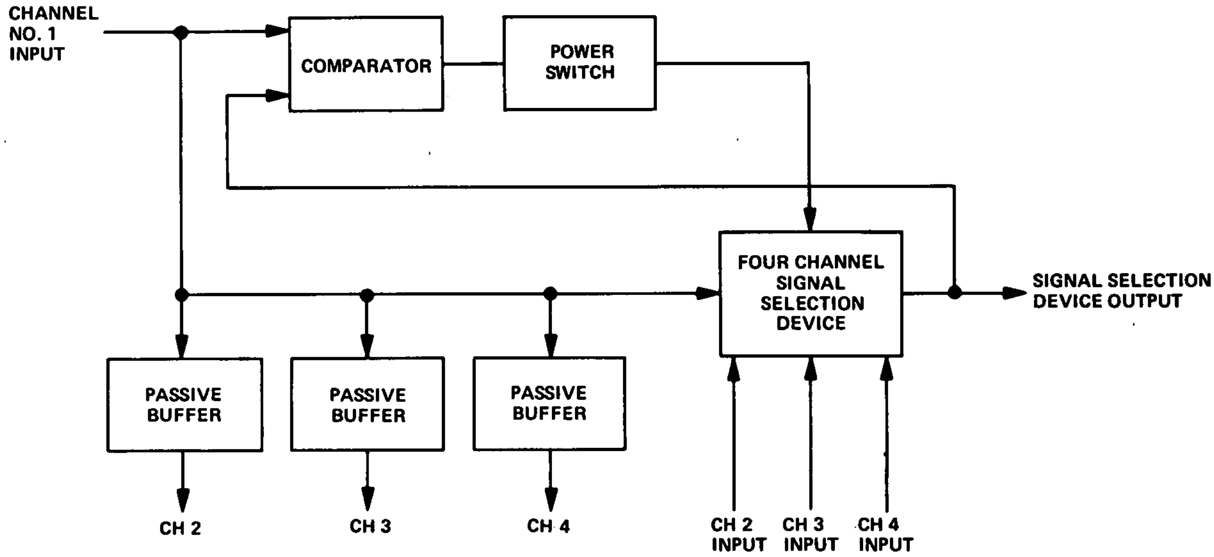

Figure 2
Signal Flow Diagram for Case 2 Signal Selection Device

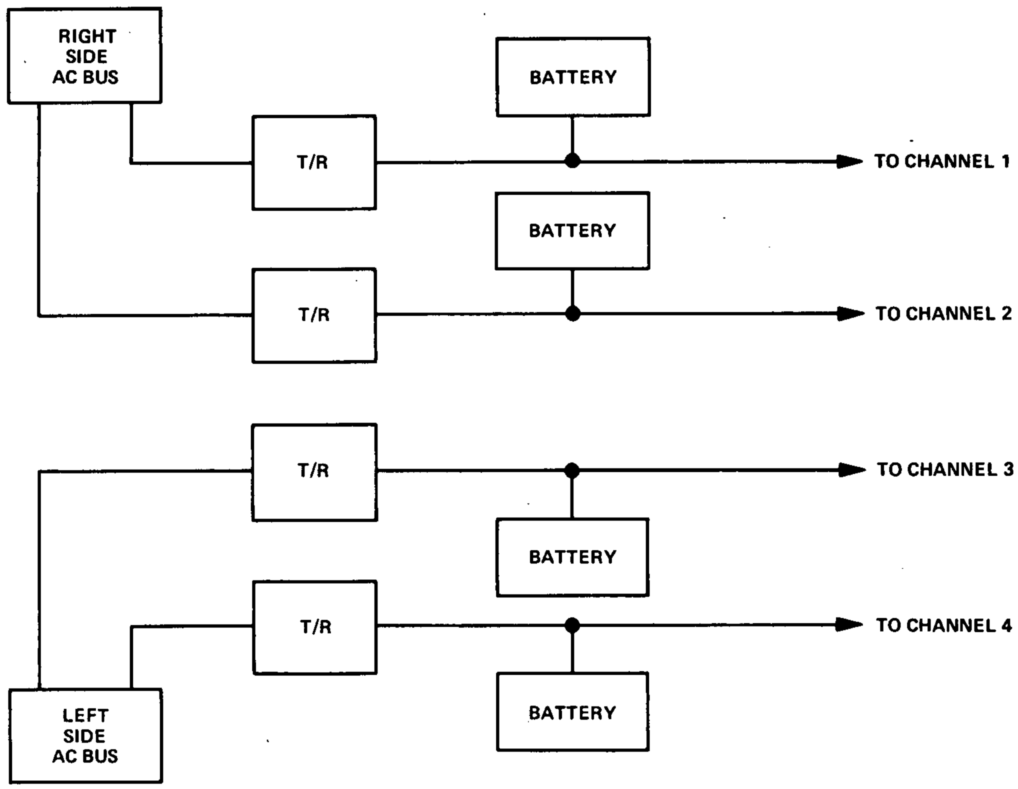

Figure 3
Isolated Quadroplex Redundant Power Distribution System

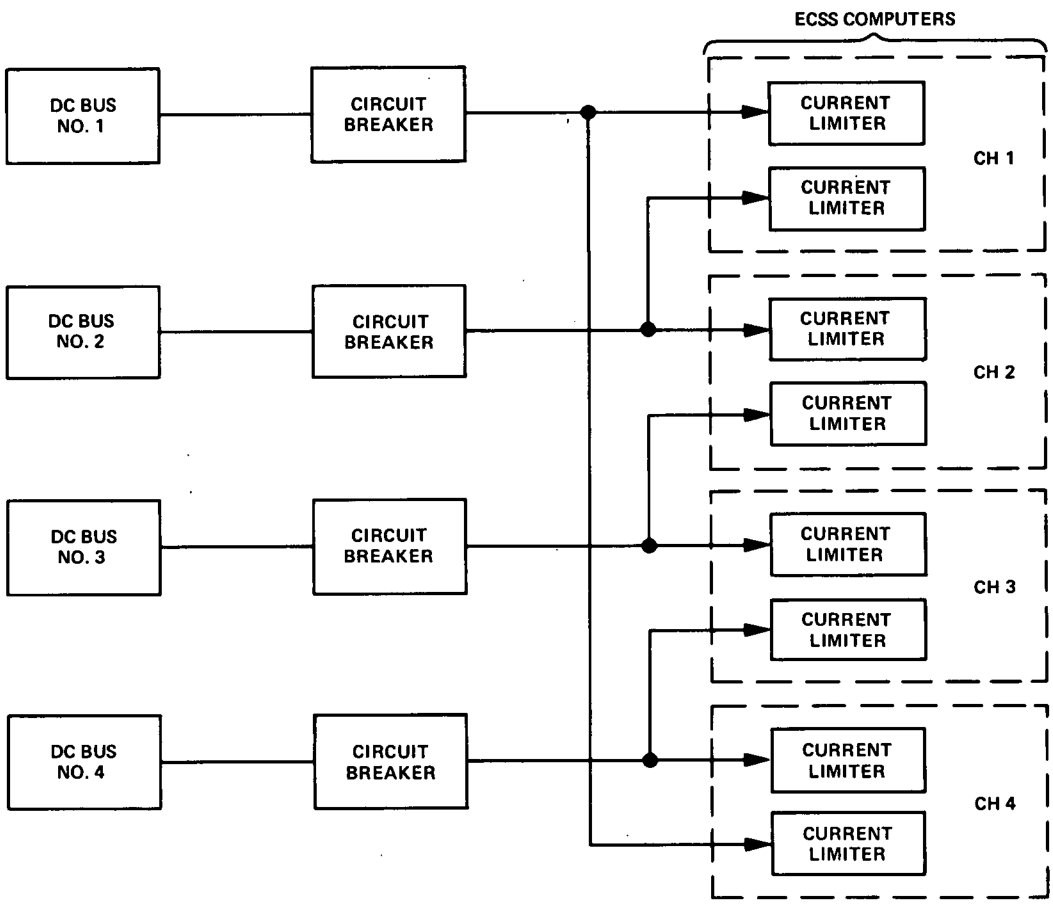

Figure 4
ECSS Power Distribution System

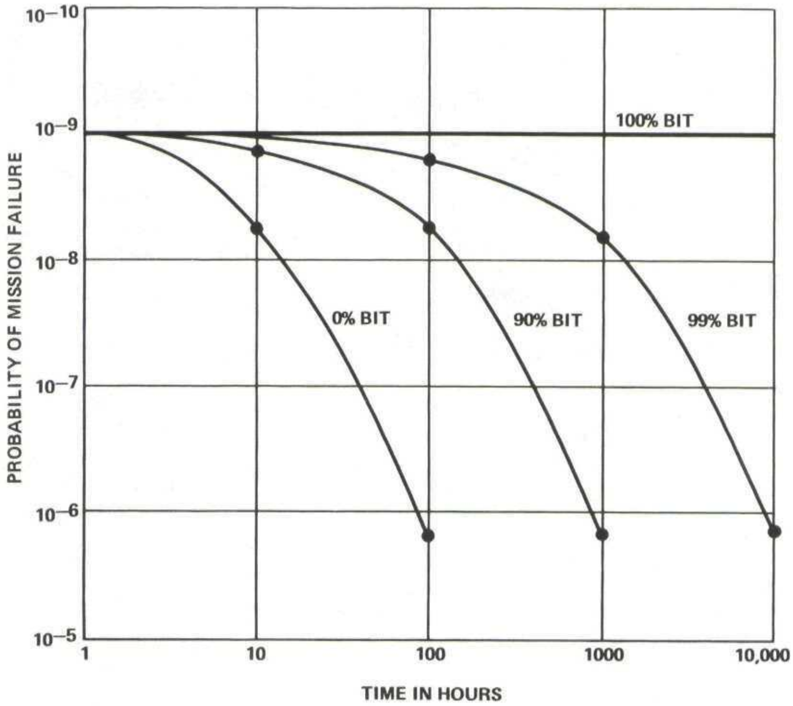

Figure 5
Effect of BIT Testing on Probability of Mission Failure

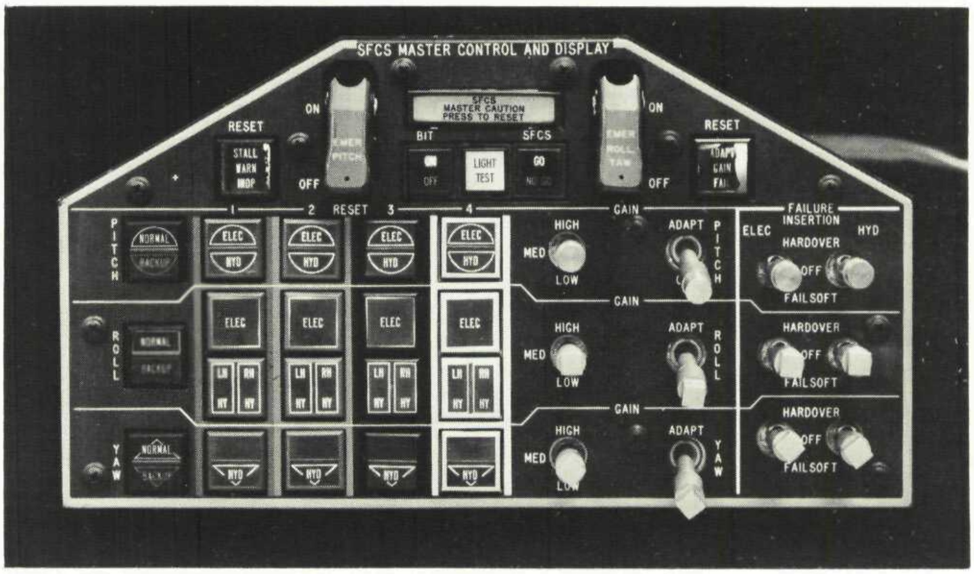

Figure 6
SFCS Master Control and Display Panel

EXPERIENCE WITH THE CONCORDE

FLYING CONTROL SYSTEM

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INTRODUCTION

This paper gives a brief description of the Concorde Flying Control System with a discussion of its performance, reliability and behaviour in flight. Possible future developments are mentioned.

SYSTEM PHILOSOPHY

The basic requirement for Concorde was for an aircraft which was precise and easy to handle throughout a flight regime far in excess of anything previously experienced in the civil field. (Fig.1).

This requirement was unlikely to be met by a Mechanical Flying Control System because of:-

The lack of precision of mechanical systems considering the long cable and rod runs that would be necessary.

The aeroelastic properties of the aircraft.

Differential kinetic heating of the mechanical linkages.

Thus a decision was made in favour of electrical signalling. At this time (about 10 years ago) it was felt unlikely that a system could be developed (or accepted by Certificating Authorities and Flight Crew) which did not have a mechanical reversion. Since a further requirement for the aircraft was a failure survival category III Automatic Landing System, adequate redundancy already existed in electrical and hydraulic supplies, control surfaces, etc.

The system that was designed, therefore, had two independant channels of electrical signalling with a mechanical back-up. Under electrical control, the mechanical linkage follows the Pilot's demands such that, in the event of reversion to mechanical mode, the change-over can be made quickly and smoothly without structural or aerodynamic disturbance. Also, because even with the precision of electrical signalling, it was improbable that pleasant handling qualities would exist throughout the whole flight envelope, a two channel autostabiliser system was added in each control axis to damp out the aircraft response to various disturbances.

SYSTEM DESCRIPTION.

The equipment in the Control System comprises:

- Conventional Flight Deck controls.
- Electrical signalling resolvers and servo amplifier chains.
- Mechanical signalling back-up chains using relay booster jacks to improve accuracy and provide an input for autopilot signals.
- Duplex Power Flying Control Units which actuate the control surfaces.
- Artificial Feel System in three axes which provides control resistance compatible with pleasant handling and returns the control linkage to neutral.
- Trim System with three axes mechanical control and electrical control in the pitch axis. This provides a datum adjust function on the neutral point of the control linkage.
- Autostabiliser System in three axes to damp the aircraft response to structural and aerodynamic disturbances.
- Safety Flight Control System incorporating anti-high incidence functions and a standby signalling system in pitch and roll axes employing control column load signals.

BASIC FLIGHT CONTROL SYSTEM.

Since Concorde is a delta winged aircraft, pitch and roll control is achieved by movement of surfaces known as elevons. Each wing has three elevons, each controlled by a duplex electro-hydraulic Power Flying Control Unit (P.F.C.U). Yaw control is achieved from a conventional rudder having two control surfaces each again controlled by duplex PFCU's (Fig. 2).

The envelope of control surface demand is shown in fig. 3. This shows that in the roll control axis differential control surface movement is incorporated between inner and outer/middle elevons.

Electrical pitch and roll control is achieved by mixing output signals from resolvers mounted on a chassis at the base of the control column and feeding the resultant demands, via servo amplifiers, to the moog servo valve of the appropriate power control unit. (Fig.4).

Electrical rudder control is achieved simply by feeding resolver demands from pedal position via servo amplifiers to the rudder power control units.

Each power control unit consists of two double acting hydraulic jacks in tandem each driving a ram piston assembly mounted on a single rod. This rod is anchored to the aircraft structure at both ends so that control surface demand is transmitted by movement of the PFCU body. (Fig.5).

The Power Control Unit can operate in electrical (two channel) or mechanical mode. In mechanical mode, a clutch assembly engages an input lever to the aircraft mechanical control runs. In both modes of operation micro-switch spring boxes feed signals to a logic unit to ensure protection against valve jams.

The principle of relay jack operation is similar to that of the main PFCU's except that the electrical mode of relay jack operation is reserved exclusively for autopilot operation.

To give the required redundancy there is duplication of resolvers, amplifiers, electrical and hydraulic supplies.

The electrical power supply used exclusively by the Flying Control System is 26V 1800 Hz derived from two static inverters driven from each of the DC essential bus bars. This supply has the advantages of being segregated both from mains power supply interrupts and also harmonic interference from 400 Hz systems.

These dedicated generation systems are designated "Blue" and "Green".

The Aircraft has 3 hydraulic systems - Blue, Green and Yellow. Each double acting Power Control Unit and relay jack is normally powered with Blue and Green hydraulics with Yellow on stand-by capable of taking over if either main supply fails.

In normal operation, the aircraft operates with the Blue electrical Flying Control System but, in the event of a Blue failure, the Green system takes over automatically. In the event of a further failure affecting the Green system, the aircraft reverts to the mechanical mode.

MONITORING.

In the event of a detected failure of the Inverters (under/over voltage, under/over frequency etc), the resolvers (open circuit, seizure etc), the signalling amplifiers or autostabilisers (interchannel discrepancy) or the hydraulic supplies (low level or low pressure) in the controlling lane, the lane will be taken off line automatically and control will revert to the other lane, if available, or to the mechanical mode. (Fig. 6).

'In line' monitoring is employed on the elevon and rudder surfaces.Each control surface has an associated monitoring chain which compares the pilot's demands, derived from monitor resolvers mounted at the foot of the control column, with the actual surface position. Compensation is made for autostabilisation at the comparators. In the event of a detected error $\geq 2.5^{\circ}$, the changeover logic switches the signalling mode the next serviceable channel.

MECHANICAL MODE.

In the mechanical mode, movement of the control column or rudder pedals activates the appropriate relay jack which boosts the mechanical feed to the power controls. (Fig.5).

Under automatic control, the relay jack is locked, thereby feeding the autopilot demands back into the pilot's controls and transmitting them through the normal electrical signalling channels.

AUTOSTABILISER.

The autostabiliser system comprises the following equipment:-

- 2 three axes autostabiliser computers.
- 6 rate gyros (2 per axis)
- 2 lateral accelerometers.
- 2 Roll relay jack position sensors.

The basic principle of autostabilisation is the derivation, from pitch, roll and and yawrate signals, of a control surface demand to compensate the aerodynamic disturbance. Air data computer information on Mach. No. and speed is used to schedule gain and authority limitation to obtain the optimum control law for the complete flight régime. (Fig. 7).

In addition to this basic function, a roll/yaw turn co-ordination is provided using roll relay jack position as the prime sensor, gain scheduled and authority limited by Mach. No.

To minimise sideslip under engine shutdown or failure conditions at high supersonic speeds a lateral accelerometer signal is computed in the yaw autostabiliser channel thus improving powerplant performance margins.

The autostabilisation signals are injected directly into the flying control servo amplifiers as DC signals and therefore, are not transmitted through the pilots controls. This function is not available in mechanical mode.

Safety Flight Control System.

The Safety Flight Control System has three functions.

- High incidence warning.
- Stabilisation at high incidence.
- Emergency piloting to cater for the case of a control run jam upstream of the relay jacks.

The behaviour of Concorde at high incidence is different from sub-sonic aircraft and this leads to supplementary forms of protection.

In addition to a conventional stick shaker signalled purely from aircraft incidence information, a "stick wobbler" operating from incidence and speed information warns of approach of minimum speed. The "wobbler" warning is obtained by pulsating the pressure in the pitch artificial feel loading jacks thus wobbling the control column against pilot resistance.

The aircraft must also be protected against dynamic manoeuvres at high incidence. This protection is known as the 'Superstabiliser' and employs the output circuitry and servo amplifier inputs of the pitch autostabiliser channel using pitch rate, incidence and speed information. An increasing "nose-down" elevon demand is made with proximity to stall.

To cater for certain control run jam failure cases, an emergency piloting mode is available. Both control columns incorporate load detectors providing an electrical signal proportional to the load applied. These signals are transmitted to the control surfaces by the stabilisers in pitch and roll axes.

Artificial Feel.

The artificial feel system has three functions:-

- To provide reaction on the controls compatible with pleasant handling.
- To provide an authority limit to the pilot to prevent reaching a dangerous configuration.
- To provide an authority limiter for the autopilot.

The variable stiffness in each axis is provided by hydraulic actuators controlled by a computer. The two actuators are permanently servoed but only the Blue is active with the Green in it's extreme position. Failure of the Blue system results in a rapid change-over to Green. (Fig. 8).

Ultimate safety is provided by spring struts in the event of a double electro-hydraulic failure.

TRIM.

The Dual Trim System is manually and electrically operated. The mechanical trim operates in parallel with the manual control system.

Electric trim has four main functions in the pitch axis only.

- Manual trim by the pilot.
- Auto-trim to zero feel loads.
- Mach trim.
- Incidence trim.

As with the feel system, rapid changeover occurs in the event of system failure.

GENERAL.

The overall principle of the Flight Control System is shown in Fig. 9.

RELIABILITY.

Figure 10 shows that the probability of losing an electrical signalling channel is 307×10^{-6}/ 3 hour flight.

Figure 11 shows the probability model for those aspects of the hydraulics generation circuit and associated selection logic inherent to electrical signalling of flying controls.

By combining all these figures, it can be seen in figure 12 that the probability of total loss of electrical signalling is 1.27×10^{-7}/3 hour flight.

It is therefore clear that the occurrence of loss of a single system would be too high for simplex to be acceptable and also that while the probability of having to revert to manual mode with a duplex system is extremely low, it is not low enough to contemplate deletion of the mechanical back-up.

Further attempts to improve the reliability of existing components or perhaps even the addition of a third electrical signalling channel such that the calculated probability of total electrical loss fell from 1×10^{-7} to 1×10^{-9} or 1×10^{-10} raises the issue of the practical meaning of these probability figures with respect to all possible failures or conditions to which the aircraft or system can be subjected. It can be argued that the mechanical back-up could not be justifiably deleted even under these circumstances because of common mode failure possibilities to an electrical signalling control system. These possibilities might be lightning strikes, electrical noise, vibration, batch problems in equipment manufacture or perhaps erroneous maintenance procedures on similar control channels.

To demonstrate immunity to failure possibilities of this nature a form of dissimilar redundancy must be inherent to the control system. Clearly present technology offers a mechanical back-up system as the simplest and most widely acceptable form of dissimilarity however known research and development into the use for instance of fibre optic transmission systems could offer dissimilar control which overcomes the obvious penalties of mechanical systems.

SYSTEM PERFORMANCE.

Flight Testing of Concorde began in 1968 with the French Prototype 001 and will be completed when the Certificate of Airworthiness is granted in 1975. Over this period Concorde will have become the most tested aircraft in Aviation history, not because its flight test programme has been beset by unforeseen or serious problems but because the concept of a commercial supersonic transport aircraft had led to the establishment of new certification standards.

Problems have been encountered with Concorde Flying Controls. However, the solution to these problems has led to a reliable and efficient control system and must be considered as minimal and forming a normal systems development programme.

Initial development of the Flying Control System gave rise to differential elevon gearing ratios between inner and outer/middle elevons in both pitch and roll axes. This modification was prompted by simulator studies an aerodynamic characteristics in the transonic regime. The inner elevon deflection ratio relative to outer/middle elevons was reduced to improve lateral behaviour at low supersonic speeds because of the effects of yaw induced by the inner elevons. At the same time the inner elevon deflection ratio in pitch was increased to obtain better distribution of hinge moments through the transonic phase. This modification was incorporated because of the narrow C.G. corridor during transonic deceleration on PFCU half body hydraulics. (Fig. 13).

Further optimisation of these gearing ratios was made through Pre-production aircraft and the final Production standard favoured a gearing ratio in roll of 0.7 between inner and outer/middle elevons, whilst the requirement for differential gearing in pitch was not confirmed from flight tests. This latter point was shown from the ability to use further aft C/G valueson Pre-production and Production aircraft with the rearward transfer of aerodynamic centre following the planned wing re-design incorporated at Pre-production stage.

It should be noted that in order to evaluate these changes, the flying control gearing ratios had to be assessed in both electrical and mechanical modes. The system implications of such changes will be discussed inthe next section. However, particular problems were found with mechanical signalling.

Flight testing has shown that with regard to passenger comfort and pilot workload, the mechanical signalling mode configuration is the worst that is experienced on Concorde and can lead to a pilot/aircraft coupling according to flight conditions. This coupling can be summarised as being essentially longitudinal subsonic/transonic and lateral at supersonic speeds. This characteristic, resulting from the inherent transmission lags and frictional and backlash errors, added to the absence of auto-stabilistion in mechanical mode, led to a compromise in the selection of gearing ratios.

A typical stability plot for the aircraft in pitch is shown in Fig. 14. It can be seen that some modes are poorly damped giving marginal, but acceptable, control with electrical signalling. The roots gave rise to a tendency toward pilot induced oscillations on Prototype aircraft (4-12 secs. period according to flight cases) and have led to optimisation of the pitch autostabiliser laws. The addition of Autostabilisation clearly leads to a better damped, more controllable aircraft.

To improve lateral stability, particularly under supersonic conditions, autostabilisation was incorporated inthe roll axis. This function restores correct stability, pleasant handling and passenger comfort under turn, fuel transfer and powerplant thrust variations. In addition, auto-stabilisation is provided in the yaw axis. The prime function of this mode is to damp out Dutch roll. However, through the use of lateral accelerometers, sideslip, ensuing from engine failure or assy-metric thrust variations, is also compensated and powerplant performance margins have been improved.

A further stabilisation term has been added to improve lateral stability in the transonic region. This is a roll/yaw turn co-ordination function. This term has provided further stability and handling improvements under full pitch-up PFCU saturation conditions.

On the subject of PFCU saturation and hinge moment limitation, this is a particular problem on Concorde and is directly attributable to the aerodynamic design features of delta winged aircraft with elevons. These problems have, however, been minimised by systems development and deflection limitation on the control surface envelope. Fig. 3 shows the envelope of elevon deflection available in pitch and roll axes. It can be seen that the maximum pitch-up deflection available is 15°. This limitation, achieved by a mechanical stop onthe front flying control chassis, ensures that when the elevons are in pitch-up saturation and the column is being pulled towards the pitch-up limit then 4° of roll control is still available to de-saturate one wing and permit a recovery manoeuvre to re-establish normal hinge moments. To compensate further for abnormal wing loadings in certain flight regimes, a system has been developed known as 'outer-elevon neutralisation.' This is basically an overspeed correction system which under 'Vmo + ' conditions neutralises the outer-elevon deflection to zero thus minimising torsional wing loads due to hinge moment.

In conclusion the general construction of the electrical flying controls has been shown from flight testing to be wholly compatible with good aircraft response. Initial response to pilot demands has been precise with good dynamic damping. The control forces have been optimised to an acceptable level throughout the complete flight regime and this combined with a reasonable static stability has led to an aircraft which is easy to fly and, generally speaking, devoid of the requirement for close pilot monitoring.

EQUIPMENT DEVELOPMENTS.

In order to achieve the performance improvements discussed in the previous section, equipment developments have been necessary. These developments initially resulted from the requirement to change the flying control gearing ratios.

The design of the electrical signalling chains was such that differential gearing between inner and outer/middle elevons could be incorporated very simply and quickly. This modification merely required the fitment of new actuation levers on the inner elevon resolver chassis and an electrical re-scaling on the relevant servo amplifiers. However, the repercussions on mechanical signalling were more involved. To incorporate differential gearing in mechanical mode involved a complete re-design of the mechanical pitch and roll mixing chassis.

Further development was also necessary to the surface position monitoring system.. Prior to differential gearing the principle of elevon monitoring was that of a voter-monitor on each wing semi-span, switching signalling mode on port and starboard elevon pairs on failure detection. The revised gearing of inner elevons relative to outer and middle elevons, however, required a new monitoring philosophy.

Rudder control and monitoring had also undergone changes from its original standard. Suspected flutter problems on prototype aircraft provoked a shelving of the twin rudder control channel definition in favour of a single rudder electrical control channel with the second rudder PFCU slaved in themechanical mode thus linking the two control surfaces. This led to an 'in-line' monitored system.

A type of in-line monitoring was therefore adopted on inner elevons to meet the requirement of differential gearing. The principle employed was to compare the absolute difference between port and starboard inner elevons with twice the roll demand established from a x 2 differential resolver mounted on the roll mixing chassis. On outer and middle elevons a system was incorporated which made a separate comparison between the two elevons in each semi-span, switching the signal mode of all four elevons in the event of a starboard or port outer/middle elevon discrepancy. The philosophy of switching symmetric pairs of elevons ensured that at all times each wing was subject to a similar signalling mode pattern.

After having adopted the master and slave principle on rudder control, further problems arose because of external temperature effects and different load saturation levels between upper and lower rudder control surfaces. Initially the lower rudder was electrically controlled with the upper rudder surface slaved. However, premature saturation of the lower control surface was limiting available upper rudder control surface deflection. This phenomenon was overcome by reversing the control mode of the two surfaces and making the upper rudder the electrically signalled surface. The problems caused by differential heating of the linkages (the upper rudder PFCU is mounted on the starboard fin surface with the lower rudder unit mounted on the port side of the fin) led to thermal compensation rods being incorporated in the mechanical linkage to offset the temperature induced deflection errors.

Following preliminary flutter tests on prototype aircraft the suspected problems did not arise so the rudder control chain was restored to the original two channel system employing in-line monitoring on each channel.

Flutter tests continued on Pre-production aircraft using electrical signalling inputs as the excitation to the aircraft structure. These inputs were provisioned by test boxes feeding dc step and oscillatory functions through spare autostabiliser inputs at the servo amplifiers. Flutter and resonance testing by this means permitted inputs of known amplitude and duration to be injected thus overcoming the hitherto unpredictable nature of manual pilot stimuli.

During C/G corridor tests on full and half-body PFCU hydraulics, problems were found with unwanted signalling mode changes resulting particularly from application of roll demand under pitch saturation conditions. This was found to be characteristic of the monitoring principle involved. To overcome this problem and to obtain better monitoring integrity, in-line monitoring was adopted on all signalling chains thus creating eight independant monitoring chains each using the in-line principle.

Despite changes provoked by provoked by performance and handling improvements, equipment reliability has been better than forecast and the majority of single system mode changes encountered in flight have been because of failures in the monitoring as opposed to the command system. A notable exception occurred during early development flying on one of the Prototype aircraft. This incident occurred during a transfer from a full to half-body hydraulic condition on PFCU's and resulted in a rolling moment on the aircraft until the full-body hydraulic state was re-established. The problem was traced to seal distortion on an inner elevon PFCU permitting "blow-by" of hydraulics across the ram-piston. The seal design was subsequently changed allowing more elastic flow of the seal thus overcoming the problem.

Design improvements have also been made to the control system changeover logic. On Prototype and Pre-production aircraft this changeover logic was achieved by using double coil multi-contact relays. However, the reliability of this unit was degraded by overheating problems and this has resulted in a re-design of the unit employing solid state switching techniques.

In conclusion the principle of the electrical Flying Control System has not radically changed from its basic design and overall reliability has been good. In short, flight testing has proven the benefits of electrical signalling on Concorde.

FURTHER DEVELOPMENTS TO FLYING CONTROL SYSTEM.

Considering the stability and handling penalties incurred by operation in mechanical mode (i.e. loss of auto-stabilisation and inherent limitations on response) and the weight of the system, clearly the most obvious development would be the deletion of the mechanical signalling chain. However, as mentioned previously, requirements for increased reliability to meet "Extremely Improbable" status, ie of no single catastrophic effect having a probability greater than 10^{-9} per hour, would probably necessitate some dissimilar redundancy. For this reason, therefore, a complete deletion of mechanical signalling cannot be proposed at this stage. Weight saving modifications have been studied for all aircraft systems, however, and within these proposals changes to the flying control system have been suggested.

The following modifications are suggested:-

- Deletion of hydraulic jacks from the roll artificial feel system.
- Deletion of mechanical control, artificial feel jacks and relay booster jack in rudder control channel.
- Deletion of mechanical signalling control to outer elevons.

The deletion of the existing roll and yaw artificial feel systems would result in the introduction of an authority limitation on control surface deflection computed from air data information. This computation would control rudder deflections, roll deflections on inner and outer elevons. The present outer elevon neutralisation function would be deleted. To reduce the probability of loss of control of both rudder control surfaces following deletion of the mechanical control, a third 26V 1800 Hz static inverter would be added which could replace a faulty main inverter. In association with these changes, control surface monitoring and signalling mode changeover logic would be changed to ensure correct mode switching and integrity retention.

Despite what was mentioned previously about the present Concorde design being unsuitable for total dependance on CCV techniques, there are some areas where a feasible partial dependance may offer real advantages.

The C/G corridor imposed by stability and handling limitations (Fig 13) has already been mentioned with regard to the limited elevon hinge moment capability on half-body PFCU hydraulics in the transonic region. This latter capability gives rise to the forward c/g limit, whilst the aft limit is dictated by instability at low and high speed with a further manoeuvre limit transonically. Since the fully aft c/g limit is the normal take-off condition to obtain maximum fuel up-lift and range, clearly a relaxation on the low speed stability limit permitting further aft c/g values at take-off could make significant improvements to the present performance and handling criteria.

With this object in mind studies have been made into the concept of a high authority, high integrity pitch control system to provide a 1% further aft capability.

Fig. 15 shows the c/g envelope improvements envisaged from this limited phase stability augmentation.

This 1% rearward shift in centre of gravity for take off would permit a further 5,000 lb approximately of fuel to be uplifted whilst remaining within limits imposed by nose wheel loading for ground manoeuverability. Therefore a possible CCV advantage to Concorde would be an extension to the existing autostabilisation function to accommodate this capability. The risk period would be short, probably not more than ten minutes, after which the aircraft would be in its normal stable flight envelope and able to achieve centre of gravity movement by the normal Concorde in-flight method of fuel transfer.

Clearly the potential improvements mentioned here are only in the proposal stage and are dependant on the authorisation of further Concorde development. However, it seems reasonable to assume that, for civil aircraft, the first excursions into CCV techniques may come from its use over limited phases of flight where it could be most beneficial with an acceptable degradation of performance in the event of failure.

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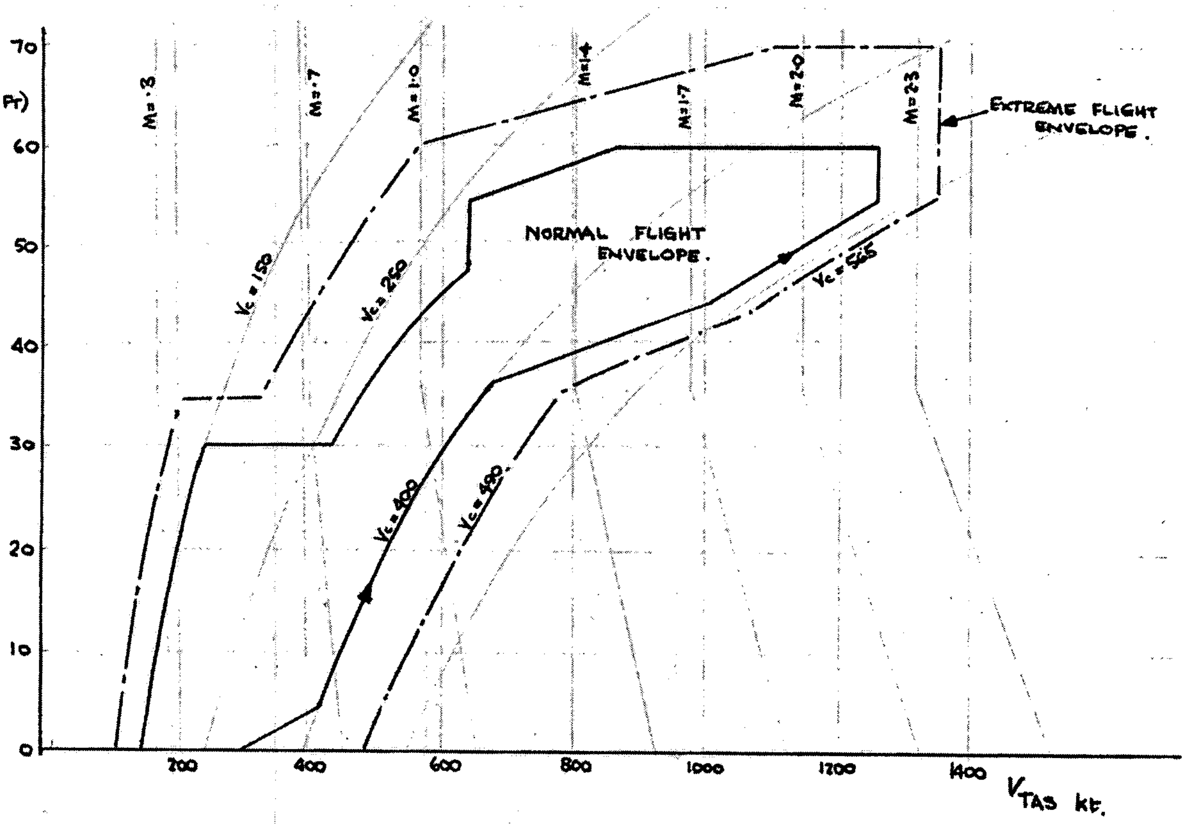


FIG. 1

FLIGHT ENVELOPE.

N.A.C.A ATMOSPHERE
γ = 1·4.

FLIGHT CONTROL–LOCATION DIAGRAM

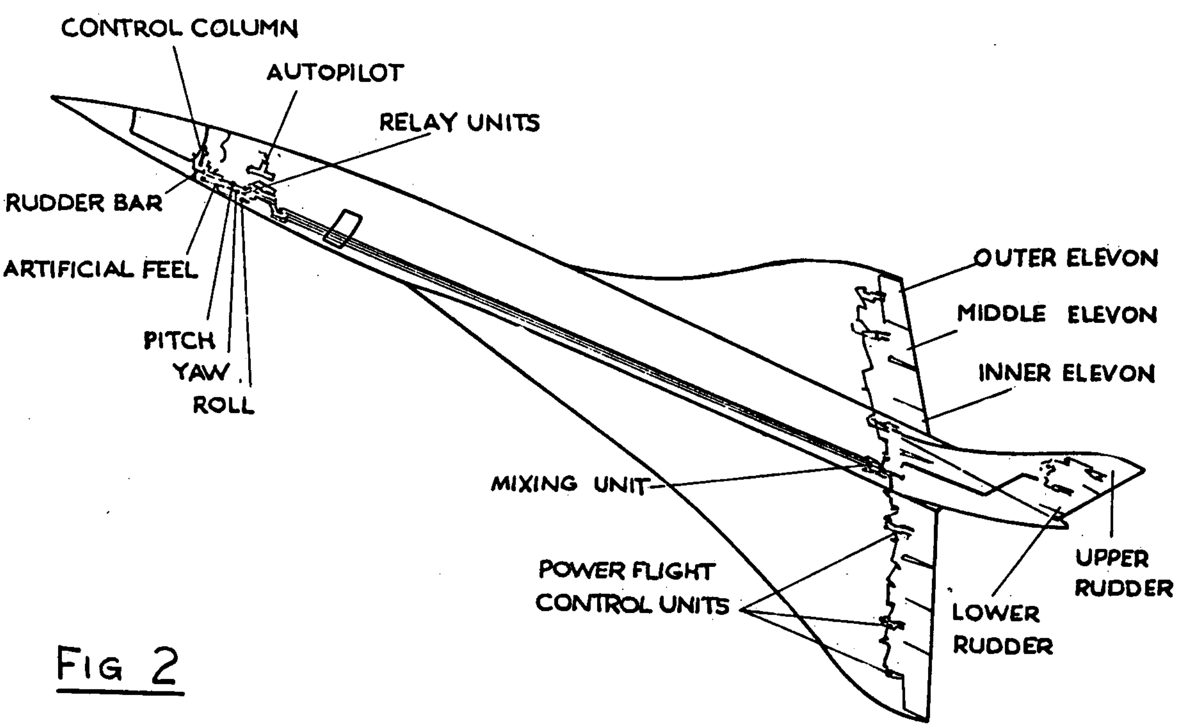
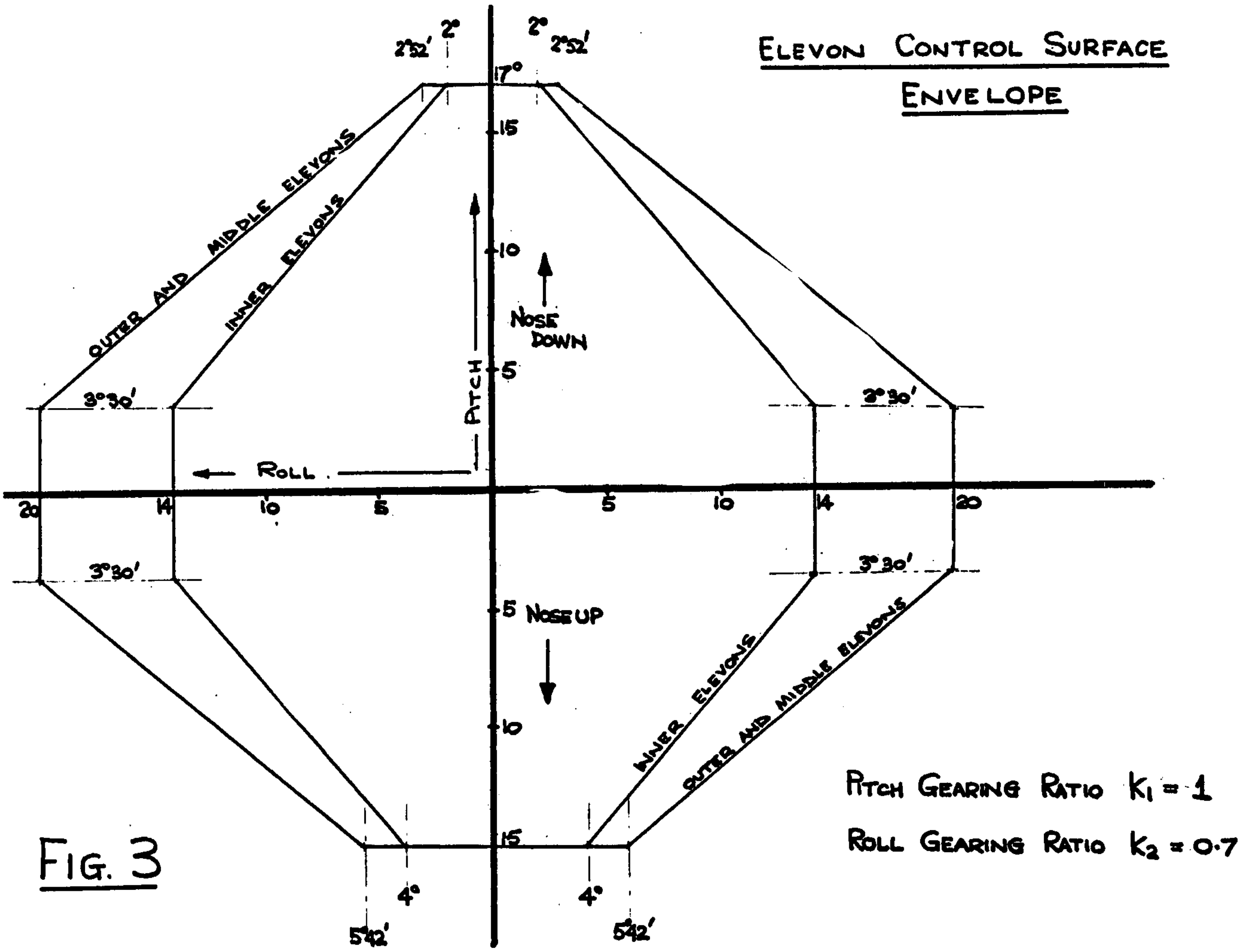
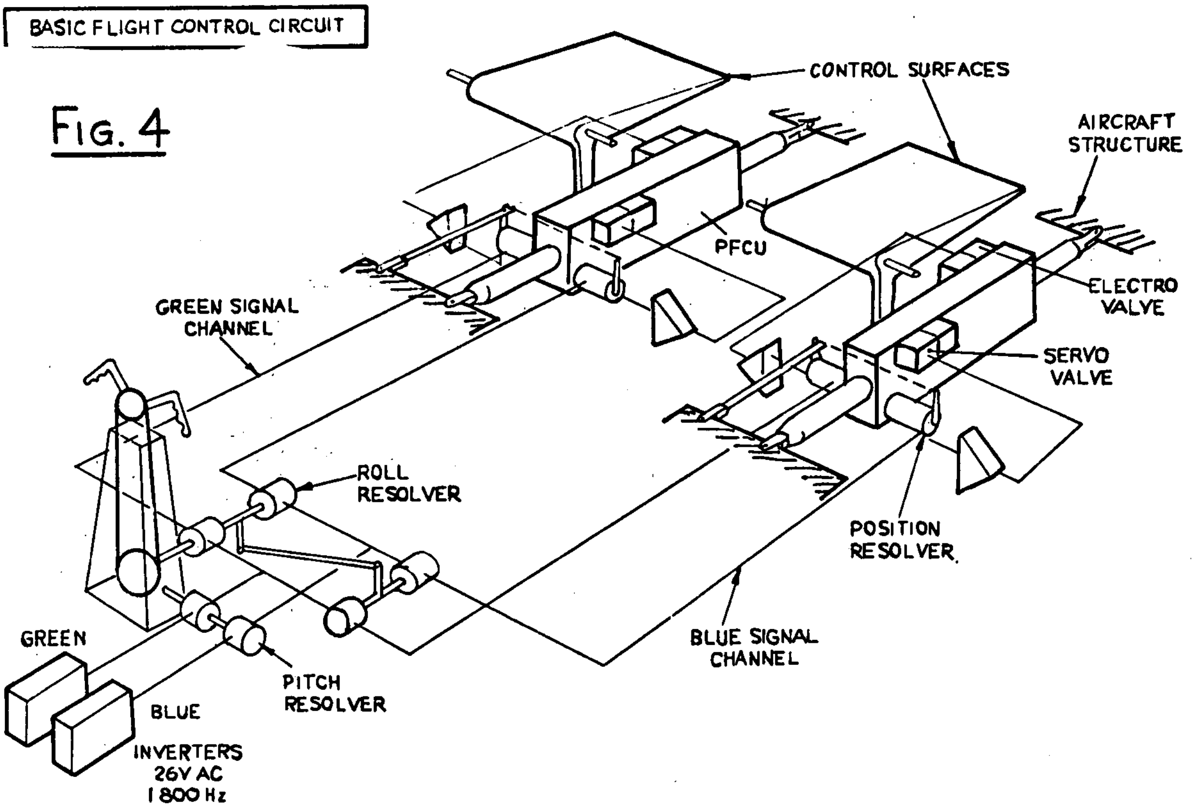


FIG 2



ELEVON CONTROL SURFACE ENVELOPE
OUTER AND MIDDLE ELEVONS
INNER ELEVONS
PITCH
ROLL
NOSE DOWN
NOSE UP
17°
2°
2°52'
3°30'
4°
5°42'
Pitch Gearing Ratio K1 = 1
Roll Gearing Ratio K2 = 0·7

FIG. 3



BASIC FLIGHT CONTROL CIRCUIT
CONTROL SURFACES
AIRCRAFT STRUCTURE
PFCU
ELECTRO VALVE
SERVO VALVE
GREEN SIGNAL CHANNEL
ROLL RESOLVER
POSITION RESOLVER
BLUE SIGNAL CHANNEL
GREEN
BLUE
PITCH RESOLVER
INVERTERS 26V AC 1800 Hz

FIG. 4

BASIC FLIGHT CONTROL CIRCUIT

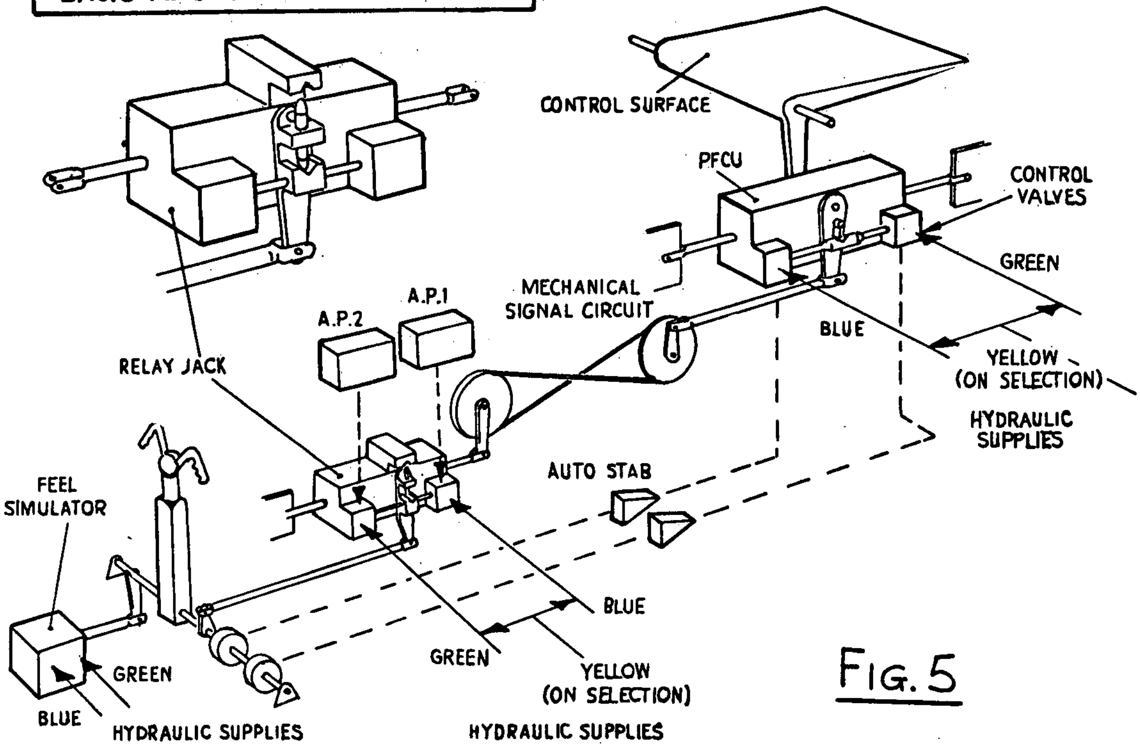
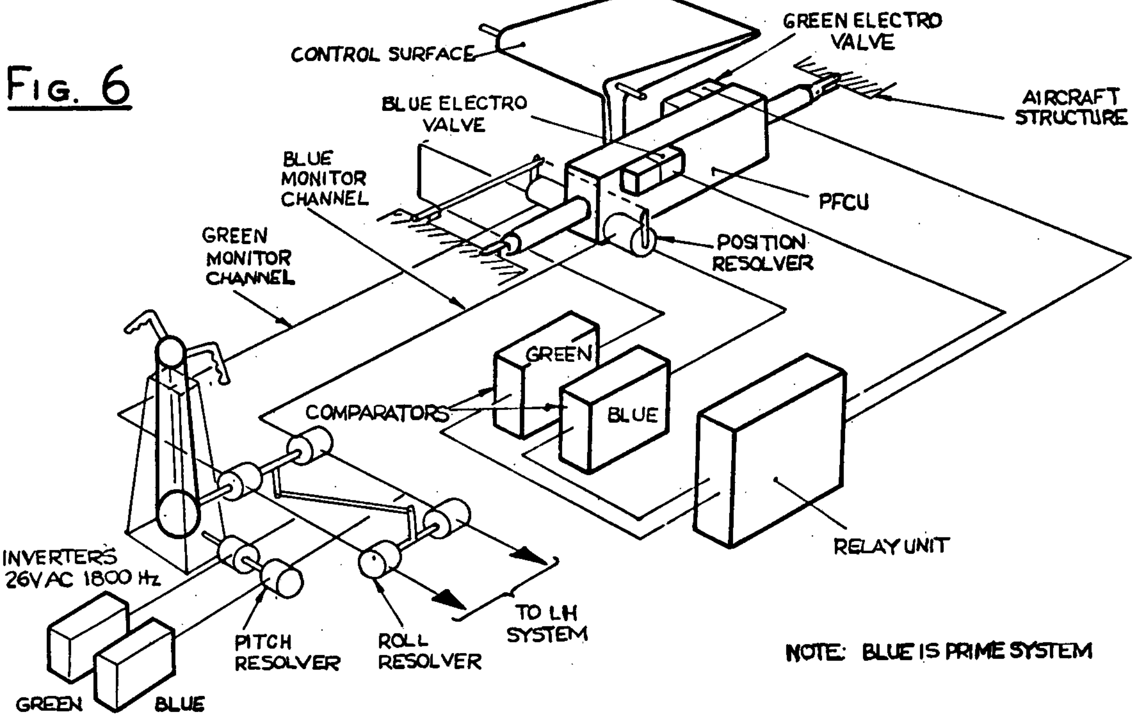


FIG. 5

BASIC FLIGHT CONTROL MONITOR CIRCUIT

FIG. 6



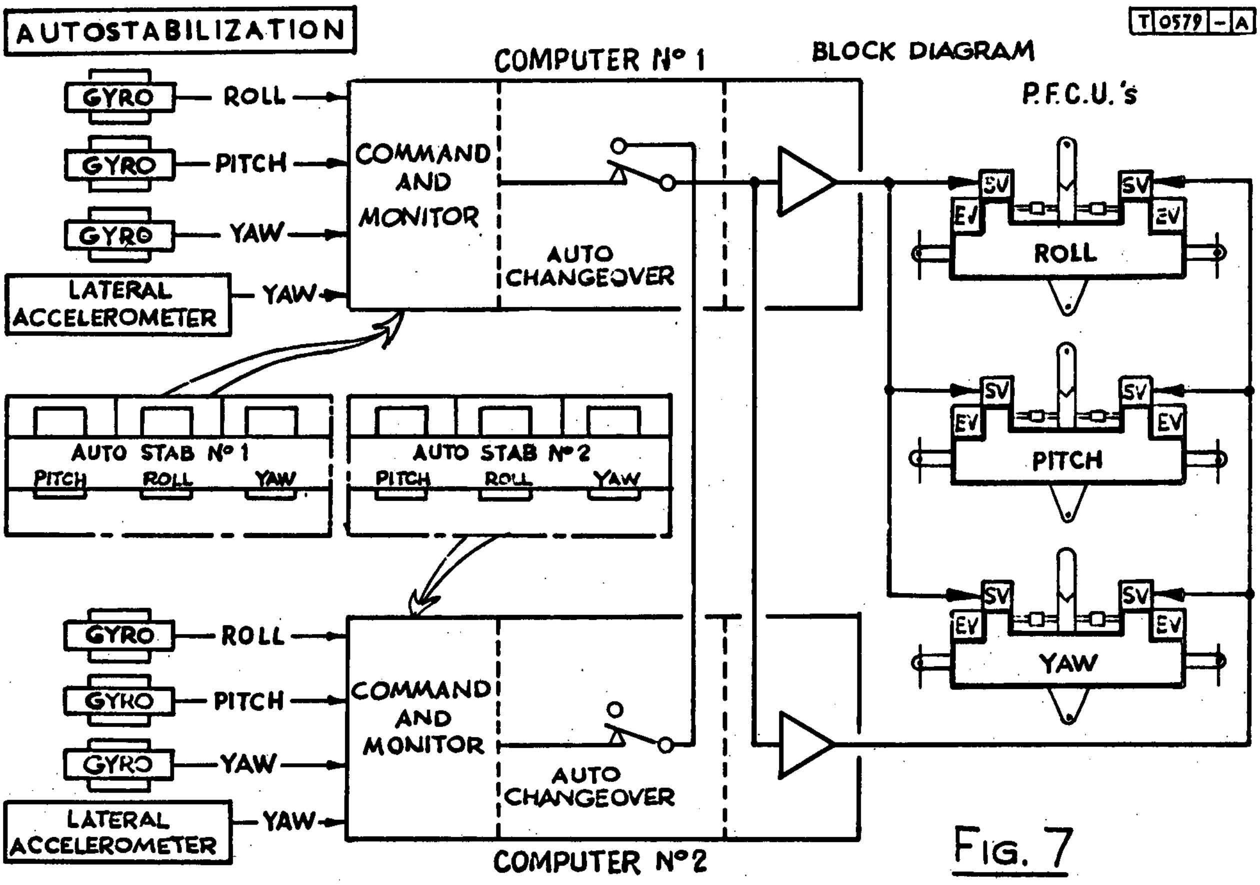


FIG. 7

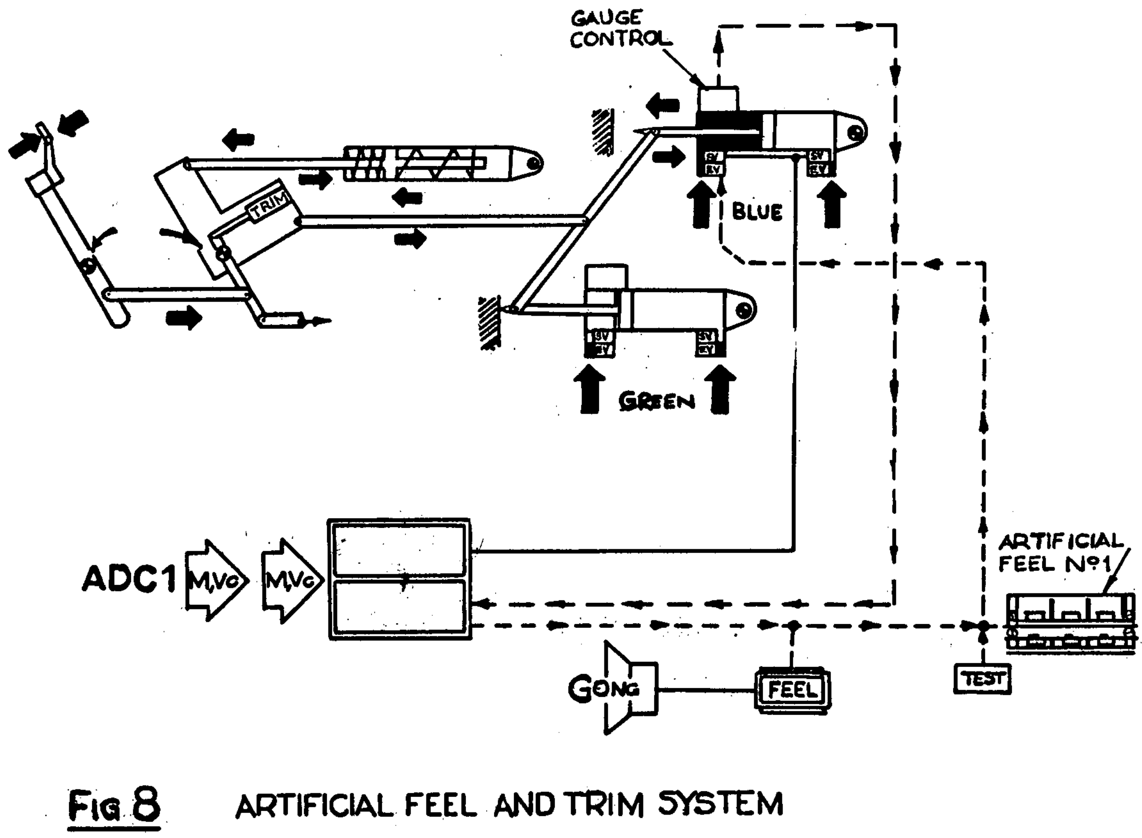


Fig 8 ARTIFICIAL FEEL AND TRIM SYSTEM

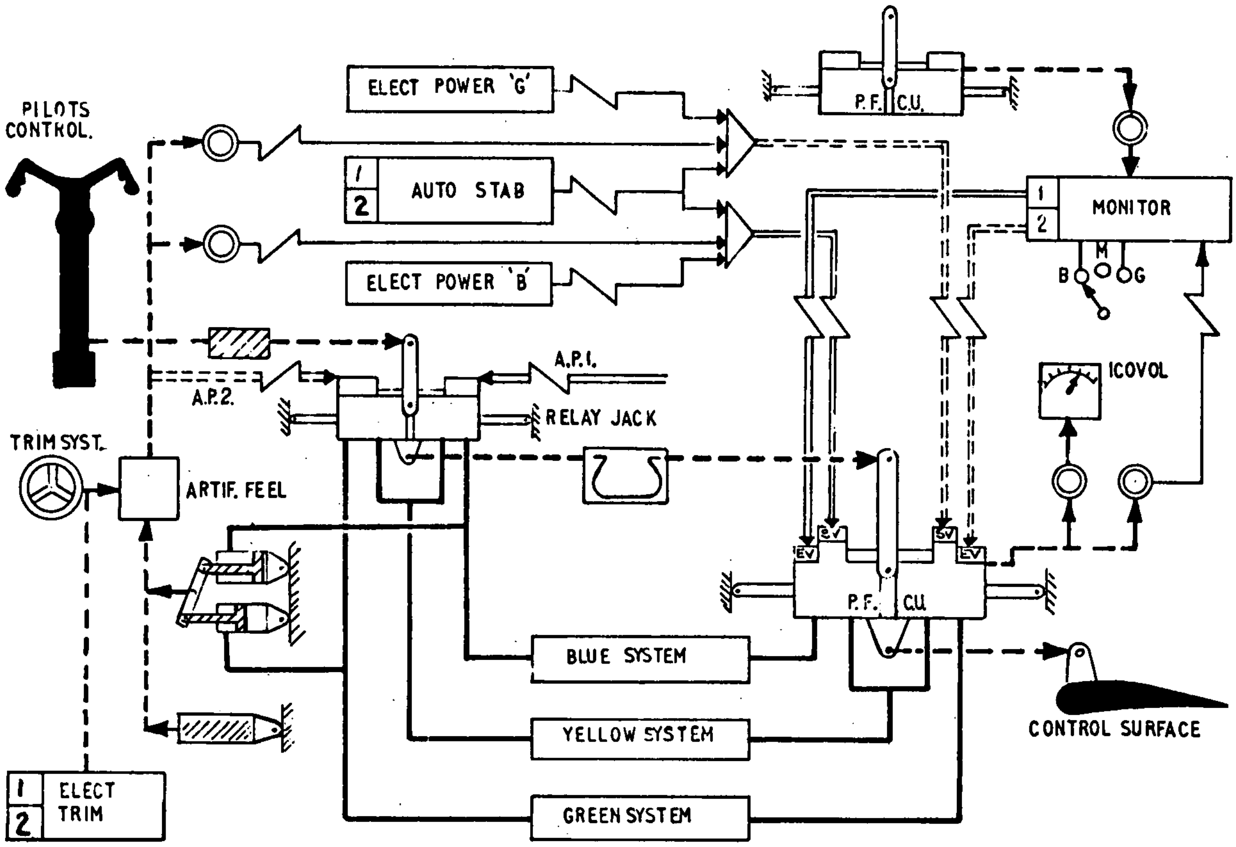
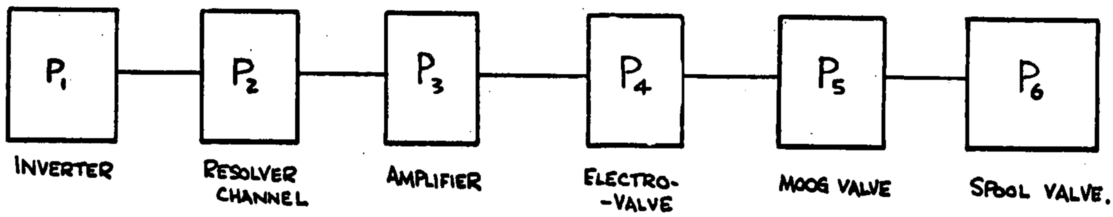


FIG. 9 FLIGHT CONTROL— PRINCIPLE

SAFETY ANALYSIS

BASIC CONTROL LOOP FOR SINGLE SIGNALLING CHANNEL



FAILURE PROBABILITIES/HR.

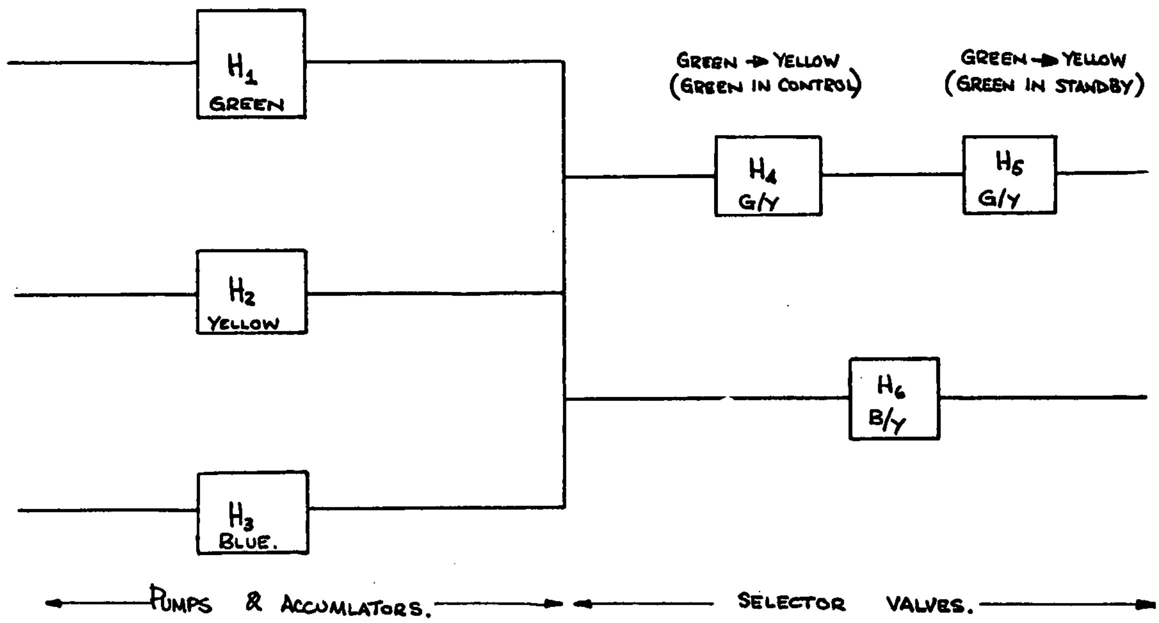
$$P_1 \sim 52 \times 10^{-6}$$
$$P_2 + P_3 \sim 44 \times 10^{-6}$$
$$P_4 \sim 3.5 \times 10^{-6}$$
$$P_5 \sim 2.2 \times 10^{-6}$$
$$P_6 \sim 0.8 \times 10^{-6}$$

$$\text{TOTAL} \sim 102.5 \times 10^{-6}/\text{HR.}$$

FAILURE PROBABILITY OF SINGLE ELECTRICAL SIGNALLING CHANNEL $\sim \underline{307.5 \times 10^{-6}} / \underline{\text{3HR FLIGHT}}$

FIG. 10

SAFETY ANALYSIS — HYDRAULICS MODEL FIG. 11



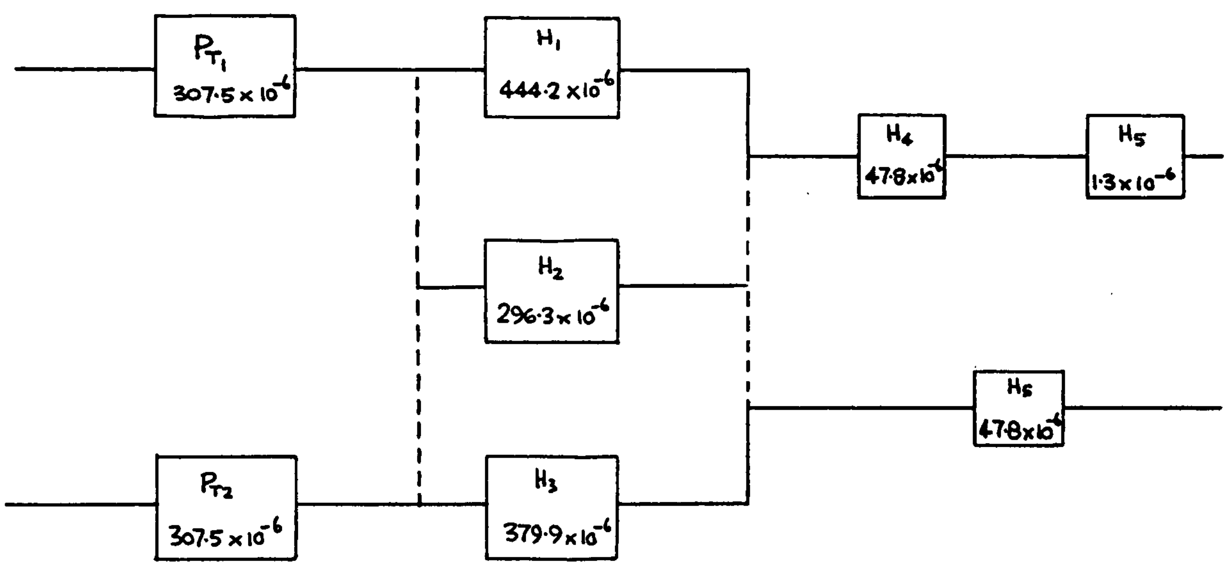
FAILURE PROBABILITIES / 3HR FLIGHT.

$H_1 \sim 444.2 \times 10^{-6}$ $H_4 \sim 47.8 \times 10^{-6}$

$H_2 \sim 296.3 \times 10^{-6}$ $H_5 \sim 1.3 \times 10^{-6}$

$H_3 \sim 379.9 \times 10^{-6}$ $H_6 \sim 47.8 \times 10^{-6}$.

OVERALL SAFETY ANALYSIS MODEL FIG. 12



YELLOW HYDRAULICS H_2 CAN BE USED TO BACK UP BLUE OR GREEN HYDRAULICS

FROM STATISTICAL ANALYSIS ON THE ABOVE MODEL :—

THE PROBABILITY OF TOTAL LOSS OF ELECTRICAL CONTROL

$= 1.268 \times 10^{-7}$ / 3HR FLIGHT

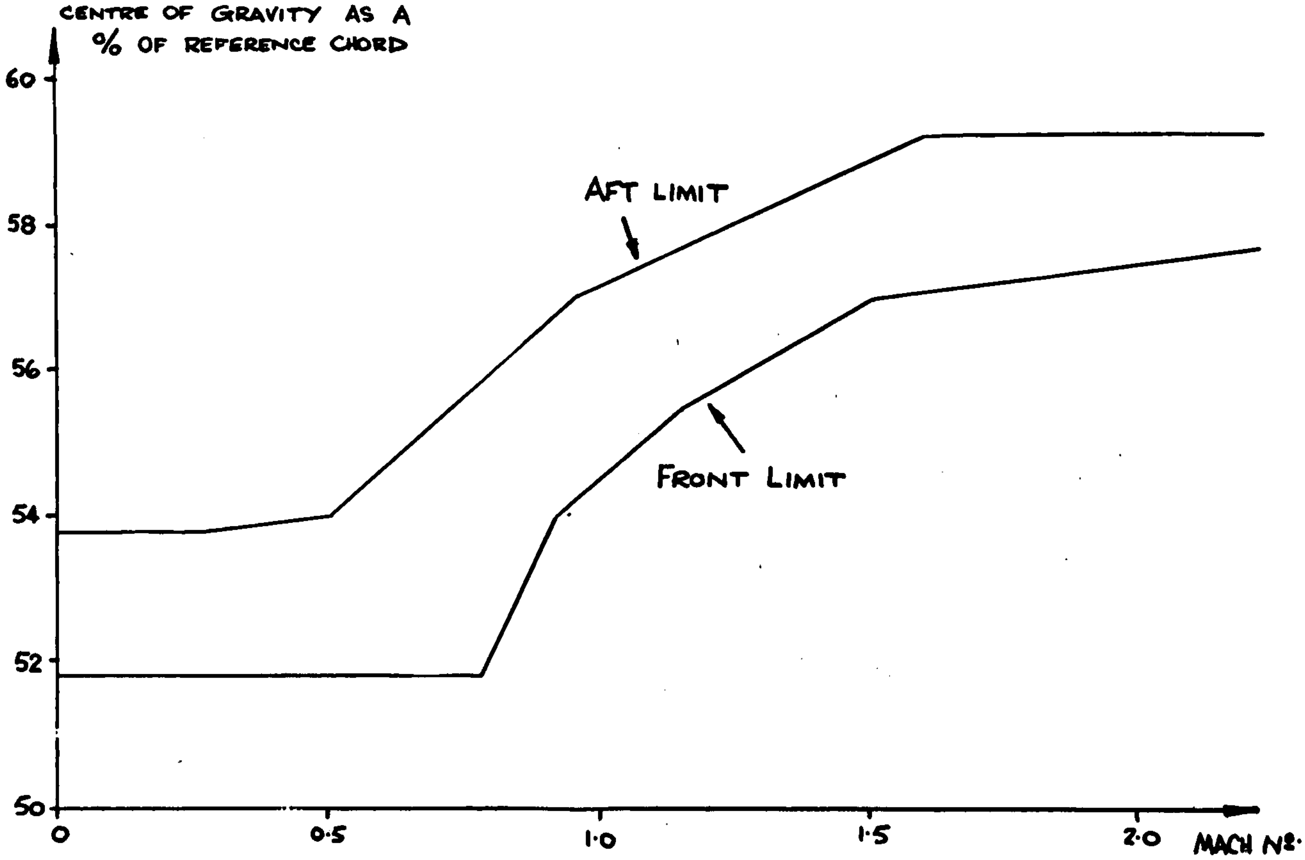


FIG 13 CENTRE OF GRAVITY FLIGHT ENVELOPE.

ROOT LOCUS θ/η SHOWING EFFECT OF PITCH AUTOSTABILISATION AT MACH 2·0

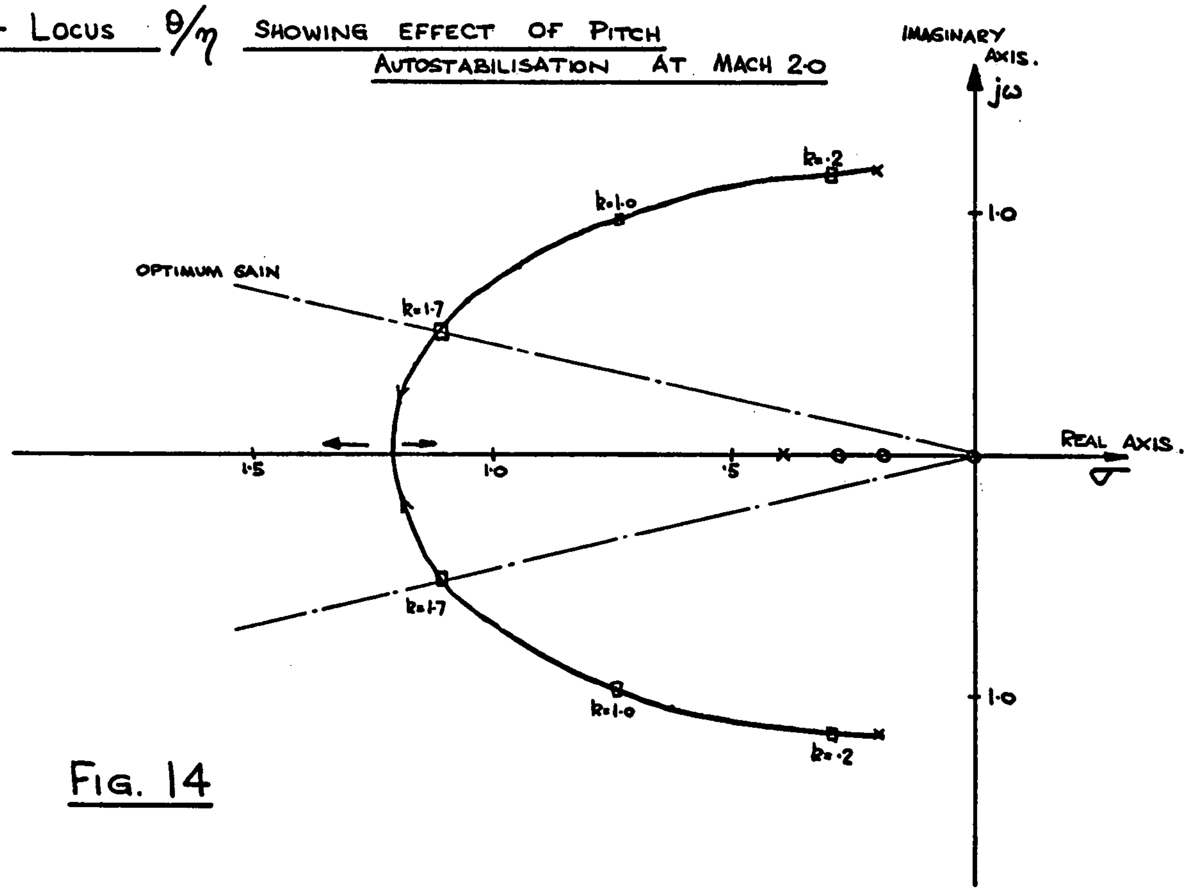


FIG. 14

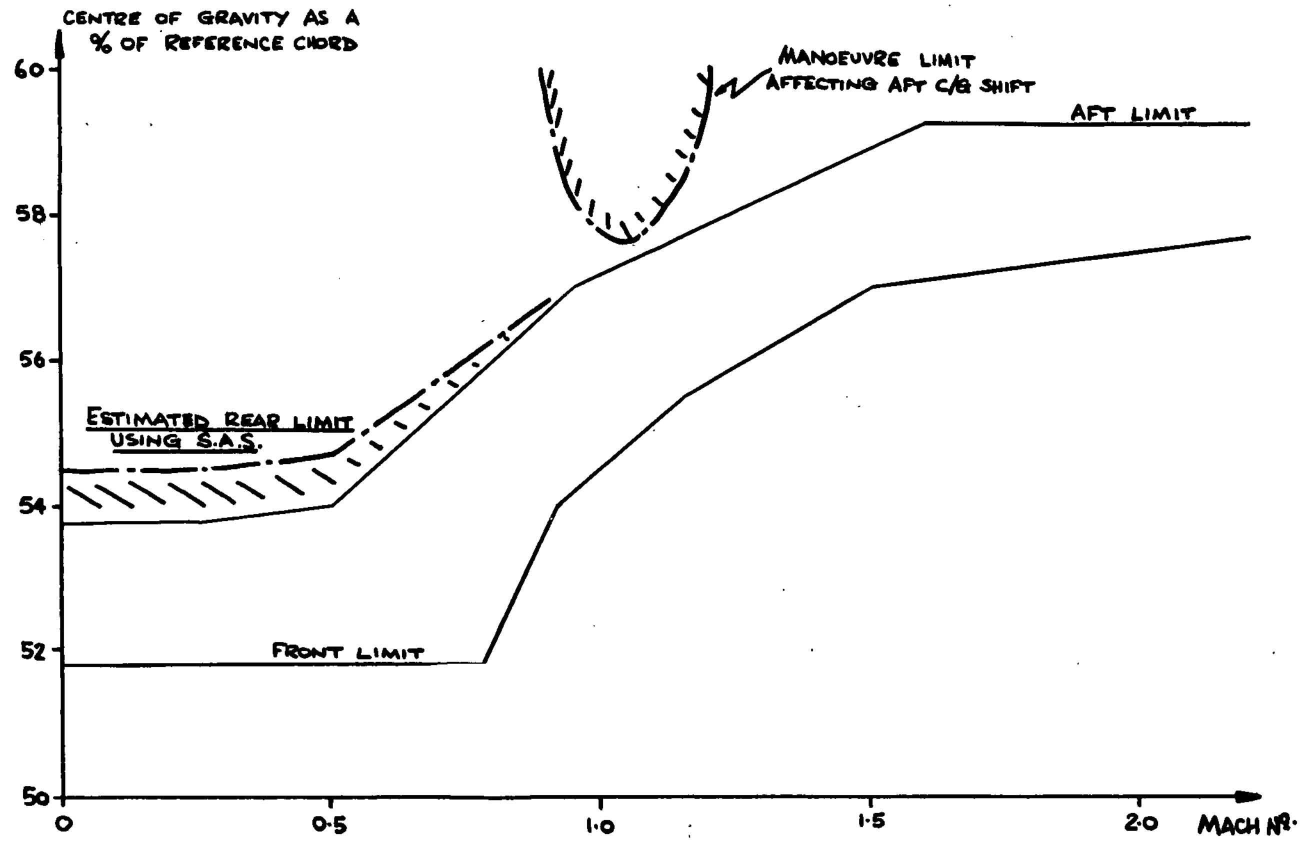


FIG 15

PROJECTED ENHANCEMENT OF C/G CORRIDOR FROM USE OF STABILITY AUGMENTATION

DESIGN OF AN ENTIRELY ELECTRICAL FLYING CONTROL SYSTEM

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SUMMARY :

After reminding the audience of the reasons for using entirely electrical flying controls, that is controls without mechanical standby systems, and defining the control modes available for a transport aircraft, we shall describe the general architecture of the system. We shall show that if safety requirements impose minimum redundancy, several precautions must be taken for the theoretical reliability achieved by this redundancy to be real. The equipment required is described briefly. From a maintenance point of view, the complexity of the system is compared with that of the flying controls on existing aircraft (A.300B, CONCORDE).

I. INTRODUCTION

More and more consideration is being given to the joint possibilities of electronics and hydraulics in aircraft design. The application envisaged (or achieved) are multiple, from the artificial feel system and yaw stabilizers of today to the flutter control system of tomorrow, or the day after. The possible effects of this on performance, structural stressing, aircraft control or the complexity of the systems involved, are numerous. It is the last point which we have chosen to discuss, very partially, by considering a fundamental phase, the phase when the aircraft is entirely dependent on electronics as it is now often entirely dependent on hydraulics.

The following lecture is the result of the experience acquired on the CONCORDE SST with a mechanical standby electrical flying control system (cf. fig. 1) and the research which the Aircraft Division of SNIAS has been carrying out for several years with a view to designing a purely electrical flying control system, essentially for use on civil transport aircraft. Nevertheless, we think that the lessons drawn from this research are at least partially applicable to military aircraft.

2. GENERAL

2.I. Motivation

For a civil aircraft the search for an entirely electrical flying control system (FBW) is dictated by economic considerations. The level of safety and performance of the existing mechanical systems (or systems with mechanical standby) are indeed satisfactory for present aircraft and only economy would allow their replacement to be justified.

The FBW must therefore allow either conventional aerodynamic aircraft to be optimized, economically, or enable improved performance aerodynamic shapes to be used.

The cost estimates which would justify the use of FBW will take the following conventional parameters into consideration.

- Possibility of enlarging the market, by improving the services rendered by the aircraft (improved confort, increased speed, possibility of serving airports with short runways, etc..)

- FBW development costs

- Cost of FBW series manufacture (including maintenance costs)

- Effects on the cost of other systems and structure

- Effects on fuel consumption

- Effects on aircraft availability

Of course, if the required degree of safety were provided and the economic assessment proved that conventional mechanical controls and FBW were on a par, preference would be given to the FBW because of its superior performance.

The applications envisaged for military aircraft are multiple :

- Improving weight and performance by reducing static margins and improving manoeuvre load distribution
- Improving performance by achieving better handling qualities
- Stabilizing vertical take off aircraft
- Damping flutter

In the transport aircraft field, it is essentially the second generation supersonic civil transport aircraft which seen to be the correct basis for the FBW, as the FBW enables naturally unstable aircraft to be controlled and frees the constructor from certain shape or loading constraints. It provides damping for structural modes and facilitates the design of "gust absorbers" which are interesting features for flexible aircraft design. It enables control surfaces to be positioned accurately (very low positioning thresholds), thus attenuating the problems which could be caused by flying in large control surface efficiency conditions. These advantages are also found when the FBW is applied to a subsonic aircraft, buy they are doubtlessly insufficiently noticeable, for aircraft like present subsonic aircraft, to justify the use of FBW designed with todays technology.

2.2. Safety objectives

We shall limit this point to a reminder of safety objectives in their most general form, that is in numerical form.

For a civil aircraft, the probability of a catastrophic incident due to systems must be less than 10^{-7} per flight hour ; a rough distribution of this risk can be made a priori, as follows : For the system providing aircraft control, it leads to a probability objective of less than 10^{-9} / flight hour for each of the two main types of failure (total failure - incontrollable runaway).

With the possible exception of an aircraft able to regain a safe landing area quickly, a triple control system at least is required to achieve these objectives. Economic considerations then call for take off being authorized with one item of electrical or electrohydraulic equipment failed. The result of this is that the system must be fourfold (or fivefold for fault isolation). This point will be discussed later.

3. CONTROL

3.I. Control modes

Three control modes are now available for civil transport aircraft:

- Automatic control : the aircraft is automatically slaved to a reference value the nature and amplitude of which are chosen by the pilot. The pilot's action is limited to this choice and to speed control via the throttles if he does not wish to use the autothrottle.
  The reference value can be held or acquired. The number of reference values available varies from one aircraft to another, but it is always fairly large (attitudes, altitude, heading, inertia guidance data, radio guidance data).
- Manual control. The autopilot is disconnected and the aircraft is controlled by the pilot
- Control wheel steering: the pilot controls the aircraft with the controls used for manual control, but the autopilot is not disconnected

These three control modes must be maintained :

- Automatic control must be maintained for its flight path guidance capabilities
- For safety reasons, it must be possible to control the aircraft manually after total autopilot failure (or at least, failure of all the flight path guidance functions).
- Once the previous two control modes exist, control wheel steering is justified because it enables the pilot to act as in manual mode but without having to disconnect then reconnect the autopilot.

The FBW is particularly well suited to the use of control wheel steering (cf. fig. 2). It enables the autopilot servo-actuators to be removed and provides good short term aircraft stability. From then on, as long as the pilot uses the manual controls, manual and control wheel steering mode characteristics can be very similar and even identical if so desired : the relationships between pilot control deflection and force and between force and aircraft response will be the same, whether the autopilot is engaged or not.

3.2. Control laws (manual)

Let us consider the use of an FBW for a supersonic transport aircraft. It is economically justified mainly if it provides :

- Improvement to L/D ratio by allowing flight at very aft c.g. locations

- Decrease in fin surface area and thus drag

- Possible structural mode damping

We shall then assume that the FBW must provide artificial aircraft stability throughout the flight envelope ; loss of this artificial stability will be considered catastrophic.

This artificial stability will be obtained by means of rate gyro and accelerometer detections.

If we consider longitudinal control in greater detail, we note that the conventional control laws (relationship between force and modulated deflection in terms of Mach, Vc and trim deflection ; superposition of stabilization deflection and the deflection controlled by stick operation ; correction of static stability by Mach and Vc detection) can provide satisfactory handling qualities.

However, investigations carried out to date show that C^* laws (vertical acceleration + pitch rate (cf. fig. 3) should provide more confortable control without increase in complexity. It thus makes the aircraft fairly insensitive to quick aerodynamic factor variations, when passing through the transonic phase for instance. This slight sensitivity to aerodynamic factors also greatly facilitates re-establishing the artificial static stability, which appears necessary for comfort throughout the flight envelope.

4. CONTROL SYSTEM ARCHITECTURE

4.I. General arrangement

The possible architecture of the system is defined in fig. 4.

The autopilot channels provide guidance command computation. These commands pass through the FBW which computes and carries out the required control surface deflections. Commands from the autopilot are limited in amplitude.

In manual mode, the control surfaces are positioned by the FBW according to pilot control deflection (or forces).

In control wheel steering mode, the FBW takes account of commands from the pilot controls as in manual mode, while the autopilot becomes synchronized.

We can see that the FBW is indispensable for controlling the aircraft ; the safety objectives defined are therefore wholly applicable.

4.2. FBW components

We have seen that the FBW must provide artificial aircraft stability. It therefore comprises :

- Pilot controls with electrical detectors

- Rate gyros and accelerometers

- Computers

- Servo-actuators and power servo-controls or electrically controlled power servo-controls.

4.3. Nature of connections between the various items of equipment

Connections with or without switching can be established between the items of equipment required for controlling one axis (cf. figs 5, 6, 7 and 8).

Based on "reasonable" MTBF assessments, we can consider that switching is only advantageous between computers and servo-actuators (or electro-hydraulic servocontrols).

4.4. Level of redundancy

For the "channels" comprised of pilot control position detectors, rate gyros, accelerometers and computers, the calculations show that five channels with external monitoring or four self monitored channels are required. In the latter case, perfect self-monitoring (undetected failure rate $< 10^{-9}$ / hour) would lead to excessive complexity. It is preferable to have partial self-monitoring (undetected failure rate $< 10^{-6}$ / hour) completed by external monitoring.

In the servo-actuators (or power servo-controls) the level of redundancy required will depend on the efficiency of the control surface (s) controlled.

4.5. Quality of redundancy

Probability calculations show that four or five computation channels are required to comply with the safety objectives. These calculations have a meaning only if they are accompanied by systematic troubleshooting.

This systematic troubleshooting requires a certain number of precautions which are conventional but which need to be reinforced :

- Geographic segregation of channels

- Fire protection in certain zones

- Protection from the propagation of electrical interference voltages which could be destructive

- Possibly, protection from ventilation failures

Other precautions, which have not yet become usual practice, must be taken to provide protection from programming errors (see below) and inadvertent disconnections of the complete system by its monitoring. This last point merits special attention.

The first monitored electronic systems caused frequent disconnects for no reason. Now that the channels are divided into portions and voters or procedures specific to certain monitoring systems are used the frequancy of such disconnections should no longer be a problem on aircraft now in service thanks to the present use of electronics.

However, monitoring systems constitute a weak point in electronic systems. The fairly low ratio between the normal tolerances of the monitored items and the maximum acceptable values of monitoring system time constants and thresholds often makes them more sensitive to failures than the systems in their principal functions would be. In addition to this, and above all, threshold settings and time constants prevent disconnections in normal operation in a certain envelope of input variation only; this envelope can become very difficult to define when the disconnection frequency must be very low and it is possible that the estimated envelope does not correspond to the real envelope.

It therefore seems, at least at electronic level, that total disconnection of the FBW must be made impossible by its very design (for example, provision of an unmonitored standby channel, or a mechanical or electrical logic preventing total disconnection).

5. EQUIPMENT

5.I. Position, angular velocity and acceleration detectors

Conventional analog detectors suffer from a certain number of defects :

- low accuracy

- presence of thresholds - hysteresis for rate gyros and accelerometers.

As far as the latter point is concerned, precautions must be taken when defining the control laws. We can hope that it will be possible to use torsion bar accelerometers and gyros, which are the simplest.

As for as low accuracy is concerned, it is mainly troublesome at servocontrol level. Digital detectors could thus be used to do away with the mechanical synchronisation required with certain types of servocontrols, but this would be at the cost of excessive electronic complexity and wiring weight. Conventional analog detectors are therefore preferred.

5.2. Cockpit controls

The FBW pilot controls, which are destined to drive some electrical detectors, will be different from those of existing civil transport aircraft.
For a supersonic aircraft flight deck, one could go as far as two sticks one to the left of the pilot and the other to the right of the copilot, or at a "desk" carrying the controls. In the first case, pitch and roll would be controlled via these sticks, which, unlike existing control columns, would not be mechanically linked together.
One is tempted to imagine control columns without articulations, but technical difficulties and habits make the use of such columns improbable with the first generation of FBW as they would require force control with very small deflections.
In any case, it will always be possible to choose pilot controls which would avoid possible problems of reconversion for pilots accustomed to conventional controls.

5.3. Computers

Digital computers (cf. fig. 9) are preferred to analog computers. In addition to computing surface deflection commands they will be required to carry out a certain number of monitoring functions. In particular, we intend requiring the computers to carry out their own monitoring. Investigations show that this self monitoring cannot be perfect. In fact, it would not be very advantageous to achieve a non-self detected failure rate $< 10^{-7}$ / hour. It would even be difficult to achieve a failure rate of $\sim 10^{-7}$ / hour, and if the undeniable advantage of achieving such an objective justifies present efforts in this direction, today the result must be considered uncertain.

Digital computers raise another interesting problem : the problem of programming.
The use of digital design computers makes one sensitive to the fancies of computers which are not perfectly programmed.
It is true that the calculations required of the FBW computers will be far simpler than those required from design computers, but since an incorrect calculation could break the aircraft in a fraction of a second, one must be certain of the FBW programming to "IO^{-9} / hour"; this will perhaps lead to simultaneous use of three different programmes, (or simultaneous use of two different programmes with switching to a standby channel should there be disagreement).

5.4. Servo-controls - hydraulic systems

We know that the FBW raises servocontrol problems because of the redundancy required and the normal tolerances between channels.

If we dispose with the use of intermediate servo-actuators, we are led to consider two types of servocontrols.

- Coupled synchronized distributor servocontrols.

- Separate servocontrols where the synchronization faults are absorbed by the spring items always present between the servocontrols.

The first type of servocontrol does not cause structural stresses, single failures (and sometimes even triple failures) of the servo-channels do not affect available power. On the other hand, distributor synchronization is complicated to achieve.

In fact, the choice of servocontrols will depend on the choice of hydraulic systems, which is considerably affected by FBW requirements. It is interesting to note that the FBW will add no further complication to existing hydraulic systems.

It should also be noted that servocontrol servo-loop must be made outside the digital computers because of the bandwidth required.

5.5. Electrical power supply

The electrical power supply to the FBW must be provided without interruption.

The best solution seems to be the introduction of static inverters between the aircraft network and the FBW, the aircraft batteries being buffer installed.

5.6. Data sources

The control laws must be adapted for each point of the flight envelope.

There are two possibilities :

either :
use of data specific to the system only (self adaptive system) ;

or :
use of data outside the system.

In fact, we can consider that both these possibilities are used in the flying controls of civil transport aircraft ; control column force stiffness which the pilot or autopilot cancels in balanced flight conditions is dependent on flight conditions such as speed and altitude measured outside the system, and trim deflection, the value of which is directly controlled by the system.

In future purely electrical flying controls, it is almost certain that external data will be used: for example why deny ourselves data which is easily available in digital electrical form, such as Mach, speed or altitude which an air data computer would have to compute anyway ?
But the usual reliability of these electrical sources of information is not as good as that of the flying controls. We therefore have to look for a self-adaptation of the laws which is eitheir permanent and usually aided by external information which enables it to be optimized, or used in the event of loss of external sources of information. It will indeed be seen that the pitch control law previously envisaged (accelerometer and gyro feedback followed by an integrator) provides self-adaptation of stick forces to load factor.

6. MAINTENANCE

6.I. System complexity

We estimate this complexity in relation to the complexity of existing aircraft flying controls by the total volume of electronics and the number of electrical actuators, items of electrohydraulic equipment and force or position detectors.

In flying controls, we include artificial stabilization devices, pitch, role and yaw controls and slat and flap controls.

A comparison is made in figures IO and 11.

We have voluntarily excluded from this comparison rate gyros and accelerometers the quantity of which is defined essentially by the handling qualities of the natural aircraft and not by the flying control system chosen.

It is to be noted that the differences between a subsonic aircraft and a supersonic aircraft make the aerodynamic formula intervene very directly. The comparison is thus really made between "conventional subsonic aircraft aerodynamics with mechanical flying control" and "delta ving with electrical flying control".

6.2. Number of equipment types

Once again we shall compare the FBW with the flying controls of a subsonic aircraft and those of a supersonic aircraft. This comparison is given in figures I2 and I3.

6.3. Fault isolation at interchangeable "box" level

The FBW is able to provide fairly efficient natural failure isolation, by means of its self-monitoring. By "natural" we mean isolation achieved practically without adding anything to the equipment which is functionnaly required, and which only requires the maintenance staff to read a warning light and possibly depress a test puch-button.

EQUIPMENT	NATURAL ISOLATION RATE
Digital computer	90 %
Rate gyro	80 %
Monitoring unit	95 %
Position detectors Accelerometer	The low failure rate of these items of equipment requires the isolation to be carried out redundantly (e.g. when a failure has been indicated by the system, use of ground equipment to check the faulty accelerometer or detector in situ using ground connections).
Amplifier - servovalve	Servovalve failure cannot be naturally distinguished from an amplifier failure within a servo-loop.

6.4. GO / NO-GO items

Failure of electric or electrohydraulic equipment in the flying controls of aircraft flying today prohibits take off in exceptional cases only. The same must be true for the FBW and this has a fundamental effect on the design of the system, in particular as far as its redundancy is concerned. This requirement has of course been taken into consideration in the comparison made above.

6.5. New items

This description of the FBW shows that the only new items of equipment which it comprises for flight control are the digital computers. For all the other items of equipment (detectors, servocontrols) the airlines have already mastered the maintenance problems.

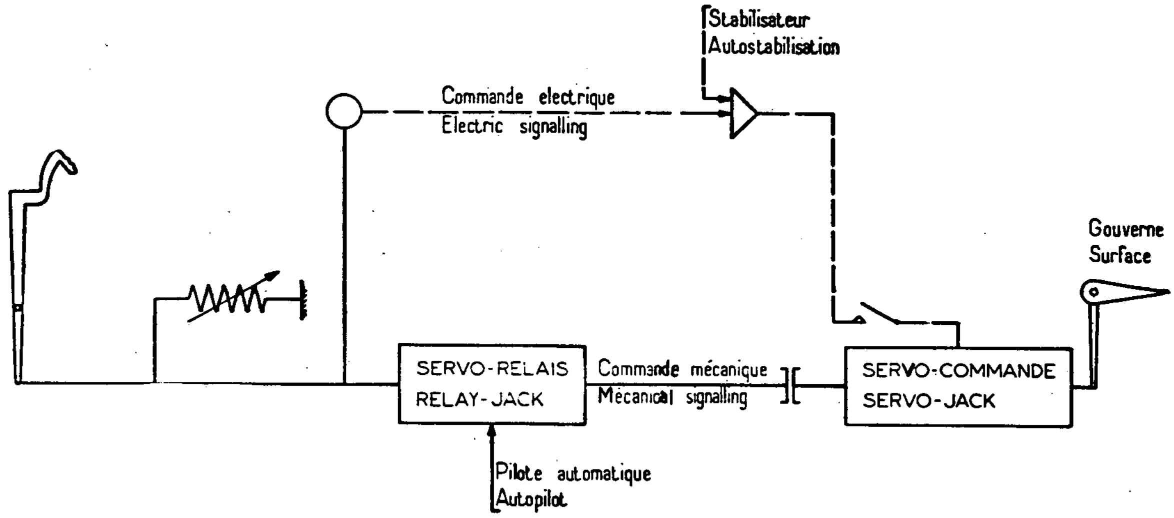
As far as the digital computers are concerned, although they are new for the main flying controls, they will already be conventional when the FBW is introduced as far as inertia platforms, autopilots, A.D.C's and perhaps management computers are concerned.

7. CONCLUSIONS

We now know how to design purely electrical flying control systems with an adequate safety level for a civil transport aircraft ; but their economic justification is at least as dependent on the aircraft on which they are to be applied as on their own design. As far as their design is concerned considerable optimization work still remains to be carried out : if we had to design FBW for commercial operation today, the safety obligations would certainly make them uselessly complex even without the help of certification requirements. But design investigations should lead to a system comprising a small number of components of different types, without special requirements for hydraulic systems, and with small requirements (buffer batteries) or no requirements at all for electrical generation. This would facilitate flight deck installations by introducing the possibility of miniaturizing pilot controls, improve control by means of its accuracy and natural adaptation to control wheel steering and, according to the aircraft on which it is used, allow flight at very aft c.g. locations, a decrease of fin and control surface areas, and structural mode damping.

PLANCHE I

FIG. I



PRINCIPE DE LA COMMANDE D'UNE GOUVERNE DU TSS CONCORDE

CONTROL PRINCIPLE FOR A CONCORDE CONTROL SURFACE

PLANCHE 2

FIG. 2

PILOTAGE MANUEL, AUTOMATIQUE ET TRANSPARENT

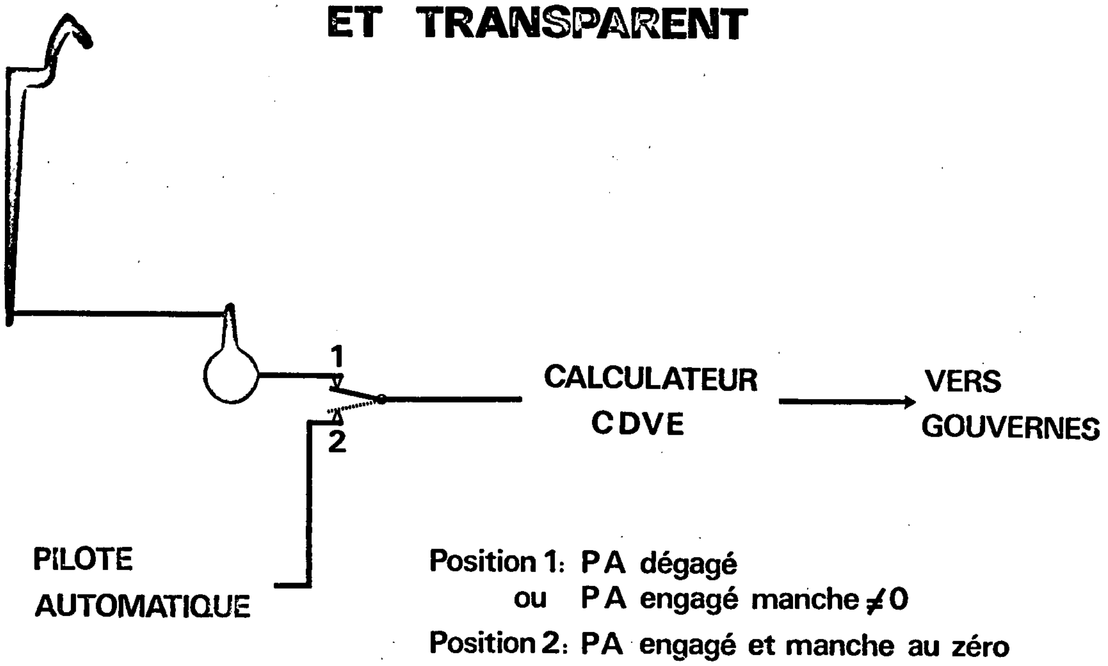
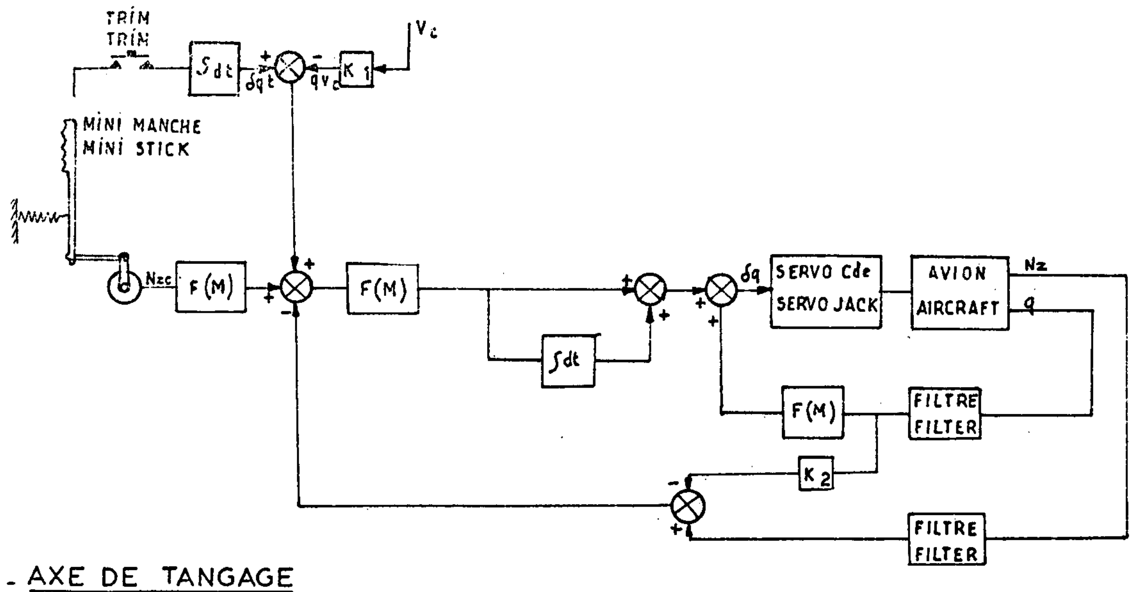


PLANCHE 3

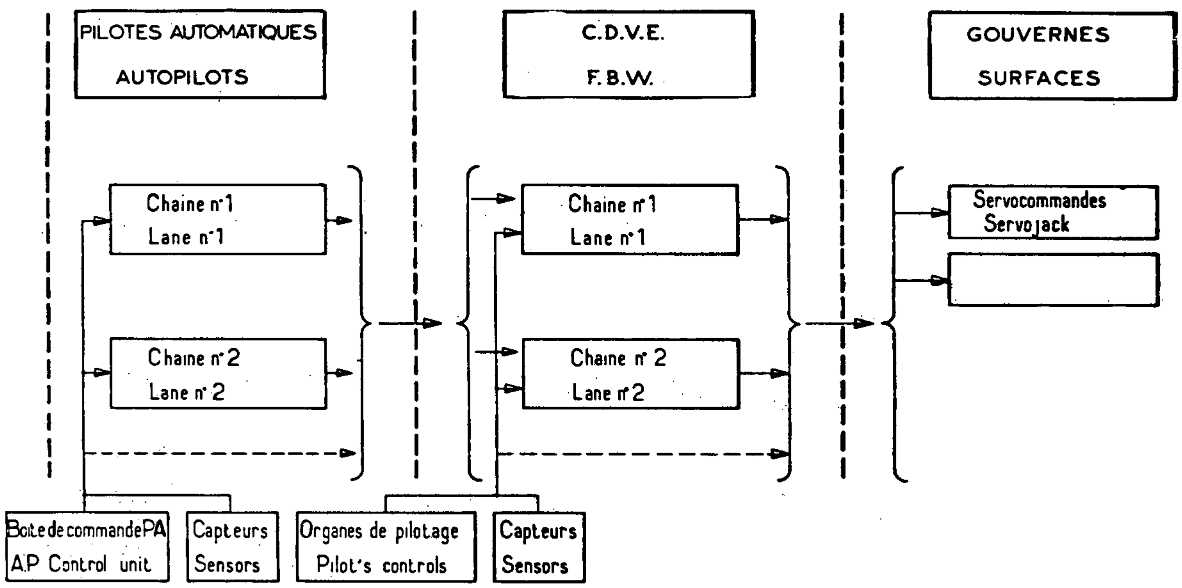
FIG. 3



- AXE DE TANGAGE
  - LOI DE PILOTAGE A RETOURS GYROMETRIQUE q ET ACCELEROMETRIQUE N_z
- PITCH AXIS
  - CONTROL LAW USING RATE GYRO q AND ACCELEROMETER N_z FEEDBACK

PLANCHE 4

FIG. 4

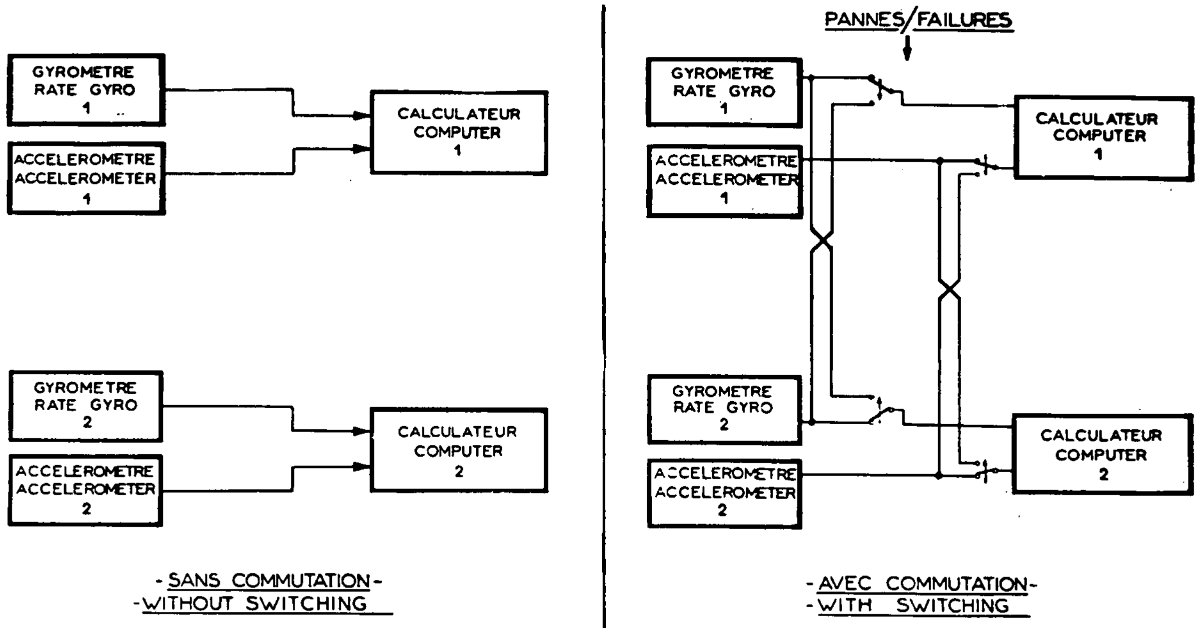


SCHEMA D'ENSEMBLE DES C.D.V.E.

F.B.W. BLOCK DIAGRAM

PLANCHE 5

FIG. 5

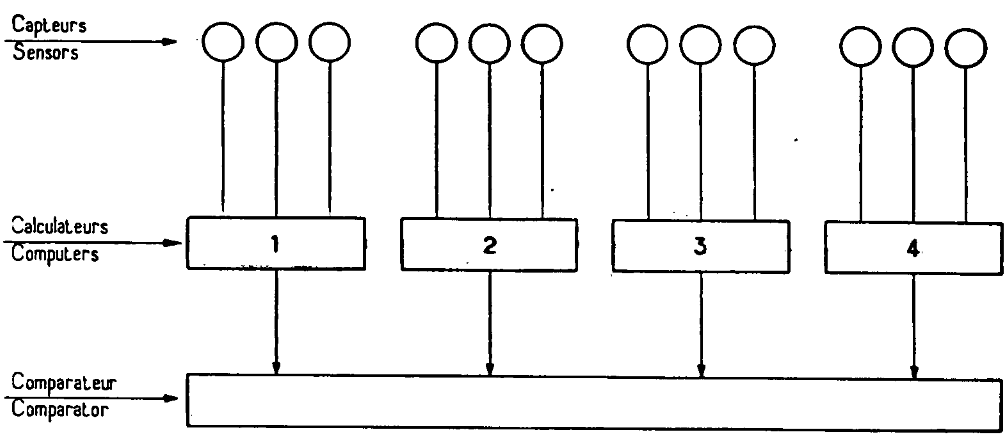


LIAISONS ENTRE EQUIPEMENTS

CONNECTION BETWEN EQUIPMENTS

PLANCHE 6

FIG. 6



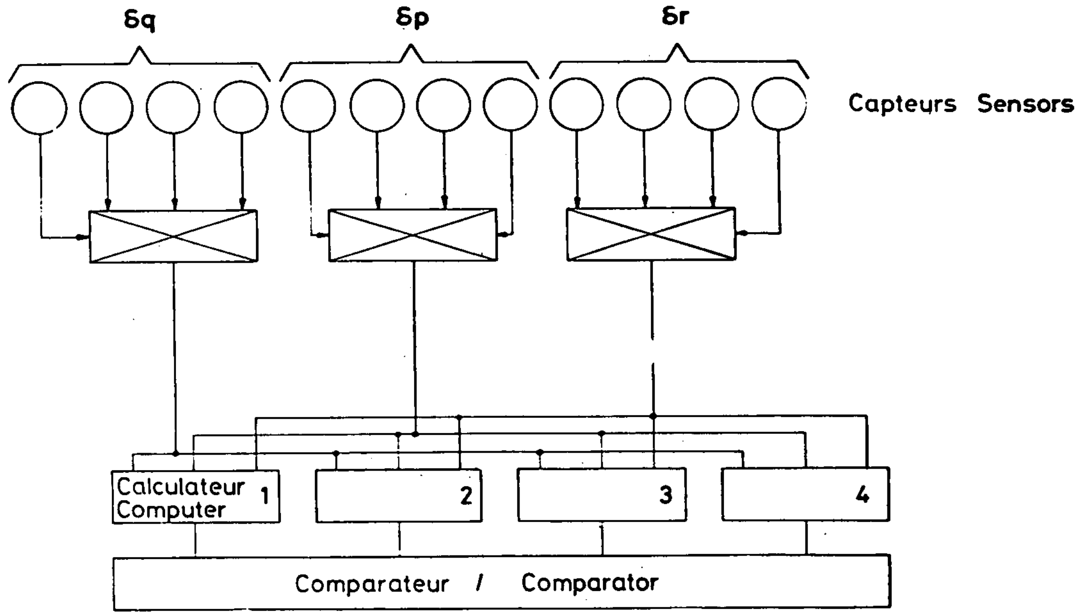
SCHEMA DE C.D.V.E. : PREMIER EXEMPLE

F. B. W. BLOCK DIAGRAM : FIRST EXAMPLE

Risque moyen/Average risk: $2{,}3 \cdot 10^{-9/h}$

PLANCHE 7

FIG. 7

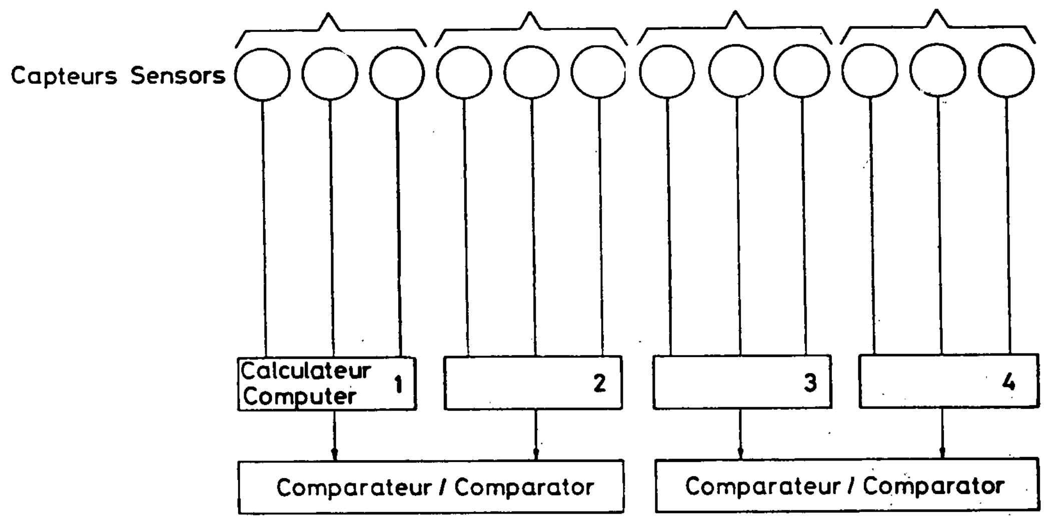


Schema de C.D.V.E. — Deuxieme exemple —
F.B.W. block diagram — Second example

Risque moyen / Average risk : $0{,}58.10^{-9}$/h.

PLANCHE 8

FIG. 8

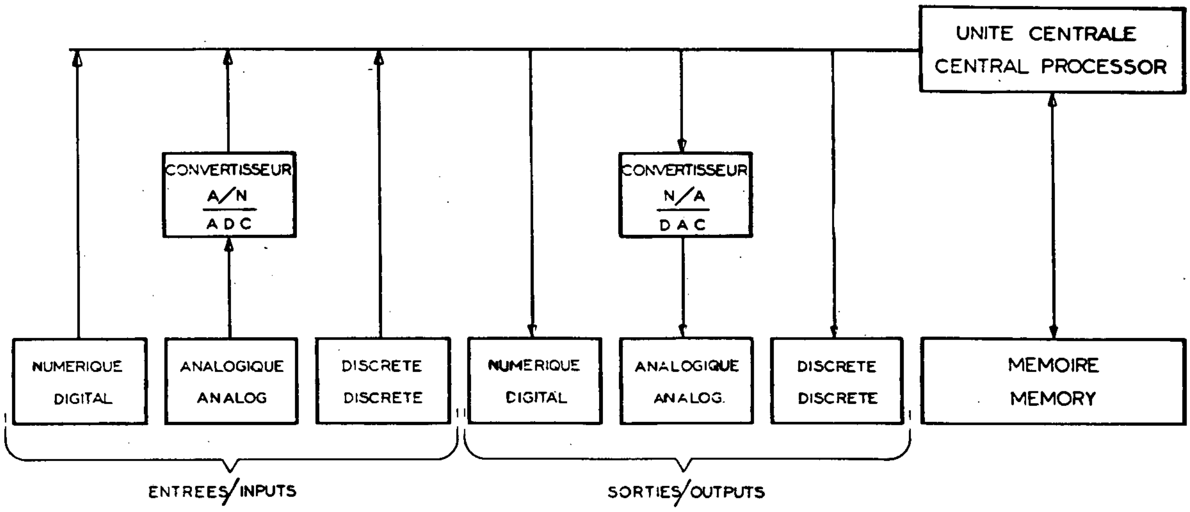


Schema de C.D.V.E. — Troisieme exemple
F.B.W. block diagram — Third example

Risque moyen / Average risk : $1{,}7.10^{-9}$/h.

PLANCHE 9

FIG. 9



CALCULATEUR NUMERIQUE
DIGITAL COMPUTER

PLANCHE IO

FIG. IO

EQUIPEMENTS EQUIPMENTS	A 300B	TSS CONCORDE	C.D.V.E. F.B.W
ELECTRONIQUE (Volume) ELECTRONICS (Volume)	1	1,6	2
SERVOVALVES (Nombre) SERVOVALVES (Number)	1	2,3	1,6
ELECTROVANNES (Nombre) ELECTROVALVES (Number)	1	1,9	1,4
DETECTEURS DE POSITION (Nombre) POSITION PICK OFFS (Number)	1	2,2	1,5

COMPLEXITE COMPAREE

COMPARATIVE COMPLEXITY

PLANCHE 11

FIG. 11

EQUIPEMENTS EQUIPMENTS	A 300B	TSS CONCORDE	CDVE FBW
MOTEURS ELECTRIQUES (Nombre) ELECTRIC MOTORS (Number)	1	0,5	0
VERINS à VIS (Nombre) SCREW JACKS (Number)	1	0	0
REGULATEURS de TENSION de CABLE (Nombre) CABLE TENSION REGULATORS (Number)	1	0,33	0

COMPLEXITE COMPAREE

COMPARATIVE COMPLEXITY

PLANCHE I2

FIG. I2

EQUIPEMENTS EQUIPMENTS	A 300B	TSS CONCORDE	C.D.V.E. F.B.W.
BOITERS ELECTRONIQUES COMPUTERS	1	1,5	0,85
SERVOCOMMANDE ELECTROHYDRAULIQUE ELECTRO-HYDRAULIC SERVOJACK	1	2	1
SERVOCOMMANDE HYDRAULIQUE HYDRAULIC SERVO-JACK	1	0	0

TYPE D'EQUIPEMENTS

TYPE OF EQUIPMENTS

PLANCHE I3

FIG. I3

EQUIPEMENTS EQUIPMENTS	A 300 B	TSS CONCORDE	C.D.V.E. F.B.W.
MOTEURS HYDRAULIQUES HYDRAULIC MOTORS	1	0	0
CAPTEURS DE POSITION ET D'EFFORT POSITION AND FORCE SENSORS	1	2,1	0.55
MOTEURS ELECTRIQUES ELECTRIC MOTORS	1	0,5	0

TYPE D'EQUIPEMENTS

TYPE OF EQUIPMENTS

THE HUNTER FLY-BY-WIRE EXPERIMENT: RECENT EXPERIENCE AND FUTURE IMPLICATIONS

by

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SUMMARY

The impact of active control technology on the design of future aircraft depends on the development of full-time and full authority control systems which have an integrity similar to that of the basic airframe. One of the major items of the R and D Programme in the UK which is aimed at providing this technology is the experimental quadruplex fly-by-wire system installed in a Hunter aircraft. Recent flight experience with this system is described in the paper.

Discussion of the implications of the future application of active control technology is restricted to the airworthiness problem: the manner of designing systems so as to ease the certification of high integrity, full-time and full authority control.

1. INTRODUCTION

At an AGARD Conference in Geilo, Norway, in September of last year, we presented two papers[1,2] describing the research and development programme in the UK which is aimed at providing more effective control technology. Much of this continuing programme is concerned with the principles and, in particular, the practice of full-time, full authority control, e.g. full time fly-by-wire.

With current in-service active control systems, the pilot is always able to revert to the basic flying controls when he requires or when the active control fails. For this reason, the aircraft, through its basic mechanical flying controls, must be controllable and flyable without the active controls. Further exploitation of active controls can be anticipated whilst maintaining this reversion facility, e.g. manoeuvre load control, improved ride, etc.

However, in considering the impact of active control technology on the design of the aircraft, the implication is that reversion to the basic flying controls will be impossible. It is assumed that full-time, full authority control will be available; and, in particular, that confidence can be established in the standards of reliability and availability required of the active control engineering.

It was realised, some years ago, that it was essential to develop and evaluate representative control systems of this kind in the context of a real aircraft. Without a real aircraft application, in the foreground of more general investigations, there is a tendency to overlook vital practical problems and a further tendency to apply a more advanced or more elegant technology than that which is realistic and practical for a high integrity, full-time control system.

A major item in the UK programme, therefore, continues to be the experiments with a two seater Hunter aircraft that has been fitted with an experimental quadruplex FBW system. Experience with this system tends to be the basis for appraisals of more advanced systems that are being studied in the UK and elsewhere.

2. DESCRIPTION OF THE HUNTER FBW SYSTEM

2.1 Aims of the flight experiment

The principal aim of the experiment was (and remains) to establish and solve the practical problems of an active control technology that would possess an integrity approaching that of the basic flying controls of a conventional aircraft. A further aim was (and remains) to examine further the extent to which feedback control can be used to modify the dynamic characteristics of the aircraft so as to produce improved pilot performance in executing difficult flight tasks.

The engineering design was made in close association with 'theoretical' investigations and this has allowed (and continues to allow) reasonable extrapolation from the results gained from the chosen practical and experimental solution. From the start of the programme, some years ago, it was necessary to define the system principles, in particular, the method of achieving the desired level of multiplexing; and to specify hardware components which were developed and tested during the programme.

For this experimental system, feedback control has been restricted, so far, to the use of rate gyros installed near the centre of gravity.

These feedback signals, together with stick and pedal pick-offs, yield full authority control in the three axes, elevator, aileron and rudder. Shaping and filtering of these signals is made using analogue components.

2.2 Multiplexing arrangements

When aircraft are designed to rely fully on active controls, failure of the control system could result in the loss of the aircraft. The integrity of the control system must therefore approach that of present day mechanical flying controls, and it is necessary to employ redundancy. A double failure survival system was therefore designed such that the system would remain fully operational after failures in two lanes in the same axis. This was (and remains) necessary in order to accomplish the integrity requirements with the expected failure rate of a single lane.

Multiplex systems based on complete isolation, electrical and geographical, between lanes and axes are attractive in reducing the chance of a failure in one lane causing a failure in another (common mode failure). In addition, it is easier with such configurations to set up the system and to diagnose faulty components. It was therefore decided to apply this type of system and to gain practical experience of the problems with such a system.

The system as installed therefore represents an example of a double failure survival 'fly-by-wire' system using no direct electrical signal equalisation or voting within the basic signal paths.

It was recognised that the Hunter aircraft was not an ideal vehicle for the practical exercise since it is subsonic and has a single engine. It was considered impractical to provide redundant hydraulic supplies and this is the reason for the retention of the mechanical controls. However, all other problems associated with a double failure survival full-time full authority control had to be solved.

2.3 Outline of the system

Referring to Fig.1 at the heart of the system are four lane packs, each of which contains one lane of pitch, roll and yaw computing, power supplies and a three axis gyro package. Pilot inputs are derived from quadruplex command pick-off assemblies. Outputs from each of the four lane packs are fed to each of the three quadruplex electro-hydraulic assemblies, one each for elevator, aileron and rudder. Mechanical consolidation occurs at the output of each actuator pack.

As stated above, there is complete electrical isolation between signals. The quadruplex actuator pack functions as both the consolidation point and the lane error (fault) detector.

Each pack consists of four individual actuators which are connected to a common output shaft of high integrity. A cam operated microswitch assembly detects a misalignment between each piston and the output rod. In the event of a misalignment the switch assembly operates to identify and present a warning of a faulty lane.

For the experimental system, failure warnings are presented to the pilot who can disengage a faulty lane on receipt of the warning, conversion from quadruplex to triplex operation being made by equalisation of the pressure in the faulty lane. The system is automatically disengaged under certain conditions but these are associated with the simplex nature of the aircraft hydraulic supply.

Provision has been made to fly a conventional autopilot with the manoeuvre demand system.

2.4 Control used in current flight experiments

The control implemented in the system being flown at the present time is shown in Fig.2. The stick pick-off signals and elevator/aileron gains are scheduled using ADS data. On detected failure of the ADS, which is duplex only, the system reverts to a low gain which is reasonably acceptable in all flight cases.

The integral term in the pitch rate to elevator control is also duplex. Failure of the integrating servo results in the output being frozen unless modified by the pilot via a direct electrical link (trim).

3. FLIGHT EXPERIENCE AND RESULTS

3.1 Experience with the quadruplex system

It was originally intended, on first flight, to take-off, fly a limited sortie and land using the fly-by-wire system. This is an essential requirement for any aircraft that is designed to be unflyable and uncontrollable without the active control system. It was, however, necessary to test in flight, the ability of the pilot to revert to the basic flying controls in the event of a failure of the simplex hydraulic supplies (one such failure has occurred). The first two flights were made for this purpose and included engagement and disengagement of the system in flight.

There was, however, full confidence in the system itself to have permitted its use in the first flight and from the third flight onwards complete sorties were made including take-off and landing. Much of the flight envelope has been covered although manoeuvres have been restricted in the flying to date to 3 g to -1 g. This restriction was dictated by the continuing need for each part of the flight envelope to establish the pilot's capability to revert to the basic flying controls in the event of a hydraulic failure. Manoeuvres have included flight and approach to a low speed stall at height (into buffet).

Pilots comments on the system have been favourable and they have considerable confidence in the system engineering and method of use. In fact, adverse pilots comments have been restricted mainly to the dynamic characteristics of the controlled aircraft, viz performance aspects. In particular, to the aircraft response to stick input which is to be a variable in future flight experiments.

3.2 Lane equalisation

There have been a number of problems in the development, implementation and use of the system which are relevant to the present discussion, the practicality of full-time, full authority control. One of these is associated with lane equalisation. Any difference in gain in multiplexed lanes will result in an erroneous displacement of the actuator outputs, depending on the characteristics of the disturbing signal. Consolidation of the electrical signals could ease this problem but leads to latent possibilities of common mode failures, i.e. failures in one lane causing failure in another lane.

Experience with the experimental quadruplex system has shown that the control laws can be designed to permit electrical separation in the absence of any integral term. The gain associated with any effective integral term is too high to permit lane equalisation without consolidation. The pitch rate integral term in the Hunter FBW system has been changed from the separated quadruplex lag-lead filter (gain 10) to a duplex integrating servo which can be switched out by the pilot. The output of this integrator is consolidated before it is applied to the four lanes of the basic system.

This change was made as a result of further 'performance' considerations in addition to the practical problem associated with lane equalisation. As discussed further in section 4 below, certain control terms are not essential to make flyable and controllable an aircraft which has been designed to rely on the active control for stability and control. Therefore, it is not essential for this integral term to have the highest integrity provided, on failure, no large false signal is applied to the elevator.

3.3 Effects of gyro noise

Another practical problem is associated with the compromise between the speed of response of this system and the effects of sensor imperfections, e.g. gyro noise, both instrumental and from pick up of structural motion. If these sensor imperfections are not considered in the design, the feedback terms can be chosen to dominate the natural dynamic characteristics of the aircraft. It is the consideration of sensor imperfections and their effects which limit the impact of feedback control.

Attention was given to this problem in the original design of the Hunter FBW system. An automatic parameter optimisation procedure was applied using a hybrid computer. In the pitch to elevator channel, for example, the pitch rate error due to an external disturbance such as turbulence was minimised. In the optimisation the effects of gyro noise, as measured in the laboratory, was constrained to be less than a value yielding just unacceptable actuator movements.

Results from first flights demonstrated the merit of the design. Under certain circumstances, however, an unacceptably large burst of noise at 10 Hz was experienced, Fig.3. This is attributed to pick-up of a structural mode excited during ground runs and operation of the undercarriage. Since this structural mode may be excited in other flight cases, e.g. in buffet, a redesign was carried out so as to attenuate more strongly 10Hz noise at the gyro. As a result, the speed of response to either a pilot's input command or an external disturbance has been decreased.

3.4 Control effectiveness

The merit of the design of the control feedback depends on the accuracy of the modelling of the aircraft's dynamics. The more dominant the feedback terms the less important the accuracy of the model used in the design. The limit imposed by consideration of sensor imperfections implies the need for more accurate modelling which, for aircraft designed to be unflyable without the active control, may not be available before first flight.

The most important term in the dynamic model of the aircraft appears to be the effectiveness of the motivator and its variation over the flight envelope. Theoretically, the gain of the control loop can be varied to maintain constant the overall control effectiveness but there are no known means of making this variation with the integrity required. In the Hunter system, the gains are changed with airspeed but this is implemented using duplex ADS and has therefore less integrity than the basic system. Failure of the ADS results in a change in gain to that suitable (as a fixed gain) across the flight envelope.

Studies continue on possible systems that would permit appropriate gain variation with high integrity but there is yet no conclusive evidence that systems of this kind will be practical. It is realised, therefore, that the impact of active control technology on aircraft design could be limited by this factor.

3.5 Performance aspects

Flight experiments in the Hunter are being aimed at a re-examination, in an experimental environment, of the possible performance benefits of full authority control. This part of the flight programme has just begun but has already produced interesting results. Fig.4 shows records obtained from a straight and level flight in moderate turbulence, with and without the FBW control. An interesting feature is the significant decrease in stick activity with the controlled vehicle although other state variables have not changed significantly. Similar results have been obtained in the lateral axis and for other flight cases and manoeuvres. With the help of the Cranfield Institute of Technology, flight records of this kind are being analysed in order to help find some measurement criteria with which to design the 'optimal' active control. In addition, flight simulator investigations are due to commence at RAE Bedford in the Autumn to help with this kind of investigation.

3.6 Future flight programme

At the time of writing this paper, the FBW Hunter has re-started its flight programme from RAE Farnborough. Following the flight clearance to 6 g to -2 g across the main part of the flight envelope, it is planned in the near future to:-

(a) flight test the miniature side stick controller (Fig.5)

(b) add further feedback terms, e.g. accelerometers

(c) explore the flight and control boundaries of the controlled vehicle

(d) support flight simulator investigations into the 'best' full authority control to use for different high performance flight tasks.

The enforced grounding of the aircraft during the last year has allowed time to prepare for these items of the programme and we anticipate a much speedier progress in the immediate future.

4. SOME IMPLICATIONS OF CURRENT INVESTIGATIONS

4.1 System structure using digital technology

The technology and techniques applied to the Hunter FBW system represented the state of the art available some five years ago. Since then, there have been rapid developments in the UK and elsewhere, particularly in the applications of digital processing. For conventional applications such as limited authority autopilots and autostabilisers, digital multiplexed systems including optical highway data transmission are available and should overcome many of the problems encountered with current in-service systems, e.g. availability and testing.

The possibility of a common mode 'software' failure, affecting all lanes in a multiplex digital solution, continues to cause doubt on the application of this new technology to the full-time, full authority active control systems. If a central digital processor is used to perform all computational tasks for flight control (and perhaps for other functions such as engine control, navigation, etc.), it is essential to test exhaustively for possible common mode failures, including 'software failures', in that part of the system essential for basic control and stability. A significant part of recent investigations in the UK, practical and theoretical, is aimed at solving this problem for application to the fighter attack aircraft requiring full-time active control for its basic control and stability.

One possible solution to this problem is to separate the computation of the high integrity part of the control from the multiplicity of other computations. It would be advantageous if the high integrity part of the control is as simple as possible, minimising the number of inputs to, outputs from and signal paths through each 'dedicated' computer. Each dedicated computer could then be subject to exhaustive and visible tests through the complete range of possible combinations of input, output and signal paths in order to establish confidence that the desired integrity will be met. It would be a further advantage if the authority of all control signals was limited other than those in the high integrity (dedicated) part of the system, even if 'performance' suffered as a result. If this can be arranged, transient effects of switching off these additional loops will be reduced.

As illustrated by the following example, the feasibility of this approach depends on including the integrity requirements in the formulation of the control laws.

4.2 Choice of control laws

Fig.6 shows the structure of a possible system permitting the pilot, via a mode selector, to select different controls in the vertical plane:-

(a) to modify the aircraft motion to external disturbances such as turbulence using different control loops (via switches SW1, SW2 and SW3)

(b) for each control loop, prefilters F are used to obtain preferred aircraft response to a pilot's stick command, both steady state and transient response.

The main loop (1) can be designed to yield the steady state and most of the transient response to a pilot's demand so that, for this input, the authority of all other closed loop feedback paths can be limited both in amplitude and rate.

For turbulence, in the short term, the amplitude of the elevator demand is small and all loops could be limited, except (1), without significant performance deterioration. For long term disturbances, automatic trim may be an advantage and the amplitude of integral feedback terms should not be limited. However, the rate of change of the integral terms can be limited; and any steady state on the integrator outputs can be transferred (if required) to the main loop (1).

As a result, it is necessary only to 'prove' the high integrity of the control loop (1) and interfaces between the auxiliary (and complex) loops and the main loop. Because this basic control loop and the interfaces can be made simple, such proof can be obtained in a practical and visible fashion.

Fig.7 gives the results of a computer study showing the turbulence performance of:-

(a) the basic Hunter aircraft

(b) using pitch rate elevator control (loop 1)

(c) adding the pitch rate elevator integral term (loops 1 and 2)

(d) adding a DLC term (loops 1, 2, 3).

The disturbance was a vertical gust of the shape shown on the lowest trace.

For each of these closed loop configurations, the prefilters F can be chosen to ensure that the aircraft motion to stick input remains the same, both in terms of steady state and transient performance. Referring to Fig.8, there is a choice between good transient response in pitch rate and in normal acceleration using elevator only. With DLC, it is possible to offer good transient response in both pitch rate and normal acceleration.

5. CONCLUSIONS

The primary emphasis in the Hunter FBW experiment and in the several practical laboratory investigations in progress in the UK is the airworthiness aspect of using full-time, full authority control. The quality engineering required to achieve integrity and availability will be more expensive than current in service systems to develop, produce and 'prove'. There is little doubt, however, that future aircraft will be designed in such a way as to rely fully on active control technology for a variety of functions, and confidence in the use of active controls for very high integrity will evolve.

REFERENCES

1 F.R. Gill, On the design and evaluation of flight control systems. AGARD Conference Proceedings 137 on 'Advances in Control Systems' (1974)

2 P.W.J. Fullam and D. Kimberley, Flight Control System Development in the UK. AGARD Conference Proceedings 137 on 'Advances in Control Systems' (1974)

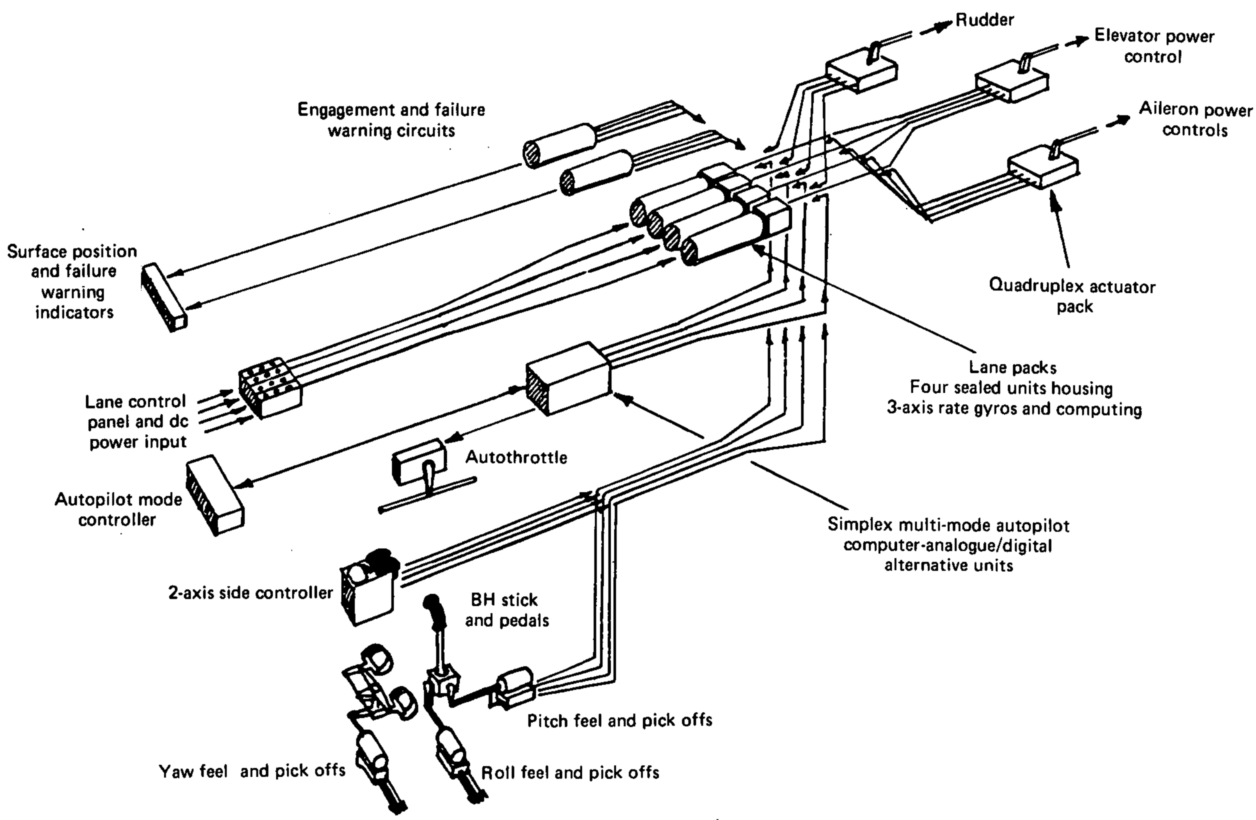


Fig.1 RAE Hunter XE 531 flight control systems schematic

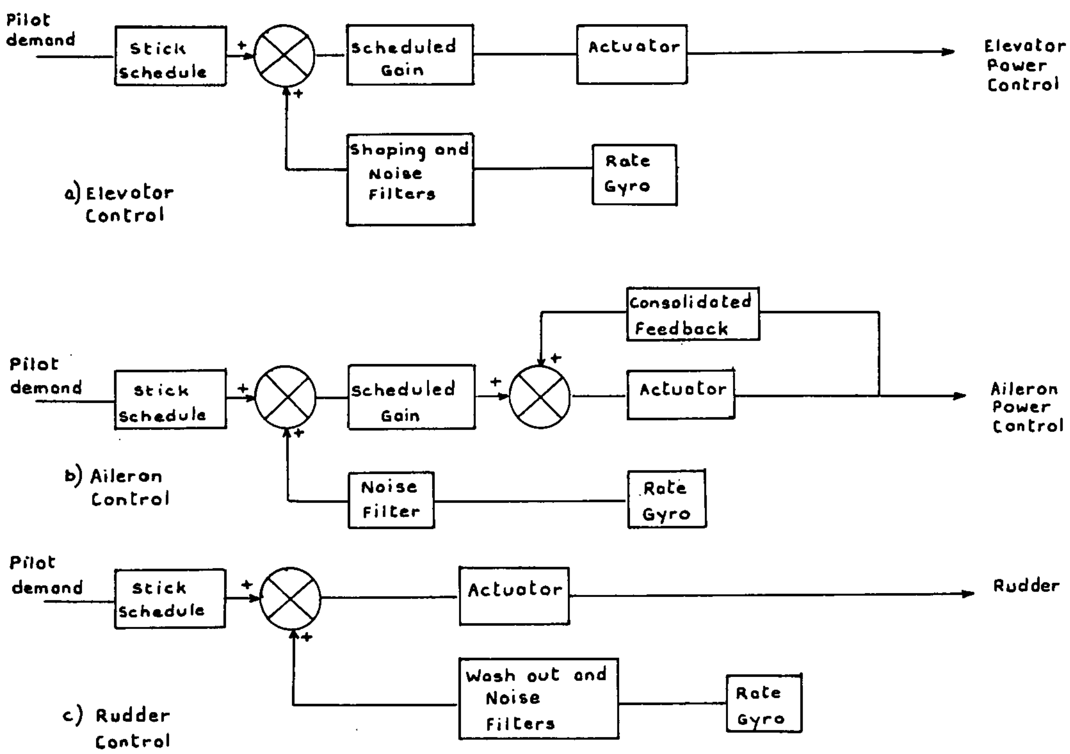
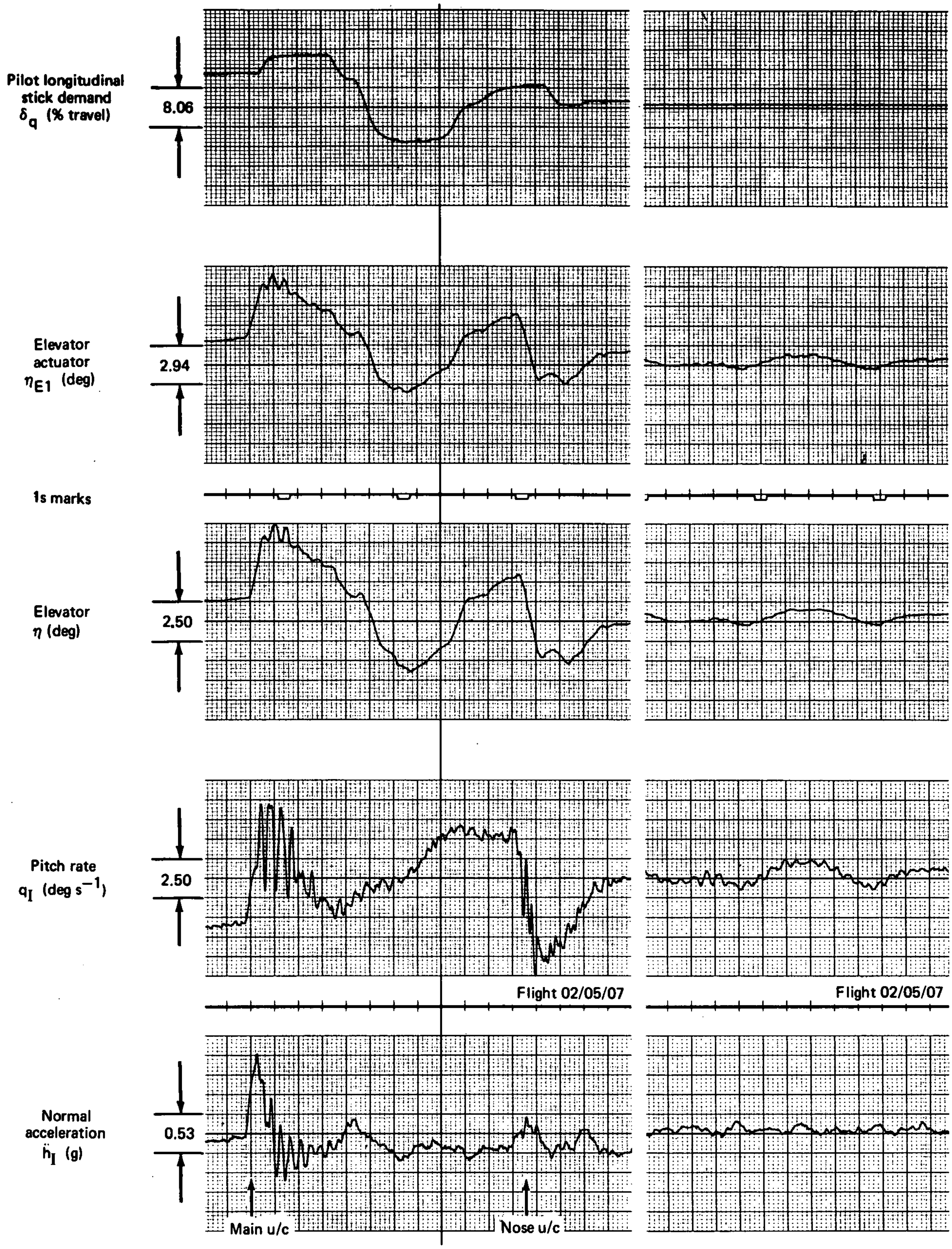


Fig.2 Active control in Hunter FBW system



(a) Touchdown and ground-roll

(b) Taxi

Fig.3 Elevator activity during landing and taxi

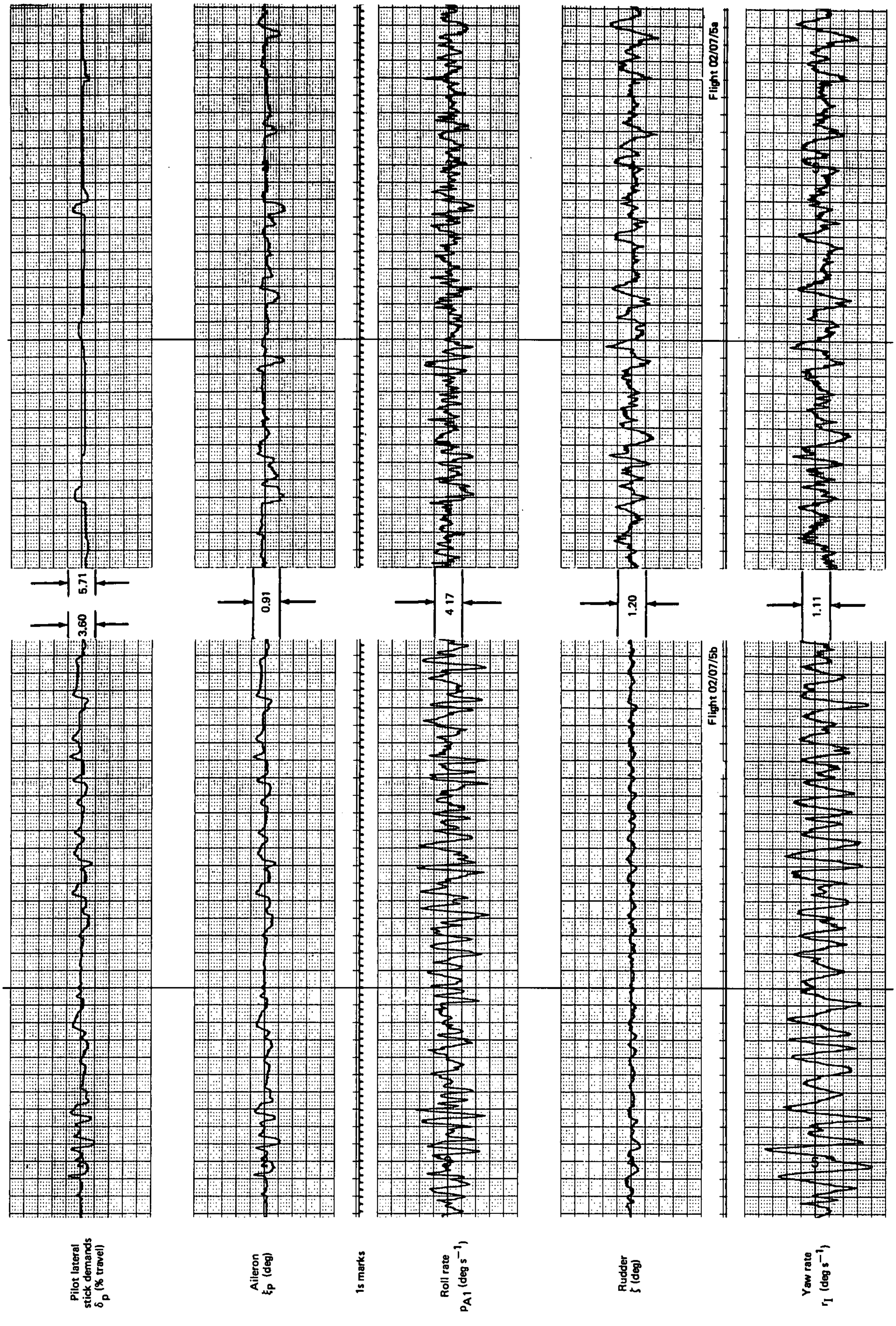


Fig. 4 Pilot lateral stick activity when flying straight and level in light/medium turbulence (FL 30 310kn ias) (No rudder pedal inputs)

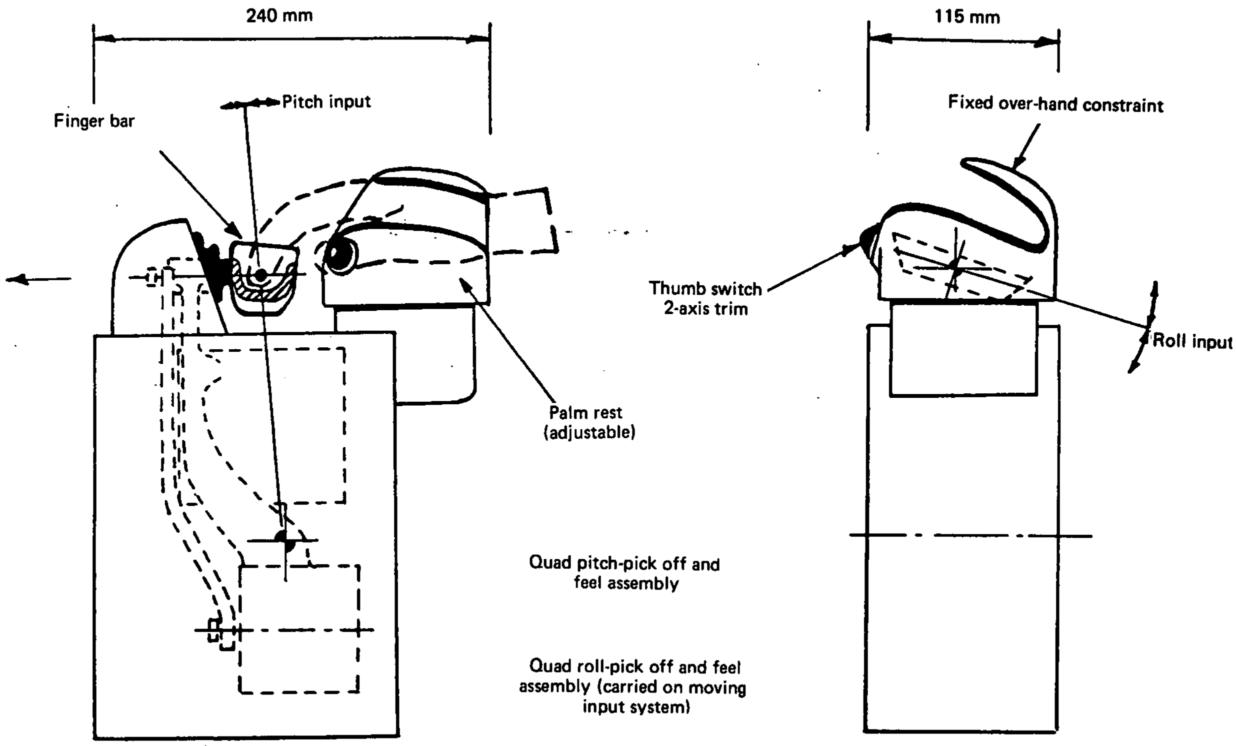


Fig.5 2-axis side controller for RAE Hunter

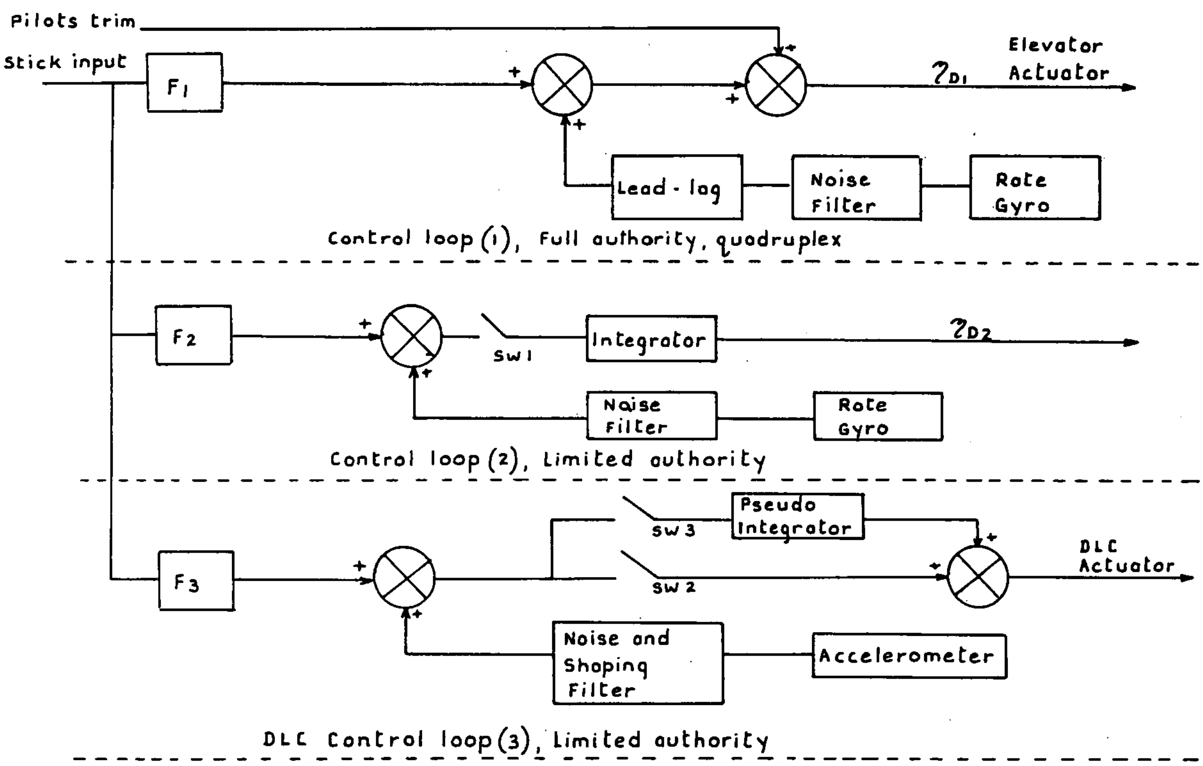
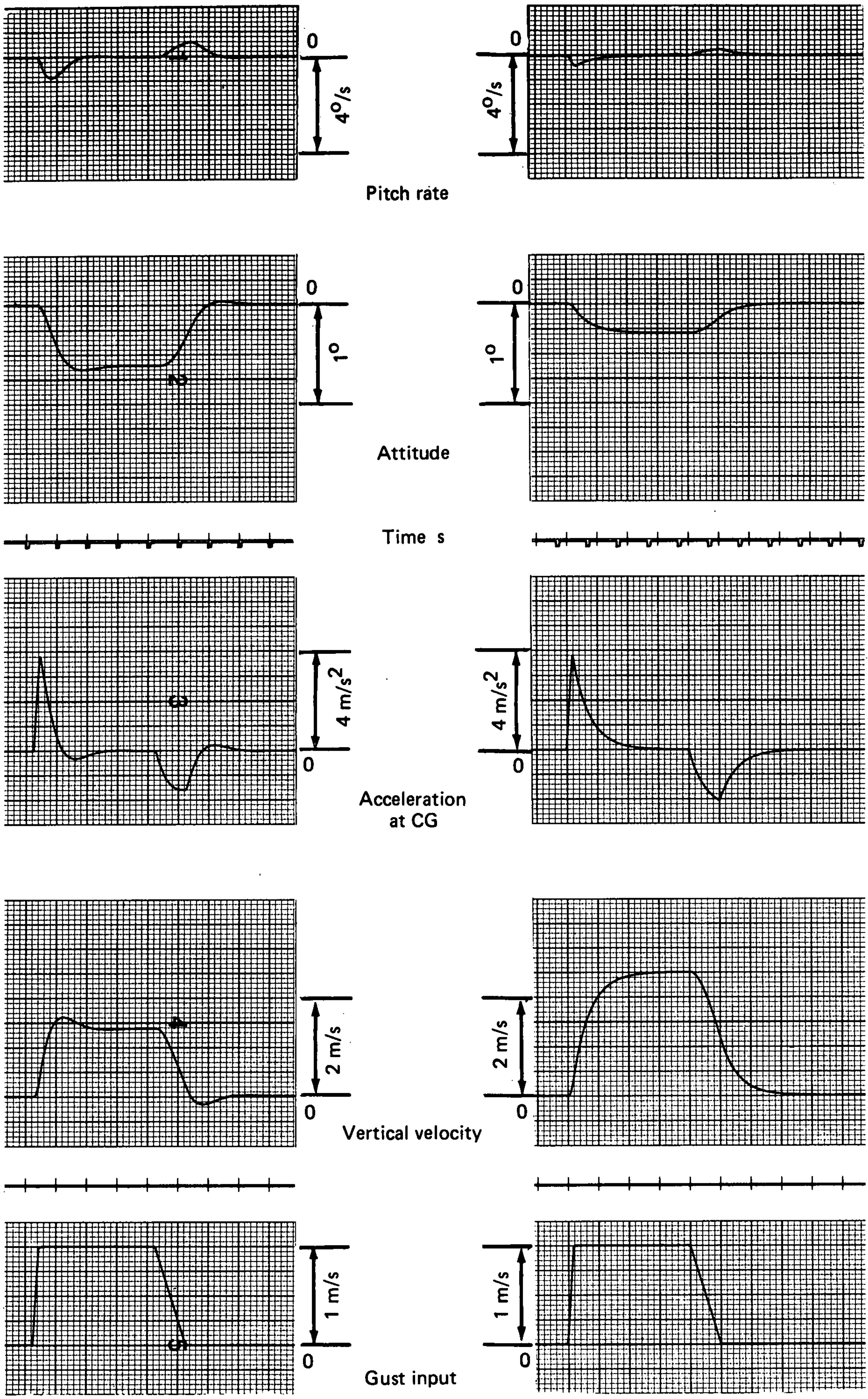
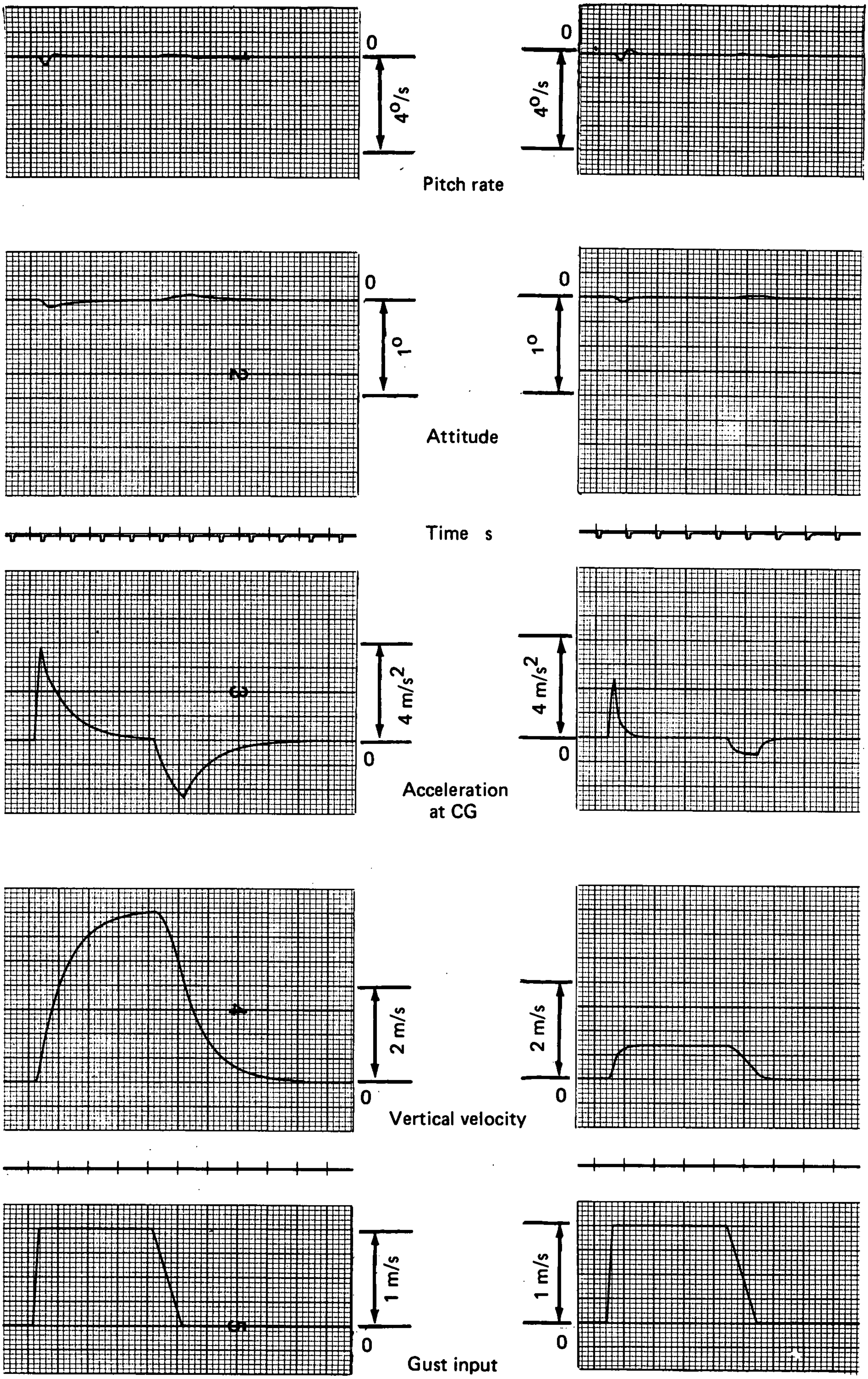


Fig.6 Structure of selectable fly-by-wire system



(a) Basic aircraft (b) Pitch rate control

Fig.7 Responses to a vertical gust



(c) Pitch rate plus integral

(d) DLC

Fig.7 (cont'd.)

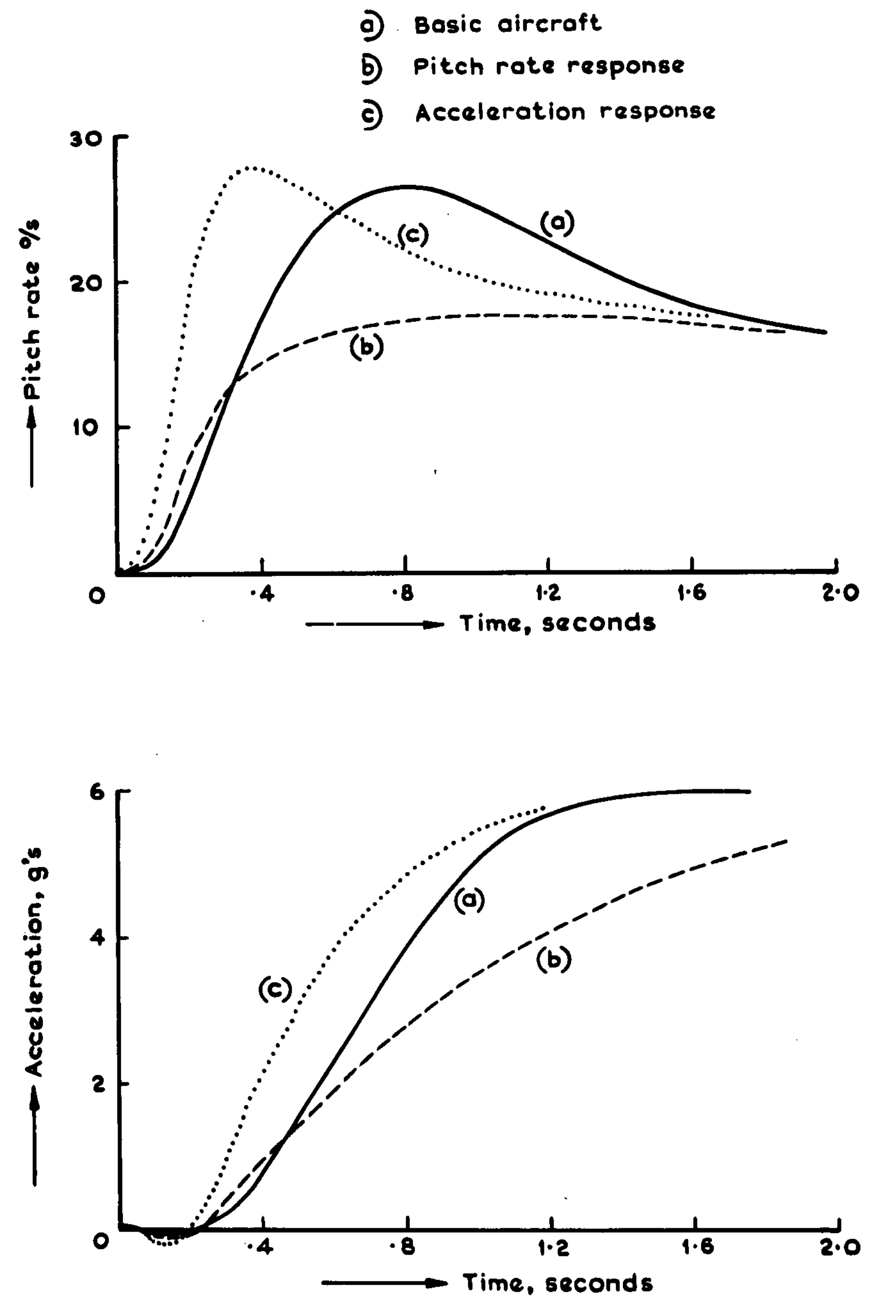


Fig.8 Aircraft response to pilot monoeuvre demand

F-8 DIGITAL FLY-BY-WIRE FLIGHT TEST RESULTS VIEWED FROM AN ACTIVE CONTROLS PERSPECTIVE

by

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SUMMARY

The results of the NASA F-8 digital fly-by-wire flight test program are presented, along with the implications for active controls applications. The closed loop performance of the digital control system agreed well with the sampled-data system design predictions. The digital fly-by-wire mechanization also met pilot flying qualities requirements.

The advantages of mechanizing the control laws in software became apparent during the flight program and were realized without sacrificing overall system reliability. This required strict software management. The F-8 flight test results are shown to be encouraging in light of the requirements that must be met by control systems for flight-critical active controls applications.

SYMBOLS

$a_1, a_2, a_3, b_1, b_2, b_3$	digital filter coefficients
$C^* = n_Z - \frac{V_{co}}{57.3g} q$, g	
G(s)	general s-plane filter
G(w)	general w-plane filter
G(z)	general digital filter
g	acceleration due to gravity, m/sec^2
K	general gain constant
K_{C^*}	C* feedback gain, deg/g
K_p	roll rate feedback gain, deg/deg/sec
K_q	pitch rate feedback gain, deg/deg/sec
K_r	yaw rate feedback gain, deg/deg/sec
n_Z	acceleration along positive Z-body axis, g
p	roll rate, deg/sec
q	pitch rate, deg/sec
r	yaw rate, deg/sec
s	Laplace transform variable
T	sample period, sec
V	velocity, KIAS
V_{co}	crossover velocity, m/sec
w	sampled-data system frequency domain variable
z	sampled-data domain transform variable
Δ	incremental change
δ	general surface command, deg
δ_e	horizontal stabilizer deflection, deg
ζ	damping ratio
θ	pitch attitude, deg
τ_r	effective roll mode time constant, sec

φ	roll attitude, deg
ψ	heading angle, deg
ω	natural frequency, Hz
$\wedge$	derived quantity

Subscripts:

d	Dutch roll mode
n	current sample
n-1	last sample
sp	longitudinal short period mode
Z	component along aircraft Z-body axis in positive (down) direction

ABBREVIATIONS

BCS	backup control system
CAS	command augmentation system
DIR	digital direct mode
KIAS	knots indicated airspeed
SAS	stability augmentation system

INTRODUCTION

To achieve maximum benefit from active controls in the design of a new aircraft, the control system will probably have to perform functions that are mandatory for the safe operation of the aircraft. A major deterrent to the application of such flight-critical active controls to aircraft has been the lack of highly reliable, large authority control systems designed to operate full time. A digital fly-by-wire control system has the potential for meeting these requirements, and, at the same time, providing design and development flexibility through software.

The NASA Flight Research Center has completed flight testing the F-8 digital fly-by-wire airplane, in which the mechanical controls were replaced with a full authority, full time digital fly-by-wire primary control system and an electrical command analog backup system. The digital system was committed for use from the first take-off and landing. It was made up of hardware developed for the Apollo guidance and navigation system. The design and development of the F-8 digital fly-by-wire system is described in references 1 and 2.

The primary objectives of the flight program were to evaluate the performance of the digital fly-by-wire system and to acquire operating experience with it. The flight program also provided an opportunity to assess the capabilities of a digital fly-by-wire system for active controls applications and to determine whether the predicted advantages of software mechanization could be realized.

This paper summarizes the results of the F-8 digital fly-by-wire flight program and the implications for future active controls applications.

FLY-BY-WIRE CONTROL SYSTEM

A single channel digital primary system and a triplex analog backup control system (BCS) replaced the mechanical control system of an F-8C test airplane (fig. 1). The components of the F-8 digital fly-by-wire control system

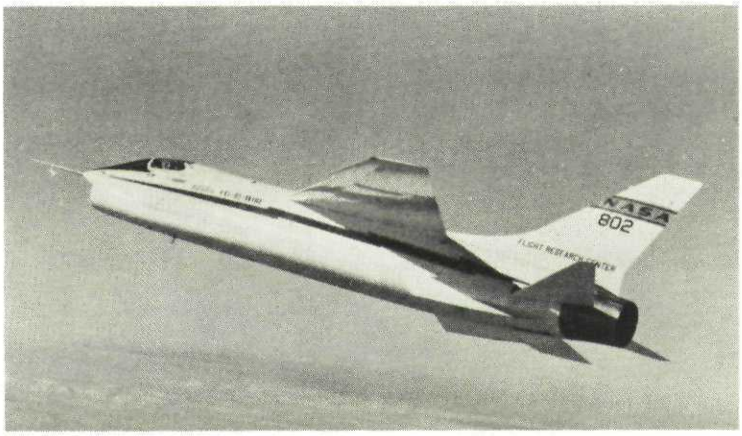


Figure 1. F-8 digital fly-by-wire airplane.

shown in figure 2 provided three-axis control of the airplane. The primary system consisted of a lunar guidance computer, inertial measurement unit, coupling data unit, and display and keyboard, all taken from the Apollo guidance and navigation system. The backup control system consisted only of surface position command electronics. Specially designed electrohydraulic secondary actuators interfaced the primary and backup electrical commands with the conventional F-8C control surface power actuators.

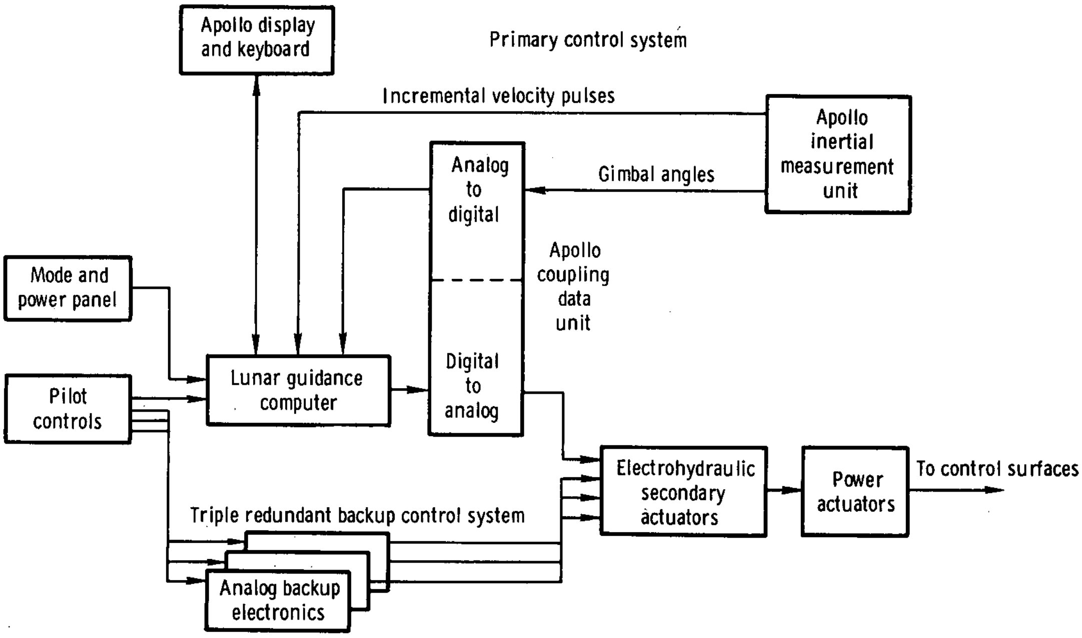


Figure 2. F-8 digital fly-by-wire control system components.

A functional schematic of the F-8 digital fly-by-wire control system is shown in figure 3. The lunar guidance computer received position inputs from the pilot's stick together with motion information from the inertial measurement unit. Surface commands were computed according to the programed control laws.

The two drive signals for each surface represented commands to the secondary actuator position loop. There was an active and a monitor servo path. If a failure were to occur in either path, a hydraulic comparator would sense the differential pressure between the active and monitor servo valves and transfer control to the backup control system. If the failure occurred upstream of the dualized path, the built-in test features of the lunar guidance computer would detect the failure and transfer to the backup control system before an unsafe command could be sent.

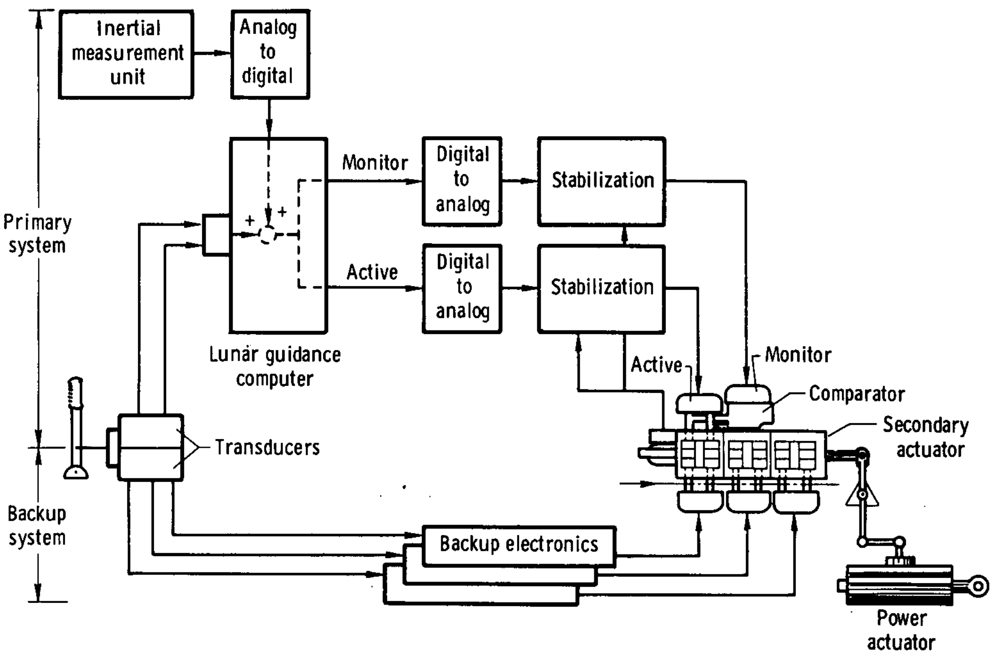


Figure 3. F-8 digital fly-by-wire control system mechanization.

The lunar guidance computer performed all the flight control computations for the primary system. The computer contained two types of memory. A 36,864-word read-only hard-wired memory contained the program for the flight control laws, as well as the executive and utility routines. A 2048-word scratch pad memory contained feedback gains, logic flags, and other constants likely to change during the flight program.

The pilot was given functional control of the computer through a mode and gain panel (fig. 4). Each axis contained a backup control system, direct (DIR) and stability augmentation system (SAS) mode. In the pitch axis, a command augmentation system (CAS) was also provided. The three gain switches were slaved by software to control system constants. There were other switches and displays for operating and monitoring the system.

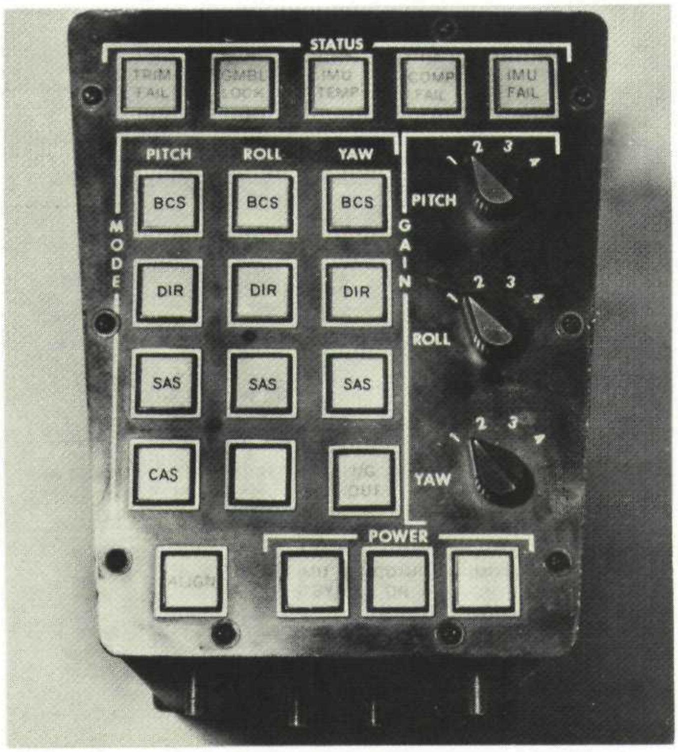


Figure 4. Mode and gain panel.

CONDUCT OF THE FLIGHT TEST PROGRAM

Figure 5 shows the nature and sequence of the flight test phases. The first flight was made using the digital fly-by-wire control system. This was significant inasmuch as it forced the designers to address the

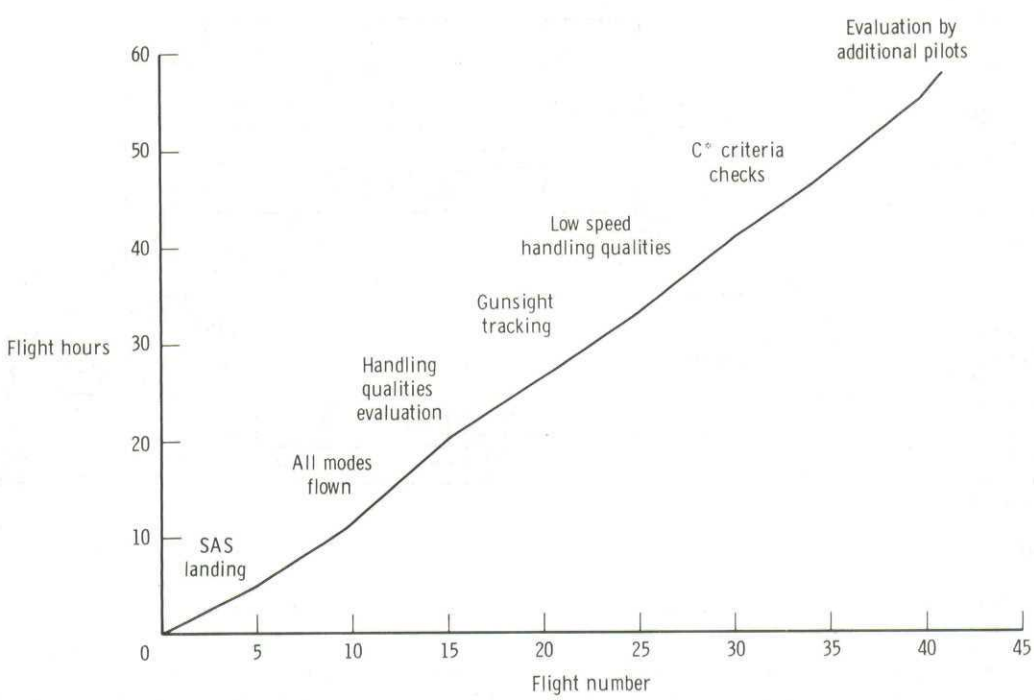


Figure 5. F-8 digital fly-by-wire flight test summary.

most critical aspects of digital fly-by-wire control before the first flight. This will also be the case for systems that incorporate flight-critical active control functions, since they must operate full time from the first takeoff. There is consequently a heavy premium on thorough hardware and software verification, accurate failure mode and effects analysis, and thorough preflight test procedures.

The evaluation of the digital fly-by-wire control system progressed rapidly, and by the eighth flight all modes had been flown. The airplane was then evaluated in a variety of tasks, including ground-controlled approaches, gunsight tracking, mild aerobatics, and formation flight. The last portion of the flight program was devoted to flying qualities assessments by additional pilots. In total, 58 hours were accumulated by six pilots during 42 flights. Most of the closed loop evaluations were made at speeds between 250 knots indicated airspeed (KIAS) and 400 KIAS and altitudes between 6000 meters and 10,700 meters. Tests at low speeds (below 200 KIAS) were made with the variable-incidence wing of the F-8C airplane in the up position.

CONTROL SYSTEM PERFORMANCE

One of the objectives of the F-8 digital fly-by-wire flight test program was to determine how well the digital fly-by-wire control system performed. This was important to establish whether the system met closed loop performance requirements and to validate the sampled-data modeling and design methods. Both aspects are important to future applications of digital fly-by-wire control.

Augmented Control Modes

The F-8 digital fly-by-wire control system contained a stability augmentation system in each axis. A command augmentation system, which used a blended normal acceleration and pitch rate feedback system commonly referred to as C* (ref. 3), was designed for the pitch axis.

Stability augmentation system modes.— The digital control law mechanization of the stability augmentation system was similar in all three axes. It is shown functionally in figure 6. Pilot inputs are shaped and summed

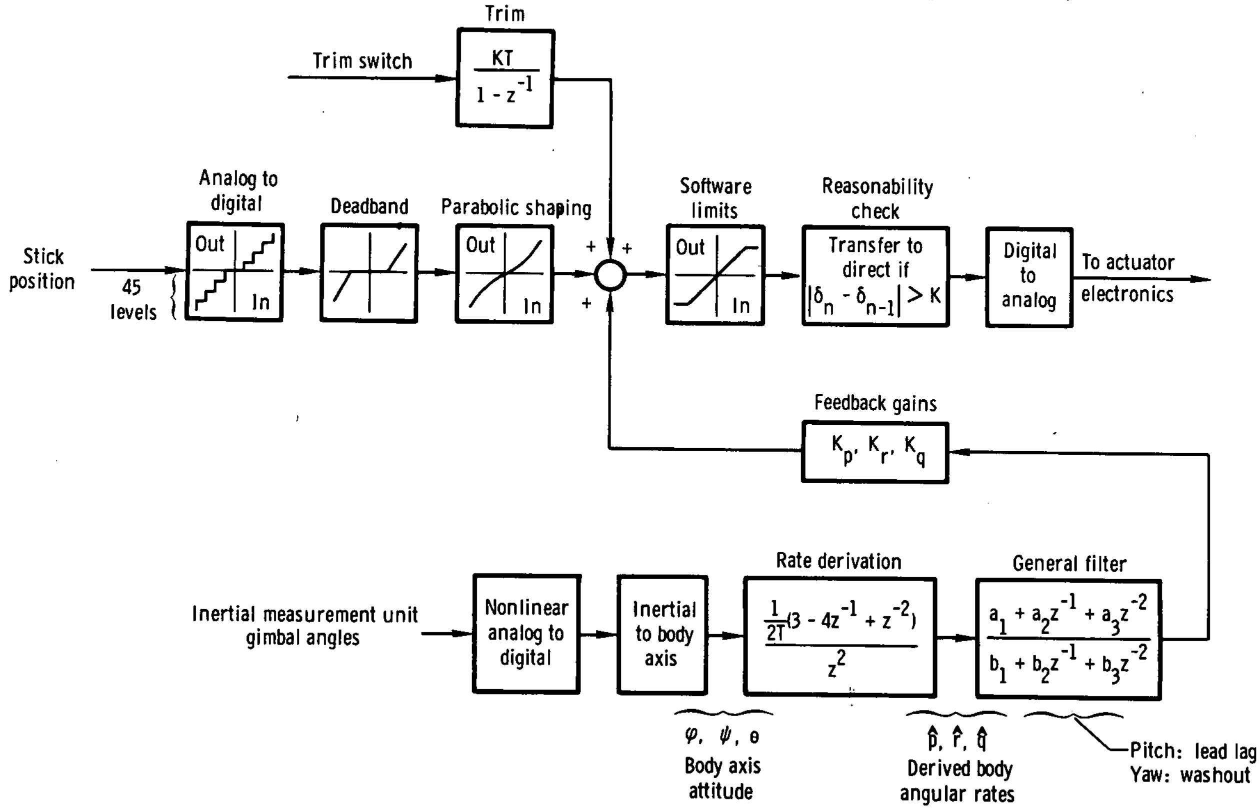


Figure 6. Digital control law mechanization for stability augmentation system modes.

with the digital series trim signal. Body axis rates were obtained by differentiating the transformed inertial platform gimbal angles (rate derivation filter). A general purpose third-order digital filter and gain was placed in the feedback path. A rate reasonability check was applied to the final command, and an automatic transfer to the direct mode resulted if the reasonability threshold was exceeded.

Sampled-data analysis methods were used to design the augmented modes. This meant that filter synthesis and feedback gain selection were accomplished directly in the discrete domain by using z-plane root loci, w-plane frequency response, and discrete time history response.

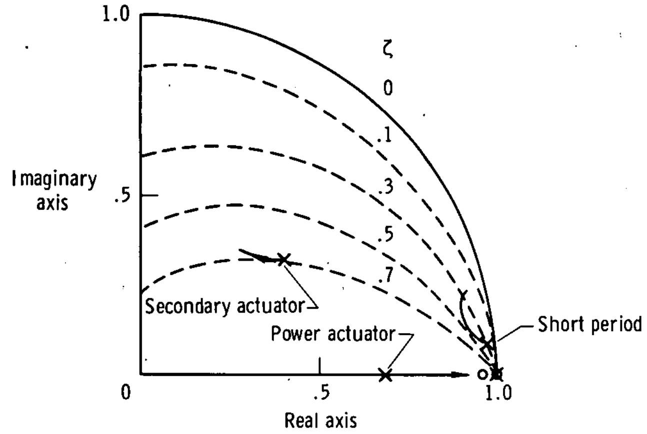
The design of the pitch stability augmentation system compensation exemplifies the discrete design procedure. A lead-lag filter was designed to improve the performance of the pitch rate loop by increasing the short period damping ratio. A w-plane frequency response was used to select compensation root locations. The w-plane compensation,

$$G(w) = \frac{w/0.1 + 1}{w^2/0.16 + w/0.286 + 1}$$

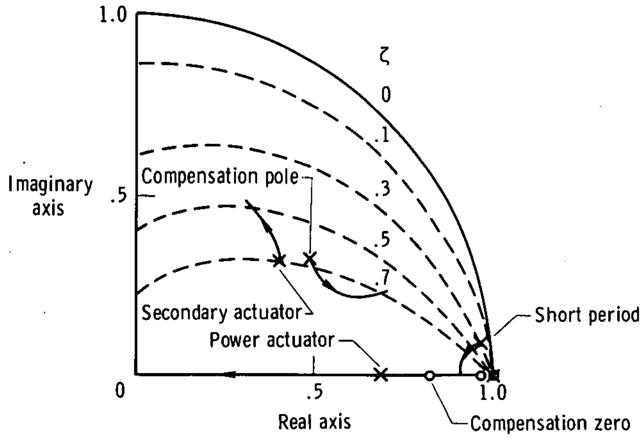
was transformed to the z-plane by using $w = \frac{z-1}{z+1}$. This yielded a discrete filter,

$$G(z) = \frac{1.023(1 + z^{-1})\ (1 - 0.818z^{-1})}{1.0 - 0.976z^{-1} + 0.349z^{-2}}$$

The z-plane root loci were also used to examine the design point. Figures 7(a) and 7(b) show the z-plane root loci for the pitch-rate-to-stabilizer transfer function without and with this compensation filter. The bending modes included in the analysis are not shown in the figure. Higher short period damping ratios were achieved by using the lead-lag filter, as one would expect in a continuous system. A comparison between the predicted



(a) Uncompensated system locus.



(b) Locus with lead-lag compensation.

Figure 7. Sampled-data system design example using z-plane locus.

effects of the compensation filter and those measured in flight is shown in figure 8, where the increment in short period damping ratio is shown for three flight conditions. The sampled-data system prediction is good. The improvement in airplane response with the compensated pitch stability augmentation system is evident in the flight time histories in figure 9. Without the compensation, there is almost no increase in damping at $K_q = 0.1$.

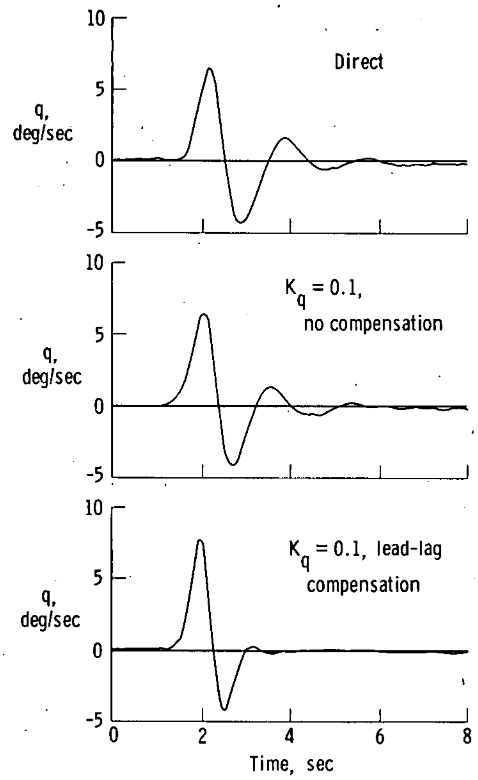


Figure 9. Effect of digital stability augmentation. Pitch stability augmentation system; 350 KIAS; 6100 m.

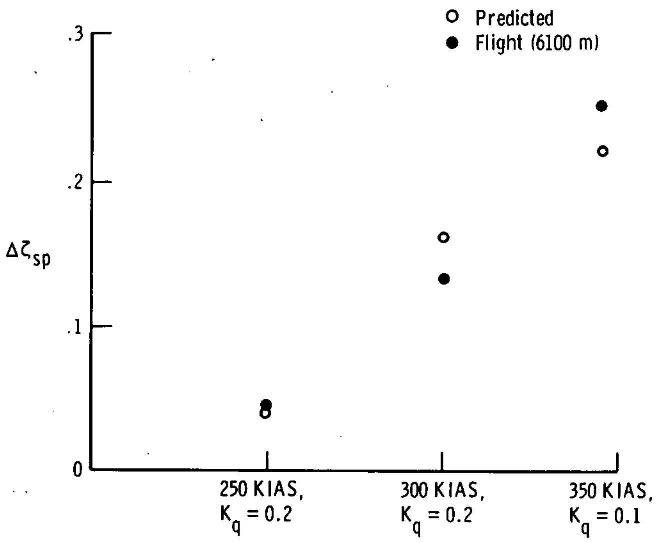


Figure 8. Improvement in short period damping ratio for lead-lag filter.

To further evaluate the sampled-data analysis method, the pitch rate feedback gain was increased in flight until the compensation root approached neutral stability. Figure 10 shows the z-plane root locus prediction of this point to be in good agreement with the flight-measured results in terms of frequency, damping ratio, and feedback gain.

Figures 11(a) to 11(c) compare predicted and measured damping due to stability augmentation system operation in each of the three axes. Agreement is generally good. The results are similar to those expected for an analog system.

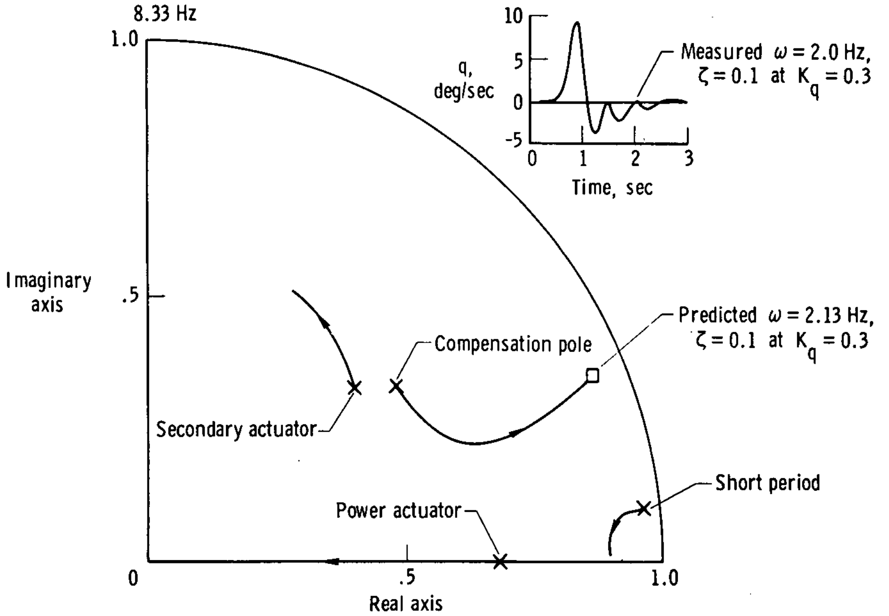
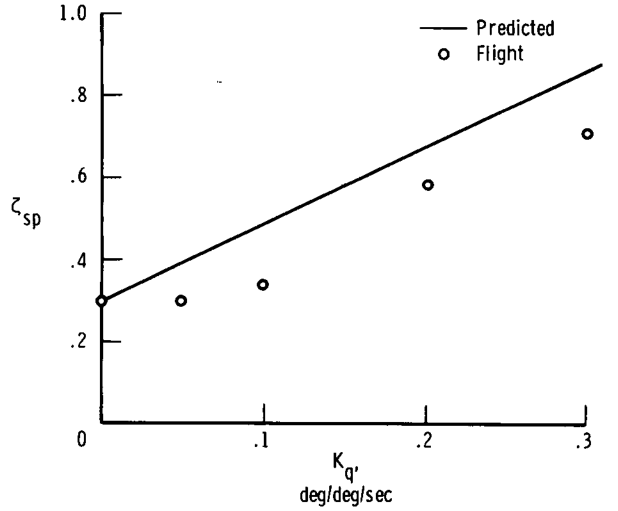
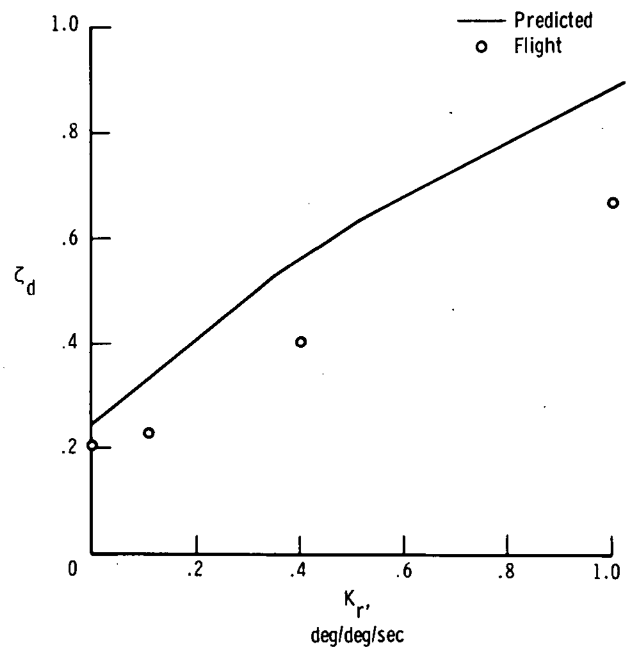


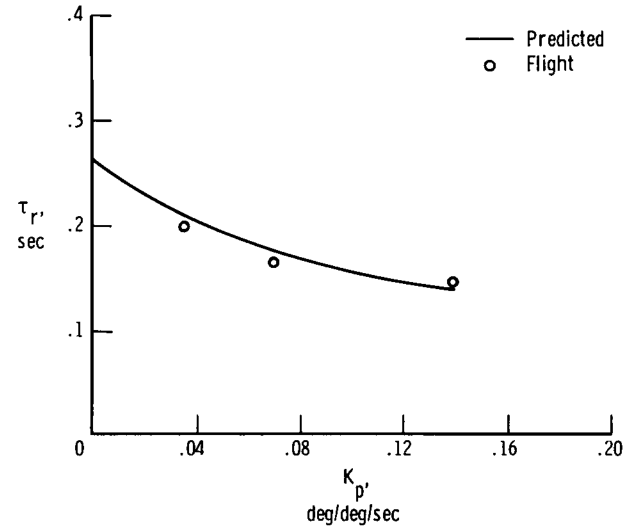
Figure 10. Prediction of system instability at high gain. 350 KIAS; 6100 m.



(a) Improvement in short period damping due to pitch rate gain. 300 KIAS, 6100 m.



(b) Improvement in Dutch-roll damping due to yaw rate gain. 250 KIAS, 6100 m.



(c) Improvement in roll mode time constant due to roll rate gain. 250 KIAS, 6100 m.

Figure 11. Comparison of in-flight and predicted digital stability augmentation system performance in pitch, roll, and yaw.

Command augmentation system mode.— A digital control law block diagram for the pitch command augmentation system is shown in figure 12. Body axis normal acceleration, derived from the inertial platform velocity pulses, is blended with derived pitch rate to form the feedback signal, C*. A forward loop integrator and bypass path provided apparent neutral speed stability. The $\cos\theta$ correction term eliminated acceleration feedback in a steady climb or descent. The pilot stick and trim interface with this mode was the same as in the direct and stability augmentation system modes.

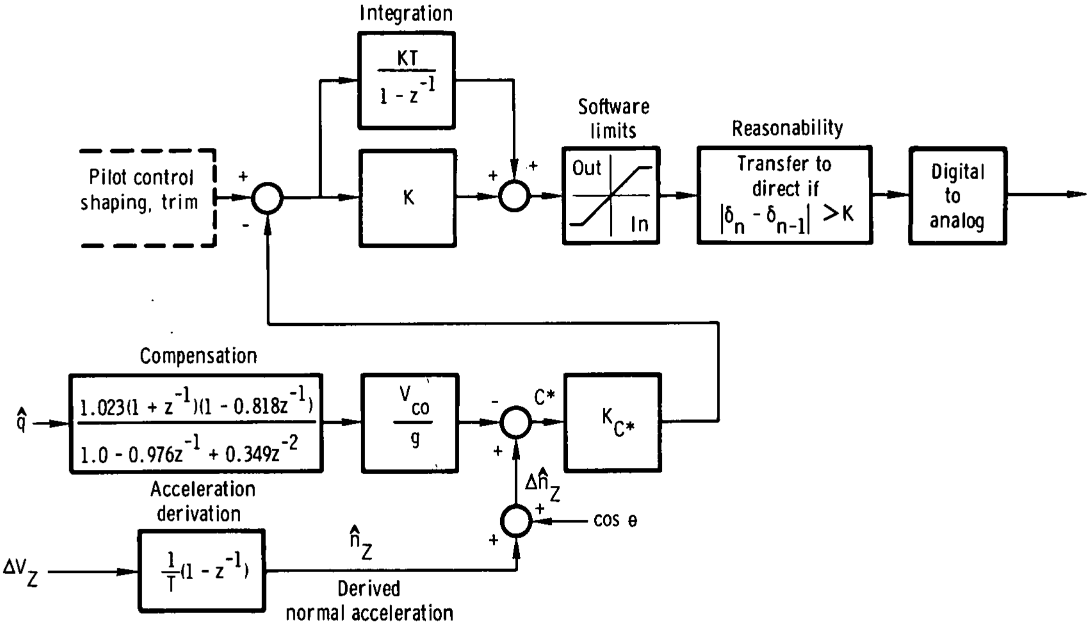
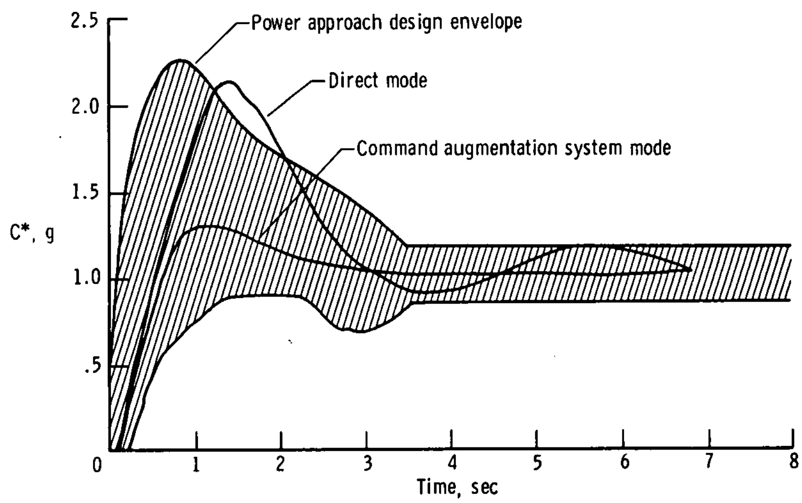
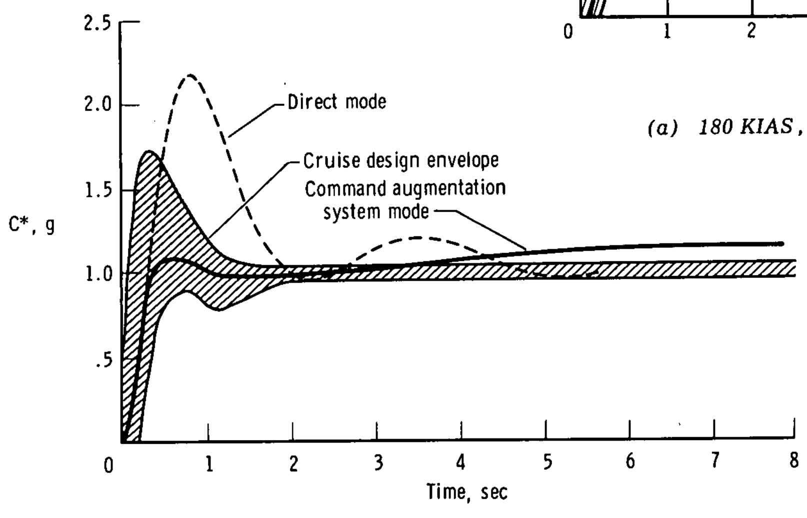


Figure 12. Digital control law mechanization of pitch command augmentation system mode.

As in the stability augmentation system modes, the performance of the digital command augmentation system was essentially as predicted by linear sampled-data system analysis. A time history criterion was used in the design of this mode. Figures 13(a) and 13(b) compare the F-8 digital fly-by-wire C* response in the direct mode and in the command augmentation system mode at 180 KIAS and 250 KIAS. The normalized responses are shown with respect to the power approach and cruise design criteria. The improvement in airplane short period response is substantial.



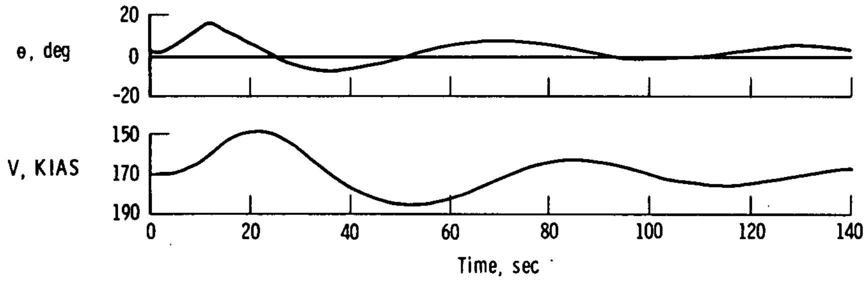
(a) 180 KIAS, 4570 m, K_{C*} = 2.0 deg/g.



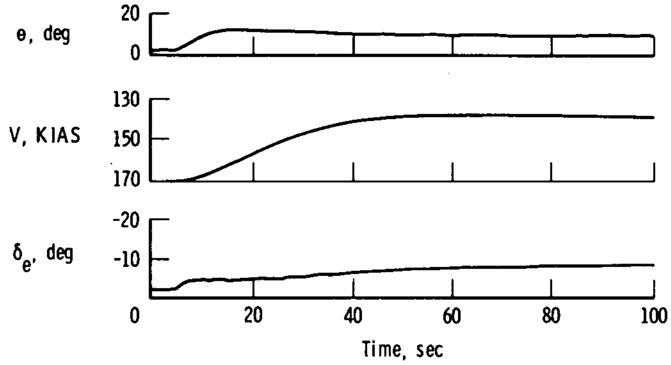
(b) 250 KIAS, 4570 m, K_{C*} = 1.5 deg/g.

Figure 13. C response of F-8 digital fly-by-wire aircraft.*

The command augmentation system mode provided the expected neutral speed stability. Figures 14(a) and 14(b) show the phugoid response of the F-8 digital fly-by-wire aircraft in the direct and command augmentation system modes, respectively. Trimmed at 180 KIAS, the aircraft was slowed approximately 10 KIAS, where the stick was again centered. The command augmentation system mode (fig. 14(b)) held zero pitch rate while the aircraft slowed to a new steady state speed of approximately 138 KIAS. Normal acceleration remained constant at nearly 1g during the maneuver, while angle of attack, which started at 3.5°, stabilized at 10°. The classical phugoid characteristics apparent in figure 14(a) are no longer present. This can create a problem if pilots expect to observe speed stability. For this reason, positive speed stability could be selected for the landing approach by removing the forward loop integrator.



(a) Direct mode, hands off.



(b) Command augmentation system mode, hands off, $K_{C*} = 2.0$ deg/g.

Figure 14. F-8 digital fly-by-wire phugoid response.

Implications of Closed Loop Results

The flight verification of the F-8 digital fly-by-wire closed loop design was encouraging from the standpoint of active controls applications. First, the discrete domain design, which is a straightforward technique, provided accurate results. The design methods used included the sampled-data counterparts of the classical continuous tools — the root locus, frequency response, and time history response. This is significant, because it means that most of an engineer's continuous system design experience is transferable to the sampled-data system design approach.

The digital mechanization has some advantages over analog designs that are particularly important for active controls applications. The designer can establish the characteristics of a given computation, such as a filter, precisely, and he does not have to account for component tolerances or aging in his design. Once mechanized, the filter characteristics do not change as long as the computer is functional. This is important in flight-critical computations.

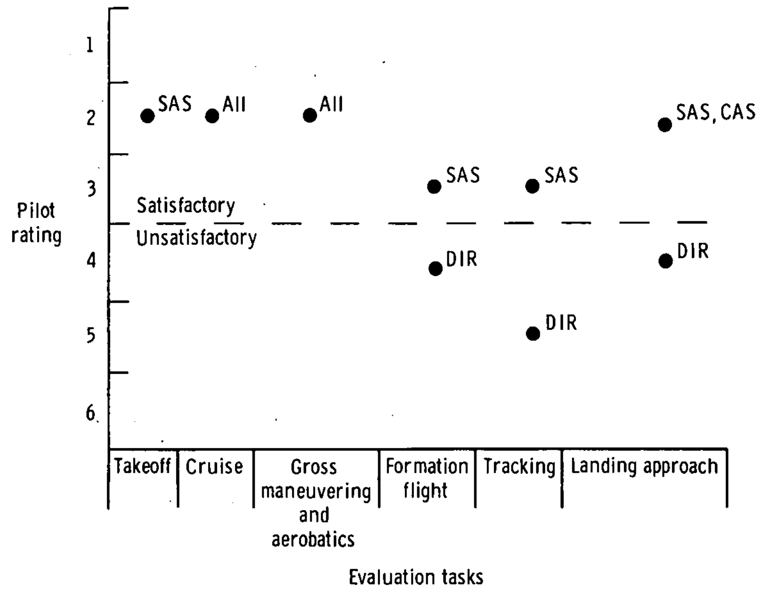
Although the digital mechanization introduces new variables, such as word length and sample rate, this does not constitute an overwhelming burden in the design process. The biggest problem that would confront designers of an all-digital active control system today would be lack of experience in trading off hardware costs and hardware capability. The computer industry itself may solve some of these problems for the designer. For example, the F-8 digital fly-by-wire three-axis control system computations were accomplished within a 30-millisecond sample period. Closed loop performance using a 14-bit plus sign data word was satisfactory with this sample time. Off-the-shelf computers are an order of magnitude faster today, and both software and hardware floating point options are available. Computer capability exceeds airplane flight control application requirements at the present time.

F-8 digital fly-by-wire closed loop flight performance did not differ appreciably from the performance that would be expected from an analog flight control system. No problems traceable to an inherent defect in the digital fly-by-wire approach were encountered in the closed loop operation of the digital fly-by-wire system. Thus, the closed loop design and performance of a digital fly-by-wire control system would not be expected to present any obstacles in an active controls application.

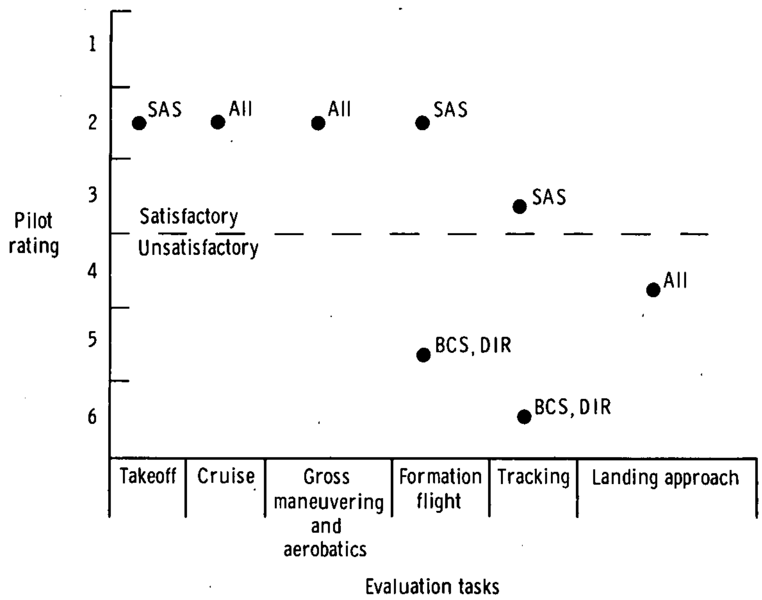
HANDLING QUALITIES

Summary of Pilot Opinions

The six pilots who flew the F-8 digital fly-by-wire aircraft found it to have satisfactory handling qualities for most maneuvers evaluated. Some problems were encountered because of the relatively coarse stick input quantization of the Apollo computer. Most tasks performed in the augmented modes (command augmentation system and stability augmentation system) were rated better than 3.5 on the Cooper-Harper rating scale (ref. 4). The pilot ratings for the F-8 digital fly-by-wire airplane's handling qualities are summarized in figures 15(a) and 15(b). Longitudinal control was satisfactory in the augmented modes for all the tasks evaluated (fig. 15(a)). Nearly all the unsatisfactory ratings were associated with the unaugmented, direct mode. Low short period damping and Apollo-hardware-related stick quantization degraded flying qualities in tight closed loop tasks.



(a) Pitch axis.



(b) Lateral-directional axes.

Figure 15. Handling qualities summary.

Lateral flying qualities were judged to be generally satisfactory in the roll and yaw stability augmentation system mode (fig. 15(b)). Problems were encountered with roll sensitivity about the stick neutral position, which resulted in unsatisfactory control for formation and tracking flight in the direct mode, and for landing approaches even in the stability augmentation system mode. This problem existed in the analog backup control system as well. Although several nonlinear stick shaping functions were investigated in both the digital and analog systems, a completely satisfactory system was never achieved.

Implications of Handling Qualities Results

The only handling qualities problems directly attributable to the digital mechanization were related to deficiencies in the Apollo hardware. The sensitivity in the roll axis appeared to be related to the fly-by-wire mechanization. A fly-by-wire control system can reduce mechanical slop and friction; however, some nonlinearities are necessary in a mechanical control system to reduce control sensitivity around stick neutral. These nonlinearities must be inserted artificially in a fly-by-wire system. This problem must be resolved whether the system is analog or digital. The mechanization of nonlinearities is trivial in a digital computer, but the desired nonlinearities must first be defined.

Future active controls applications, which may employ multiple blended control surfaces, are not likely to be able to draw on valid specifications of pilot control shaping requirements. The use of unconventional controls such as force sidesticks may require increased shaping sophistication. The versatility of a digital mechanization will be a decided advantage where the tailoring of stick shaping is likely to extend into the flight test stage of the program. Software flexibility will allow entirely new shaping networks to be substituted with no hardware impact. This was in fact taken advantage of during the F-8 digital fly-by-wire program when a parabolic shaping routine was added in software late in the design process. Thus, a good strategy for future applications would probably be to use linear stick controller transducers like those in the F-8 digital fly-by-wire aircraft and to provide all shaping in software.

In conclusion, the satisfactory pilot ratings of the aircraft's flying qualities, in conjunction with the good closed loop performance of the control system, indicate that a digital fly-by-wire control system can provide airplane flight control as well as or better than conventional systems.

SOFTWARE EXPERIENCE

The software experience acquired during the F-8 digital fly-by-wire program is important to flight-critical active controls applications. The very factors that make digital mechanizations desirable – flexibility and versatility – make the ensurance of software integrity difficult. The software in the F-8 digital fly-by-wire control system was single string inasmuch as only one software program was executed. In a redundant digital fly-by-wire system, software is still single string if the same program is contained in each computer. The requirements of the F-8 digital fly-by-wire software and the processes required to develop and manage this software are representative of those that would be required by flight-critical applications.

Software Flexibility

The advantages of the software mechanization of the control laws over hardware implementation became apparent during the refinement of the control system. It was possible for the control systems engineers themselves to test many different configurations easily, with no hardware impact. This hands-on capability greatly improved the ground operation. When many gain changes had been made and the precise configuration was in doubt, it was only necessary to dump the contents of erasable memory on magnetic tape to acquire a complete description of any given configuration. This capability is unique to digital mechanizations. It was also possible to revert to the baseline configuration merely by reloading memory with the baseline punched tape. Reversion to the original configuration could be guaranteed, with no possibility that experimentation had inadvertently damaged some components, which is always a concern when hardware changes have to be made.

The software features associated with the Apollo display and keyboard also improved the ground operation by making it easier to verify changes and to conduct the preflight tests. Originally designed for the astronauts in the Apollo program, the display and keyboard gave the control system engineers and ground crew convenient access to the computer. Sophisticated interface software permitted the operator format rules to be simple. Even those not intimately acquainted with the Apollo hardware learned to operate the system easily.

It was possible to read, load, or monitor any point in the software computations with only a dozen keystrokes or so. In addition, the digital control system variables were in engineering units, and most could be read or monitored in decimal format. This greatly reduced human error, because it eliminated the unscaling and conversion of octal numbers. The monitor feature of the display and keyboard was indispensable during the flight program. In this mode, the control system parameters displayed on the keyboard were updated once per second, resulting in an intelligible and dynamic presentation that was particularly useful in the checkout and refinement of the control system.

The benefit of the software most apparent to the pilots was the mode and gain panel mechanization. By having the gain switch mechanization in software, the digital flight control system could be checked out rapidly and safely in flight. In all, 105 parameters could be connected via software to the three gain switches. With this gain mechanization, different control system parameters could rapidly be selected and optimized. More important, the gain switches allowed the designer to make use of the pilot's capabilities. Nominal values of critical gains that were established during the simulation phase were placed on the gain switches along with larger and smaller values. The pilot could change the gain values at any time. For example, one of the gain switches was for pitch gearing. During the first flight, when the effects of pitch quantization and sensitivity had not yet been established, the pilot took off in the nominal gain position. After 13 minutes of flight, he reduced the gearing because of pitch control sensitivity at 300 KIAS. Before landing he evaluated three gain positions, finally selecting the nominal gain value 3 minutes before touchdown.

Apart from its research value, this type of gain selection and evaluation gave the pilot an important degree of freedom. A switch arrangement like this is not unique to a digital flight control system, but the ability to designate so many parameters without hardware impact is. Adjustments about a nominal gain position would also be useful in a prototype active controls aircraft, since a backup control system might not exist.

Software Management

The flexibility and versatility of digital flight control system software carries with it the need for software management and control. Two aspects of the F-8 digital fly-by-wire flight test program are of significance to full authority, man-rated digital flight control system software. First, not a single software programing error was discovered during the flight test program. Much of the credit for this is due to the thorough verification procedures and facilities developed for the Apollo software, which were used in the F-8 digital fly-by-wire program on a smaller scale. Secondly, not a single incorrect constant propagated to a flight tape that was used to load the lunar guidance computer. These results are significant because an active control system must achieve the same level of reliability as the basic airframe. The software, in turn, is central to an active control system's reliability. Software control falls into two general categories – management of system constants and management of the program structure (coded instructions). The F-8 digital fly-by-wire program had to address both problems.

Management of system constants.– As mentioned previously, control system constants such as filter coefficients, loop gains, and logic flags were contained in the erasable memory of the computer. During the flight test program several filter and gain configurations were flown, each of which required a new erasable memory load tape. Sum checks and built-in data transmission checks in the lunar guidance computer made it possible to ensure that the desired octal numbers were loaded into the computer. Making sure that the 394 erasable memory constants loaded were those actually desired, however, required the careful control procedure outlined in figure 16.

The off-line diagnostic program (the third step) was the key to reducing the effort involved in ensuring a correct computer load tape. The lunar guidance digital computer is a fixed-point machine, so there were magnitude restrictions on most parameters due to program scaling. A variety of other restrictions combined to create a formidable set of rules for the set of control system constants.

One task performed by the diagnostic program was to check each of the 394 constants against a list of reasonable values. This reasonability list was constructed from considerable experience that was acquired with an iron bird simulation before the first flight tape was made. The limits encompassed the expected or allowable operating range of each variable. Deviations from reasonability limits were flagged by the program as major errors and had to be corrected or signed off by the engineer responsible for them.

The program also reconstructed digital filter forms from their coefficients and computed their vital characteristics, such as root location, steady state gain, and absolute root magnitude in the z-plane. This was helpful in the case of digital filters, the characteristics of which are not as obvious as those of continuous filters.

The procedures outlined in figure 16 normally took approximately 2 days and could also be used in the refinement stage of a prototype digital control system.

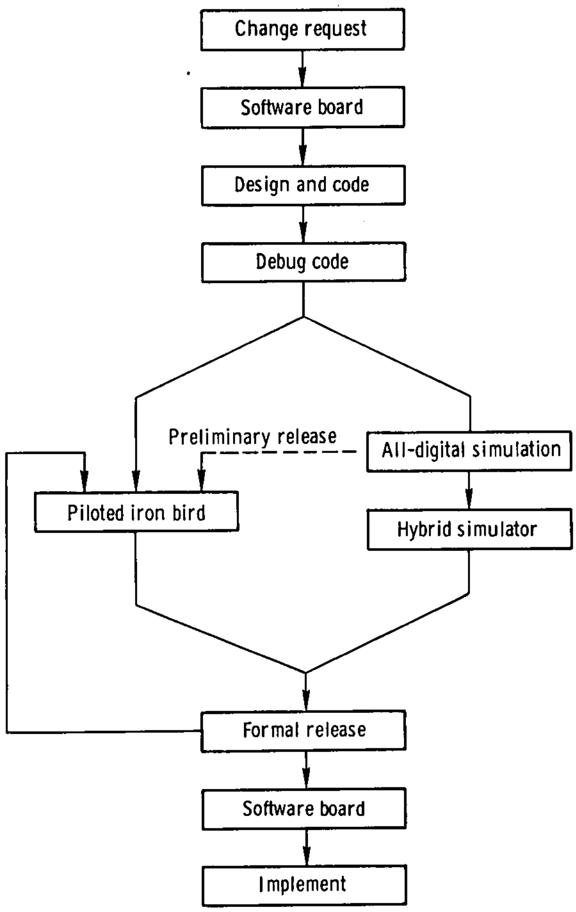


Figure 17. Programing change procedure.

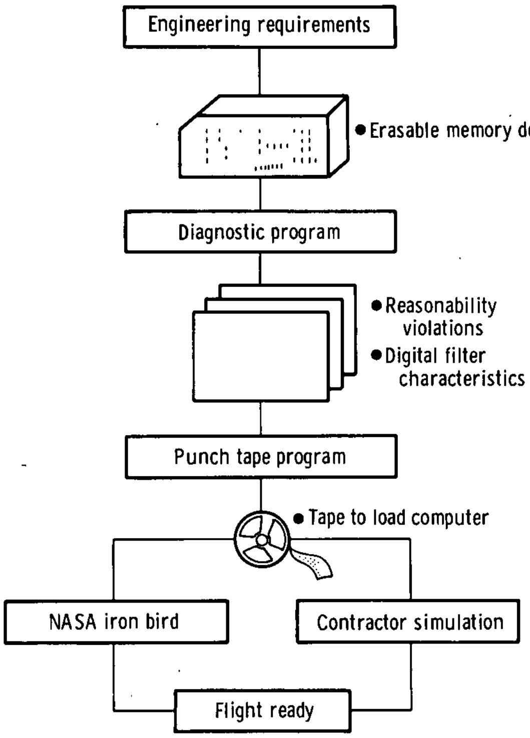


Figure 16. Procedure for new erasable memory load.

Management of program changes.— Changes could be made to the program structure only by adding code to the erasable memory of the lunar guidance computer, because the main memory was hard wired.

Figure 17 lists the procedures used to control software programing changes during the flight program. These procedures were used three times after the hard-wired memory was manufactured but before the first flight. The three special purpose programs written into the erasable memory consisted of pitch and roll parabolic stick shaping, yaw pedal deadband, and a special failure mode monitor.

The members of the software control board mentioned in figure 17 represented control system engineering, project management, flight operations, and the pilots' office. The board was necessary because of the variety of the functions performed by the computer and the potential impact of any change.

The verification steps were more extensive and time consuming for program changes than they were for control system constant changes, consisting not only of checking out the new code but of rerunning former, documented tests on related code to ensure proper program interaction, if any. Detailed records were kept of the all-digital simulation runs generated during original software verification for comparison with subsequent runs with modified code. This permitted the turnaround time for additions to the code to be short.

Implications of Software Results

The success of the production and management of flight-critical single string software in the F-8 digital fly-by-wire program is significant to future applications. Much of this success was due to the verification procedures and facilities that were established for the Apollo program, even though they were scaled down for the F-8 digital fly-by-wire application. This reveals the value of having a systematic software development plan early in the digital flight control system design process.

The management of day-to-day control system gain and filter changes in the refinement stage was aided significantly by the use of an off-line diagnostic program. Such a process would appear to meet the stringent requirements for control system changes in a flight-critical active control system development program.

The availability of a well engineered interface with the computer (display and keyboard) greatly enhanced the design and development process. Ground support equipment for current computers does not have many of the features of the Apollo display and keyboard that were valuable in the F-8 digital fly-by-wire program.

Although no large scale software changes could be made in the F-8 digital fly-by-wire system because the main memory was hard wired, experience was gained in altering the software structure by using the erasable memory. The testing of new code and the reverification of indirectly affected code was time consuming and had to be carefully controlled.

The benefits of software in such cases cannot be sold on the basis of ease of implementation when the verification task is included. Nevertheless, for flight-critical functions the value of the ability to make changes without hardware impact is significant.

Finally, active controls applications should be able to take advantage of airborne processors that have floating-point hardware and higher order languages. This equipment should make digital mechanizations even more attractive.

OPERATIONAL FACTORS

Digital System Reliability

The F-8 digital fly-by-wire control system utilized a single highly reliable digital computer. This configuration would probably not be used in an active control system. F-8 digital fly-by-wire reliability experience is nevertheless applicable to active controls technology in terms of failure detection and in terms of the features of the digital mechanization that led to a high level of confidence in this system. First, no failure was permitted that would have resulted in the generation of a hazardous control surface command. Second, any serious failure within the digital system had to be detected. In the F-8 digital fly-by-wire airplane, the failure warning signals were used to transfer control to the analog backup control system. In a redundant digital control system, operation would continue on the remaining good digital channels after a component failure.

No hardware failures occurred in the primary digital flight control system on any flight. This is not surprising in view of the demonstrated in-service reliability of the Apollo guidance and navigation equipment. The discrepancies noted in the digital fly-by-wire control system, excluding the actuators and their drive electronics, are listed in table 1. Three component failures occurred in two systems during the 2500 hours of operation (items 4, 5, and 10). Item 4 would have had no impact on normal flight operation. The failure monitor in item 5 was added to the system during the flight program for protection against a potentially hazardous single point failure in the lunar guidance computer interface hardware. The monitor circuit itself failed before use in flight, although it failed in the safe mode (transfer to the backup control system). The roll stick circuit failure (item 10) would have caused a downmode to the backup control system in flight, as it did on the ground. There were no unresolved anomalies.

TABLE 1.–DIGITAL SYSTEM DISCREPANCIES DURING GROUND OPERATION

(a) Discrepancies.

Item	Discrepancy	Reason for discrepancy
1	Computer restarts	Procedural error
2	Computer time-of-day wrong	Procedural error
3	Inertial measurement unit test result out of specification	Inertial measurement unit degradation for navigation
4*	Yaw direct light cycling on-off	Failed transistor in mode panel
5*	Backup control system downmode for rudder inputs	Failure in relay in external fail monitor
6	Computer locked in loop	Procedural error
7	Failure of preflight test	Damage to punched tape
8	Aileron offset	Procedural error
9	Roll digital to analog drift during backup control system self-test	Truncation during repeated primary/backup control system moding
10*	Backup control system downmode for aileron inputs	Failed resistor in external stick electronics

*Primary electronics failures.

(b) Summary.

Component	Failures
Apollo hardware	0
Primary electronics	3

Preflight Procedures

Two series of tests were performed on the digital system before flight. The first was a set of tests performed in the hangar the day before flight. Electric and hydraulic power were external. The second set of tests was performed immediately before flight, with engine-supplied electric and hydraulic power. Virtually all the hangar tests except specialized inertial measurement unit checks and detailed surface deflection measurements were repeated. Although the digital system's flight-line preflight was not optimized and required ground crew action, it took only 10 to 15 minutes.

One sensitive test was the computer activity check. A program in the erasable memory checked the computer's duty cycle indirectly, by measuring idle time over a several-second interval. If, in a given configuration, the duty cycle was consistent within a few percent over several time intervals, proper software operation was confirmed to a high level of confidence.

During the investigation of the anomalies that occurred in both the iron bird and the F-8 digital fly-by-wire airplane, it became apparent that it was possible to determine the health of the digital control system rapidly and confidently. The state of the digital control system could be determined in less than 5 minutes by running a self-test and by monitoring the internal control system parameters on the display and keyboard in the flight control modes. The ability to be assured quickly and confidently of proper control system performance is of paramount importance to active control systems. The repeatability of the test results of the F-8 digital fly-by-wire program inspired enormous confidence in the operational readiness of the system before flight.

CONCLUSIONS

The F-8 digital fly-by-wire flight program demonstrated the feasibility and desirability of a digital fly-by-wire mechanization for flight-critical airplane control functions. The conclusions related to active controls applications are as follows:

(1) A digital fly-by-wire control system possesses the computational ability and versatility needed for advanced control system applications. Computer hardware and software advances are leading airplane flight control applications.

(2) Existing digital design tools are satisfactory for the accurate closed loop synthesis of airplane flight control systems.

(3) A digital fly-by-wire mechanization can meet pilot flying qualities requirements. In the multiple or blended surface applications likely in active controls configurations, the nonlinear shaping of pilot control inputs will require further attention. The software mechanization of stick shaping will be advantageous in these applications.

(4) Reliable flight-critical software can be produced and maintained through the flight test and refinement program. The advantages of software flexibility were realized in practice.

(5) The digital fly-by-wire mechanization proved to have the capacity for thorough and repeatable self-test and fault detection.

REFERENCES

1. Deets, D. A.; and Szalai, K. J.: Design and Flight Experience With a Digital Fly-by-Wire Control System Using Apollo Guidance System Hardware on an F-8 Aircraft. AIAA Paper No. 72-881, Aug. 14-16, 1972.

2. Deets, Dwain A.; and Szalai, Kenneth J.: Design and Flight Experience With a Digital Fly-by-Wire Control System in an F-8 Airplane. Advances in Control Systems. AGARD-CP-137, May 1974, pp. 21-1—21-10.

3. Tobie, Harold N.; Malcom, Lawrence G.; and Elliott, Elden M.: A New Longitudinal Handling Qualities Criterion. Naecon/66; Proceedings of the Annual National Aerospace Electronics Conference. IEEE, 1966, pp. 93-99.

4. Cooper, George E.; and Harper, Robert P., Jr.: The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities. TN D-5153, 1969.

USE OF ACTIVE CONTROL TECHNOLOGY TO IMPROVE RIDE QUALITIES OF LARGE TRANSPORT AIRCRAFT

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SUMMARY

This paper describes the analyses, construction,and flight testing of two systems, "Beta-vane" and modal suppression augmentation system (MSAS), which were developed to suppress gust induced lateral accelerations of large aircraft. The Boeing 747 transport was used as the test vehicle. The purpose of the Beta-vane system is to reduce acceleration levels at the "dutch roll" frequency whereas the function of the MSAS system is to reduce accelerations due to flexible body motions caused by turbulence. Data from flight test, with both systems engaged, shows a 50-70 percent reduction in lateral aft body acceleration levels. Furthermore, this paper suggests that present day techniques used for developing dynamic equations of motion in the flexible mode region are limited. These techniques produce results which are satisfactory for analyzing dynamic loads and stability problems, but may be insufficient for development of active control systems operating in the same frequency region.

INTRODUCTION

The aft fuselage section of long slender airplanes is a position of relatively high lateral acceleration levels in moderate to heavy turbulence. These accelerations can be considered as being due to contributions from a rigid airplane with the elastic effects superimposed. Initially, because of the experimental nature of the program, two different approaches to gust alleviation were undertaken. One system worked the flexible body frequencies (MSAS system - Section I) whereas the second system worked primarily rigid body frequencies (Beta-vane system - Section II).

SECTION I

MODAL SUPPRESSION AUGMENTATION SYSTEM (MSAS)

INTRODUCTION

This section of the paper will describe in detail the analysis, construction and flight testing of a modal suppression augmentation system. This system was designed to reduce aft body lateral accelerations in the 1-3 Hz region when flexible body motions are perturbed by turbulence. Due to the problems associated with the lateral dynamic equations of motions as discussed in the following section (that is, comparison of analytical and measured transfer functions showed a variation in the flexible mode region), a technique was developed which involved 'curve fitting' transfer functions to experimental data. This method then allowed a modal suppression system to be developed without dependence on the analytical equations. Furthermore, by including the yaw damper actuator with the experimental data that was analyzed via the curve fit method, the problem associated with precise mathematical modeling of the structural compliance feedback-actuator system was avoided.

COMPARISON OF ANALYTICAL AND EXPERIMENTAL DATA

Results from 747 flight testing in turbulence indicated that aft end lateral motion was composed of the following two components:

(1)	Rigid Airplane (dutch roll) (50% contribution)	---	0.2 Hz
(2)	Elastic Effects (50% contribution)	---	1.0 - 3.0 Hz

Within the 1.0 - 3.0 Hz band of frequencies the analytical equations predict five free-free modes, all of which are composed to some degree of wing, nacelle and body motions. These modes (corrected with results from the ground vibration test) are shown in Figure 1. Flight test data reduced via the 'curve fitting' technique (explained in the following section) is also shown in Figure 1.

Based on their composition the modes are identified as (1) outboard nacelle vertical bending, (2) fundamental wing bending, (3) inboard nacelle side bending, (4) aft body bending, and (5) outboard nacelle side bending. In addition to the above set of modes, a stabilizer mode at 3.16 Hz, a fore body mode at 4 Hz and a vertical fin bending mode at 6 Hz are of concern in conjunction with the development of the MSAS filter.

Although the analytical equations were reasonably close to measured value and thus sufficient for flutter studies, the development of an active control system, however, requires not only that the characteristic equation be correct but also that the residues of the transfer function (the zeros shown in Figure 1) be properly described.

From Figure 1, it is seen that even though the roots of the system (poles) are identified and reasonably close to those obtained via flight test data, it is obvious that the associated zeros are misaligned. Various attempts in the form of refinement in both structural and aerodynamic representation did not succeed in changing the general picture. Further work along these lines still remains to be pursued.

As a practical solution to the problem, a curve fitting technique was applied to the measured transfer-functions to identify the zeros and poles of the system to be controlled.

CURVE FITTING TECHNIQUE

From initial experimental data, the aft body was found to resonate at 1.8 and 2.4 Hz whereas the fundamental frequency of the fore body was 4 Hz. Furthermore, the aft body could be perturbed by gusts striking the fin or gusts exciting the engine nacelles producing wing-body coupling suggesting that either the ailerons or rudders could be used for the active control system. Due to the complexity associated with developing a system in conjunction with the ailerons, a rudder suppression system was chosen. Though the command signal to the rudder is rate limited at 13 deg/sec compared to a 50 deg/sec requirement for the load alleviation system developed for the B-52, it was determined that this lower rate limit would satisfy the requirement.

The experimental data was obtained by excitation of the lateral airframe degrees of freedom in the 1-7 Hz region via the upper and lower yaw damper servos and their respective rudders. The forcing function itself was a continuously changing constant amplitude sine wave frequency sweep, in the 1-7 Hz range, which was produced on a computer and stored on magnetic tape. Using the experimental data in conjunction with a Fast Fourier transform data reduction package, Bode plots for various sensor locations on the aircraft could be obtained.

The 'curve fitting' technique is based on the two papers given in references 1 and 2. These algorithms were programmed on the CDC 6600 during the development of the Boeing SST and were used in the design of 3rd and 4th order prefilters in conjunction with the Horowitz Circle technique. After a few attempts at deriving transfer functions from the experimental data, the following deficiencies in the computer program were observed:

1. The program could not handle 14th order systems.

2. The small non-linearities associated with the amplitude and phase curves were sufficient to make the computer program limit cycle.

3. The program was very sensitive to end point conditions.

Transfer functions that matched the experimental data were obtained by incorporating the following procedures:

1. The transfer functions were assumed to be of minimum phase (no right half plane zeros). Therefore, only the amplitude was input to the program.

2. A pole or pole-zero combination is always included on either side of the band of frequencies that is of interest.

The transfer functions showed clearly that although the analytic equations could be manipulated so that the modes would have the correct frequencies, the zeros associated with these analytic equations (and therefore the phase) were not correct for the 2.1 and 2.4 Hz modes. The effects of the different zero locations on a control system will now be shown.

A root locus diagram of an accelerometer control system based on the analytical equations is shown in Figure 2. The control system adds approximately twice the damping to the 2nd and 4th modes; these two modes contribute 80% of the flexible energy. This system was flight tested and results showed that the 4th mode was destabilized and the 2nd mode increased in frequency as the gain of the control system was increased. This same control system based on the airplane transfer function obtained via the curve fit computer program has the root locus diagram shown in Figure 3. Notice that the loci are almost the same as those obtained in flight. This experimental verification of the curve fit technique showed that this method could be used with confidence.

The complete design technique in the development of the MSAS is the following:

1. EXCITE AIRPLANE VIA RUDDERS - CONSTANT AMPLITUDE SINE WAVE 1.0 TO 7.0 HZ.

2. CURVE FIT TRANSFER FUNCTION TO AFT BODY SENSORS.

3. ROOT LOCUS METHODS TO DESIGN FILTER.

4. EXCITE AIRPLANE VIA RUDDERS, WITH/WITHOUT MSAS, TO VERIFY SUPPRESSION OF MODES.

5. FLY MSAS IN TURBULENCE TO VERIFY CONTROL SYSTEM.

Notice that this procedure does not allow analytical verification of gust suppression; it only substantiates analytically whether the control system adds damping to the modes.

Two control systems were designed and flight tested using the above procedure. The first system used an aft body mounted lateral accelerometer sensor whereas the second system used two yaw rate gyros, one aft body and one at the cg. Figure 4 shows the reduction in aft body acceleration (Body Station 2300) for the two systems when the sine wave forcing function is fed to the lower rudder and the control systems are commanding the upper rudder. The accelerometer system was not chosen because the 2.4 Hz mode destabilized at high 'q' conditions. In addition, to obtain equal reduction in acceleration levels during turbulence, the accelerometer system required more rudder than the gyro system suggesting that the gust zeros for the two systems were quite different.

DESCRIPTION OF FINAL MSAS SYSTEM

The MSAS system is a single channel augmentation system working via the lower yaw damper servo. A block diagram of the control system is shown in Figure 5. The augmentation system provides damping to the 1.8, 2.1, and 2.4 Hz aft body lateral modes without disturbing the dutch roll mode. The salient features of the system are the following:

1. Two lateral yaw rate gyros.
2. Single channel 'real time' monitoring.
3. Scheduling of filter gain with calibrated air speed (CAS).
4. Output of system limited to ± 0.8 degrees of rudder (yaw damper authority is ± 3.5 degrees of rudder).
5. Operation of system limited to flaps "up" condition.

Figure 6 represents a functional block diagram of the computational path.

1. MSAS Damping Signal

   The MSAS signal is derived from the subtraction of two yaw rate signals. The location of the sensors is the following:

   a. Aft End Gyro:

      Body Station 2280, WL190, RBL20

   b. CG Gyro:

      Body Station 1307, WL195, RBL5

   Due to the placement, the aft end gyro is sensitive to dutch roll and flexible mode frequencies whereas the cg gyro is sensitive only to dutch roll frequencies. Upon subtraction of the two yaw rate signals, the remaining signal contains only flexible mode frequencies.

2. Band Pass Filter

   At flaps up condition, the yaw rate signal passes through a band pass filter into the yaw damper servo amplifier. The band pass filter is composed of R-C components, operational amplifiers and multipliers. The transfer function of the filter can be expressed in Laplace form as the following:

$$\left(\frac{750\,K_{CAS}}{S+50}\right)\left(\frac{16\,S^2}{(S+6)^2}\right)\left(\frac{14}{S+14}\right)^2\left(\frac{1.56S^2+9.4S+320}{S^2+15.9S+420}\right)\left(\frac{.726S}{S+1}\right)$$

   A Bode plot of the filter is shown in Figure 7.

   The functions of the band pass filter are:

   a. To wash out the steady-state yaw rate signals and to eliminate null offset of sensors.

   b. To reduce high frequency signal amplitudes so as to minimize coupling with the higher structural modes.

   c. To obtain the proper phasing between yaw rate signal and lower rudder so as to add damping to the aft body lateral flexible modes.

   Figure 8 represents the transfer function of yaw rate/lower rudder at BS-2300 whereas Figure 9 shows the effects of the MSAS filter on the above dynamics. The reason for the complexity of the filter is that the 1.8 mode required 'lag' and the 2.4 mode 'lead' in order for the system to add the maximum damping to these modes. Although various body stations were investigated, sensor positions aft of the cg, along the floor 'water line,' showed that there was no change in the phase relationship between the 1.8 and 2.4 cps mode.

Figure 10 represents a functional block diagram of the monitoring system and pre-engage mode. The function of the monitor system is the following:

a. Checks the principal gains and phase characteristics of the filter.

b. Detects failure of either gyro.

c. Detects failure of the limiter.

d. Detects failure of gain scheduler.

The purpose of the pre-engage mode is to verify that the MSAS electronic unit, including monitor, is operating correctly.

TEST RESULTS

A system corresponding to the filter shown in Figure 7 was flight tested (no monitor system, etc.). After initial calibration and stability criteria were satisfied (6 db gain margin and 60° phase shift), the system was flown in turbulence. Figure 11 shows one of the many time histories obtained. Figure 12 represents the cumulative accelerations for the time history plots of Figure 11. The MSAS system reduces the aft body flexible mode content by approximately 50% (although Figure 12 shows a 66% reduction). Figure 13 shows the cumulative acceleration at the pilot station. It may be noted that there is very little 1.8 and 2.4 Hz content at the pilot station and very little 4 Hz content in the aft end.

A production type unit has recently been certified (including monitor system, etc.) together with the Beta-vane system. The combined systems will then be installed on a production airplane for in-service evaluation.

SECTION II

BETA-VANE SYSTEM

INTRODUCTION

This section discusses a method devised for the 747 airplane of reducing those accelerations due to gust induced rigid airplane motions. As was pointed out in the previous section, the level of RMS accelerations due to turbulence is approximately 50% due to rigid body motions and 50% due to flexible motions (Figure 14). Consequently, a system designed to reduce the rigid body accelerations offers only half the potential reduction in the total level.

SYMBOLS

δ	vane rotation
U_B	longitudinal body axis velocity
V_B	lateral body axis velocity
V_P	total velocity
W_B	vertical body axis velocity
P_B	body axis roll rate
R_B	body axis yaw rate
L_1	longitudinal distance from cg to vane station
H_1	waterline distance from airplane principal axis to vane station
A_y	accelerometer output
Θ	pitch angle
ϕ	roll angle

METHOD OF SOLUTION

The method used for gust alleviation on the 747 in the frequency range 0 – 1 Hz is shown in Figure 15. The basic sensor is a relative wind vane which is used to sense lateral gusts; the output of the vane is used to drive the 747 upper rudder in a sense that reduces the airplane tendency to turn into the gust. The wind vane output signal is composed of the rapid change due to the lateral gust plus changes due to airplane motion from past disturbances. An approximate separation of these signals is accomplished through deriving airplane motion from lateral acceleration, yaw rate and roll attitude as shown in Figure 15. The resulting signal which is proportional to the lateral gust input is put through a band pass filter before being summed with the existing yaw damper signal to drive the upper rudder. The purpose of this filter is to remove steady-state sensor errors and to prevent excitation of the flexible body modes. The approximate location of the wind vane and other system components on the 747 airplane is shown in Figure 16.

ANALYSIS

For the purposes of the analysis, it was assumed that the lateral dynamics could be considered independently and that only lateral gusts were present. The assumed form of these gusts was the typical Von Karman spectrum.

The vane output can be described as:

$$\delta = \frac{GUST}{V_P} + \frac{V_B}{V_P} + \frac{L_1 R_B}{V_P} + \frac{H_1 P_B}{V_P} \tag{1}$$

where the last three terms give the sideslip angle at the vane location.

To derive a signal proportional to the gust input use is made of a lateral accelerometer mounted at the vane station. The accelerometer output is:

$$A_Y = \dot{V}_B + U_B R_B - W_B P_B + L_1 \dot{R}_B + H_1 \dot{P}_B - g \cos\theta \sin\phi \tag{2}$$

consequently,

$$\frac{V_B}{V_P} + \frac{L_1 R_B}{V_P} + \frac{H_1 P_B}{V_P} - \frac{1}{V_P}\int W_B P_B \, dt = \frac{1}{V_P}\int (A_Y + g\phi - V_B R_B)\, dt \tag{3}$$

or approximately,

$$\frac{V_B}{V_P} + \frac{L_1 R_B}{V_P} + \frac{H_1 P_B}{V_P} \simeq \frac{1}{V_P}\int (A_Y + g\phi - V_P R_B)\, dt \tag{4}$$

It is therefore possible to rewrite (1) as:

$$\delta - \frac{1}{V_P}\int (A_Y + g\phi - V_P R_B)\, dt = \frac{GUST}{V_P} \tag{5}$$

all the left side terms of (5) are available and this equation is the basis for mechanization of the system as shown in Figure 15.

The analysis was made using a Boeing derived computer program which accepts matrix inputs. This program provides root locus plots of the system and power spectral densities of designated parameters in response to given forcing functions. The complete analysis included consideration of the lateral airplane dynamics, roll, autopilot, yaw damper and gust suppression system. The performance of the gust suppression system was investigated throughout the flight envelope of the airplane with the intent of determining optimum system gain for reduction of the rear fuselage lateral acceleration and also to determine system stability. Some particular results of the analysis are shown in Figure 17. Figure 17 shows a root locus plot for different gains of the gust suppression system with the corresponding RMS 'g' levels at an aft body station shown in Figure 18. It can be seen that a reduction of about 30% in the RMS 'g' level can be obtained at the bucket of the curve shown in Figure 18. This particular gain affects the airplane stability very slightly as can be seen in Figure 17. Similar results were obtained for various airplane altitudes and speeds, the value of gust suppression gain remaining essentially the same for minimum 'g' levels.

The reason for the change in airplane stability is the approximate form adopted for compensating the vane output for airplane motion. Theoretically, this signal could be perfect, in which case, the root locus shown in Figure 19 results for all system gains. The approximate method of compensation was chosen for practical implementation.

TEST RESULTS

A system corresponding to that shown in Figure 15 was constructed and test flown in the 747 airplane. Initial flights were made to calibrate the wind vane sensor and to investigate airplane handling with the system engaged in calm air. Pilot comments were that the operation of the system had undetectable effect on handling characteristics in either normal or emergency maneuvers. Subsequently, several flights to investigate performance during turbulence were made. Typical data from one such flight is shown in Figures 20 and 21. Figure 20 shows a typical gust signal command measured at the input to the Beta filter summing amplifier while Figure 21 shows the RMS 'g' levels recorded at the aft body station with the system ON then OFF in sequence. The reduction in acceleration levels with the system ON is of the same magnitude as that predicted.

SERVICE EVALUATION

To obtain more data on the system, it has been installed on a commercial carrier airplane with a limited instrumentation package. Because this installation operates at a reduced gain while information is being collected, the results do not show such a large reduction in acceleration levels as those obtained during Boeing tests. A typical example of some of this data is shown in Figure 22, where a comparison of the acceleration levels at an aft body station during turbulence is shown with the system ON and OFF.

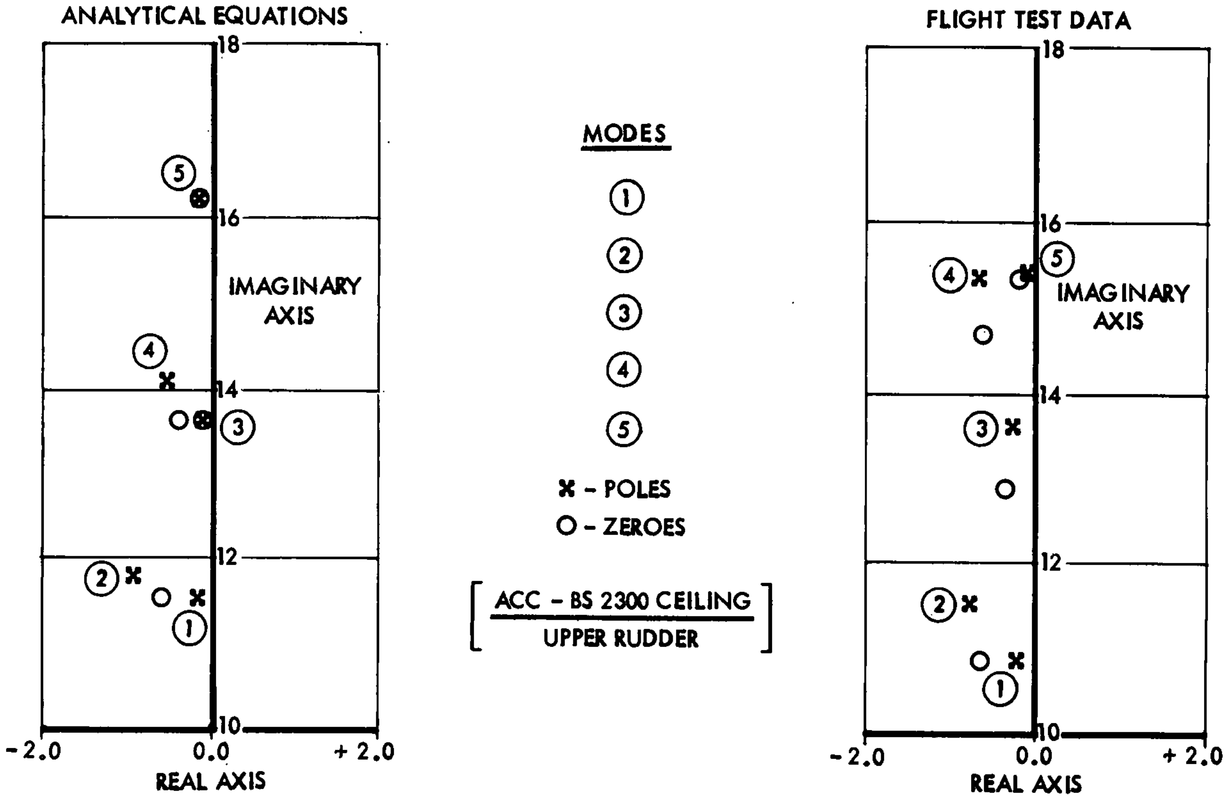
CONCLUSION

The development and testing of the Beta-vane and MSAS systems have been described. Data from flight test have indicated that a 50-70 percent reduction in aft body lateral acceleration levels can be achieved with the above systems. Non-linear filtering and different sensors will be the subject of future research.

REFERENCES

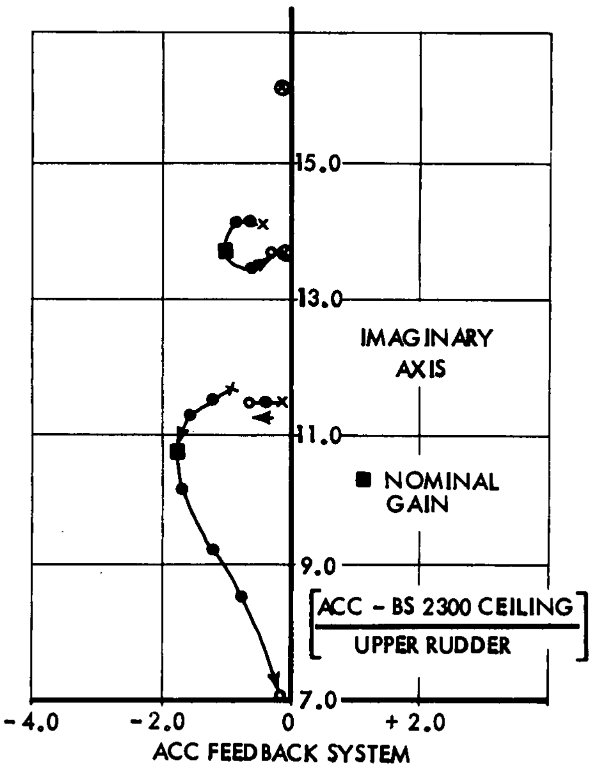
1. E. C. Levy, "Complex Curve Fitting," IEEE Transactions on Automatic Control, vol AC-4, pp 37-44; May 1959.

2. C. K. Sanathanan and J. Koerner, "Transfer Function Synthesis as a Ratio of Two Complex Polynomials," IEEE Transactions on Automatic Control, pp 56-58; January 1963.



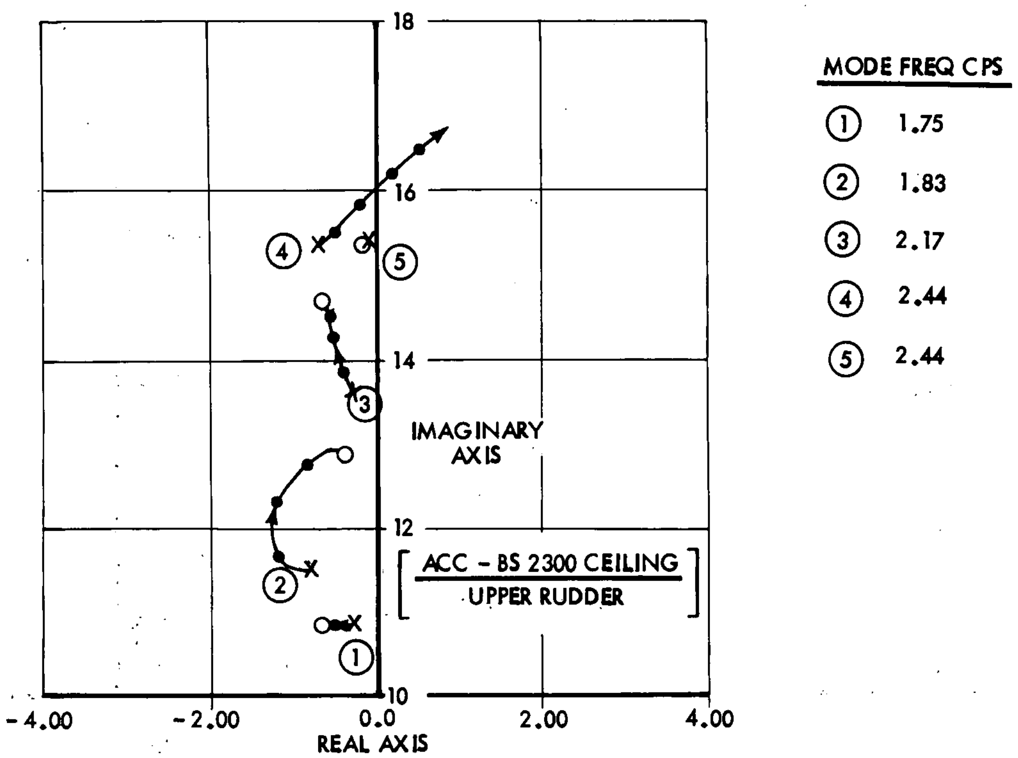
COMPARISON OF ANALYTICAL AND FLIGHT TEST DATA

FIGURE 1



ROOT LOCUS PLOT OF CONTROL SYSTEM BASED ON ANALYTICAL EQUATIONS

FIGURE 2



ROOT LOCUS PLOT OF CONTROL SYSTEM BASED ON CURVE FIT COMPUTER PROGRAM

FIGURE 3

TRANSFER FUNCTION

LATERAL ACCELERATION (AFT BODY) / RUDDER (LOWER)

- ALTITUDE = 31,000 FT
- MACH = .86

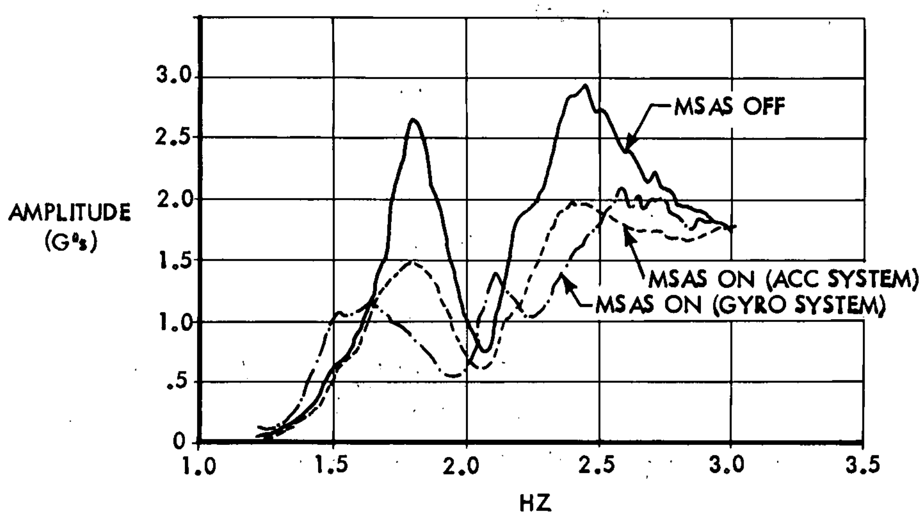
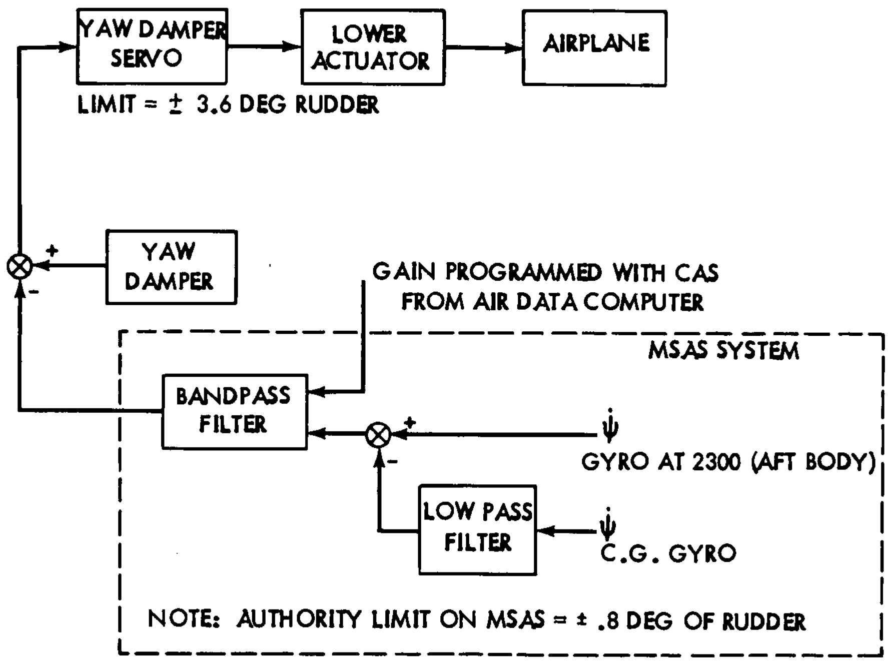
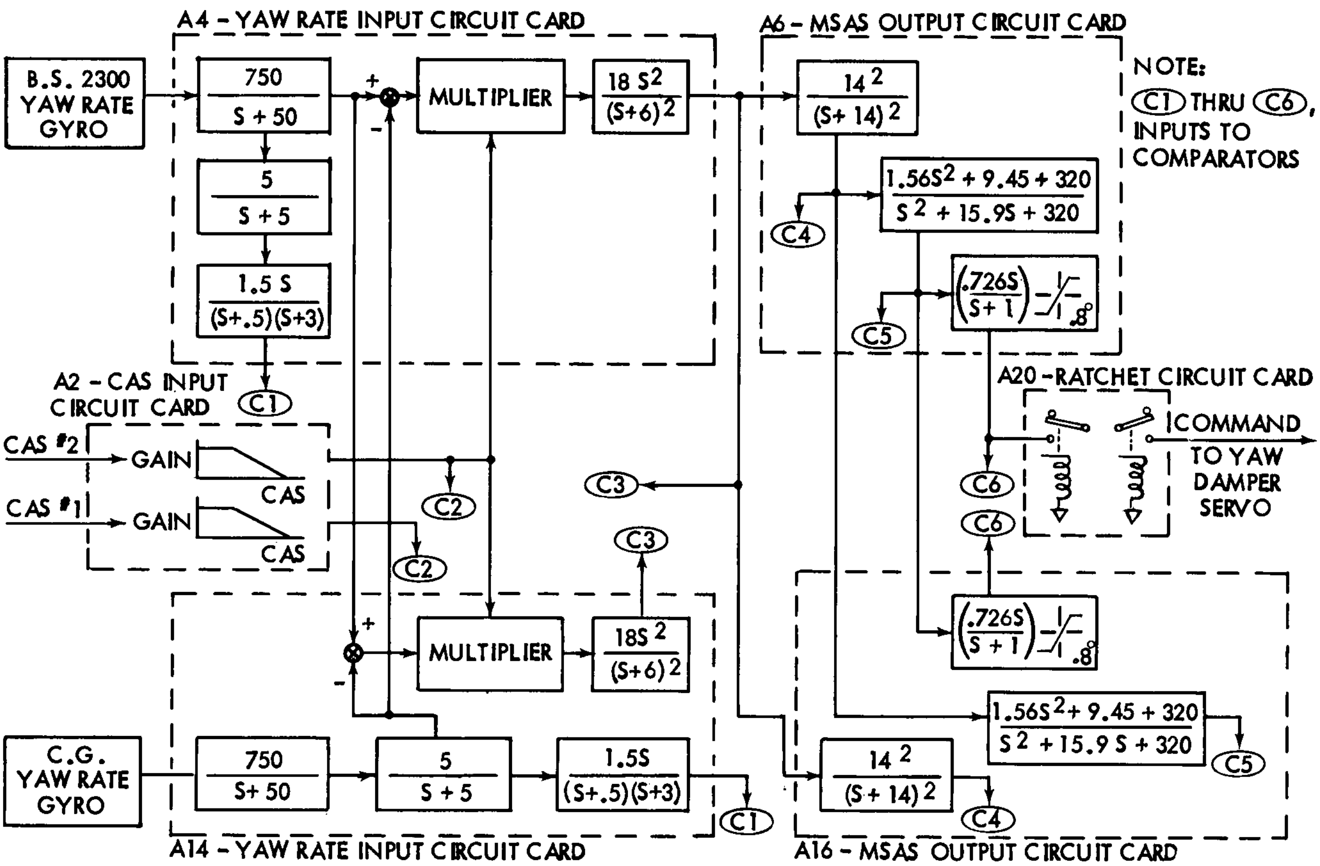


FIGURE 4



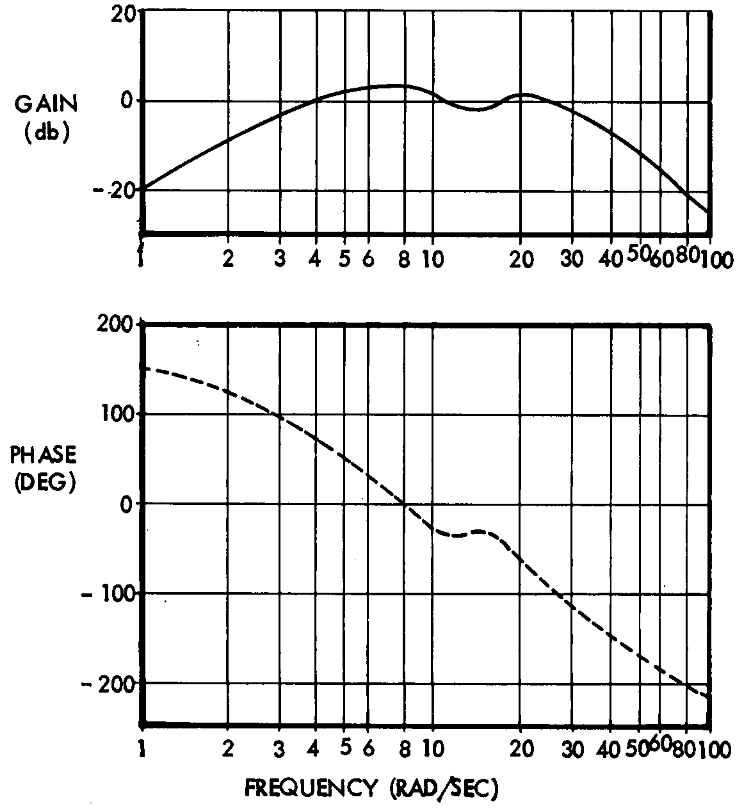
BLOCK DIAGRAM OF MSAS CONTROL SYSTEM

FIGURE 5



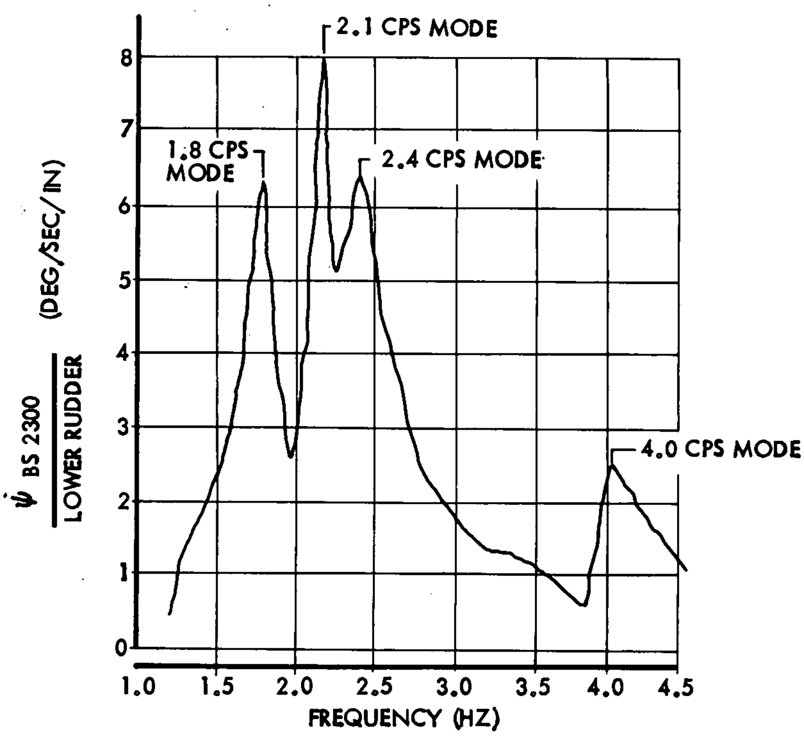
SYSTEM DIAGRAM MODAL SUPPRESSION AUGMENTATION SYSTEM

FIGURE 6



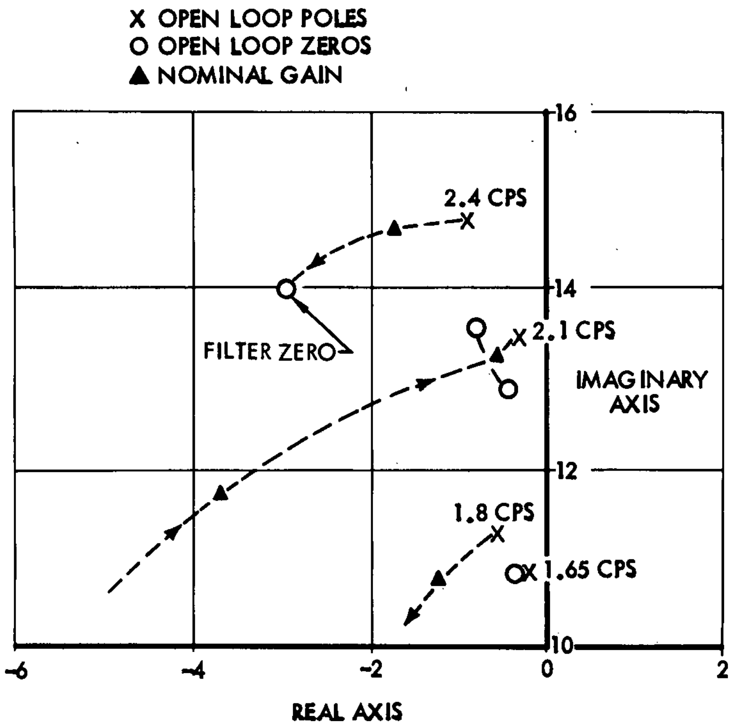
GAIN AND PHASE PLOT OF MSAS FILTER

FIGURE 7



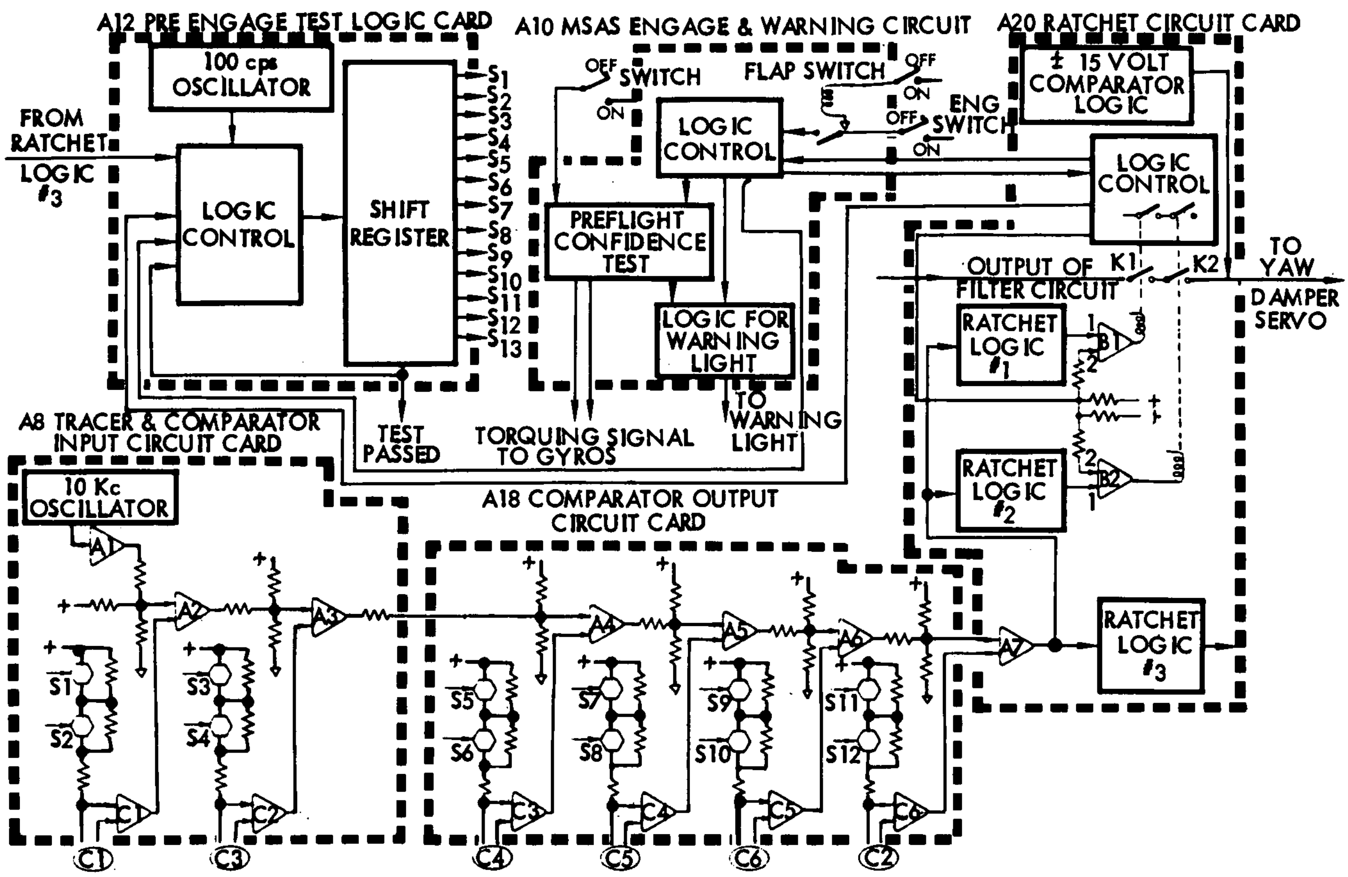
AMPLITUDE RESPONSE - $\dot{\psi}_{BS\ 2300}$/LOWER RUDDER INPUT

FIGURE 8



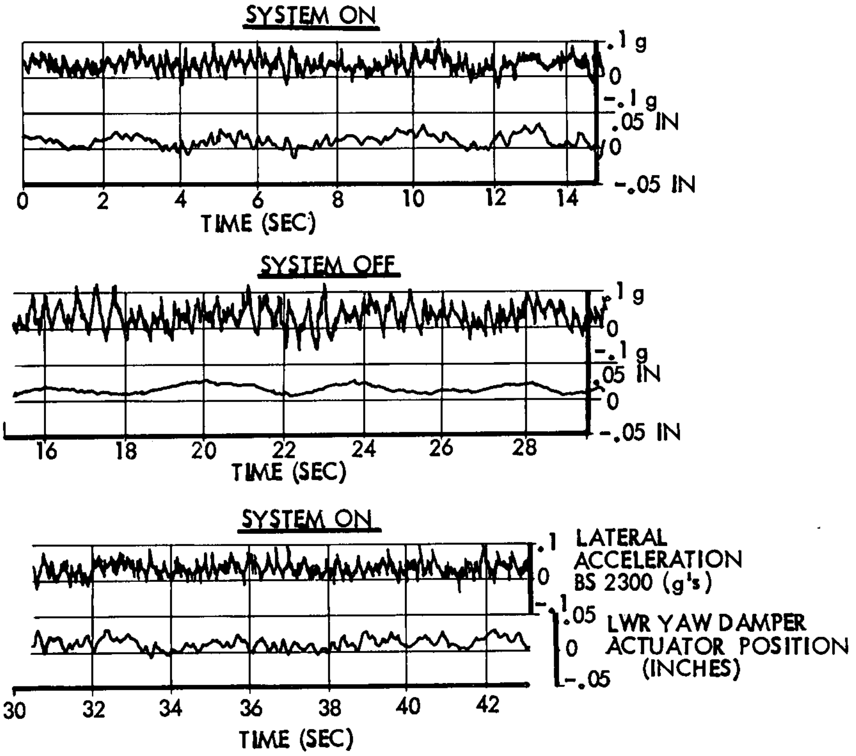
ROOT LOCUS PLOT OF CONTROL SYSTEM

FIGURE 9



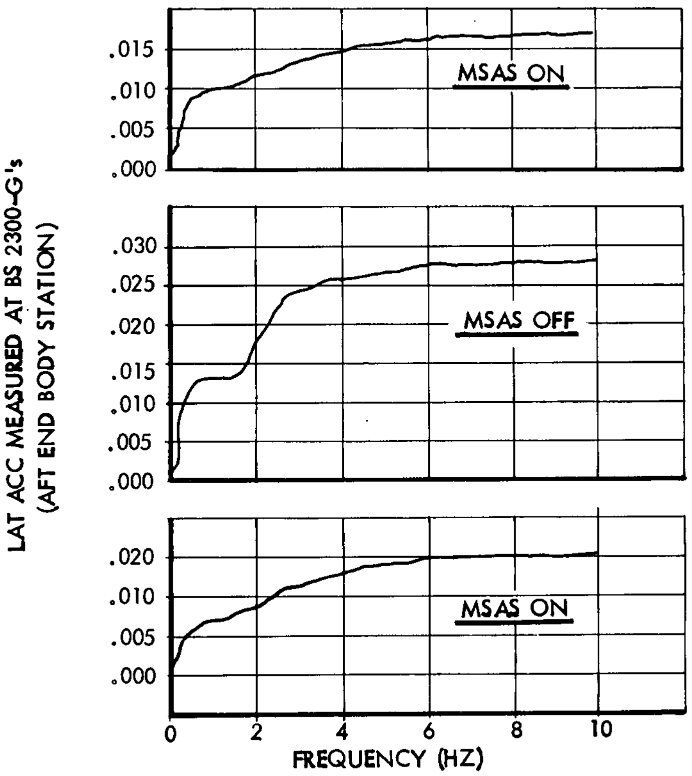
FUNCTIONAL BLOCK DIAGRAM OF MSAS MONITORING SYSTEM

FIGURE 10



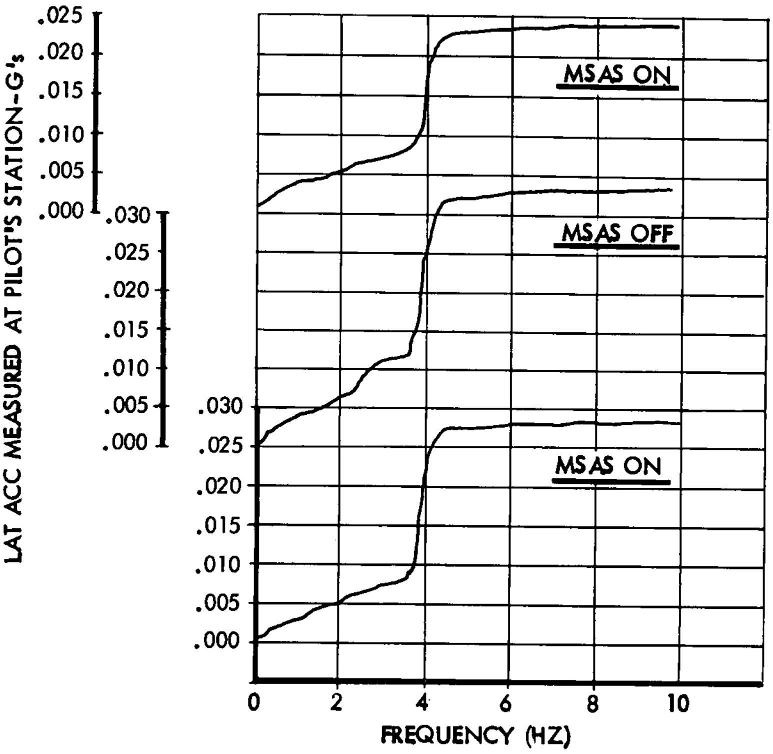
AIRPLANE RESPONSE IN TURBULENCE WITH AND WITHOUT MSAS

FIGURE 11



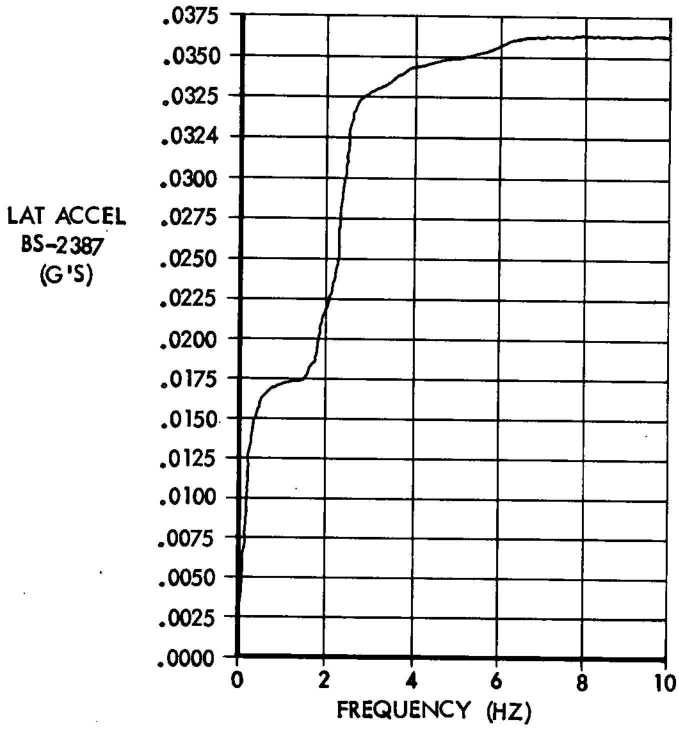
ACCUMULATIVE RMS - TURBULENCE

FIGURE 12



ACCUMULATIVE RMS - TURBULENCE

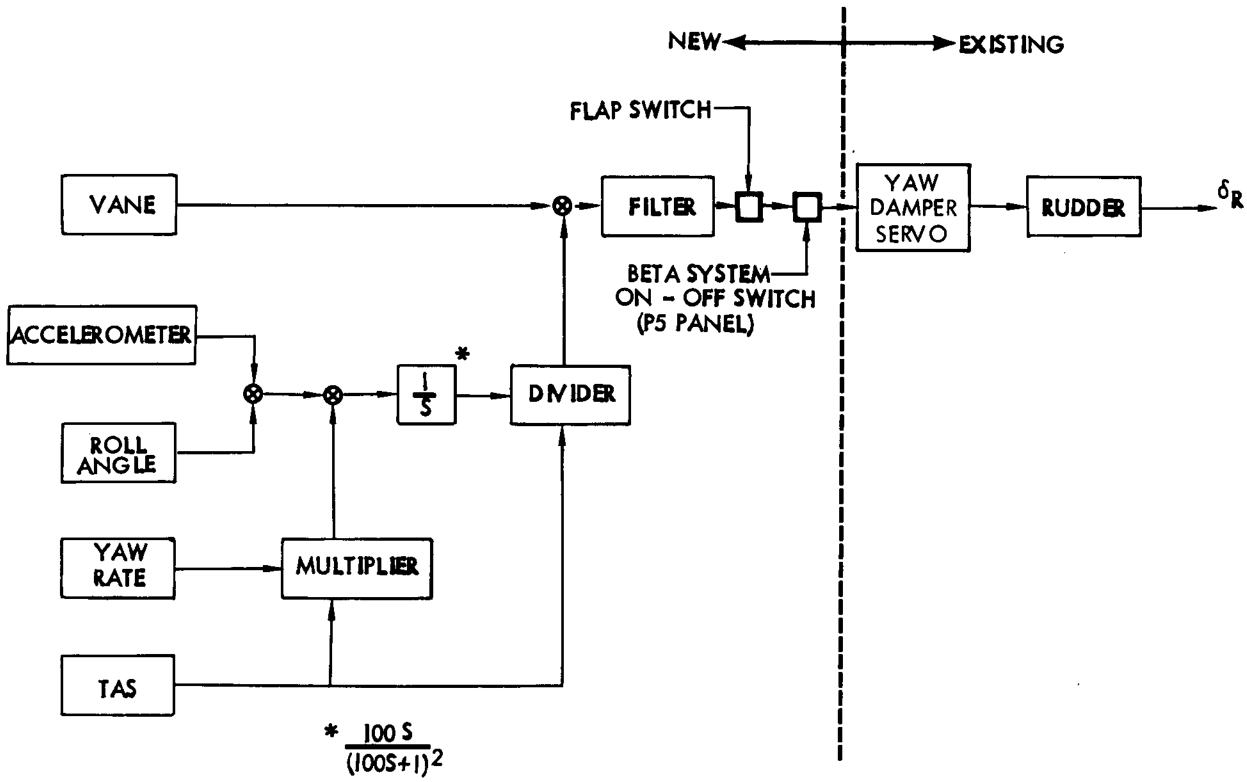
FIGURE 13



CUMULATIVE RMS - QANTAS BETA

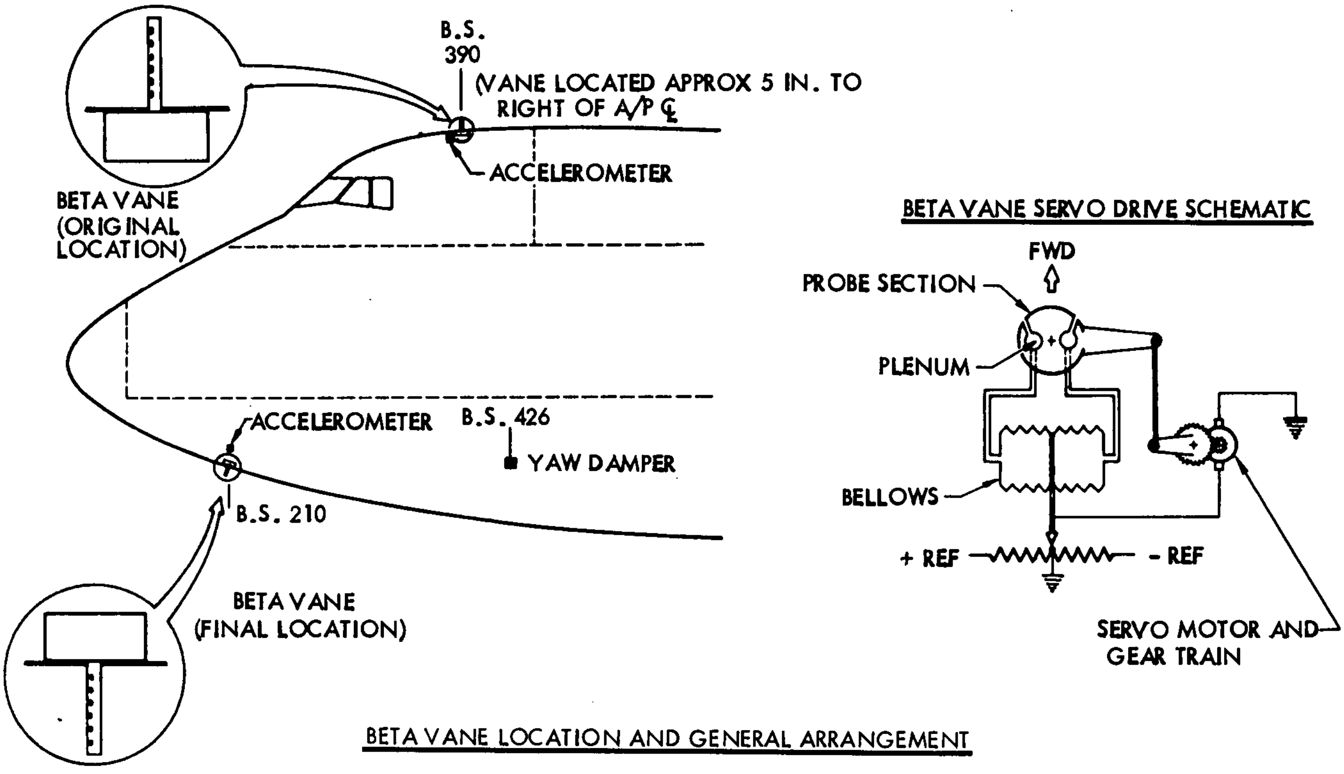
SYSTEM EVALUATION IN TURBULENCE

FIGURE 14



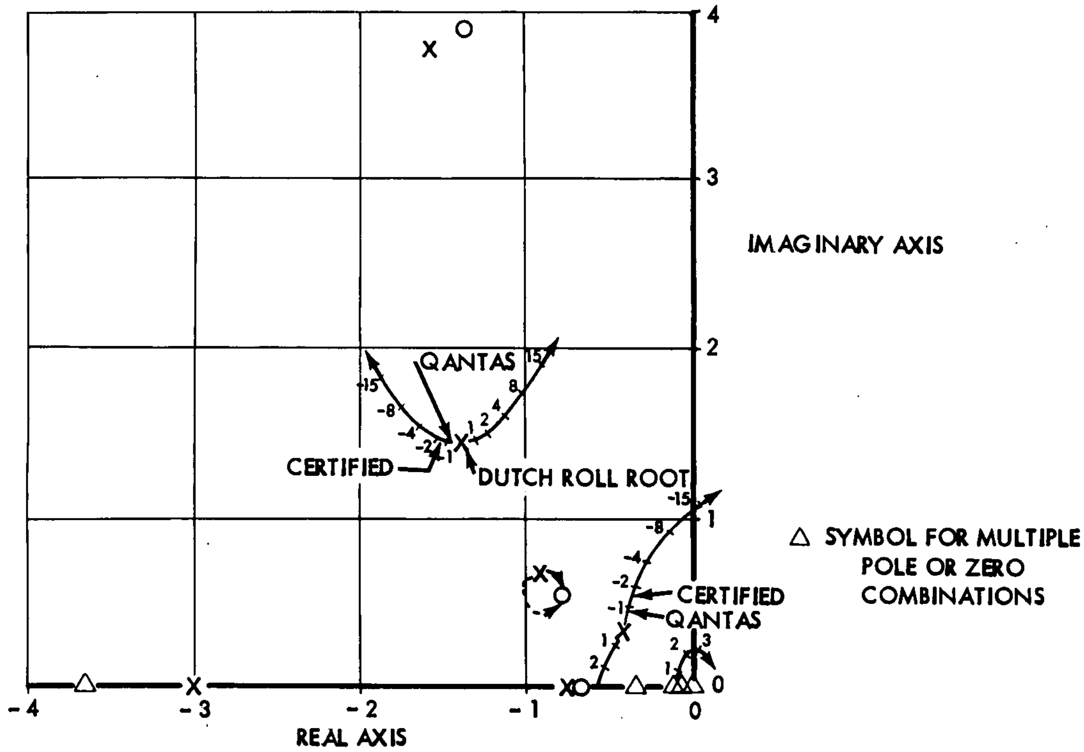
BETA MODIFIED YAW DAMPER BLOCK DIAGRAM

FIGURE 15



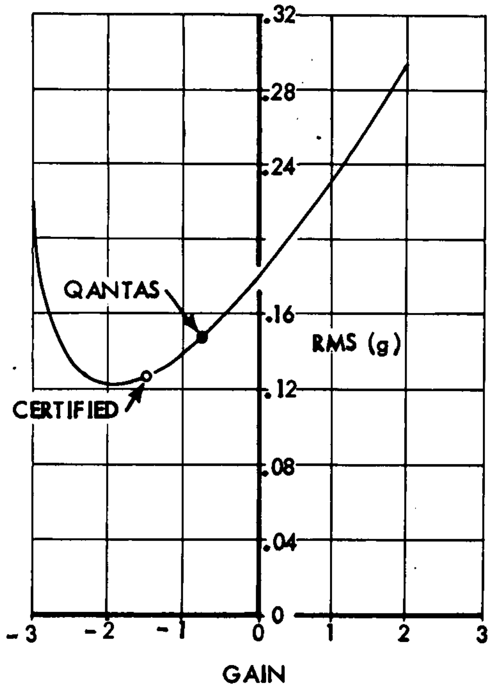
BETA VANE LOCATION AND GENERAL ARRANGEMENT

FIGURE 16



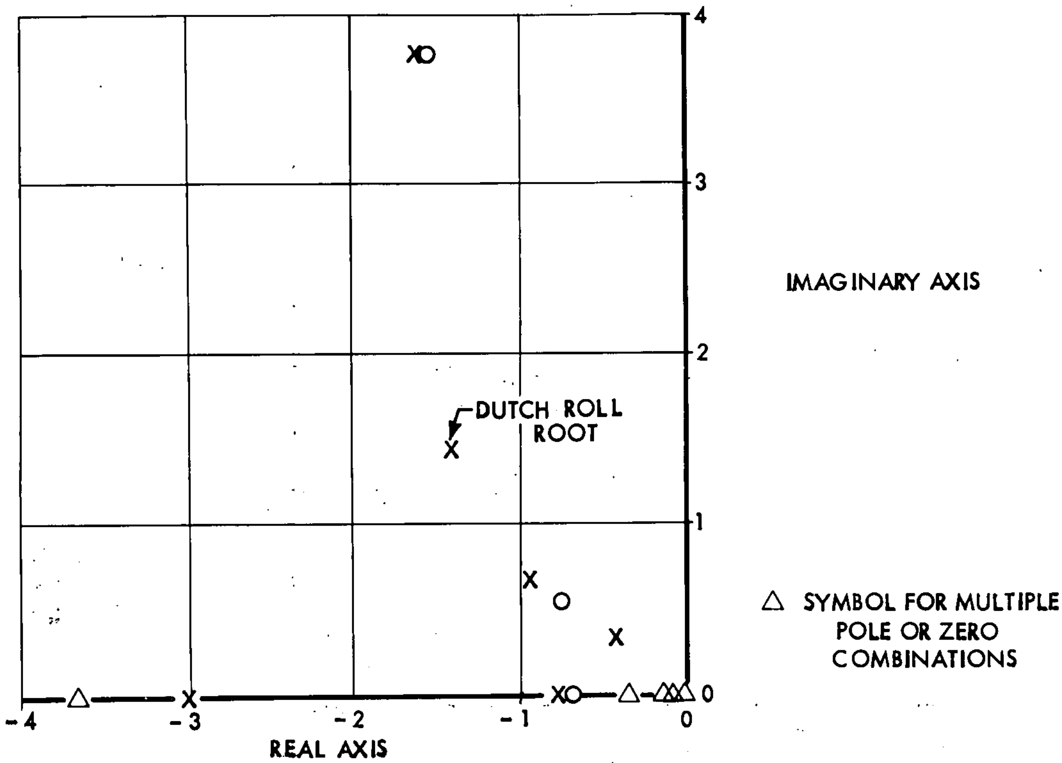
ROOT LOCUS FOR PARTIALLY COMPENSATED SYSTEM WITH NOMINAL YAW DAMPER GAIN

FIGURE 17



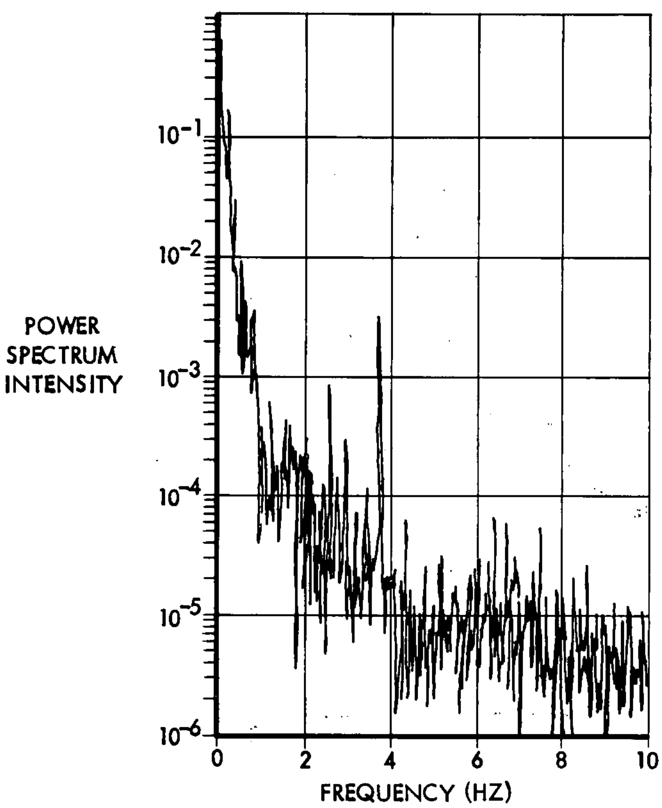
CUMULATIVE ROOT MEAN SQUARE ACCELERATION (BS 2300) VERSUS SYSTEM GAIN
Actual Compensation (Beta System)

FIGURE 18



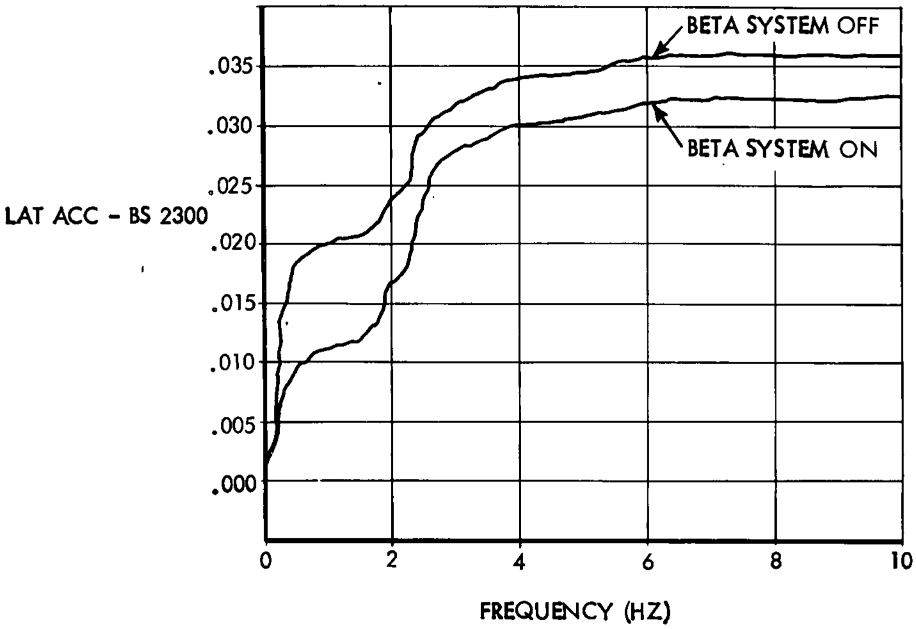
ROOT LOCUS FOR PERFECTLY COMPENSATED SYSTEM
Reduced to Airplane Roots Only

FIGURE 19



POWER SPECTRUM OF COMPENSATED BETA SIGNAL MEASURED IN TURBULENCE

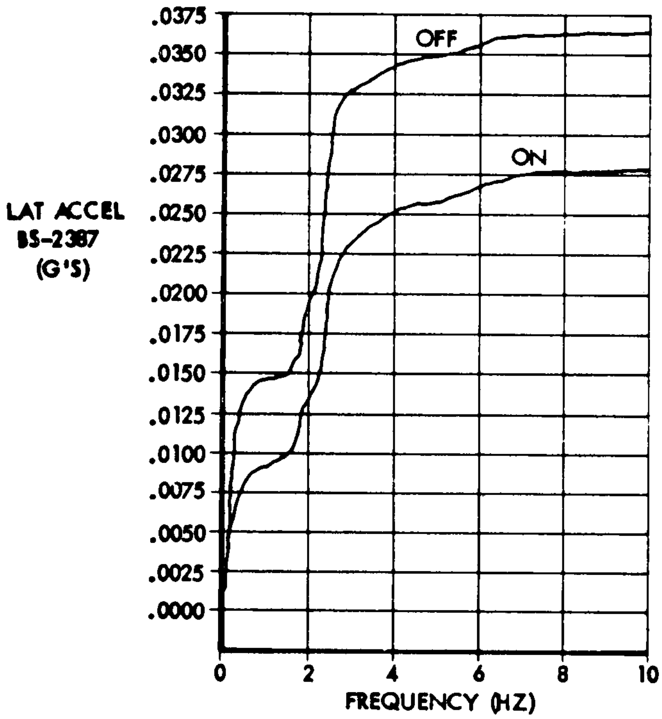
FIGURE 20



CUMULATIVE RMS

Beta System In Turbulence

FIGURE 21



CUMULATIVE RMS QANTAS BETA SYSTEM EVALUATION IN TURBULENCE

FIGURE 22

THE C-5A ACTIVE LIFT DISTRIBUTION CONTROL SYSTEM

by
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SUMMARY

This paper presents the technical details of the development of the Active Lift Distribution Control System (ALDCS) for the C-5A aircraft. The text is developed from a structural loads, flutter-control system interaction viewpoint, in such a way that the unique aspects of the analysis, aeroelastic wind tunnel test, and flight test portion are bound together to indicate the system design characteristics and performance.

The purpose of the ALDCS is to reduce gust and maneuver incremental wing root bending moments while minimizing the effects of the control system on torsion, flutter, and flying qualities. In other words, these criteria are based on axial load reduction as a means of improving wing fatigue endurance without significantly affecting existing flutter margins or handling qualities.

Even though this is a retrofit system which was required to use as much existing hardware as possible, throughout flight test all design goals have been met. The system is currently planned to be manufactured and installed on the fleet during the next several years.

INTRODUCTION

Load alleviation studies for the C-5A airplane were begun in 1967 during the design stages of the C-5A. A brief history and description of the earlier systems studied are included as an introduction to this paper.

Loads Alleviation and Mode Stabilization (LAMS) Program - The LAMS program was the first of the studies but was basically academic in nature as far as the C-5A was concerned. The LAMS program, see Reference 1, was directed at reducing airplane fatigue loads by changing the response characteristics of the C-5A during turbulence. The load reduction and ride improvement worked through a combination of aileron, spoiler, and elevator deflections to reduce the short period and modal response. The aileron and spoilers, when activated by vertical accelerations, altered the wing lift distribution providing a direct source of load reduction. The inboard elevators provided an indirect wing load reduction by increasing pitch damping.

The studies performed on the C-5A were conducted primarily by Minneapolis Honeywell, Inc., under contract to Boeing and the U.S. Air Force Flight Dynamics Laboratory. Lockheed participated by providing the C-5A data, a mathematical model, and support to the analysis effort to demonstrate the applicability of the analysis methods and techniques of the LAMS program to another large flexible airframe. The study results concluded that a LAMS type control system could reduce structural fatigue damage rates caused by turbulence without significantly affecting aircraft stability or handling qualities.

Maneuver Lift Distribution Control System (MLDCS) - Maneuver load alleviation work was conducted by Lockheed in late 1969 and early 1970 to study various means of reducing maximum wing upbending moments on the C-5A. The basic goal for this system was to reduce design wing root bending moments by 10% without affecting the airplane handling qualities or performance. Again, the ailerons were used to relieve the wing bending moments, thus using existing hardware with minimum new components.

Since the goal of the system was to reduce the maximum upbending moments for static strength, the MLDCS was designed to be activated only after a 1.5g load factor was reached. The frequency characteristics of the system were such that little aileron activity would be obtained above the first wing bending frequency. Normal accelerometers used to activate the system were located at the wing first bending node line, making the system independent of nearly all aircraft flexible response. Flight test of this system showed that the system design load reductions could be obtained with no indications of system instabilities. However, during the MLDCS development, it became clear that some form of fatigue loads reduction was highly desirable. This fatigue load relief could be obtained by aileron uprig. Therefore, after comparing aircraft performance with no drag penalties during takeoff, climb, and cruise versus costs of construction and installing the MLDCS into the C-5A fleet, it became apparent that a simplification of the system was desirable. As a result of this conclusion a passive load alleviation system was studied.

Passive Lift Distribution Control System (PLDCS) - The primary objectives of the PLDCS were similar to the MLDCS but, in addition, it would provide service life improvement by reduced 1.0g mean bending moments. The PLDCS concept is a fixed aileron uprig system with specific amounts of uprig as a function of airplane configuration and flight condition. Studies indicated that the static load reduction objective could be attained with a two position system having five degrees uprig above twenty thousand feet and ten degrees uprig below twenty thousand feet. The C-5A fleet has been using the PLDCS since November 1971. The structural loads improvement obtained with the PLDCS is illustrated in Figure 1.

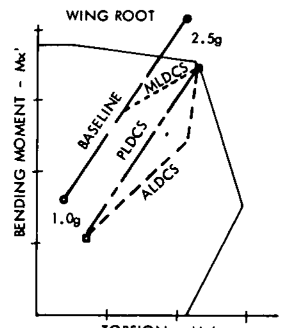

FIGURE 1 C-5A lift distribution systems wing structural loads improvement

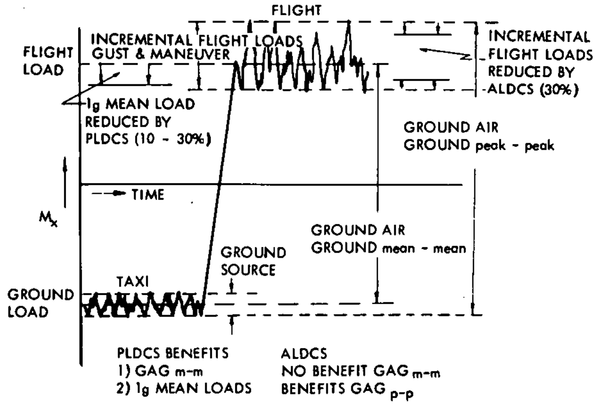

FIGURE 2 Airplane ground — flight load sources as affected by PLDCS and ALDCS

Active Lift Distribution Control System (ALDCS) - In 1971-72 an Air Force established Independent Review Team (IRT), made up of U.S. Air Force, Lockheed, and other structures personnel in the field, reviewed the wing fatigue characteristics of the C-5A. Many recommendations were made by this group for increasing the fatigue life of the C-5A wing both for the short and long range fleet management. Many of the more obvious things such as changing airplane usage, revised fuel distribution and revised aileron uprig (PLDCS) were recommended and incorporated into the overall management of the C-5A fleet. Another recommendation of the IRT was the development of an Active Lift Distribution Control System (ALDCS) which would offer, as an interim fix until a more permanent solution could be implemented, significant fatigue damage reduction to the wing structure. This Active Lift Distribution Control System is the subject which will be dealt with in detail in this paper.

The ALDCS provides control surface command signals through the existing Stability Augmentation System (SAS) and primary servo-actuation system as a means of reducing fatigue damage on the C-5A wing due to maneuver, gust, and peak-to-peak ground-air-ground load sources (Figure 2). The system interfaces with existing C-5A control surfaces, actuators and servos, modified SAS and CADC (Central Air Data Computer) computers, and new hardware as shown in Figure 3. The first of a four phase program (Phase A) was awarded in May of 1973 for the detail development of the ALDCS. The schedule, dictated by Phase A, required that a prototype ALDCS computer be available for the C-5A Vehicle Simulator within an eight month period. Phase B of the contract, awarded in September 1973, allowed for flight test of the system and called for first flight of the C-5A with ALDCS to be in March of 1974, ten months after the initial development had been authorized. The first flight of the system took place within two weeks of the scheduled date and by the middle of the initial flight test program (May 1974) the ALDCS program was on schedule. Phase C of the ALDCS program, scheduled for July of 1974, calls for the production of the ALDCS computers to retrofit all aircraft in the C-5A fleet. The first systems are to be operational one to three years later. Figure 4 is the schedule of the ALDCS principal milestones. Phase D calls for the retrofit of all aircraft in the C-5A fleet to be completed approximately two to four years after the production contract (Phase C) award.

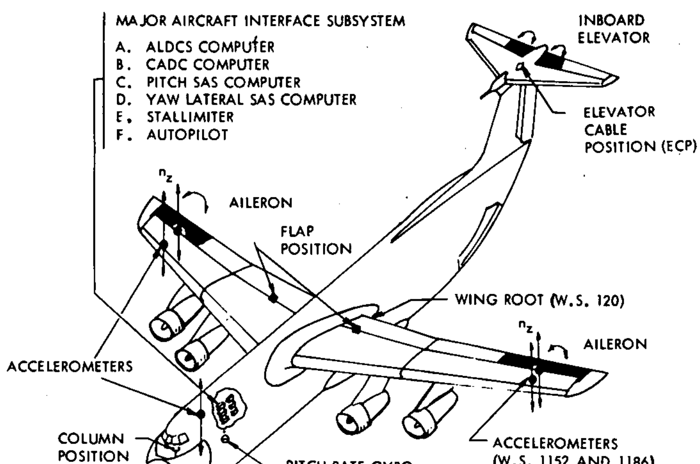

FIGURE 3 C-5A ALDCS major airplane components interface

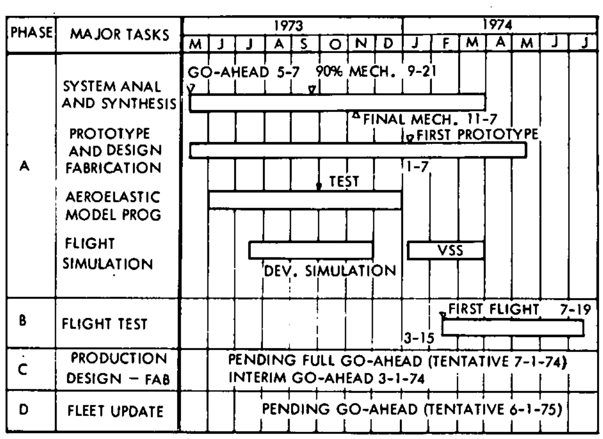

FIGURE 4 C-5A ALDCS development program schedule

Since the purpose of the ALDCS is to reduce wing fatigue damage during fleet usage, the design goals for the ALDCS are basically to reduce incremental wing root bending moment by 30% without significantly changing performance, flutter margins, and handling qualities of the C-5A. Following is a list of goals and criteria to which the system was designed:

o These general design goals were considered in the design of the ALDCS computer, sensors, and interfacing of the system with existing aircraft hardware and systems:

  oo The system shall operate on a full time basis within the design speed/altitude envelope of the C-5A.

  oo The system shall be designed to fail-safe concepts and no single failure of the ALDCS shall affect the normal operations of the pitch and yaw/lateral stability augmentation systems.

  oo The ALDCS shall be designed to interface with existing C-5A systems and shall use existing C-5A hardware where possible with minimum new components.

o These design goals of the ALDCS were specified for gust and maneuver load sources. The load reduction goals were defined in terms of wing root (W.S. 120) loads for conditions throughout the C-5A design speed/altitude envelope. The criteria were defined relative to the unaugmented aircraft.

  oo The maneuver incremental wing root bending moments with ALDCS operative shall not exceed 70% of the free aircraft values.

  oo The root-mean-square wing root bending moments with ALDCS operative shall not exceed 70% of the free aircraft values for root-mean-square gust velocities up to 5 feet per second. The shape of the gust spectra was defined by the Von Karman gust spectra with scales of turbulence varied as a function of altitude, as defined by Lockheed derivations from C-141A gust measurements (Reference 2).

  oo The associated root-mean-square wing root torsion moment shall not increase more than 5%.

  oo The system shall incorporate a variable aileron gain as a function of load factor and shall be designed so that, with load factors in excess of 1.9, the ALDCS incremental aileron deflection is removed such that, at design limit load factor of 2.5, the system is again in the PLDCS configuration. This is necessary to prevent the generation of a wing front beam shear flow problem.

o The ALDCS/airframe combination shall meet stability margin criteria so as to preclude any of the following:

  oo Adverse structural modal coupling or limit cycle tendencies.

  oo Significant degradation of existing handling qualities and flutter margins.

  oo Adverse coupling with existing flight control systems.

o Stability margin design goals were:

  oo 6 dB minimum gain and 45 degree minimum phase margin for all ground test and flight modes with a 10 dB goal for flight.

  oo 60 dB per decade attenuation (roll-off) and infinite phase margin for flight modes beyond control mode natural frequencies.

o The ALDCS shall be made inoperative for the flaps down condition. This decision was based on the fact that loads associated with the clean aircraft are higher than those associated with the flaps down condition. The peak loads which can be expected with the flaps down portion of a mission (ALDCS inoperative) are no higher than those with the flaps up portion of a mission (ALDCS operative). Therefore, the most damaging portion of the fatigue loads spectra, peak-to-peak ground-air-ground loads, are not reduced if the ALDCS is made operative in the flaps down configuration. Further testing and analyses of the flaps down condition would greatly increase the development and flight test costs of the ALDCS with a relatively small benefit to be gained.

The ALDCS design features, shown in Figure 5, are expanded as follows:

o The ALDCS is a fail-safe dual channel design which actively operates the ailerons and inboard elevators through the lateral and pitch Stability Augmentation System where a malfunction of the ALDCS will not affect normal operation of these systems. The design is a "full time" system which operates throughout the ALDCS design limit speed/altitude envelope. The ALDCS operates in conjunction with the passive LDCS and uses the existing servo actuators of the control surfaces.

o The ALDCS pitch channel uses the existing pitch rate gyro and forebody accelerometer associated with the C-5A Stability Augmentation System and autopilot systems. The location of these sensors is approximately twenty feet behind the cockpit in the upper lobe of the fuselage flight deck. The ALDCS computer is mounted in the same compartment. These sensors provide signals to the ALDCS computer pitch channel which commands the inboard elevator deflections through the pitch Stability Augmentation System.

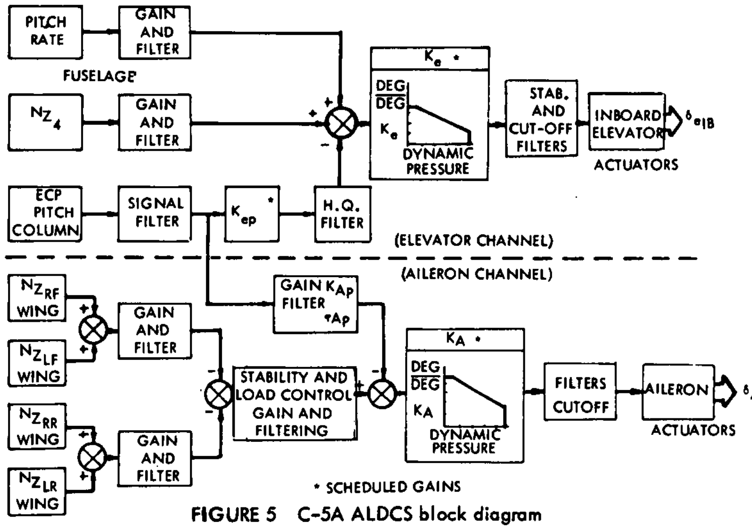

FIGURE 5 C-5A ALDCS block diagram

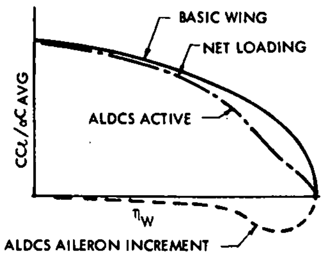

FIGURE 6 Effects of ALDCS on wing airload distribution

- o Additional sensors in the form of wing mounted accelerometers are used by the ALDCS computer to command symmetrical aileron deflection which suppresses response of the first wing bending mode and provides wing airload distribution changes so as to reduce maneuver bending moments. The accelerometers (two in each wing) are mounted at 89% semi-span and are located on the front and rear beams of the wing. This location is near the mid-span location of the ailerons. Each pair of wing accelerometers are summed proportionally front to rear (40% to 60%) to optimize the chord-wise location and are summed equally between wings to permit only symmetrical vertical input signals to the ALDCS computer.

- o The wing acceleration signals and the pilot induced column position changes provide the ALDCS information to compute aileron deflections. These symmetrical aileron deflections are achieved through the lateral Stability Augmentation System.

- o The gains of the ALDCS computer are dependent on the Central Air Data Computer (CADC) which is original equipment on the aircraft. The basic aileron and elevator gains are scheduled as a function of dynamic pressure. The CADC signal is also used to turn the ALDCS off in the event that the design limit Mach number (0.825) and/or speed (350 KCAS) of the ALDCS are exceeded. Similar information from the CADC is used to re-engage the ALDCS when the aircraft reenters the ALDCS design speed/altitude envelope.

- o Stallimiter disengages the ALDCS in the event that the aircraft approaches stall.

- o A flap position of more than five degrees is used as a signal to disengage the ALDCS to eliminate any possible interference between automatic flight control systems operating in the terminal phase of flight.

- o Aircraft vertical acceleration for maneuver is part of the ALDCS logic which assures that there will be no aileron deflection due to ALDCS command present for a net load factor of 2.5g. This load factor is computed by summing the average of the wing and fuselage forebody accelerations. The acceleration computation is a rough approximation of the aircraft center of gravity acceleration with the forebody accelerometer and wing response equally weighted.

LOADS ANALYSIS

The two different load sources, maneuver and gust, require separate design approaches since the aircraft response is initiated differently and the desired ALDCS effects on aircraft response are different. For maneuver, the ALDCS cannot affect the overall response significantly but can only affect the wing loads which result from a given aircraft response. In addition, a very small portion of the wing loads associated with maneuver can be contributed to the response of the aircraft flexible wing modes. Consequently, the wing load relief for maneuver must be obtained strictly by changing the wing airload distribution by deflecting the aileron to reduce the proportion of the lift contributed by the outer wing. Figure 6 shows these air load distribution changes. The wing lift loss on the outer wing is insignificant compared to the total wing lift and is compensated for by a very slight increased angle of attack of the aircraft. The aircraft nose-up pitching moment associated with trailing edge up aileron deflection is easily compensated for by a slight trailing edge down elevator deflection. This feature is designed into the ALDCS and makes it nearly impossible for the pilot to detect the operation of the ALDCS in the aircraft handling qualities during maneuver. The simplicity of the system in maneuver, however, brings about a problem for which there is no compensation within the design requirements of the ALDCS. The trailing edge up aileron accomplishes the primary purpose of reducing wing root bending moment but also produces a wing leading edge up torsion moment. Only a certain amount of this torsion moment can be allowed and have the wing loads stay within a design wing bending-torsion envelope. Consequently, a "fader" is designed into the ALDCS to remove the aileron deflections commanded by the ALDCS at load factors approaching the 2.5g design value.

A quite different situation is present for relieving loads associated with turbulence. First, there is no requirement that the aircraft respond as a result of a gust encounter. In fact, the loads resulting from aircraft response due to gusts are quite often higher than the direct gust loads themselves. This response is primary short period motion and, therefore, can be reduced by proper elevator commands. The ALDCS uses aircraft pitch rate and forebody load factor to command elevator deflection to reduce the aircraft pitch response. Second, a significant proportion of the gust fatigue wing load spectra can be attributed to response of flexible wing modes, primarily the fundamental wing bending mode. The ALDCS aileron command frequency response is shaped so the maximum gain is available for the frequency range associated with the first wing bending mode (i.e., 0.7 to 1.2 Hz depending on fuel loading). Figure 7 shows a transfer function and the corresponding output spectrum for wing root bending response to gust. Torsion increases associated with aileron deflections commanded by the ALDCS still exist for gusts as they did in maneuver, but are compensated for by less aircraft response due to the ALDCS elevator commands. The overall effect on rms gust loads is to significantly reduce wing bending moments with little effect on torsion moments for a given gust environment. At one point in the design of the ALDCS, significant increases in aftbody bending loads were evident due to the use of elevator in reducing aircraft pitch response. The addition of a filter in the elevator channel of the ALDCS changed the phasing of the elevator command such that a reduction in aftbody bending moment is produced which more than compensates for the loads produced by the elevator in controlling this response.

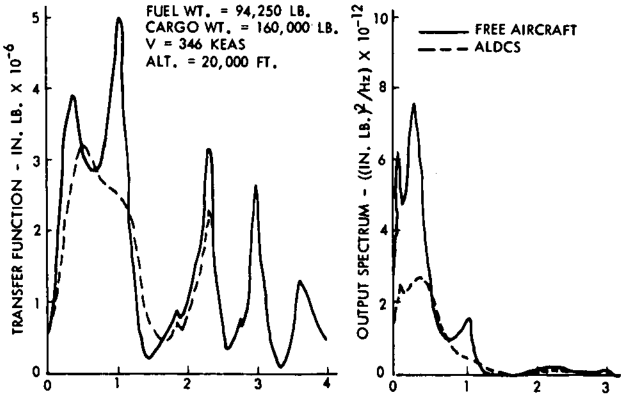

FIGURE 7 Wing root bending moment for one FPS RMS gust

The ALDCS loads development tasks involved three basic analyses. Below is a brief description of the analyses, i.e., computer programs used to develop the ALDCS. It should be noted that the same mathematical model was used for the loads program as was used for the major portion of the stability work.

- An analog computer program was developed to analyze the ALDCS. The structural model consisted of the basic pitch and plunge degrees of freedom plus six selected aircraft flexible modes. The complete nonlinear control systems including actuator characteristics, hinge moment limits, and surface deflection limits were simulated. The aerodynamics for the model was linear strip theory which was derived from a much more elaborate aerodynamic model. The five panel system, used for the analog model, was used only for aerodynamic delay and penetration effects. The primary objective of this program was to provide good qualitative results for quick turnaround parameter variations during the development of the ALDCS mechanization. This program computed aircraft response and loads due to control system inputs at a limited number of stations. Discrete gust and continuous turbulence forcing functions were also considered in this program.

- The primary loads analysis program used for ALDCS design was a digital program which simulated the structural model with three rigid body degrees of freedom and the first 15 flexible aircraft symmetrical modes. Linear strip theory aerodynamics was used with unsteady aerodynamics computed by Kussner and Wagner functions. The program overall size capability, including the control system representation, was 22 degrees of freedom of sixth order equations of motion. The output of the program indicates stability characteristics and/or loads associated with either turbulence or control surface input. Automatic control systems can be evaluated either closed or open loop.

- The maneuver loads analysis program combines a structural model which has the C-5A flexible characteristics defined quasi-statically and a very detailed representation of the C-5A wind tunnel and flight test aerodynamic data. Loads are available at any point on the aircraft structure. This program was used to define the effects of ALDCS on steady maneuver loads. These quasi-static maneuver loads were used for mean and maneuver incremental loads for fatigue analysis.

The analog discrete analysis produced aircraft response to discrete and continuous turbulence and maneuver. This analysis, although it did not contain sufficient flexible degrees of freedom to give good quantitative loads, was adequate to give good qualitative results for aircraft response, load ratios, and control system non-linear effects to compare system on and off characteristics. The ability to make parameter changes immediately made the analog program a valuable tool in identifying trends produced by various ALDCS parameters and checking the effects of these trends on a wide variety of aircraft forcing functions. Discrete transient gust, though not used for design of the C-5A, was used to evaluate the ALDCS response for high frequency forcing functions. It was found that the discrete gust encounter offered no real problem to the ALDCS in reducing wing load for a variety of gust gradient inputs. A continuous gust time history of white noise shaped by the Dryden Gust Spectra was used in the analog program to check for non-linearities in the system and its ability to produce the desired load reduction. Peak count of aircraft response parameters and wing loads were studied for various combinations of autopilot, Stability Augmentation System, ALDCS and basic aircraft configurations. The output spectra and derived root-mean-square loads for various gust levels indicated that, for the root-mean-square gust levels normally encountered by the C-5A, the aileron actuator characteristics and deflection limits produce no apparent non-linear effects.

One important requirement of the ALDCS, as previously mentioned, is to reduce maneuver load associated with the fatigue spectra of the airplane; however, it should not produce new design load conditions associated with the 2.5g design condition. The increased torsion associated with the aileron deflection commanded by the ALDCS would give loads outside of the wing root bending-torsion envelope. See Figure 1. It is, then, necessary to remove the aileron before the 2.5g load condition is reached so as to be assured that the limit load envelopes are not exceeded. Using the response characteristics of the C-5A, a g load factor level of 1.9 was used as the point at which the ALDCS would start to remove the aileron deflection which it had commanded. A time delay of one half second was built into the system to eliminate the possibility of nuisance disconnects associated with gust response. The 1.9g level followed by the time delay allows the system "fader" to remove the incremental aileron deflection it has commanded before the aircraft can obtain a 2.5g net load factor. This value was picked based on an aft center of gravity condition at high dynamic pressure. It is necessary to keep that load factor as high as possible in order to influence handling qualities the least and to cause as few as possible system off conditions. A lower load factor level of 1.7g was selected to re-engage the ALDCS after it had sensed a load factor turning it off. This value was later reduced to 1.3g's when it was found that the system could be forced into a mild limit cycle by maintaining a constant load factor near 1.9g forcing the "fader" to cycle on and off.

In addition to the previous conditions, the abrupt maneuver condition (i.e., one commanded by an abrupt pilot elevator input) had to be considered. The analysis indicated that the aircraft response to elevator inputs did not appreciably change for step inputs to ramp inputs of approximately 1.0 to 1.5 seconds duration. The ramp input is in the time range that normal pilot inputs can be expected. For the abrupt maneuver condition another ALDCS feature was required to obtain load relief since the wing accelerometers for a pitch-up condition would be slightly behind the aircraft center of gravity. A slight delay in aileron command was expected; therefore, the load relieving aileron deflection could not be expected in time to reduce the wing loads significantly in this situation. An "aileron to stick" cross feed was added to the ALDCS. This cross feed commands an aileron deflection whenever the control column is moved. To compensate for the pitching moment associated with the "stick commanded" aileron deflection a lead term is also inserted into the elevator command. This cross feed produces the desired wing load reduction for the abrupt maneuver condition comparable to those associated with the steady maneuver loads.

The analog analysis served an important purpose in the development of the ALDCS, but the large digital analysis with more aircraft degrees of freedom was used to derive the ALDCS gains required for power spectral gust. The program was also used to design the major portion of the filter characteristics in the ALDCS mechanization to assure that the stability characteristics met the criteria for the system.

The initial ground rule for designing the ALDCS was to control the response of the first and second wing bending modes, (i.e., frequencies up to 2.75 Hz). Initial side studies indicated that there was little to be gained in attempting to control the response of the second wing bending mode since only a very small portion of the turbulence loads could be attributed to that mode. This resulted in a system design simplification since the frequency range of the aircraft short period and the first wing bending mode could be mechanized by a single circuit. In addition it was possible to reduce the system gain sufficiently between the first wing bending frequency and the 8.5 Hz and outer wing torsion mode so that no elaborate filtering was required at the basic aileron frequency. In spite of this fact, the 8.5 Hz mode required more care in the mechanization of ALDCS than any other frequency range in which the ALDCS was involved. The aileron mode required that two accelerometers be placed in each wing, one on the front beam and one on the rear beam. The output contribution of these accelerometers were weighted such that the total accelerometer response of this mode approximated that of the node line of the mode (i.e., approximately zero). The node line for this mode is nearly parallel to the aileron hinge line and lies approximately 10 inches aft of the wing elastic axis. The two-accelerometer design was dictated because of the problems associated with physically mounting the accelerometers outside of the wing or in the fuel tanks.

Stability analysis results show (Figures 8 and 9) that the design stability goals of 10 dB were obtained for all flight conditions within the C-5A ALDCS flight envelope. The most critical condition occurred at minimum reserve fuel. The scheduling of gains with dynamic pressure made the system relatively insensitive to speed. Minimum stability margins were nearly constant for all values of dynamic pressures. The elevator channel of the ALDCS proved to have the smallest gain margin. Phase margins were smallest for the aileron loop of the ALDCS but were never less than the 45 degree design value.

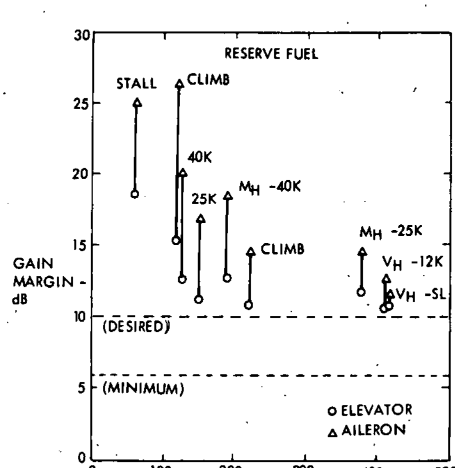

FIGURE 8 C-5A ALDCS stability gain margins

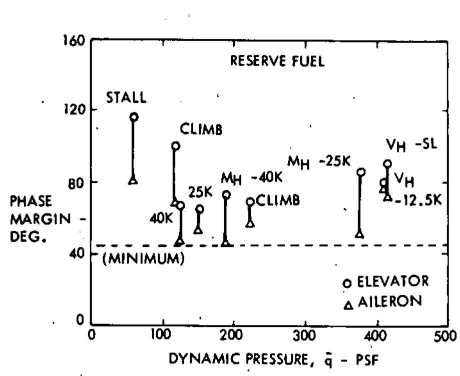

FIGURE 9 C-5A ALDCS stability phase margins

In addition to the basic airplane with ALDCS a functioning autopilot had to be considered as a part of the total system. Analysis was conducted using the autopilot system combined with the ALDCS. The addition of the autopilot to the ALDCS showed no significant detrimental effects on the aircraft.

The power spectral density gust analysis (summarized in Figures 10 and 11) with the effects of ALDCS engaged showed that the wing root bending moment design goal ratios have been met for all flight conditions in the C-5A ALDCS flight envelope. In addition, the lowered aircraft pitch response has caused the torsion moments to remain at the same value or lower in spite of the fact that the ailerons are being used to reduce the wing response.

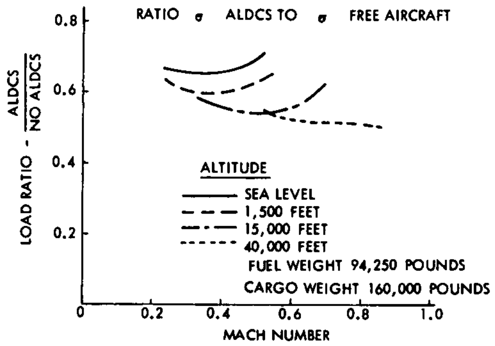

FIGURE 10 Wing root RMS bending moment ratio between ALDCS-on and ALDCS-off

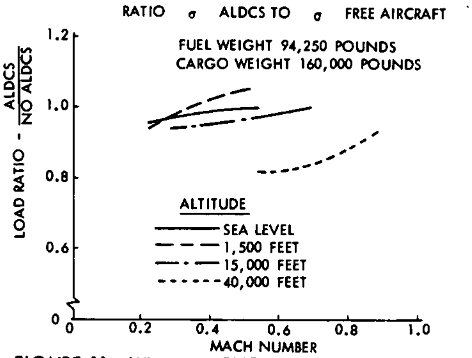

FIGURE 11 Wing root RMS torsion moment ratio between ALDCS-on and ALDCS-off

FLUTTER ANALYSES

The flutter characteristics of the C-5A with ALDCS were evaluated by digital analysis, by Direct Analogy Electrical Analog Computer (DAEAC) analysis, and by high speed aeroelastic model wind tunnel tests. All three phases of the investigation required the development and utilization of procedures and techniques not required on previous production projects. The three techniques were utilized to implement more than one approach to the problem in such a way that they complimented each other. For example, the analog computer analysis provided early but limited results. The digital program provided the more comprehensive analysis. Without extensive prior flutter analysis of the C-5A with active control/ feedback airplane systems, the model tests provided additional results over a large Mach number and dynamic pressure envelope.

Airplane digital vibration analyses were conducted. Some 52 component modes plus rigid body modes were computed. The wing, horizontal stabilizer, and vertical stabilizer component modes used in the analysis were uncoupled cantilever modes. These component modes were synthesized to obtain complete airplane symmetric free-free modes. The ALDCS is designed to be effective only for symmetric airplane motion; therefore, the antisymmetric flutter characteristics are the same as the no ALDCS condition. Twenty-seven complete airplane symmetric free-free modes plus rigid body pitch, vertical translation, and fore and aft translation were used in the analyses.

Oscillatory aerodynamic loads were applied to the wing and horizontal stabilizer using modified strip theory. Most of the analyses were conducted using theoretical two-dimensional values for the lift and moment on the primary surface due to rotation of the control surface. Later analyses used experimental steady state three-dimensional values for the lift and moment on the primary surface due to rotation of the control surface.

The ALDCS equations were synthesized with the basic airplane equations of motion. Additional generalized coordinates of aileron rotation, elevator rotation, aileron actuator input signal, and elevator actuator input signal were added. The aileron actuator command displacement was considered proportional to the command transducer displacement. The equations of motion were then synthesized with the ALDCS equations to yield the final equations.

Flutter Analysis Results - As the analysis program proceeded and the ALDCS development continued, the task of defining the important parameters and conditions of the ALDCS for stability from flutter continued to lead the program in many directions and caused a continual update of the analysis. The more obvious parameter variations to investigate were the effects of Mach number, altitude, and airplane wing fuel loading. Additional, but not as obvious, parameters were ALDCS gain and phase stability margins and certain failure mode cases.

The results considered more pertinent and of more general interest are summarized in the next few paragraphs.

The effect of Mach number on the flutter speeds was investigated for empty fuel and 30% full fuel conditions at 5000 feet altitude using ALDCS gain schedules associated with velocities of 200 and 350 KEAS and for zero gains (ALDCS-off). The addition of the ALDCS resulted in significant changes in the zero fuel flutter characteristics. See Figure 12. The 3.5 Hz and 5.5 Hz ALDCS-off flutter modes were replaced with 1.6 Hz and 6.4 Hz flutter modes, both with higher flutter margin than the 3.5 Hz basic mode. A more critical ALDCS-on 1.1 Hz flutter mode was also obtained, which was discovered as the analyses continued to be the most critical nominal ALDCS-on mode. This 1.1 Hz mode, which is essentially independent of Mach number, is associated with airplane pitch mode and not with the first wing bending mode. The flutter speed associated with this mode is between 310 and 320 KEAS giving a minimum margin of 110 knots above the 200 KEAS gain schedule. The results obtained using gains associated with 350 KEAS are similar to the 200 KEAS gains but the flutter margin for the 1.1 Hz pitch mode was increased to 140 knots above the 350 KEAS gain schedule. The Mach number effects were also calculated for 30% full fuel condition. For both the 200 KEAS and 350 KEAS gains, the basic 1.8 Hz ALDCS-off flutter mode was stabilized by ALDCS. Again as with 0% fuel condition, the 1.1 Hz mode appeared with approximately the same flutter margins as for the 200 and 350 KEAS gains.

The effects of altitude on the flutter speeds were calculated for the same fuel conditions and ALDCS gain schedules as in the Mach number effects study. For the empty fuel results with 200 KEAS gains, three flutter modes were obtained for ALDCS-off. These flutter modes with frequencies near 2.9 Hz, 3.2 Hz, and 4.6 Hz were obtained at all altitudes. They were not obtained with the ALDCS-on, but three other flutter modes were obtained with frequencies near 1.1, 1.7, and 6.4 Hz. The 1.1 Hz pitch mode again has the more critical flutter speeds. With 350 KEAS gains the same three flutter modes are obtained with the ALDCS-on at altitudes below about 25,000 feet. The ALDCS is not designed to function at 350 KEAS for altitudes greater than approximately 25,000 feet; therefore, no ALDCS-on runs were made at higher altitudes. The only flutter mode obtained for 30% full fuel with no ALDCS is a 1.7 Hz mode. With 200 KEAS gains the ALDCS stabilizes this mode but introduces three additional flutter modes with frequencies near 1.1, 3.0, and 4.2 Hz. The 3.0 and 4.2 Hz modes have relatively high flutter speeds with, again, the 1.1 Hz pitch mode significantly lower than the ALDCS-off flutter speeds but still having a margin greater than 100 knots above the 200 KEAS gains. The 350 KEAS gain schedule runs obtained similar modes as with the 200 KEAS gains.

The results of fuel variations showed that the ALDCS stabilizes the basic ALDCS-off flutter modes. The 1.1 Hz mode, again the lowest flutter mode with ALDCS-on, has a relatively constant flutter speed at all fuel conditions, indicating that this mode is associated with the empennage and not basically the wing. Similar trends resulted for both 200 and 350 KEAS gains.

Investigation of the effect of aileron and elevator ALDCS increased gains on the flutter speed at near 0% wing fuel resulted in an 8.8 Hz wing flutter mode with flutter speeds below the 350 KEAS nominal gain speed for gains more than 2.0 times the nominal aileron gains; and the 1.1 Hz pitch mode below the 350 KEAS nominal gain speeds for gains more than about 2.5 times the nominal elevator ALDCS gains. See Figure 13. Addition of wing fuel rapidly improves the flutter speed of the 8.8 Hz mode but again has little or no effect on the 1.1 Hz pitch mode.

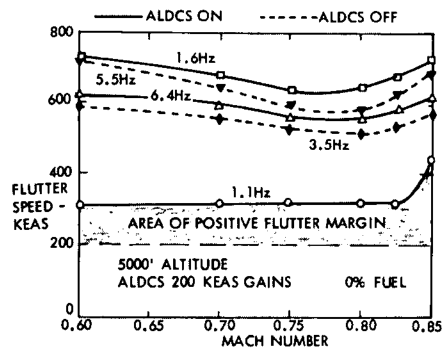

FIGURE 12 Effect of mach number on flutter speed

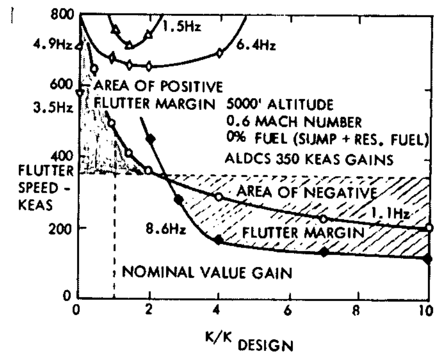

FIGURE 13 Effect of aileron and elevator gains on flutter speed

Variations in aileron or elevator ALDCS phase show that large deviations of phase from the nominal values yield flutter modes with low flutter speeds. As expected, if the aileron phase is varied 60° or more, the basic wing bending mode is driven unstable. When the elevator phase is varied as much as approximately -45° the nominal 1.1 Hz mode is driven to a very low flutter speed. See Figure 14.

Various ALDCS failure modes were analyzed with no significant comments or any notable degradation in the flutter boundaries with the incorporation of ALDCS. The ALDCS is a dual redundant system which uses comparators to disengage the system for most failure conditions affording the most positive flutter prevention possible.

AEROELASTIC/FLUTTER MODEL PROGRAM

The incorporation of the ALDCS into the C-5A necessitated a very comprehensive investigation into its effects on the flutter stability characteristics and a preview into the overall load reducing effectiveness. Theoretical flutter and loads analyses were under way precontractually to establish, at the earliest possible time, these ALDCS structural characteristics. The time frame for the development of the ALDCS into the C-5A had been established on a very tight schedule spanning some ten months to the first test flight. This abbreviated development time precluded the design and fabrication of a new aeroelastic wind tunnel model; therefore, an existing 1/22nd scale complete C-5A flutter model that had been designed and tested at high speeds for compressibility effects during the original basic C-5A design was selected to perform flutter and aeroelastic loads tests.

The ALDCS flutter model tests were planned to investigate for adverse ALDCS/structural coupling and to determine if the flutter margin of the basic C-5A was compromised. The ALDCS aeroelastic model loads tests were to determine the dynamic aileron effectiveness in reducing wing loading and the effectiveness of the proposed sensor locations in the wing and fuselage. In addition, they were to investigate, if possible, optimization of the ALDCS gains and phase and to obtain static aerodynamic test data for correlation with analyses and flight test.

Model Flutter Tests - The C-5A ALDCS speed envelope is bracketed by an airplane speed of 350 knots calibrated airspeed (model dynamic pressure of 50q) and a Mach number boundary of 0.825. See Figure 15. The flutter model tests covered this speed envelope to ensure that the basic airplane's dynamic stability was not degraded by the ALDCS jeopardizing the airplane's already established flutter margins. Time being of essence, the test configurations were limited to two wing fuel loadings (0% and 33% fuel). The 33% loading has fuel primarily in the outer wing. Since the 0% case was originally the most critical fuel loading tested, it was chosen for these tests as the correlation flutter condition between the earlier C-5A tests and the present ALDCS C-5A model.

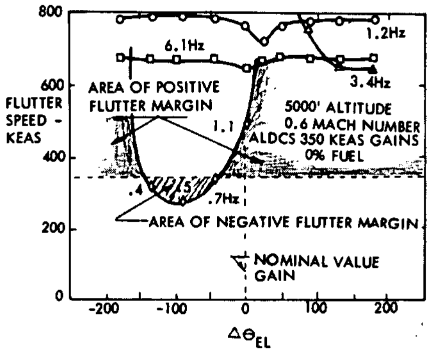

FIGURE 14 Effect of elevator phase angle on flutter speed

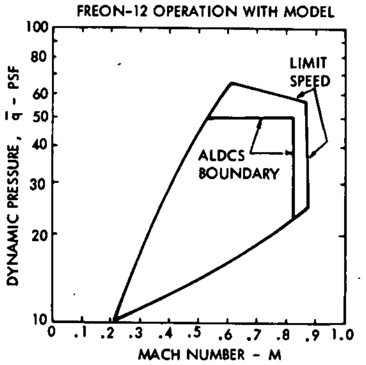

FIGURE 15 C-5A aeroelastic model, flutter boundry MACH - Q scaling

Model Loads Tests - The model program was equally important as a tool in testing the ALDCS loads effectiveness. Static and dynamic air loads were to be measured on the model wing and aileron. To validate the model's aerodynamic and elastic properties, as compared to the full scale airplane, several basic aerodynamic tests were to be performed. Static wing loading tests were conducted by varying angle of attack using the horizontal stabilizer and a lift cable between the model and tunnel floor, and by symmetrically deflecting the ailerons through several angles. Dynamic loads were measured by control surface step and ramp inputs, sinusoidal oscillation of the ailerons and horizontal stabilizer, and model excitation through the tunnel airstream oscillator vanes.

Model Description and Scaling - The basic flutter model was the original C-5A high speed complete model which was scaled for a Mach number ratio of one. This model was rescaled into an aeroelastic Froude number model to correctly represent the acceleration and gravitational forces of the full scale airplane. In other words, the rescaling permitted the model to be tested at the same lift coefficient as the airplane. The basic model construction consisted of hollow and solid metal spars located along the elastic axis of the wing, pylons, fuselage, fin and horizontal stabilizer simulating the desired elastic properties. These spars were covered by balsa wood fairings to achieve the proper aerodynamic shape for the entire model. The fairings were divided into sections which were attached at a single point to the spar and ballasted to simulate the proper section mass properties. Lead weights were attached to the wing spar inside the wing sections and arranged to represent the mass properties of specific wing fuel loadings. The ailerons and horizontal stabilizer were constructed to simulate fully the properties of the full scale airplane. The horizontal stabilizer of the model was used as an active control instead of the inboard elevators as on the airplane. A few additional design details on scaling are presented in Table I.

TABLE I MODEL DESIGN SCALE RATIOS

Item	Symbol	Value
Geometry	b_m/b_a	1/22
Velocity	V_m/V_a	1/4.69
Density	ρ_m/ρ_a	2.65
Dynamic Pressure	q_m/q_a	1/8.3
Frequency	ω_m/ω_a	4.69
Deflection	δ_m/δ_a	1/22
Weight	W_m/W_a	1/4017
Stiffness	EI_m/EI_a	1/1,944,325
Froude Number	F_m/F_a	1/1

Aircraft design point - 415q at 5000 feet for a Mach number of 0.58 (350 KEAS)

Model Hydraulic Powered Controls - As mentioned previously, the model had fully operative ailerons and horizontal stabilizer. These control surfaces were powered by a unique onboard completely self-contained hydraulic system (except for cooling water and electrical power). This hydraulic system drove miniature flapper vane actuators attached to each aileron and a more conventional linear piston actuator attached to the horizontal stabilizer. The hydraulic system consisted of a hydraulic pump with a self-contained reservoir. The model hydraulic system was complex and complicated and completely simulated the full scale airplane.

Model Control System - The model actuators were controlled by separate servo valves, one for each aileron and one for the horizontal stabilizer. These servo valves utilized pressure and position feedback signals from transducers and were controlled by specially designed servo amplifiers. The airplane aileron and inboard elevator transfer functions were modified to reflect the time scale ratio between the model and the airplane. The response characteristics of the control surfaces were quite satisfactorily simulated to 35 Hz in respect to gain ratio and phase lag response. See Figures 16 and 17. As explained earlier, the horizontal stabilizer, which was remotely trimmable when the basic C-5A model was originally tested, was used for these tests to simulate the inboard elevator because of simplicity and cost. The horizontal stabilizer travel, scaled to the elevator travel, was approximately one third. However, using the horizontal stabilizer in place of

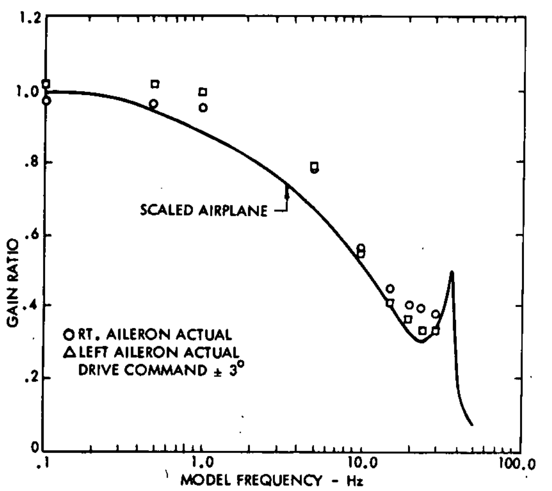

FIGURE 16 C-5A aeroelastic model typical actuator gain ratio

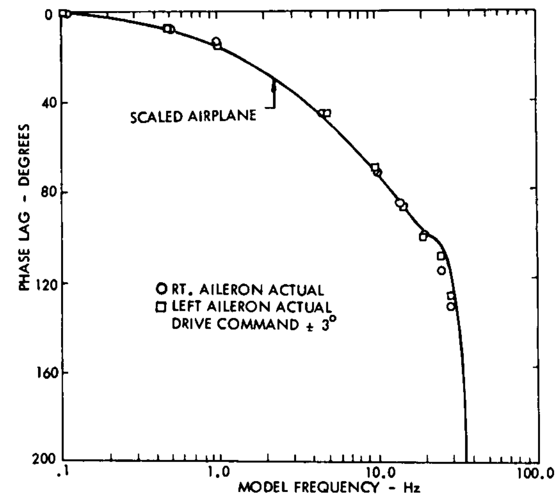

FIGURE 17 C-5A aeroelastic model typical aileron actuator phase lag response

the inboard elevators could develop into a serious flutter or, at best, a serious divergent problem if the hydraulic power was lost. To prevent this condition and the possible loss of the model, the piston type actuator was designed double ended with a spring over each end to force the horizontal stabilizer to remain trimmed with loss of hydraulic or electrical power. As a result of this spring-actuator design, it was more difficult to match the inboard elevator transfer function; nevertheless, the match was considered very acceptable to above 20 Hz and reasonably acceptable to 35 Hz.

Model ALDCS Control Laws - The model ALDCS mechanization was designed to simulate the airplane through the same type sensors and control laws. As with the airplane, the model utilized the pitch Stability Augmentation System (SAS) as a principal part of the mechanization. The model aileron SAS in the normal airplane roll or lateral mode was not represented primarily because the ALDCS is a symmetrically activated system. But, the aileron SAS, as modified for the airplane ALDCS, was represented in the model control laws. The model aileron channel had two wing accelerometers, one located on each wing at the appropriate wing station and chord location. The left and right wing accelerometers were summed to measure only symmetric wing acceleration. The wing signals were then channelled through the appropriate control laws to activate the aileron actuators. The elevator channel of the ALDCS was represented on the model, as on the airplane, primarily by two sensors, a pitch rate gyro and a fuselage accelerometer. These two sensors were used by the ALDCS computer to command horizontal stabilizer deflections to control model pitch response.

Both wing and pitch ALDCS control laws were simulated by an analog computer located in the tunnel control room that took the sensor signals from the model and transformed them to simulate the mechanization of the ALDCS to activate the control surface servo valves. The ALDCS scheduled gains had to be corrected manually for the model whenever the tunnel speed conditions were changed by adjusting three potentiometers on the computer. The ALDCS could be readily switched from engaged to disengaged as the test program dictated or in case of an emergency or instability.

The test plan called for the investigation of stability margins of the ALDCS relative to gains and phase. It proved to be too difficult to devise a method for accomplishing this during the test, not in the gain change which was relatively simple but in the phase change. A phase change for a particular frequency would cause an excessive gain increase at the lower frequencies. Time during the test period was not sufficient to work out an acceptable means to accomplish the phase variations.

Model Instrumentation - Including the ALDCS instrumentation mentioned in the previous paragraphs, the C-5A ALDCS model consisted of 39 channels of measurement or control functions. The measurements on the model were made using strain gages, accelerometers, position indicators, and a pitch rate gyro.

Epoxy-backed foil strain gages were bonded to pre-determined locations on the wings, horizontal stabilizers, and fin to measure bending and torsion moments. The model's right wing had additional strain gages to measure the wing bending and torsional moments at wing stations comparable to the full scale airplane sensor positions: Model W.S. 9.0 (198), W.S. 14.31 (315), W.S. 28.86 (635), and W.S. 41.80 (920). Smaller epoxy-backed foil strain gages were used to measure aileron hinge moments. High output piezo-resistive accelerometers were used to measure wing and fuselage acceleration. Aileron and horizontal stabilizer positions were measured using miniature potentiometers and precision resistor bridge network. The pitch rate gyro was connected to a modulator/demodulator unit also mounted in the model which furnished the output signal for the pitch rate gyro.

The pitch attitude of the model was measured using a d.c. servo-balance accelerometer. The hydraulic control system instrumentation consisted of the previously described control surface position indicators and strain-gage type pressure transducers wired differentially to measure servo-valve output differential pressure. Three channels were devoted to servo valve input signals and two channels were used for hydraulic pump overheat and low oil warning indicators. A miniature d.c. motor connected to the horizontal stabilizer pitch mechanism was used for pitch-trim of the model. The outputs from the signal conditioning amplifiers were connected to FM magnetic tape recorders for a permanent data record as well as to heat pen type strip chart recorders for monitoring and "quick-look" data reduction and analysis. Some of the strain data output was printed by an IBM line printer at each data point.

Model Laboratory Test - The basic model was thoroughly tested for conformity to design requirements. Each spar was individually tested to measure bending and torsion stiffnesses and each spar and section fairing was measured for weight, unbalance and inertia. Model hydraulic system components were individually tested and then the entire system was mocked up with all electronics wired prior to installation in the model. The hydraulic motor was originally shock mounted because it was feared that the motor's starting torque and running would cause excessive vibrations in the model, but these fears proved unfounded.

After the model was completely assembled, a model ground vibration test was conducted with a satisfactory comparison of structural frequencies and mode shapes with the full scale airplane. The model strain gages, especially the bending and torsion gages located along the wing, were calibrated. Gage outputs were plotted as a function of applied load and the gage sensitivity was determined from these plots.

Model Tunnel Installation - The model was tested in the NASA Langley, 16 foot Transonic Dynamics Tunnel located at Hampton, Virginia. The tunnel test medium is Freon-12, a gas approximately four times more dense than air. This heavier gas allows the model to be tested at the higher Mach numbers at lower dynamic pressures permitting simpler model construction and less risk of model damage.

The model was suspended on the two flying cable mount system developed by NASA engineers, see Figure 18. The front cable was in the vertical plane forming a truncated "vee" from pulleys mounted in the model to the wind tunnel floor and roof approximately twenty-five feet upstream of the model. The aft cable had to be rigged in the horizontal plane to clear the model's "Tee-tail" plane. It was attached from the model's aft pulleys to each side of the wind tunnel sidewalls approximately twenty-five feet downstream. Hydraulic actuator powered snubber cables capable of holding the model firmly in the center of the wind tunnel in times of emergencies were attached to the model at one fuselage point. The two cables of the free-flight mount system still contributed their normal restraint when the model was snubbed.

Model Static Loads Program - The initial tests were programmed to measure the static loads of the model as a measure of the model's static aerodynamic similarity to the full scale airplane. The basic model C_{L_A} plotted as a function of fuselage angle of attack was obtained by restraining the model at its c.g. by a cable attached through a load cell to an air piston below the tunnel floor while pitching the model with the horizontal stabilizer. Wing bending, torsion and model lift signals were recorded as a function of pitch angle. Static aileron effectiveness was plotted as a function of wing bending and torsional moments per degree of aileron deflection and for $C_{\ell_{\delta_A}}$ as a function of dynamic pressure for averaged trailing edge up and down aileron deflections.

Model Dynamic Loads Program - The dynamic loads portion of the tests was conducted by using the ailerons and horizontal stabilizer as an excitation source. This means of excitation was also planned as one of two methods of exciting the full scale airplane. The model ailerons were oscillated from 2 to 20 Hz with and without static aileron uprig (PLDCS configuration) and the horizontal stabilizer was oscillated from 0.5 to 20 Hz. The ailerons were also oscillated with the model restrained in pitch with the snubber cables and by fore and aft vertical fuselage cables between the model and tunnel floor and ceiling. Restraining the fuselage while oscillating the ailerons would be used for correlating wing oscillatory strain with analytical results by eliminating load relief through the fuselage response. A third means of exciting the model was with the tunnel airstream vanes, Figure 18. The vanes were mounted in pairs on either side of the tunnel side walls at the entrance to the tunnel at the settling chamber. The vanes, when oscillated, provided a sinusoidal gust field to the model through shed vortices off each vane tip. The tunnel vanes were set to oscillate through 6° amplitude and were driven from 0.7 to 16 Hz either symmetrically or antisymmetrically. The vanes were normally driven symmetrically to excite the ALDCS symmetric response modes but were driven antisymmetrically on several occasions to ensure that, in fact, the ALDCS did not respond to antisymmetrical excitation. The tunnel vanes excited the model very well at the lower frequencies but did not provide much excitation above 14 Hz. Motion picture data which was taken throughout all the tests best demonstrated the effectiveness of the ALDCS to the gust response. During all the dynamic tests, data was taken with the ALDCS active and inactive.

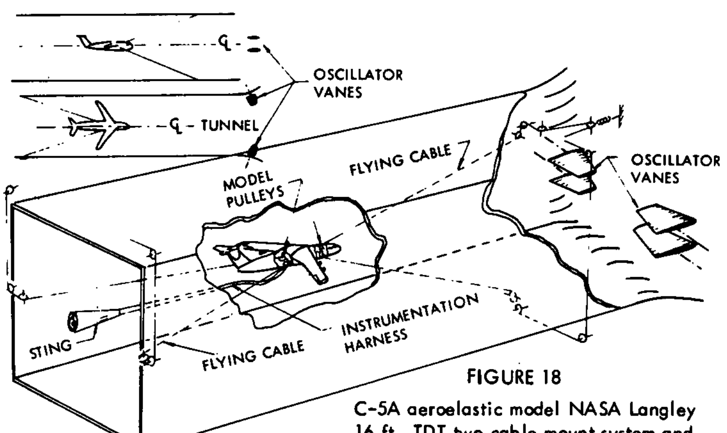

FIGURE 18
C-5A aeroelastic model NASA Langley 16 ft. TDT two cable mount system and tunnel oscillator vanes

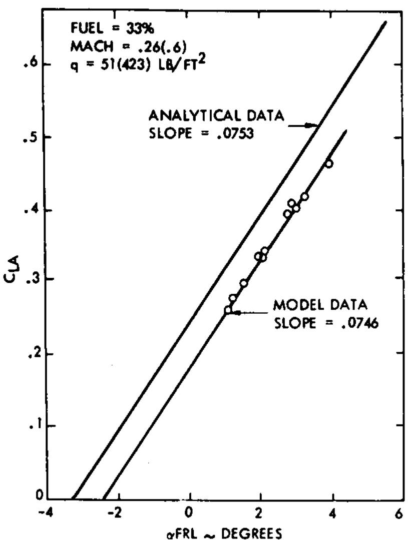

FIGURE 19 C-5A aeroelastic model trimmed lift vs. angle of attack

Model Flutter Test Program - The model was tested to the ALDCS Mach number and dynamic pressure limits to investigate any adverse effects the ALDCS might have on the existing flutter stability of the basic airplane. At each predetermined test point the ALDCS was engaged and disengaged. An aileron step input was executed for each condition and the model damping response characteristics were recorded.

Model Static Loads Results - The model was tested first for static loads, to determine the static lift characteristics of the model and to correlate with the C-5A airplane's analytically derived lift data. For the static lift portion of the test the model was restrained with a single vertical cable as explained earlier to measure the model lift versus the model angle-of-attack. The total model lift and the wing bending and torsion moments along the wing were recorded for each pitch angle. The tests were conducted for 33% full wing fuel at two dynamic pressures, 41 lb/ft^2 (341 equivalent airplane) and 51 lb/ft^2 (423). Excellent agreement was obtained for the change (slopes) of lift, wing bending moment and torsion versus model angle-of-attack between the model and analytical airplane data, see Figure 19. The curves of the model data showed a rather small discrepancy in the angle of zero lift, and bending and torsion moment of approximately one degree angle-of-attack. The dynamic loads and flutter test results are dependent on the lift curve slope rather than the curve intercept. This small discrepancy is probably accounted for by slight misalignment of the model sections or fuselage reference line.

Test results were plotted as curves of bending and torsion moments at various wing stations, respectively, versus angle-of-attack of the fuselage reference line, see Figures 20 and 21. The slopes of the bending moment and torsion curves were used to generate distributed bending and torsion moment per degree of angle-of-attack along the wing.

Static load test results with aileron uprig (passive LDCS configuration) favorably compared the model aileron effectiveness with the analytical airplane data, see Figure 22. The tests were conducted for aileron uprig settings 0°, 5°, and 10°. The model roll coefficient was determined by extrapolating the bending moment and torsion data to the model center line. Difficulties in scaling control surface gap and seals to the full scale airplane cause some loss in model control surface effectiveness.

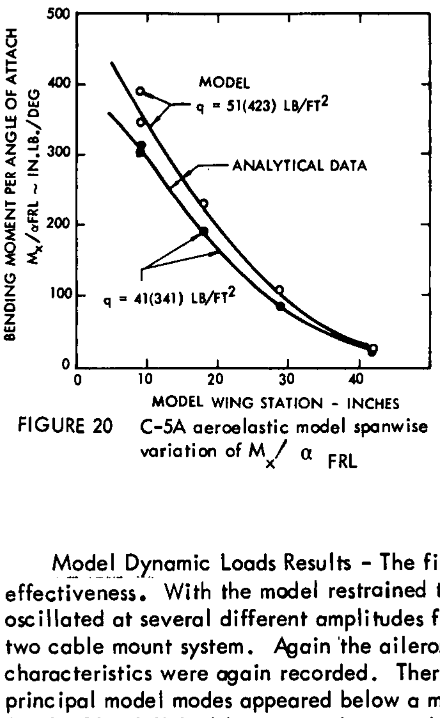

FIGURE 20 C-5A aeroelastic model spanwise variation of M_x/α_{FRL}

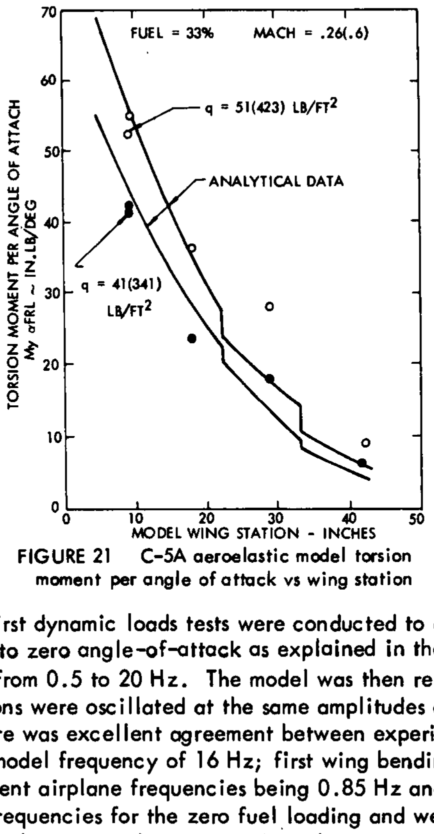

FIGURE 21 C-5A aeroelastic model torsion moment per angle of attack vs wing station

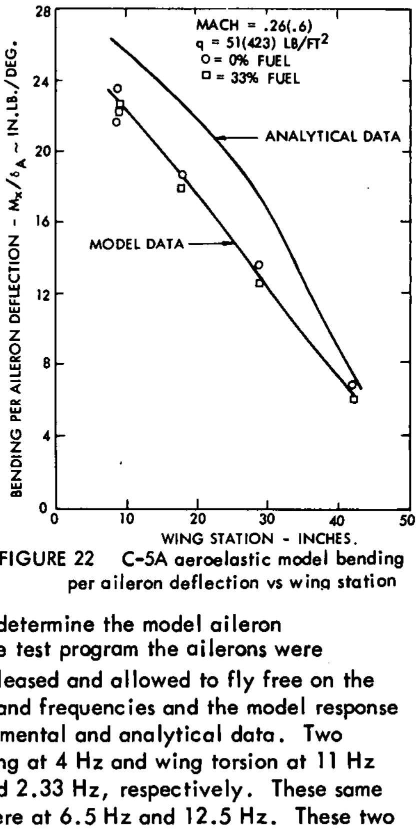

FIGURE 22 C-5A aeroelastic model bending per aileron deflection vs wing station

Model Dynamic Loads Results - The first dynamic loads tests were conducted to determine the model aileron effectiveness. With the model restrained to zero angle-of-attack as explained in the test program the ailerons were oscillated at several different amplitudes from 0.5 to 20 Hz. The model was then released and allowed to fly free on the two cable mount system. Again the ailerons were oscillated at the same amplitudes and frequencies and the model response characteristics were again recorded. There was excellent agreement between experimental and analytical data. Two principal model modes appeared below a model frequency of 16 Hz; first wing bending at 4 Hz and wing torsion at 11 Hz for the 33% full fuel loading with equivalent airplane frequencies being 0.85 Hz and 2.33 Hz, respectively. These same modes appeared at slightly higher model frequencies for the zero fuel loading and were at 6.5 Hz and 12.5 Hz. These two modes are particularly significant since the lower wing bending mode is the prime structural mode the ALDCS was designed to control and the higher wing torsion mode is the basic wing flutter mode. Both of these modes will be shown for the full scale airplane later in this paper in the sections on ground vibration and flight flutter tests. Additional dynamic response tests of the model were made by inducing step and ramp inputs into the aileron commands. Flight test data will be available too late in the program to present comparisons in this paper.

It seems appropriate in this section to discuss the model test results with analytical data. In comparing the aileron position as a function of frequency during an aileron sweep, it was observed that, with ALDCS-off, the aileron deflection steadily decreased with increasing frequency with the analytical and test data tracking exceedingly well. With ALDCS-on, both analytically and experimentally, the aileron position had a sharp dip in amplitude at the first wing bending mode and a less pronounced dip at the torsion mode. Both data responded similarly with the analytical data showing slightly more delta deflection at both modes, see Figure 23. In plotting the model wing tip acceleration for the same aileron frequency sweep, the ALDCS reduced the wing bending mode by approximately 60% in acceleration. The analytical acceleration data predicted a similar 60% reduction in wing tip acceleration but the analytical data with ALDCS-off response was 30% less than the experimental data, see Figure 24. The 11 Hz torsion mode for the model showed no change with or without ALDCS for nominal gains but, for an increased gain, the experimental data showed a very slight degradation very similar to the analytical data. The 1.6 gain increase was the only attempt at checking gain changes of the ALDCS on the model stability.

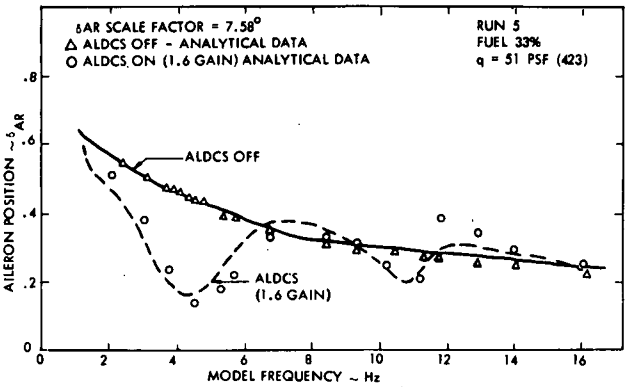

FIGURE 23 C-5A aeroelastic model aileron frequency sweep aileron deflection

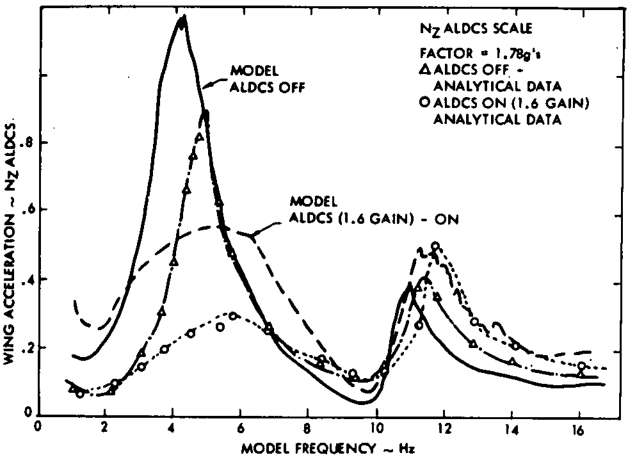

FIGURE 24 C-5A aeroelastic model aileron frequency sweep wing acceleration

Plotting wing root bending moment for the same aileron sweep produced very much the same comments as in describing the wing tip acceleration results; again an effectiveness of more than 60% in wing root bending moment was achieved with ALDCS on.

Since the wing torsion moments were a design consideration for ALDCS, it was important that the torsional moments not be increased as a function of ALDCS. Using the same aileron frequency sweep, a plot was made of the wing torsion moments. The model and analytical data essentially showed elimination of the torsion moments as a resultant of first wing bending and significant improvement of torsional moments at the model wing 11 Hz torsion mode. However, the analytical data showed comparable first wing bending mode (4.5 Hz) torsional moments to be greater than the model data but less at the 11 Hz mode, see Figure 25. Similar comparison can be made for the other wing stations recorded.

The model experimental data recorded for the zero fuel loading condition was very much the same as for the 33% fuel loading and any further explanation would be repeating previous description, see Figure 26. However, the zero fuel data, in almost every instance, showed the ALDCS to be as effective in achieving load and response reductions.

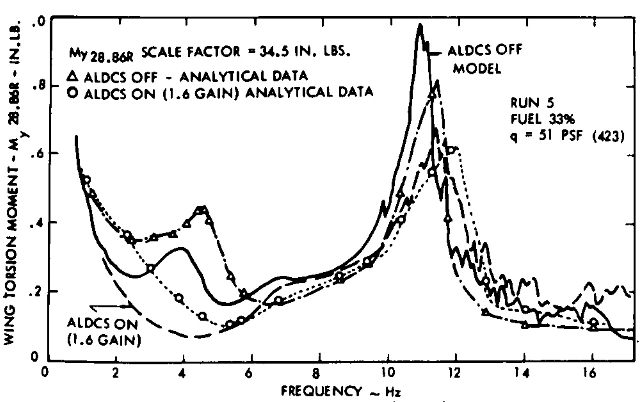

FIGURE 25 C-5A aeroelastic model aileron frequency sweep wing torsion moment

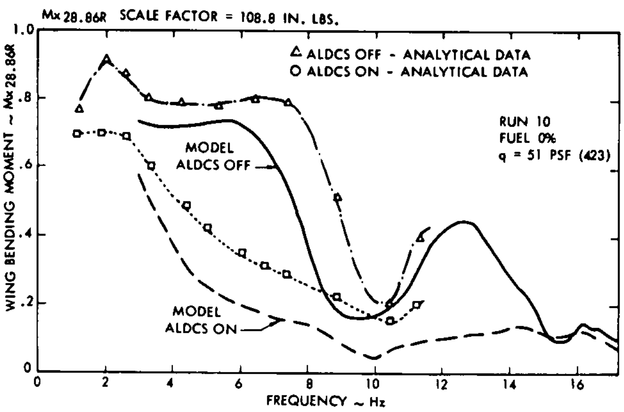

FIGURE 26 C-5A aeroelastic model aileron frequency sweep wing bending moment

Additional dynamic loads tests were conducted by exciting the model with the horizontal stabilizer and the wind tunnel aerodynamic vanes upstream of the model. The vanes which simulated a sinusoidal gust field to the model were oscillated from 0.5 Hz to 16 Hz. The vane produced a gust velocity decreasing exponentially as a function of frequency at a tunnel dynamic pressure of 41 lb/ft^2 (341 lb/ft^2 equivalent airplane dynamic pressure) from approximately 2.4 ft/sec at 1.0 Hz to less than 0.1 ft/sec at 10 Hz. The tunnel vanes excited primarily the first wing bending mode and the lower rigid body plus cable mount modes. The results of the ALDCS to the vane excitation were very comparable to the aileron frequency sweeps. The horizontal stabilizer did not excite the model as well as either the ailerons or tunnel vanes.

The tunnel airstream oscillators were phased to excite the model antisymmetrically. Since the ALDCS is designed to function only for symmetric wing excitation, no analysis had been performed for antisymmetric excitation, since any antisymmetric condition analytically would not be affected by ALDCS; therefore, this condition was investigated experimentally to ensure that no adverse results would develop through the ALDCS with antisymmetric excitation. As designed, the ALDCS had no effect when the model was excited antisymmetrically.

The ALDCS responsive model mode evaluation was the only part of the test plan not fully completed. The original plan, as explained earlier, was to attempt to determine how sensitive the stability of certain modes was to gain and/or phase by varying the control laws. The gain through the ALDCS had previously been increased by a factor of 1.6 but a convenient or meaningful way to shift phase at one mode without greatly affecting gains of other modes could not readily be executed. The determination of gain and phase margins are vital to the overall stability of the ALDCS and this type of testing should be thoroughly investigated for future programs.

Up to this point in the test plan all the conditions were conducted with 33% wing fuel. Several aileron frequency sweeps were now made for the 0% fuel loading at the model test design point. Several more wing modes were excited with the ALDCS off as compared to the 33% wing fuel loading. With ALDCS on, the first wing bending mode, as with the 33% fuel loading, was greatly suppressed; however, the higher modes were significantly affected. In the case of the second mode, it was either completely suppressed and either another mode 14.6 Hz (3.2 Hz airplane) was excited or the mode was shifted from 12.7 to 14.6 Hz. The possibility that one mode was suppressed and another mode excited seems more probable. The third and fourth modes reappeared at approximately the same frequency and amplitude as with ALDCS off. The 0% fuel loading is an unrealistic flight condition and all of the higher modes become more heavily damped with the addition of fuel.

As an experiment to determine the sensitivity of the 0% fuel wing modes to ALDCS, an earlier discarded control law was set up on the computer and an aileron frequency sweep was made. Once again the wing first bending mode was heavily suppressed; the 14.6 Hz (3.2 Hz airplane) mode, however, approached an alarming amplitude as its critical frequency was passed; continuing to increase the frequency, the 16.3 Hz (3.48 Hz airplane) mode began diverging at such a rapid rate that the ALDCS was cut off to avoid destroying the model. The experiment was discontinued.

Model Flutter Test Results - The Mach number and dynamic pressure boundary were expanded to cover the C-5 flight conditions to determine if a destabilization of any modes might occur with ALDCS as a function of Mach number or density. The model having the equivalent of 33% wing fuel was flutter tested to the ALDCS speed limits. The tunnel density was initially decreased to an equivalent pressure level of 40,000 feet. With the model unsnubbed, the Mach and dynamic pressure were increased while maintaining a constant pressure level up to the ALDCS design boundary. The model became increasingly less stable in a model mount Dutch roll type mode and could not be flown unsnubbed much above approximately 0.75 Mach number; nevertheless, it was flown to higher Mach numbers snubbed. Preliminary stability analyses of the model on the flying cable mount had been conducted, utilizing the model target mass and inertial data, to aid in determining the model pulley locations and cable tensions. The model flight characteristics were predicted to be marginally stable on the free-flight mount throughout the Froude number scaled C-5A/ALDCS operational boundary. Stability analyses, conducted later for the final model mass properties as opposed to the target data, predicted this higher Mach number mount stability problem. Since no solution to this instability was evident from the analysis parameter study and since test time was at such a premium, no attempt was made to experimentally change the mount system. It should be noted here that no model SAS was used to control the Dutch roll mode. However, the test was considered as a valid indication of ALDCS margin stability snubbed or unsnubbed since it had been determined from previous tests that fuselage restraint had little effect on the wing symmetric or antisymmetric flutter modes. Several pressure density lines were similarly tested down to near sea level. No degradation of any modes was noted with ALDCS on throughout the speed altitude boundary.

The last portion of the flutter tests was conducted for zero fuel. The 0% fuel had been proven from the original C-5A tests to be most critical standard fuel loading from a flutter standpoint. It was, therefore, chosen for these tests as the correlation flutter condition between the earlier C-5A tests and the present ALDCS C-5 model. Several flutter points were obtained in a 13 Hz (2.77 Hz airplane) antisymmetric mode, see Figure 27. This mode and the Mach-dynamic pressure boundary plotted check extremely well with the original model test results and substantiated the validity of the flutter results of this test.

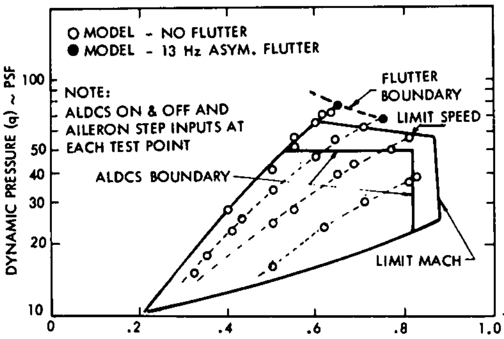

FIGURE 27 C-5A aeroelastic model zero fuel flutter test points

The model flutter stability for 0% and 33% fuel conditions was tested with ALDCS off and then on at each increasing Mach-dynamic pressure test point with the model being excited by a 5° aileron step for each condition until flutter or the desired maximum speed was reached. The ALDCS appeared to have no flutter destabilizing characteristics but it is interesting to note that, at each test point that flutter occured, the model fluttered antisymmetrically after the ALDCS was turned on. The ALDCS, being a symmetric configuration, would not degrade an antisymmetric condition but may actually have damped a near flutter symmetric mode, thereby permitting the antisymmetric mode to flutter.

In summary, the C-5 aeroelastic/flutter model test with ALDCS appears to be an excellent representation of the full scale airplane and to have provided very comprehensive data as to the loads effectiveness and flutter/dynamic stability of the ALDCS as it is designed and used on the model. The model wing first bending moments were reduced comparable to the analysis and flight test data while the model torsional moments were only slightly affected with no serious degradation to the flutter parameters.

FLIGHT TEST DEVELOPMENT PROGRAM

The ground and flight test program for the ALDCS involved seven basic parts:

- o Ground Vibration Test - Open and closed frequency response data and investigation of any adverse coupling of any ALDCS characteristics with the basic airplane system were conducted.

- o Flight Development - The major portion of this phase was performed first and was used to determine if the system was functioning as designed and to determine if the loads reductions were as predicted (i.e., were the system gains approximately correct).

- o Flight Flutter - After the system checkout and preliminary gain checks, the flutter portion of the test was conducted as final substantiation for freedom from flutter or other instabilities and to clear the system for more detailed development.

- o Final Flight Development - After the aircraft with ALDCS was cleared from flutter more detail frequency response runs were conducted at higher speeds.

- o Flight Demonstration - This phase of the flight test program was conducted simultaneously with the final development phase. The objective of this phase was to show that the ALDCS achieved (1) proper load relief for maneuver, (2) that the system worked satisfactorily in conjunction with the automatic control systems originally on the aircraft and did not significantly alter handling qualities.

- o Flight Simulated Failure Modes Effects - The flight test program was to show that simulated failures associated with the ALDCS would not cause response of the aircraft such that the pilot could not control the airplane easily or cause degradation of stability. This portion of the program was conducted simultaneously with the 4 initial tests named above.

- o Flight Dynamic Response Test - The last phase of the ALDCS flight test program and one of considerable importance to the overall evaluation of the ALDCS effectiveness is the dynamic gust response tests which are scheduled for completion in July 1974.

The following paragraphs go into some detail in presenting the results of the ground and flight test program listed above. It should be noted, however, that the failure tests demonstrated that there are no failures that can occur in the ALDCS which will give the pilot* significantly more work load or subject the aircraft to significantly more response than similar failures not associated with the ALDCS. Actually in some failure conditions the aircraft response is not as large when ALDCS is operative.

The flight test aircraft was C-5A Air Force serial number 66-8305 (Lockheed ship number 0003). The aircraft was a fully instrumented aircraft which has been used for many programs during engineering tests and demonstration of the C-5A, but was primarily instrumented to demonstrate the aircraft loads. The aircraft was equipped with one of the three prototype ALDCS computers built for this development program. All of the ALDCS computers were checked on the Vehicle System Simulator prior to installation on the airplane to verify uniformity of fabrication. Instrumentation was added to the test aircraft so that various points of the ALDCS computer could be constantly monitored by telemetry throughout the tests.

The flight test program was initially estimated to be approximately 154 flight hours. The major portion of the test program was conducted at the Lockheed-Georgia Company flight test facility. The flight hours estimated for these home based tests were 60 hours, actually 61.25 hours were required; however, timewise the program was completed one week ahead of schedule after starting two weeks late. Inflight refueling and dynamic gust response flight tests were being conducted at the Air Force Flight Test Center, Edwards AFB, California, during the writing of this paper.

Airplane Ground Vibration Test - Prior to first flight of the C-5A airplane with ALDCS, a ground vibration test was conducted to ensure that, on the ground, there would be no adverse effects of the ALDCS on the airplane resonant modes and to demonstrate that no hydraulic-electrical-structural coupling with ALDCS would cause control instabilities.

The C-5A, because of its tremendous size and weight, was tested on its landing gear. The tires and struts were deflated as much as practical to lower the airplane's rigid body frequencies. However, vibration analyses had to be conducted with the airplane supported on its gears because the rigid body gear frequencies significantly coupled with the normal airplane structural modes.

The instrumentation used on the test was purposely planned to duplicate the equipment used on the initial C-5A shake test, plus the addition of certain ALDCS instrumentation. The principal excitation system consisted of dual electro-hydraulic shakers, each capable of 2400 pound force output, attached vertically to the fuselage nose jack pads. A second excitation system consisting of two electromagnetic shakers, each capable of 150 pounds force output, was attached to the wing tips. Accelerometers were attached to the airplane wing tips, empennage, fuselage, and engines to measure airplane response. The ALDCS sensors, the ALDCS computer outputs to the Stability Augmentation System (SAS) through which the control surfaces are commanded, and the control surface position signals were also monitored and recorded.

Prior to commencing the externally forced excitation through the shakers a thorough open and closed loop frequency response test was conducted. A sinusoidal force signal was input through the SAS to activate the ALDCS in the open loop and then closed loop mode. The ALDCS response characteristic had been thoroughly tested by this time on the analog computer and on the Vehicle System Simulator ("iron bird") but this was the first opportunity to test the ALDCS gain and phase response versus frequency on the C-5A airplane.

*A real-time pilot in-the-loop flight simulation program was conducted for development of the flight control system, evaluation of the resulting aircraft flying quality characteristics, and for pilot reaction to simulated ALDCS failure modes.

After the internal response tests, frequency sweeps were made with the external shakers from 0.5 Hz to 30 Hz. Frequency sweeps were made for the basic airplane, basic airplane with SAS, basic airplane with SAS and ALDCS, and basic airplane with SAS, ALDCS, and autopilot. To ensure sufficient gain margins, additional sweeps were made for increased ALDCS gains. Both the aileron and elevator system gains were doubled individually, first on the aileron sweeps and repeated for the elevator sweeps; then both system gains were increased by 1.4 for the final frequency sweeps. The ground vibration tests showed excellent agreement with the original basic airplane tests with no indications of undesirable structural coupling between ALDCS and the airplane.

Airplane Loads Flight Test - The initial development portion of the flight test program was started April 18, 1974. The initial flight of the C-5A, with ALDCS installed, was really a shakedown flight for the basic aircraft since numerous changes had been made to the airplane during the several months lay-up prior to the first ALDCS flight. The flight did, however, accomplish several tests with the ALDCS on, but disengaged (i.e., open loop). This allowed the engineers time before the second flight to thoroughly check the open loop response of the ALDCS before it was actually engaged to affect the aircraft response. The open loop tests included ground checks of elevator and aileron deflections, frequency responses where the ailerons and inboard elevators were oscillated through the frequency range of the ALDCS, and various flight maneuvers to load factors up to 2g. The open loop response of the ALDCS first flight did indicate that all outputs from the computer were very near to the desired frequency response characteristics.

During the second flight and the flights thereafter, the ALDCS was flown closed loop after the initial open loop tests were made to determine that the ALDCS computer was functioning properly. The initial development flights included tests which were run ALDCS-on but disengaged prior to actual test points with ALDCS-on and engaged. This practice allowed engineers the opportunity to check the ALDCS computer output prior to actually performing the ALDCS engaged test. The object of the initial development flights was to determine that the analytical gains K_A (aileron channel) of the ALDCS computer were sufficient to produce the desired load reductions. Figure 28 shows the load changes for a maneuver condition for a typical flight test point. This particular set of data was obtained by flying a series of coordinated turns at the indicated load factors. The effect of the ALDCS was to reduce incremental wing root bending by 42% while the corresponding torsion was increased by 29%. This phase was completed prior to the flutter tests so that the upper limit of the ALDCS gains could be defined for use in the flutter tests. The initial development flight verified the analytical work that had been done for the system mechanization. The flutter tests were to clear the aircraft to fly throughout the C-5A ALDCS flight envelope with the ALDCS operative.

After it was determined that the maneuver loads criteria had been satisfied, one of the most important aspects of the ALDCS to investigate was its effect on the basic C-5A handling qualities. The primary term in the mechanization involving handling qualities is the compensating elevator deflection required to cancel the pitching moments caused by aileron deflections. Several test conditions at widely separated flight conditions and aircraft loading configurations were considered in this evaluation. Considerable data analyses and pilot judgment were required to determine that analytically derived values of K_e (elevator channel gains) were acceptable.

Flight test results and pilot evaluation indicated that there were no significant changes in C-5A handling qualities. The basic C-5A aircraft short period and phugoid damping were very high before the addition of the ALDCS; therefore, the increased damping associated with the ALDCS was not detectable by the pilots. A concern of installing the ALDCS originally was that the C-5A roll power authority would be degraded since the ailerons would have somewhat less available travel while in a turn; however, roll degradation proved not to be significant. Finally, stick force 'g' characteristics and ease of trimming the aircraft at a new speed was not altered significantly. A summary of the influence of the ALDCS on handling qualities of the C-5A can be indicated by saying that the pilots did not know whether ALDCS was on or off for flight conditions within the normal flight envelope.

Final development work was actually combined with the demonstration portion of the flight test program since the initial development work had shown no need for any significant gain or system changes. Many tests were performed which verified the desired steady maneuver load conditions by flying the aircraft in wind-up turns and obtaining steady load factors of various values. This type of maneuver was also used to determine the stick-force per 'g' data to evaluate handling qualities. Slow pull-ups and push-overs were also flown to evaluate the maneuver load reduction. Stability and system frequency response characteristics were evaluated by aileron and elevator frequency sweeps and compared with similar analytical data. These comparisons show very good correlation, see Figure 29. The demonstration of load reduction for

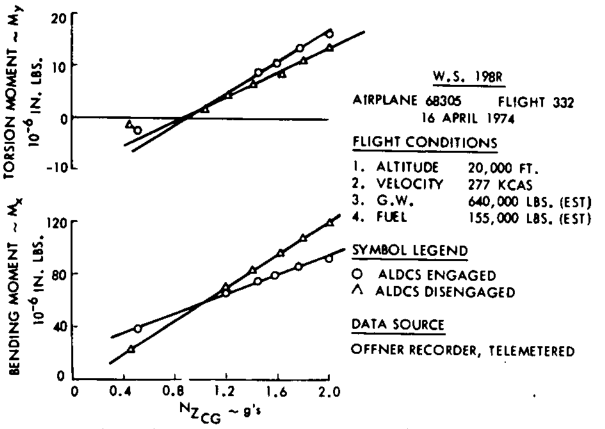

FIGURE 28 Effect of ALDCS on wing bending and torsion - recorded flight test data

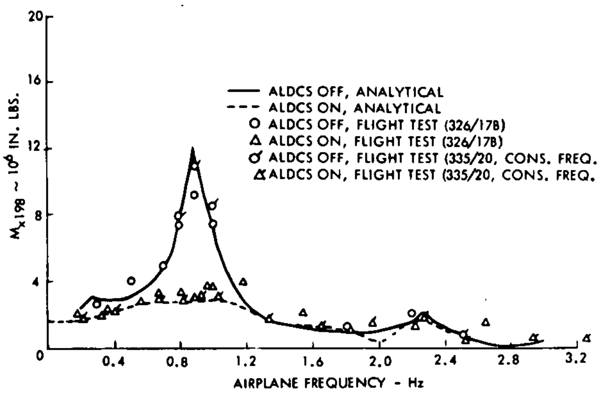

FIGURE 29 Wing bending moment comparison for aileron frequency sweeps - recorded flight test data

abrupt maneuver initially indicated that the requirement for the stick-to-aileron cross feed circuit would not be required because the load reductions for aircraft forward center of gravity positions (the initial tests) were greater than anticipated by analog analysis. The more critical condition, high dynamic pressure and aft center of gravity, more closely agreed with analog results. Therefore, the analytical values of the cross feed gain and delay functions (K_A and ψ_A) were used in the system.

The ALDCS fader, the portion of the mechanization which removes the aileron command after a load factor of 1.9g is obtained, is the only portion of the system which caused adverse pilot comment. The purpose of the fader, as mentioned previously, is to remove the ALDCS commanded aileron deflection when the aircraft reaches 1.9g to permit sufficient time to have all of the deflection removed before 2.5g is reached. A time delay built into the fader requires that the aircraft load factor remain at 1.9g or above for one half second to prevent the ALDCS disengagement due to turbulence. The disengagement at 1.9g does give a transient into the aircraft when the ailerons are commanded to return to the PLDCS position instantaneously. The inboard elevators are also commanded to return to the normal turn position but with the one half second time delay they do not return at maximum rate. The fader, depending on aircraft center of gravity and flight condition, caused the aircraft to experience as much as a 0.2g change in load factor. This transient, due to the very few number of times it will occur in service, was not considered to be excessive and the fader configuration was not changed. As indicated before, however, the 1.7g load factor at which the ALDCS was reengaged was changed to 1.3g because during one flight the system was forced into a limit cycle. It was resolved that the transient associated with disengaging the ALDCS was large enough to excite the basic wing bending mode which, due to oscillation, lowered the computed load factor to a value less than the 1.7g and the system would immediately be reengaged causing the cyclic motion. The lower reengage load factor setting has no significant effect on the system and was probably lowered considerably more than required to avoid the limit cycle.

The ALDCS/autopilot compatibility was another flight test demonstration flown. This was primarily a handling qualities test. The portion of the ALDCS which had to be checked was the value of Kp (pitch column signal) required for the system. This portion of the ALDCS was similar to the autopilot and was set to zero when the autopilot was on. Values of Kp were tried between zero and the value currently used when ALDCS only is operative. It was concluded that the least ALDCS effect on C-5A handling occurred with K_p set to zero.

The values of K_e (inboard elevator channel gain) and K_A (aileron channel gain) in the system were defined primarily by maneuver and gust loads analyses and were verified by early flight tests for maneuver loads. Frequency response sweeps seemed to verify that the gust analyses would substantiate the operation of the ALDCS in a gust environment.

The dynamic response test program, scheduled for completion in July 1974, will obtain data during continuous turbulence inputs with ALDCS on and off to demonstrate the capability of the system to produce gust load reductions. The scheduled program consists of twelve flight conditions, two fuel weights, two altitudes, and three speeds.

Airplane Flight Flutter Tests - The flight flutter tests were conducted for the same test conditions as those of the original C-5A flight flutter tests. The flight flutter instrumentation was in addition to the ALDCS monitored channels and was specifically planned to monitor strategic locations on the airplane. Accelerometers were located on the engines, wing tips, and empennage. All control surfaces were instrumented with position potentiometers; strain gages and ALDCS channels were used for secondary monitoring. Thirty channels selected prior to each flight were telemetered. Flutter, loads, and control system engineers recorded the telemetered data and directed and approved each test point before proceeding to the next condition. In addition, all flights were accompanied by an Air Force chase airplane with a pilot and an engineering observer.

All flight flutter tests were conducted for two standard fuel sequences (i.e., 100% to 67% and 40% to 10%). The heavy fuel case was tested at 22,400 feet and the light fuel case was tested at 22,400 feet and 35,000 feet. An incremental speed build-up was made for each test condition to speeds in excess of the ALDCS speed limits as an added stability margin. The ALDCS gains were also increased by 25% to ensure that sufficient margins would be allowed for system tolerances and for any slight adjustments that may become desirable.

Several methods were employed to excite the airplane structure and control surfaces during the flutter program. The aerodynamic rotating vanes that were used in the original C-5 flight flutter tests were reinstalled on the ALDCS airplane as the primary source of excitation. The rotating vanes were a double wedge paddle that rotated about a center shaft mounted between two pylons. These pylons were mounted on top of the wing at the wing tip closing rib and 21% wing chord. The vanes were synchronized between both wing tips and were driven by hydraulic motors. The vanes were approximately 12" by 18" and produced about 1500 pounds of lift and drag force on the wing tip at maximum dynamic pressure obtained during the flight flutter tests. The vanes were driven through a frequency sweep at each test point incremental speed from 0.5 Hz to 25 Hz. Since the vane force increased as a function of dynamic pressure, each accelerometer monitored was normalized by vane drag force and plotted as acceleration per drag force versus airspeed. The vane frequency sweep time in the original C-5A flutter tests was optimized as a sixty second logarithmic sweep. However, for the ALDCS sweeps the logarithmic sweep time was increased to ninety seconds to allow more time in the very low frequencies for structural amplitude buildup.

A second method of exciting the airplane was through applying a sinusoidal drive signal to the Stability Augmentation System controlling the control surfaces in the same manner that open and closed loop responses had been conducted. However, this method of excitation was used only in the initial buildup of airspeed and check-out of the ALDCS functions. The third method of structural excitation was by pilot induced control surface pulses. The pulses, primarily aileron and elevator inputs, were executed at each test point and followed by frequency sweeps of the rotating aerodynamic vanes.

At each incremental speed test point, the following sequence of events occurred. Control surface pulses followed by vane frequency sweeps were made with the airplane in the basic configuration with ALDCS disengaged; SAS on, ALDCS engaged; and with SAS, ALDCS, and autopilot engaged. The entire flight test program, including ALDCS development and flutter, was monitored and directed through telemetry. Running plots of acceleration versus airspeed were made for selected instrumentation channels.

The ALDCS boundary logic (350 KCAS or Mach number 0.825) is scheduled to disengage the ALDCS at 350 KCAS at the test altitude of 22,400 feet. But the flutter margin was to be cleared to 370 KCAS. The boundary logic was tested by increasing the airspeed past the 350 KCAS limit (0 to +5 knot tolerance) until the boundary logic disengaged the ALDCS, then decreasing airspeed until the ALDCS re-engaged (0 to 5 knot tolerance). Once the logic was proven, the ALDCS speed override switch was disengaged and the flutter test with active ALDCS was conducted at incremental speed points of 350 and 370 KCAS.

The first test configuration at 22,400 feet with a nominal heavy standard fuel loading showed principally the first wing bending mode at 1.0 Hz, a torsion 2.5 Hz mode, and several higher modes including the aileron rotation/outer wing torsion mode of 8.5 Hz. Significant effects in reducing the wing bending moments and wing tip accelerations were recorded with ALDCS engaged versus the basic airplane at the wing first bending frequency of nominally 1.0 Hz with little or no change noted in the wing torsional moments. Similar load reductions were noted for both fuel conditions tested at the 22,400 foot test altitude, but the load reducing capability of the ALDCS showed even greater effects on the first bending mode for the 35,000 foot light fuel loading test condition, see Figures 30, 31, and 32.

Modes of 2.5 Hz and higher, excluding the 8.5 Hz aileron mode, were not effected by the ALDCS in either bending or torsional moments. The aileron rotation/outer wing torsion mode did indicate a very slight increase (10% or less) in activity with ALDCS engaged but did not increase its activity with airspeed and was, therefore, not considered significantly degrading to the system. The 1.0 Hz pitch mode that was the most predominant mode in the flutter analyses with ALDCS did not appear during the flight test program.

Certain failure modes that would not cause the two comparator channels to switch the ALDCS off and were undetectable by the flight crew were flight flutter tested with frequency sweeps for stability.

No conditions were noted during the ALDCS development phase, load determination, or the flutter tests that would indicate that the system, as designed and installed on the C-5A airplane, will significantly degrade its stability.

Concluding Remarks - In this paper the development of the C-5A ALDCS to date in regards to background history, design, loads and flutter analyses, aeroelastic wind tunnel model tests and airplane flight test development programs have been described. The background history is described in the earlier load alleviation studies that have been conducted on the C-5A. The ALDCS design requirements, goals, and functions have been discussed. The loads analyses is thoroughly outlined, describing methods and results which give assurance that the ALDCS will provide the wing load relief required without degrading performance or handling qualities. The flutter analyses have investigated all of the various parameters such as wing fuel, compressibility effects, and ALDCS gains and phase that may affect the flutter characteristics of the C-5A. A vivid description with details of the aeroelastic wind tunnel model program has been presented with results correlated to the theoretical loads analysis. The data obtained indicates that much useful information can be obtained from aeroelastic model tests.

The flight test program has been discussed with reference to the airplane ground vibration test through the flight ALDCS development, loads and flutter programs. The ALDCS results presented show that the desired C-5A wing load relief is achieved without any degradation to performance, existing handling qualities, stability or flutter margins of the airplane.

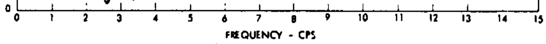

FIGURE 30 C-5A/ALDCS flight flutter test symmetric frequency sweep vs wing tip vertical amplitude relative response

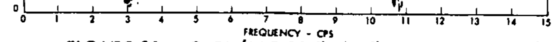

FIGURE 31 C-5A/ALDCS flight flutter test symmetric frequency sweep vs wing tip vertical amplitude relative response

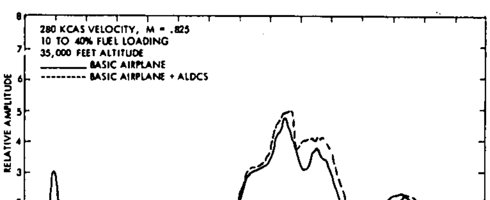

FIGURE 32 C-5A/ALDCS flight flutter test symmetric frequency sweep vs wing tip vertical amplitude relative response

References

1 Burris, P. M., Bender, M. A., "Aircraft Load Alleviation and Mode Stabilization (LAMS), C-5A System Analysis and Synthesis", Technical Report, AFFDL-TR-68-162 (Nov. 1969)

2 Firebaugh, John M., "Evaluation of Spectral Gust Model Using VGH and V-G Flight Data", Journal of Aircraft, Volume 4, Number 6 (Nov. - Dec. 1967)

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15. Abstract

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